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INFLUENCE OF LOADING DISTRIBUTION ON THE PERFORMANCE OF HIGH PRESSURE TURBINE BLADES

by

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B. Eng. (Mechanical Engineering)

M. Eng. (Applied Mechanics)

A THESIS SUBMITTED TO THE FACULTY OF GRADUATE STUDIES AND RESEARCH
IN PARTIAL FULFILMENT OF THE REQUIREMENTS FOR THE DEGREE OF

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IN
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**INFLUENCE OF LOADING DISTRIBUTION
ON THE PERFORMANCE OF HIGH
PRESSURE TURBINE BLADES**

SUBMITTED BY DANIEL CORRIVEAU, B. ENG., M. ENG.
IN PARTIAL FULFILMENT OF THE REQUIREMENTS FOR
THE DEGREE OF DOCTOR OF PHILOSOPHY

THESIS SUPERVISOR, S.A. SJOLANDER

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EXTERNAL EXAMINER

Carleton University
Ottawa, Canada
July 2005

Abstract

Midspan measurements were made in a transonic wind tunnel for three High Pressure (HP) turbine blade cascades at both design and off-design incidences. Comparisons with two-dimensional numerical simulations of the cascade flow were also made. The baseline profile is the midspan section of a HP turbine blade of fairly recent design. It is considered mid-loaded. To gain a better understanding of blade loading limits and the influence of loading distributions, the profile of the baseline airfoil was modified to create two new airfoils having aft-loaded and front-loaded pressure distributions. Tests were performed for exit Mach numbers between 0.6 and 1.2. In addition, measurements were made for an extended range of Reynolds numbers for constant Mach numbers of 0.6, 0.85, 0.95 and 1.05.

At the design exit Mach number of 1.05 and at design incidence, the aft-loaded airfoil showed a reduction of almost 20% in the total pressure losses compared with the baseline airfoil. Based on the analysis of wake traverse data and base pressure measurements combined with numerical results, it was found that the poorer loss performance of the baseline mid-loaded profile compared to the aft-loaded blade could be attributed to the former's higher rear suction side curvature, which resulted in higher flow velocity in that region, which, in turn, contributed to reducing the base pressure. The lower base pressure at the trailing edge resulted in a stronger trailing edge shock system for the mid-loaded blade. This shock system increased the losses for the mid-loaded baseline profile when compared to the aft-loaded profile.

On the negative side, it was also found that as Mach numbers were increased beyond the design value the performance of the aft-loaded blade deteriorated rapidly. Under such conditions, the front-loaded airfoil showed generally inferior performance compared with the baseline airfoil.

At off-design incidence, the aft-loaded blade maintained a superior loss performance over a range of incidences extending from about -5.0° to $+5.0^\circ$ at design outlet

Mach number. For incidences outside that range, the baseline mid-loaded profile had a better loss behavior than the front-loaded and aft-loaded blades.

Résumé

Des mesures ont été prises dans le plan médian de trois grilles d'aubes de turbine à haute pression dans une soufflerie transsonique pour des angles d'attaque nominaux et non-nominaux. Le profil de base correspond au plan médian d'une aube de turbine à haute pression de conception récente. Elle est considérée comme ayant une charge aérodynamique mitoyenne. De manière à accroître notre connaissance des limites de charge aérodynamique et l'influence de la distribution de la charge, le profil de l'aube de base a été modifié de manière à créer deux nouvelles aubes ayant des charges aérodynamiques arrière et frontale. Des essais ont été effectués pour des nombres de Mach à la sortie de la grille variant de 0,6 à 1,2. De plus, des mesures ont été prises pour une plage étendue de nombres de Reynolds à des nombres de Mach constants de 0,6, 0,85, 0,95 et 1,05.

Au nombre de Mach nominal de 1,05 et à incidence nominale, l'aube avec charge aérodynamique arrière a démontré une réduction de près de 20% de perte de pression totale comparé à l'aube de base. En se basant sur une analyse des relevés de sillage et les mesures de pression de base combinées avec des résultats numériques, il a été démontré que l'efficacité inférieure de l'aube de base à charge aérodynamique mitoyenne comparée à l'aube à charge aérodynamique arrière pouvait être expliquée par la courbure arrière plus élevée de l'aube de base. La présence d'une courbure arrière plus élevée sur l'extrados a résulté en une vitesse d'écoulement plus élevée dans cette région. Une vitesse d'écoulement plus élevée au bord de fuite contribua à son tour à réduire la pression de base. La présence d'une pression de base inférieure au bord de fuite a produit un système d'ondes de choc plus fort. Ce système de choc a augmenté les pertes pour l'aube de base à charge mitoyenne en comparaison à l'aube à charge arrière.

Cependant, il fut aussi vérifié que pour des nombres de Mach supérieurs au nombre de Mach nominal, les pertes de pression totale de l'aube à charge arrière aug-

mentent rapidement. L'aube à charge frontale a démontré une performance généralement inférieure en comparaison avec le profil original.

À des angles d'attaque non-nominaux, l'aube à charge arrière maintient une performance supérieure pour des angles d'attaque variant de $-5,0^\circ$ à $+5,0^\circ$ au nombre de Mach nominal. Pour des angles d'attaque au-delà de cette plage, l'aube de base à charge mitoyenne a démontré des performances supérieures comparé aux deux autres aubes.

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List of Symbols

Symbols

A	Area or calibration constant
A^*	Throat area
$AVDR$	Axial Velocity Density Ratio, $\left(= \frac{\int_0^1 (\rho_2 C_{(ax)_2})_{MS} d(y/s)}{\int_0^1 (\rho_1 C_{(ax)_1})_{MS} d(y/s)} \right)$
AVR	Axial Velocity Ratio, $\left(= \frac{\int_0^1 (C_{(ax)_2})_{MS} d(y/s)}{\int_0^1 (C_{(ax)_1})_{MS} d(y/s)} \right)$
a	Leading edge ellipse major axis
b	Leading edge ellipse minor axis
B	Calibration constant
C	Blade chord length
C_b	Base pressure coefficient, $\left(= \frac{(P_b - P_2)}{q_2} \right)$
C_p	Static pressure coefficient, $\left(= \frac{((P_1)_y - P_{1r})}{q_{1r}} \right)$
C_{ps}	Static pressure coefficient, $\left(= \frac{((P_s)_x - P_1)}{q_1} \right)$
C_{po}	Total pressure coefficient, $\left(= \frac{((P_{o2}) - P_{o1})}{q_2} \right)$
C_x	Axial chord length
d	Leading edge diameter
E	Anemometer output voltage
E_R	Measured calibration voltage
E_C	Calculated voltage using King's law
H	Blade span or shape factor
i	Incidence, $(= \alpha_1 - \beta_1)$
$i_{effective}$	Effective incidence, $(= i - i_{des})$
i_s	Stall incidence

K_c	Compressibility correction factor
K_p	Compressibility correction factor
K_{TE}	Trailing edge correction factor
K_1	Coefficient
K_2	Coefficient
L	Length-scale of the energy containing eddies
m	Transducer calibration curve slope
M	Mach number
Ma	Mach number
m/A	Mass flow rate per unit area
n	Calibration constant
N	Number of sample
o	Throat opening
P	Pressure
P_b	Base pressure
P_o	Total pressure
P_s	Static pressure
q	Dynamic pressure, $\left(= \frac{1}{2}\rho V^2\right)$
R	Radius of curvature or gas constant
Re	Reynolds number, $\left(= \frac{\rho V_2 C}{\mu}\right)$
Re	Reynolds number based on unit length (D=30.5cm)
s	Blade pitch or spacing
SES	Sum of error square
t	Trailing edge thickness or blade thickness or time
t_m	Blade maximum thickness
T	Temperature
T_I	Integral time scale
Tu	Turbulence intensity
u	Fluctuating velocity component or velocity
U	Flow velocity
U_R	Velocity calculated from the pressure difference

U_C	Velocity obtained from calibration equation
V	Flow velocity or transducer voltage
w	Distance between bounding streamlines
W	Inlet duct width
We	Wedge angle
x	Axial distance
y^+	Distance in wall coordinates
Y'_{TET}	Trailing edge loss coefficient
Y_{te}	Trailing edge loss coefficient
Y_p	Profile loss coefficient
Y_t	Midspan total pressure loss coefficient, $\left(= \frac{(P_{01} - P_{02})}{q_2} \right)$
z	Output voltage at zero absolute pressure
Z	Compressible Zweifel loading coefficient, $\left(= \frac{s}{C_x} \cos^2 \alpha_2 (\tan \alpha_1 + \tan \alpha_2) \frac{\gamma M_{2s}^2}{\left[1 + \frac{\gamma-1}{2} M_{2s}^2 \right]^{\gamma/(\gamma-1)} - 1} \right)$

Greek Symbols

α_o	Coefficient of resistivity
α	Flow angle measured from the axial direction
β	Metal angle measured from the axial direction
β	Pitch angle
δ	Boundary layer thickness or deviation
δ^*	Boundary layer displacement thickness
$\Delta \phi_{TET}^2$	Change in trailing edge kinetic energy loss coefficient
$\Delta \phi_p^2$	Change in profile kinetic energy loss coefficient
γ	Specific heat ratio
ε_u	Normalized standard deviation
μ	Air dynamic viscosity
ω	Frequency
Ω	Axial Velocity Density Ratio, AVDR

ϕ_1, ϕ_2	Fundamental quantities
ρ	Air density
ρ_u	Autocorrelation coefficient
τ	Time delay
θ	Boundary layer momentum thickness or misalignment angle
θ_u	Uncovered turning angle
ξ	Energy loss coefficient, $(= (h_2 - h_{2s})/(h_{o2} - h_2))$
χ	Profile loss incidence parameter
χ'	Profile loss incidence parameter
ζ	Stagger angle measured from the axial direction or energy loss coefficient

Subscripts

o	Total
1	Cascade inlet
2	Cascade outlet
2m	Measured
ax	Axial
c	Chord
crit	Critical
des	Design
e	Free stream condition
is	Isentropic
le	Leading edge
max	Maximum
min	Minimum
MS	Midspan
r	Reattachment or reference
s	Isentropic or separation
t	Transition inception or trailing edge

T	Transition completion
te	Trailing edge
TE	Trailing edge
tr	Transition

Abbreviations

AMDC	Ainley and Mathieson and Dunham and Came
CFD	Computational Fluid Dynamics
DFVLR	Deutsche Forschungs- und Versuchsanstalt für Luft- und Raumfahrt
HP	High Pressure
HS1A	High Speed 1A
HS1C	High Speed 1C
HS1D	High Speed 1D
IBLT	Inlet boundary layer thickness
LDG	Loading
LP	Low Pressure
M	Midspan Loss
P	Profile Loss
S	Secondary Loss
T/S	Tollmien/Schlichting

Chapter 1

Introduction

Stiff competition for market share in the gas turbine industry combined with the ever more stringent requirements for clean, efficient engines, as required by governments around the world, has pushed gas turbine manufacturers to optimize every component of the engine. Gains can still be made in terms of reduction in losses as well as in weight of the engine in the High Pressure (HP) turbine.

This can be achieved by carefully designing blade profiles such that minimum losses are generated within the passage while at the same time maximum loading is developed in order to reduce the number of airfoils required per stage. To succeed in designing such optimized blades, it is necessary to determine the maximum allowable amount of diffusion on the blade surface. The location of the diffusion region together with its extent must also be considered. The diffusion limit is imposed by the need to avoid separation of the boundary layer on the blade surface which is associated with high losses, especially if the separated flow does not reattach. To create a successful blade design, one would like among other things to raise the peak Mach number on the blade's suction surface as much as possible without incurring a complete boundary layer separation.

Designing highly-loaded HP turbine blades within the diffusion limits requires a knowledge of the effect of pressure gradient on boundary layer separation and transition under turbomachinery conditions. The importance of the pressure gradient on the boundary layer transition has been recognized for a long time. However, given the fact the Reynolds numbers encountered in HP turbines are much higher than in Low Pressure (LP) turbines, transition is not as large an issue for these airfoils. On the other hand, a good understanding of the effect of the pressure gradient on boundary layer separation of highly-loaded HP turbine airfoils is of great importance

as separation is frequently observed on the surface of these blades. Even nowadays boundary layer separation under adverse pressure gradient is not fully understood and a significant amount of research is still being performed in order to develop a better knowledge of the phenomenon.

In addition to requiring a good understanding of the viscous effects, designing highly loaded HP turbines demands a good understanding of shock-boundary layer interaction since the flow through the HP turbine stage of modern aero-engines is transonic.

Experimental studies in turbine aerodynamics usually involve either cascade or cold flow rotating rig testings. Although the flow in a cold flow rotating rig is closer to that of an actual engine, the flow in a planar cascade has the advantage of a simpler flow structure, which facilitates a basic understanding of the flow physics. Cascade tests are used, among other things, to examine the effects of variations of flow conditions and blade geometry on losses. In addition to the midspan losses, secondary and tip clearance flows can also be studied in cascades. The focus of this study is primarily on the midspan flow and the resulting losses in a plane cascade.

The research project upon which this thesis is based is part of a collaboration involving Pratt & Whitney Canada and Carleton University. The program has been initiated in order to gain a better understanding of the flow physics through HP turbine blade passages at transonic Mach numbers and low Reynolds numbers. The ultimate objective of the program is to improve the aerodynamic performance of turbine blades through the development of improved design rules.

The present study has investigated the influence of the loading distribution on the aerodynamic performance of HP turbine blades. Detailed cascade measurements were performed in the Pratt & Whitney Canada high speed wind tunnel at Carleton University. Midspan loss measurements at both design and off-design incidences were made on three cascades for a range of transonic outlet Mach numbers and Reynolds numbers. The three blade profiles consist of a baseline mid-loaded blade, HS1A, an aft-loaded blade, HS1C, and a front-loaded blade, HS1D. Blade HS1A is the midspan section of the HP turbine blade of a small gas turbine engine of recent design. Using Pratt & Whitney Canada design tools, the author modified this blade profile in order to obtain aft- and front-loaded profiles having the same inlet and outlet velocity

triangles and Zweifel loading coefficients.

The thesis has been divided as follows. The next chapter reviews the work published in the open literature pertaining to the subject of this thesis. In Chapter 3, a description of the Pratt & Whitney Canada high speed wind tunnel is given together with a presentation of the instrumentation used. The geometry of the three blade profiles used for this research project is also described. The measurement and data reduction procedures are described in Chapter 4, together with an explanation of the experimental error analysis. Computations performed with a three-dimensional Navier-Stokes code were also used to assist in interpreting the experimental results. This code is described in Chapter 5. Chapter 6 presents a study that was performed in order to quantify the flow quality in the high speed wind tunnel. The results obtained at design incidence are presented and analyzed in Chapter 7. The results at high Reynolds numbers are discussed first and then the effect of Reynolds number is investigated. In Chapter 8, results obtained at off-design incidences are discussed. Finally, conclusions drawn from the present study and recommendations for future work are given in Chapter 9.

Chapter 2

Literature Review

2.1 Introduction

The objective of the current study is to investigate the influence of the chordwise loading distribution, Mach number, incidence and Reynolds number on the performance of three HP turbine blades. This chapter reviews the work published in the open literature pertaining to the current study. It begins with a general description of the flow and loss breakdown in turbine blade passages. The aerodynamic parameters influencing the losses at both design and off-design incidences are then discussed. Finally, a review of the different profile loss correlation for axial turbines is presented.

2.2 Flow Description Within an Axial Turbine Blade Row

The flow structure in a turbine passage is very complex and to some extent is not fully understood, as exemplified by the continual publication of new studies such as those of Benner et al. (2004) and Pullan (2004). Therefore, before examining specific issues related to turbine cascade flow, it is worthwhile to review briefly the main features of the flow generally.

Figure 2.1 is a diagram showing schematically the main features of the flow structure in a turbine rotor passage. Among the main flow features is the horseshoe vortex which results from the separation of the endwall boundary layer around the blade leading edge. The horseshoe vortex has two legs: the pressure side leg and the suction side leg. As the pressure side leg of the horseshoe vortex enters the blade passage, it is driven towards the suction side of the adjacent blade, due to the cross-

Figure 2.1: Diagram of the Flow in a Turbine Rotor Passage (Adapted from Jeffries, 2001).

passage pressure gradient. The passage vortex is the result of the pressure side leg of the horseshoe vortex and the endwall boundary layer rolling up due to the passage pressure gradient.

For a rotor, when present, the tip clearance causes part of the flow to be driven through the gap by the pressure difference between pressure and suction surfaces of the blade. This flow results in the formation of what is commonly known as the tip leakage vortex on the suction side of the blade. However, in the case of a stator blade row or a cascade there is no tip clearance and the flow structure at both ends of the blade would correspond to the hub flow shown in Figure 2.1.

2.3 Loss Breakdown for an Axial Turbine Blade Row

The overall losses in a turbine blade row are influenced by a number of factors. The losses have been traditionally broken down in several components for convenience (see for example Denton (1993) or Mee et al. (1992)). The different loss components in a turbine blade row can be listed as follows:

1. **Boundary layer losses**
2. **Trailing edge losses**
3. **Shock losses**
4. **Endwall or secondary losses**
5. **Tip leakage losses**
6. **Annulus losses**

Each of these components is influenced by the flow conditions, the geometry of the blades as well as by some of the other sources of losses.

The **boundary layer losses** correspond to the useful mechanical energy degraded within the blade surface boundary layers. In transonic flow, this would also include the additional losses arising from the interaction of shock waves with the blade surface boundary layer.

The **trailing edge losses** arise due to the finite thickness of the blade's trailing edge. As the pressure and suction surface flows reach the trailing edge, they experience a sudden increase in area because of the trailing edge finite thickness. The mixing of the flow that ensues at the trailing edge and downstream in the wake has a significant impact on the losses.

Shock losses start to become significant as the sonic condition is approached. When the free stream flow goes through the shock waves present in the blade passage a reduction in the flow total pressure occurs. This contributes to the losses.

As explained in the following section, the sum of these three loss components corresponds to the midspan losses. Some researchers or organizations also refer to these three loss components together as the profile losses.

Endwall or secondary losses correspond to the losses generated in the annulus boundary layers between the blade leading edge and trailing edge, together with the losses generated by the mixing out of the passage vortex as it is convected downstream.

Tip leakage losses are a consequence from the flow over the rotor blade tips or the hub clearance of stator blades. Tip clearance flow has three sources of losses associated with it: viscous losses within the gap, mixing of the gap flow with the passage flow and finally mixing of the tip leakage vortex downstream in the flow.

The **annulus losses** are due to the boundary layer losses on the hub and casing outside the blade row.

There are many other sources of losses in turbomachines. Among those, there is the unsteady losses, partial admission losses and disk windage losses. Most of these are relatively small in most circumstances and are usually estimated as a net decrease in stage efficiency, as opposed to a total pressure loss.

2.4 Losses in Two-Dimensional Transonic Cascade Flow

The present research focusses on two-dimensional transonic cascade flow. The interest is in midspan losses. Therefore, only the following types of losses will be considered in this research:

1. **Boundary layer losses**
2. **Trailing edge losses**
3. **Shock losses**

The extent of the secondary flow will also be briefly examined since it can have a significant impact on the assumption of two-dimensional flow.

Figure 2.2: Variation of Midspan Loss Components with Mach Number for a HP Turbine Blade (Mee et al., 1992).

Mee et al. (1992) performed a study on the midspan losses in a transonic high pressure turbine cascade. Using detailed measurements of the boundary layer on the aft portion of the suction surface of the blade as well as wake traverse data, they were able to isolate the individual loss components. Their results indicate that the relative contribution of each loss component varies in importance depending on the Mach number. The breakdown in losses as obtained by Mee et al. (1992) for their high pressure turbine blade is shown in Figure 2.2 and the details of the cascade used are given in Table 2.1.

It can be observed from the figure that the boundary layer losses are the major source of energy dissipation at low Mach number. The mixing losses (trailing edge losses) account for a significant proportion of the losses as well at low Mach numbers. With increasing Mach number, the shock losses gain in relative importance. Near the sonic condition, the contributions from the three loss components are approximately equal. For exit Mach numbers greater than 1.0, the mixing losses increase significantly to account for more than half of the losses at an exit Mach number of 1.2. It is worth noting, as mentioned by Mee et al., that shock losses occurring downstream of the traversing plane are accounted as mixing losses. The reduction in shock losses

Parameter	
Chord Length, C	100.0 mm
Axial Chord, C_x	76.9 mm
Aspect Ratio, H/C	3.0
Inlet Metal Angle, β_1	42.8°
Outlet Metal Angle, β_2	68.0°
Trailing Edge Thickness, t	2.0 mm
Solidity, s/C	0.84
Inlet Mach Number, M_1	0.31
Exit Mach Number, M_2	0.92
Inlet Total Temperature, T_{o1}	285K
Inlet Turbulence Intensity, Tu	4.1%
Exit Reynolds Number, Re_2	1,000,000

Table 2.1: Blade Geometry and Test Conditions Used by Mee et al. (1992).

observed for the higher Mach numbers is due to the fact that the shocks are more oblique as the Mach number increase.

The different loss mechanisms are seldom independent from one another. As an example, the trailing edge shock of a blade impinging on the suction surface of an adjacent blade will contribute to both the shock losses and probably the boundary layer losses. The direct shock losses arise from the mid-passage flow crossing the trailing edge shock. Additional profile losses can arise from separation of the boundary layer resulting from the adverse pressure gradient cause by the impinging shock on the suction surface.

2.5 Aerodynamic Influences on Losses at Design Incidence

2.5.1 Mach Number

The need for high stage work requires high rotor speed in order to limit the aerodynamic loading. Furthermore, high HP turbine rotor speeds are required in

order to generate the high pressure ratio needed from the HP compressors. The presence of high rotor speed results in high transonic Mach numbers in modern engine turbine stage.

This, in turn, results in the development of shock waves in the turbine passages that contribute to the losses in two ways. First, through the shock-boundary layer interaction that can have a significant impact on the boundary layer losses. The second problem associated with the presence of shock waves in the blade passages is the direct losses associated with the entropy rise across the shock. This is usually categorized as the shock losses. This component of loss will be discussed in more detail below.

Usually, when a weak shock interacts with a boundary layer a separation bubble will form at the foot of the shock. For laminar flow, most of the time transition occurs within the separation bubble. The presence of a separation bubble always causes additional dissipation within the bubble. However, if the separated flow reattaches the additional loss is usually fairly small. In the event a strong shock wave is present complete boundary layer separation may occur. Thus, increases in boundary layer losses are likely to arise from shock wave-boundary layer interaction in turbines.

The problem of shock boundary layer interaction has been the subject of much analytical and experimental work. This is mainly due to the difficulty in predicting the effect of the shocks on transition and separation. Delery and Marvin (1986) published a comprehensive review of the phenomena.

The interaction between an impinging shock wave and a blade surface boundary layer can be summarized briefly as follows with the help of Figure 2.3. This figure shows a typical shock boundary layer interaction. This schematic shows the compression waves upstream of the incident shock which combines to form the separation shock. The compression waves and the separation shock are the result of flow deflection around the separation bubble. After passing through the separation shock the boundary layer flow encounters the incident shock wave followed by its reflected expansion waves. As the flow reattaches, a second series of compression waves forms and coalesce to generate the reattachment shock. The reattachment shock is the result of flow realignment with the surface after the separation bubble.

The second problem associated with the presence of shock waves in a turbine

Figure 2.3: Schematic of Shock-Wave Boundary Layer Interaction (Adapted from Japikse and Baines, 1994).

passage is the development of losses arising from the presence of the shocks themselves. As the Mach number increases a shock system develops as follows in the blade passage. Shocks appear in the blade passage at subsonic outlet Mach numbers as low as 0.6. Depending on the inlet flow angle and the blade geometry at the leading edge, a supersonic bubble can form on the suction side close to the leading edge. This supersonic bubble is usually closed through a compression shock.

For transonic but subsonic outlet Mach numbers in the vicinity of 0.9, the flow is strongly accelerated along the suction side. As a result of the flow expansion, another supersonic region develops just downstream of the throat and closes with a shallow oblique shock as shown in Figure 2.4.

With increasing Mach number, the shallow oblique shock moves downstream until it reaches the trailing edge at an outlet Mach number of 1.0, as shown in Figure 2.5. Also, shown in this figure is the nearly normal shock that forms at the throat at sonic condition.

For increasing supersonic exit flow Mach number, the nearly normal shock at the throat becomes more oblique and its impinging point on the adjacent blade rear suction surface moves downstream towards the trailing edge as shown in Figure 2.6.

Figure 2.4: Shock System in a HP Turbine Blade Passage at an Outlet Mach Number of 0.9 (Detemple-Laake, 1989).

Figure 2.5: Shock System in a HP Turbine Blade Passage at an Outlet Mach Number of 1.0 (Detemple-Laake, 1989).

Figure 2.6: Shock System in a HP Turbine Blade Passage at an Outlet Mach Number of 1.25 (Detemple-Laake, 1989).

It is at this stage that the most significant shock boundary layer interactions occurs in the turbine passage. In parallel to that, the shallow oblique at the trailing edge shock becomes more oblique and is now part of the trailing edge shock system to be described in the following section. The shallow oblique shock propagates in the wake and therefore does not interact with the adjacent blade surface boundary layer.

As mentioned previously, the presence of shock waves in a blade passage results in compression and shock losses which are not desirable features for turbines. However, the disadvantages of having compression and oblique shocks, which generate moderate losses provided the upstream Mach numbers are kept to low supersonic values, is more than offset by the added benefit of high stage loading.

There exists relatively few data on the performance of turbine blades in the transonic flow regime in the open literature as summarized in Table 2.2. Among the studies available in the open literature, those pertaining specifically to HP turbine blade are a minority, as shown in the table.

Table 2.2: Summary of Experimental Investigations on the Effect of Mach Number and Incidence on Losses.

Author(s)	Type of Test	Flow Condition	Loss Component(s) Investigated	Parameters Investigated						Conditions for Experimental Investigations				Comments
				$I_i(^{\circ})$	Re	Ma	Tu	IBLT	LDG	Incidence ($^{\circ}$)	Re x 10^{-5}	Ma	Tu (%)	
Baines et al. (1986)	HP Turbine Nozzle; Linear Cascade	Steady	M			✓				design	5.0	0.3-1.2	4.0	-three blade profiles scaled to the same pitch
Boyle et al. (2002)	Turbine Vane; Linear Cascade	Steady	M			✓	✓			design		0.3-0.9	1.0, 10.0	
Delempre-Laae (1991)	HP Turbine; Linear Cascade	Steady	M			✓				design		0.9-1.25		-shock wave boundary layer interaction and secondary flow structures depend on exit Mach number and inlet flow angle
Graham & Kost (1979)	Linear Cascade	Steady	M	✓		✓				-20.0 to 8.0		0.7-1.4	1.0	-measurements for two cascades
Haller & Camus (1984)	HP Turbine Rotor; Linear Cascade	Steady	M			✓				design	8.5	0.9-1.25	0.5	-study effect of film cooling on performance
Hashimoto & Kimura (1984)	LP Turbine Vane; Linear Cascade	Steady	M	✓		✓				-25.0 to 25.0		0.2-1.1		-forward loaded profile
Hoholsel et al. (1987)	Linear Cascade	Steady	M	✓	✓	✓	✓		✓	-38.0 to 12.0	1.0-11.0	0.3-0.95	0.8, 2.7, 5.1, 7.1	-type of loading influence total pressure losses -aft loaded blades lead to lower losses
Jeffries (2001)	Linear Cascade	Steady	M	✓		✓				-10.0 to 15.0		0.5-1.2	4.0	-same blade profile as Kiock et al (1986)
Jouini et al. (2002)	HP Turbine; Linear Cascade	Steady	M	✓		✓				-10.0 to 14.5		0.5-1.25	4.0	-study influence of leading edge metal angle
Kiock et al. (1986)	Linear Cascade	Steady	M			✓				design		0.3-1.3		-measurements with same profile in four different profiles
Li et al. (2002)	Steam Turbine; Linear Cascade	Steady	M	✓		✓				-25.0 to 35.0		0.4-1.15		-propose a modified shock loss correlation
Mee et al. (1992)	Linear Cascade	Steady	P, M			✓	✓			design	5.0-20.0	0.6-1.2	4.1	-investigate contribution of different loss components
Moustapha et al. (1993)	Turbine Vane; Linear Cascade	Steady	M			✓				design		0.3-1.3		
Oldfield et al. (1981)	Linear Cascade	Steady	M	✓		✓				-40.0 to 40.0		0.3-1.1		-study three cascades having different solidity
Song et al. (2004)	Steam Turbine; Linear Cascade	Steady	M	✓		✓				-60.0 to 30.0		0.2-0.8		-study three cascades having different solidity
Stastny et al. (1997)	Linear Cascade	Steady	M	✓		✓				-70.0 to 30.0		0.6-1.5		

Abbreviations: P : Profile loss   I : Incidence   Tu : Turbulence Intensity
M : Midspan loss   Re : Mach number   IBLT : Inlet boundary layer thickness
S : Secondary loss   Ma : Mach number   LDG : Loading

2.5.2 Base Pressure

This section reviews the shock system at the trailing edge that results from the finite trailing edge thickness. Such a shock system is illustrated schematically in Figure 2.7. Large trailing edge loss can arise because of the low base pressure formed immediately behind the trailing edge. The flow expands around the trailing edge to this low pressure. It is then recompressed by the strong trailing edge shock located at the point where the pressure and suction side flows meet. Intense viscous dissipation at the edges of the separated flow region immediately behind the trailing edge occurs. Energy dissipation occurs also due to the presence of strong shocks that form in this region. This gives rise to significant entropy generation.

Denton (1993) has shown using a control volume analysis for incompressible flow that the trailing edge losses could be related to the base pressure coefficient and the boundary layer parameter as follows:

$$Y_{te} = -\frac{C_b t}{o} + \frac{2\theta}{o} + \left(\frac{\delta^* + t}{o}\right)^2 \quad (2.1)$$

where θ and δ^* are the boundary layer parameters at the trailing edge, and C_b is the base pressure coefficient, which is a function of the base pressure (see Nomenclature).

The first term on the right side of Equation 2.1 corresponds to the loss arising from the presence of low base pressure at the trailing edge. This portion of the trailing edge losses is directly proportional to the base pressure and the thickness of the trailing edge.

The second term in this equation represents the losses due to entropy generation in the blade boundary layer. This term indicates that the trailing edge losses are also a function of the boundary layer momentum thickness.

Finally, the third term in Equation 2.1 accounts for the loss due to the sudden area expansion in the trailing edge region.

It should be noted that the contribution from the base pressure in compressible flow is expected to be even larger than what is predicted by Equation 2.1.

Xu and Denton (1987, 1988) have shown that the state of the boundary layer has a significant influence on the base pressure which in turn affects the magnitude

Figure 2.7: Trailing Edge Shock System.

Figure 2.8: Sieverding et al. Base Pressure Correlation (Sieverding et al., 1980).

of the trailing edge losses, as shown in Equation 2.1. Xu and Denton argue that the entrainment power of the rear suction and pressure surface shear layer fluid might be responsible for the presence of low base pressure under certain circumstances. Xu and Denton explain that the base pressure tends to decrease with decreasing boundary layer shape factor at the trailing edge. Fuller boundary layer profiles are usually associated with lower shape factors and turbulent boundary layers usually have more momentum within the boundary layer and are thus more capable of entraining the base region fluid.

As shown from Equation 2.1, knowledge of the base pressure is required for the calculation of the trailing edge losses. One way to obtain an estimate of the base pressure consists of using a correlation for the base pressure developed by Sieverding et al. (1980). The Sieverding et al. correlation relates the base pressure to the outlet static pressure P_{s2} , the trailing edge wedge angle We_{te} and the unguided turning angle θ_u . The correlation is shown graphically in Figure 2.8.

Figure 2.9 can be used to explain the variation of the base pressure with outlet

Figure 2.9: Evolution of the Base Pressure as a Function of the Downstream Static Pressure (Sieverding et al., 1980).

static pressure shown in Figure 2.8. Region (a) corresponds to subsonic outlet flow. As the pressure ratio P_{s2}/P_{01} is reduced (increasing M_2) supersonic flow starts appearing at the trailing edge (b). With further reductions in the pressure ratio (c), the sonic condition is reached. At the sonic condition, a normal shock is present at the throat (dotted line in diagram of Figure 2.9). For increasing transonic Mach number greater than one, the normal shock will become oblique to form the trailing edge shock system. Region (d) corresponds to the displacement of shock (S1) impingement point on the adjacent blade suction surface closer to the trailing edge. In region (e), the shock is making its final approach to the trailing edge. Finally, in region (f) the shock (S1) is interacting with the wake. Based on the knowledge of base pressure combined with the use of the Sieverding et al. correlation, important information can be deduced

regarding the trailing edge shock system in transonic and supersonic flow.

2.5.3 Reynolds Number

When small gas turbine engines operate from sea level takeoff to cruise conditions, the High Pressure (HP) and Low Pressure (LP) turbines see a significant decrease in the Reynolds number. For example, Kozu and Yashima (1989) who tested a small turbofan engine, similar in size to those of Pratt & Whitney Canada, in an altitude test facility quote Reynolds numbers as low as 2.8×10^5 for the high pressure turbine and of 4.4×10^4 for the low pressure turbine. The presence of low Reynolds numbers in the LP and HP turbine stages delays the transition from a laminar to turbulent boundary layer and can cause separation to occur.

Hourmouziadis (1989) in his review paper on the aerodynamics of LP turbines describes clearly the effect of the Reynolds number on transition and hence losses. Figure 2.10 shows the variation of the losses and the behaviour of the boundary layer with Reynolds number. At high Reynolds numbers, transition occurs near the leading edge. Turbulent separation may occur near the trailing edge (a). In such cases, the mixing losses (3) produced are added to the shear layer losses (2). As the Reynolds number is decreased, turbulent separation disappears (b) and the transition point moves towards the trailing edge. As the Reynolds number is further decreased, transition moves beyond the laminar separation point and a separation bubble is formed (c). In this situation the losses may be a minimum, being generated in the wall shear layers, at the trailing edge, and in the wake of the blade only.

As the Reynolds number is further reduced, transition moves closer to the trailing edge to the point where the turbulent free shear layer cannot reattach (d). This results in a dramatic increase in the losses (1). The largest part of the losses in such cases is attributed to the mixing of the separated shear layer in the wake.

At this point, any reduction in the flow Reynolds number can only lead to the region of laminar separation to become even larger. Losses continue to increase as the laminar separation point approaches the leading edge (e).

The above description is based on experimental data that covers a Reynolds numbers range from about 1×10^5 to 4×10^6 . This range of Reynolds numbers has

Figure 2.10: Reynolds Number Effects on Turbine Cascade Performance (Hourmouziadis, 1989).

been the focus of several experimental investigations.

Patterson and Hoeger (1986) tested three LP turbine vanes in a subsonic cascade wind tunnel over a Reynolds numbers range of 1.0×10^5 to 10.0×10^5 . Figure 2.11 shows a summary of their results. They found that at the lowest Reynolds number (1.0×10^5) the trailing edge became the dominant contributor to the very high losses. With increasing Reynolds numbers, they observed a displacement of the transition point towards the trailing edge and a reduction in the level of losses.

Vijayaraghavan and Kavanagh (1988) also investigated the effect of Reynolds number on the measured losses in an axial turbine cascade. The profile used consists of a section of an HP turbine rotor blade. The Reynolds numbers were varied from 3.3×10^5 to 7.0×10^5 . This is somewhat higher than the Reynolds number usually encountered in LP and HP turbines of small engines at cruise conditions. For this range of Reynolds numbers no significant effects of Reynolds number were observed. The transition of the suction surface boundary layer was completed at approximately the same streamwise location. On the pressure side, a separation bubble was formed close to the leading edge for all test conditions.

The studies mentioned represent the scope of the work done on low Reynolds

Figure 2.11: Loss Coefficient Versus Reynolds Number (Patterson and Hoeger, 1986).

numbers flow in turbines up to the year 1990. They form the basis of review papers such as those of Mayle (1991), Walker (1993) and Hourmouziadis (1989).

However, as mentioned previously, the LP and HP turbine blades of small gas turbine engines, such as those designed by Pratt & Whitney Canada, operate at Reynolds numbers as low as 0.5×10^5 at cruise conditions. For such low Reynolds numbers, the prediction methods for transition presented by Mayle (1991) or Walker (1993) may not be applicable. These prediction methods were developed for Reynolds numbers above 1.0×10^5 . In 1991, Mayle emphasized the need for additional investigations on the boundary layer development over turbine blades at very low Reynolds numbers.

This gap in the data in the low Reynolds number range has started to be filled in recent years. Several studies on the effect of very low Reynolds numbers on the losses in HP and LP turbines have been published.

Ladwig and Fottner (1993) studied the effect of Reynolds number on losses for an HP turbine blade in a cascade wind tunnel. Tests were performed for an outlet Mach number of 0.59 and outlet Reynolds numbers varying between 0.5×10^5 and 10.0×10^5 . Results obtained by Ladwig and Fottner for the variation of the total pressure loss coefficient with Reynolds number are shown in Figure 2.12. As shown by the earlier studies, for outlet Reynolds numbers greater than about 5.0×10^5 , the losses are essentially independent of Reynolds number.

Murawski et al. (1997) performed a study on the aerodynamics of a cascade of LP turbine blades at low Reynolds numbers. Their objective was to assess the effects of the Reynolds number on the suction surface flow of the blades. They performed tests from a Reynolds number of 3.0×10^5 down to a Reynolds number of 0.5×10^5 based on exit conditions. For the blade profile they used, separation was observed at all test Reynolds numbers. However, for Reynolds numbers above 1.0×10^5 , the boundary layer reattached to form a separation bubble. The onset of transition was observed to move slightly rearward as the Reynolds number was increased. Furthermore, a shrinkage of the separation zone occurred as the Reynolds number increased. Figure 2.13 shows the variation of the loss coefficient with the Reynolds number. There is a sharp increase in loss coefficient when the Reynolds number is below 100,000. This was caused by a strong separation without reattachment for this

Figure 2.12: Variation of the Pressure Loss Coefficient With Reynolds Number (Ladwig and Fottner, 1993).

particular blade profile resulting in significantly larger wakes.

Matsunuma et al. (1998) performed a study at low Reynolds numbers in an annular turbine stator cascade. The tests were conducted for Reynolds numbers varying between 4.7×10^4 and 30.3×10^4 . Using flow visualization, they were able to determine that the separation line on the suction surface moved closer to the leading edge as the Reynolds number was reduced. They also noticed an increase in the size of the passage vortices at the tip and the hub as the Reynolds number decreased. Finally, as observed in all previous studies on the effect of Reynolds number mentioned, the total pressure loss increased for lower Reynolds numbers.

A summary of the experimental investigations which are included in this review of previous work on the effect of low Reynolds numbers on the boundary layer can be found in Table 2.3.

For the current research project a Reynolds number study was performed for a range of flow conditions. However, because of the limitations of the wind tunnel, the Reynolds number could only be lowered enough for the effects on losses to be seen for exit Mach numbers below about 0.6.

Table 2.3: Summary of Experimental Investigations on the Effects of Reynolds Number on Losses.

Author(s)	Type of Test	Flow Condition	Loss Component(s) Investigated	Parameters Investigated						Conditions for Experimental Investigations				Comments
				$i(^\circ)$	Re	Ma	Tu	IBLT	LDG	Incidence ($^\circ$)	Re x 10^{-5}	Ma	Tu (%)	
Baughn et al. (1995)	Linear Cascade	Steady			✓		✓				0.67-1.44		1.0, 10.0	-turbulence intensity has a large effect on the local heat transfer -effect of turbulence on heat transfer depends on Re
Curtis et al. (1997)	LP Turbine; Linear Cascade	Steady; Unsteady	P		✓				✓	0	0.7-4.0			-unsteady effect important for LP turbine -benefit of laminar suction surface B.L. offset by higher trailing edge losses
Hodson et al. (1994)	LP Turbine; Turbine Stator	Unsteady			✓						0.9-1.8			-steady flow prediction method might be sufficient for LP turbine blade design
Hoheisel et al. (1987)	Linear Cascade	Steady	P	✓	✓	✓	✓		✓	-38.0 to 12.0	1.0-11.0	0.3-0.95	0.8, 2.7, 5.1, 7.1	-type of loading influence total pressure losses -aft loaded blades lead to lower losses
Matsunuma et al. (1998)	Turbine Stator; Annular Cascade	Steady	M, S		✓			✓		design	0.47-3.03	0.031-0.197	0.6	-total pressure loss increases with the decrease in Re
Murala et al. (1997)	Linear Cascade	Steady	P	✓	✓					-11, -16, -21	0.4-1.6 Inlet	~0.1	1.3	-length of separation bubble forming on the airfoil increases with Re -total pressure loss increases with decreasing Re
Murawski et al. (1997)	LP Turbine; Linear Cascade	Steady	P		✓		✓			design	0.5-3.0	low subsonic	1.1-1.8	-losses increase sharply when Re is less than 100,000
Patterson & Hoeger (1988)	LP Turbine; Linear Cascade	Steady	P		✓				✓	design	1.0-10.0	0.78	4.0	-as Re decreases profile losses increase significantly -aft loading reduces friction losses
Rivir (1996)	LP Turbine; Linear Cascade	Steady	P		✓		✓			design	0.5-20.0		0.5, 1.0, 10.0	-momentum thickness transition location insensitive to turbulence level -transition length sensitive to Tu
Schulte & Hodson (1998)	LP Turbine; Linear Cascade	Steady; Unsteady	P		✓		✓			design	0.8-2.2 design: 1.3	Low subsonic	0.5 & 4.0	-total pressure loss increase significantly as Re decrease -losses are lower and almost independent of Re when moving bars
Vijayaraghavan & Kavanagh (1988)	Linear Cascade	Steady	P	✓	✓		✓			0.0, -7.0, -14.0	3.3, 5.4, 7	low subsonic	0.8, 2.0, 4.8, 6.4	-increase in loading with incidence -losses do not change appreciably with turbulence level

Abbreviations: P : Profile loss I : Incidence Tu : Turbulence Intensity
M : Midspan loss Re : Mach number IBLT : Inlet boundary layer thickness
S : Secondary loss Ma : Mach number LDG : Loading

Figure 2.13: Variation of the Loss Coefficient With Reynolds Number (Murawski et al., 1997).

2.5.4 Loading Distribution

In addition to the other factors discussed, the onset of boundary layer separation will depend on the free stream pressure gradients. Several studies have been performed on the subject. However, actual measurements in accelerating/decelerating flows remain scarce. Most of the work in this area, with few exceptions, was performed at low turbulence intensity and only deals with measurements to determine the onset of transition. A comprehensive set of measurements was presented by Abu-Ghannam and Shaw (1980). Walker (1993), Mayle (1991) and Hoheisel et al.(1987) are among some of the researchers who studied the effect of pressure gradient on turbine performance.

Pressure gradients can be categorized as being favorable or adverse. An adverse pressure gradient is one in which the pressure increases in the direction of the flow. Adverse pressure gradients can be very detrimental to the boundary layer since it can lead to boundary layer separation. If the boundary layer is laminar at separation, transition can then happen in the free shear layer. It is very difficult for a

separated laminar boundary layer to reattach in an adverse pressure gradient flow before reaching the trailing edge. On the other hand, a favorable pressure gradient will prevent boundary layer separation. In such cases, transition can occur in the attached boundary layer.

Both Walker (1993) and Mayle (1991) have conducted extensive literature surveys on the subject of pressure gradient in boundary layer flow. In their reviews, they discuss the existing models to predict transition in both adverse and favorable pressure gradients.

A few experimental studies exist on the effect of loading distribution on the performance of actual turbine blades. However, the conclusions of these studies are often contradictory. Patterson and Hoeger (1986) studied the effect of both the velocity distribution and Reynolds number on the performance of three LP turbine vanes. Their results were obtained mainly at low transonic Mach numbers. They concluded that the aft-loaded profile yielded the worst performance for the conditions investigated. The performance of the first and second stage stator vanes of an LP turbine was tested for both front and aft-loaded profiles by Hashimoto and Kimura (1984). It was found that the front-loaded designs yielded lower losses and a wider range of usable incidence for high subsonic Mach numbers. Hoheisel et al. (1987) looked at the effect of free-stream turbulence and blade pressure gradient on the losses for three turbine cascades. They studied a front-loaded profile together with two aft-loaded ones. The three designs had the same overall aerodynamic loading. Tests performed at low Mach numbers showed that lower losses can be obtained with an aft-loaded pressure distribution, provided that the rearward diffusion is carefully controlled. More recently, Howell et al. (2001), presented results for high-lift and aft-loaded LP turbine profiles with unsteady incoming wakes. They showed that aft-loaded LP turbine blades tend to have lower losses due to a reduction in the extent of turbulent boundary layer on the suction surface. It was concluded that the aft-loaded profile offered this advantage mainly in unsteady flow. However, in another study published by González et al. (2002), no definite advantage in shifting the loading to the rear of the blade could be seen in terms of reducing the losses in unsteady flow. For steady flow, the aft-loaded profile had higher losses. However, based on their results, they conclude that an optimum axial position for the peak

Mach number probably exists.

From the open literature, it can be seen that it remains unclear which type of blade pressure distribution is the best in term of losses, thus motivating the research conducted in this thesis.

2.5.5 Free Stream Turbulence

Both Hourmouziadis (1989) and Mayle (1991) mentioned the importance of conducting all experimental works with appropriate levels of turbulence and length scales in order to reproduce, as closely as possible, engine conditions.

Binder et al. (1989) performed a study on the turbulence level found in a multistage LP turbine. The turbulence intensity downstream of the rotor at three planes is shown in Figure 2.14(a). The planes of measurements are shown in Figure 2.14(b). For the Reynolds number range studied (1.2×10^5 to 4.93×10^5), the turbulence intensity is shown to vary between 1.0 and 8.0% at midspan. The highest turbulence level is observed at low Reynolds numbers. This suggests that even higher values of turbulence intensities could be reached in turbine rows of small engines where the Reynolds number can reach values as low as 0.5×10^5 .

Several studies have been performed on the effect of turbulence intensity on the performance and the boundary layer development of turbine blades. Among these studies, the one by Hoheisel et al. (1987) presents the most interesting results with regards to the variation of losses with Reynolds number and turbulence intensity. Their results, presented in Figure 2.15, show that the turbulence intensity influences the losses only for the higher range of Reynolds numbers tested. At low Reynolds numbers of about 1.0×10^5 , their results show a sharp increase in the losses but hardly any influence of the turbulence intensity.

Several researchers, such as Gostelow and Blunden (1989), have shown that the turbulent spot production rate increases with higher turbulence intensities. Moreover, investigations by Mayle (1991) and Abu-Ghannam and Shaw (1980) as well as many other studies have shown that the effect of the turbulence intensity on the boundary layer is to change the Reynolds number at which transition begins.

(a) Radial Turbulence Distribution

(b) Measurements Planes

Figure 2.14: Turbulence Distribution in Various Turbine Stages (Binder et al., 1989).

Figure 2.15: Effect of Reynolds Number on Measured Loss Coefficient at Different Turbulence Levels for Blade Profiles T104, T105 and T106 (Hoheisel et al., 1987).

2.5.6 Axial Velocity Density Ratio (AVDR)

Loss measurements in two-dimensional flow was one of the objectives of this research project. In order to quantify the two-dimensionality of the flow the Axial Velocity Density Ratio (AVDR) is often used. The Axial Velocity Density Ratio is a measure of the two-dimensionality of the flow through a blade passage. It is given by the following equation

$$AVDR = \frac{\int_0^1 (\rho_2 C_{(ax)_2})_{MS} d(y/s)}{\int_0^1 (\rho_1 C_{(ax)_1})_{MS} d(y/s)}. \quad (2.2)$$

It represents essentially the ratio of the mass flow rate per unit spanwise distance at the passage outlet to that at the blade row inlet. Hence, when calculated from measurements made at midspan, it becomes a measure of the average contraction or expansion of the streamtube at this location. In a nominally two-dimensional turbine blade passage flow the AVDR should be equal to 1.0. However, this represents only a necessary but not sufficient condition in order to ensure two-dimensional flow in the

cascade as explained by Rodger et al. (1992).

The effect of AVDR on the measured midspan performance of compressor blades has received relatively more attention than the effect on turbine performance. Stark and Hoheisel (1981) concluded that the AVDR has a very strong influence on the blade pressure distributions as well as on the loss coefficients and outlet flow angles. Similar conclusions were reached both experimentally and analytically by Pollard and Gostelow (1967), Starken and Lichfuss (1975) and Pollard and Horlock (1963) for example. In order to assist in the evaluation of the effect of AVDR on compressor blade performance, Starke (1981) developed simple formulae to calculate the effect of AVDR on losses and outlet flow angles in terms of the wake momentum thickness and the diffusion ratio.

In the case of turbine blades, the effect of AVDR on the measured performance has not been documented as well as for compressor blades. The influence of AVDR on turbine cascade results under low-speed conditions has been examined by Rodger et al. (1992). They found that AVDR had a significant effect on the losses at higher incidence where boundary layer separation was present as shown in Figure 2.16. In this figure, it can clearly be seen that losses decrease with increasing Axial Velocity Ratio (AVR). However, the effect of AVDR on transonic cascade results is still somewhat unclear. Kiock et al. (1986) concluded that the AVDR had no effect on the losses or deviation in the range of 0.9 to 1.0. Sharma and Graziani (1983) concluded that the AVDR in itself could not explain the effect of the endwall flow on the aerodynamics and heat transfer on midspan blade surface. More recently, Jouini et al. (2001), (2002) observed that the AVDR increased with incidence for measurements made in a transonic cascade. They also concluded that the losses are affected by varying the AVDR.

Figure 2.16: Variation of Midspan Losses with Axial Velocity Ratio (AVR) (Rodger et al., 1992).

2.6 Aerodynamic Influences on Losses at Off-Design Incidence

2.6.1 Effect of Incidence on Losses

Turbine blades are usually designed such that minimum losses are produced when the engine is operating at the design point. This corresponds to the situation where the inlet flow velocity vector is aligned with the blade metal angle or the design inlet angle for the cases where it differs from the inlet metal angle.

However, under certain circumstances such as at take-off, at idle, or at operation at various cruise altitudes, the engine's operating point may differ from the design operating point. When the engine operates at off-design conditions, the oncoming flow angles for the different stages will vary from their design values. This results in additional losses through the blade row commonly referred to as incidence or off-design losses.

A typical variation of losses with incidence is shown in Figure 2.17 from Mayle

Figure 2.17: Incidence Effects on Blade Boundary Layer Behaviour and Blade Midspan Losses (Mayley, 1991).

(1991). As shown in this figure, the losses increase moderately at low values of positive incidence. Under these conditions, small separation bubbles may form on the suction surface of the blade. These separation bubbles contribute to the moderate increase in losses. As the incidence is increased further, the losses start to increase much more rapidly due to the appearance of massive boundary layer separation on the suction surface of the blade. For negative incidences, significant increase in losses can be observed at relatively higher negative incidences when compared to positive incidence operations. At high negative incidences, the rapid rise in losses is the result of massive flow separation on the pressure surface.

2.6.2 Influence of Aerodynamic Parameters on Losses at Off-Design Incidence

Several studies have been published over the years in the open literature regarding the effects of off-design incidence on the profile losses as measured in turbine cascades. However, a significant portion of these studies were performed at subsonic

conditions for LP turbine blades. This is partly explained by the fact that subsonic cascade wind tunnels are much more common and much less expensive to operate. Furthermore, it is much more common for an LP turbine stage, where the Mach number encountered are usually subsonic, to operate at significant off-design incidences than an HP turbine stage which is located right at the exit of the combustor.

Among the studies performed at subsonic Mach numbers in which the effect of incidence on losses was examined, those of Vijayaraghavan and Kavanagh (1988), Goobie et al. (1989), and Tremblay et al. (1990) are worth mentioning. Vijayaraghavan and Kavanagh (1988) measured midspan losses over a limited range of incidences. They found that the losses varied slightly for the range of incidences investigated. Goobie et al. (1989) and Tremblay et al. (1990) studied the effect of incidence on the midspan losses using the same blade profile. It was found that for this profile the rise in losses at mild positive incidence was due to entropy production in the leading edge region whereas at high positive incidence trailing edge separation accounted for a significant portion of the midspan losses.

The effects of Mach number on losses at off-design incidence have been studied by relatively few researchers. This is revealed by looking at Table 2.2 which lists studies performed on the effect of Mach number on losses. It can be seen that only a small number of investigations include both the effect of Mach number and incidence.

Graham and Kost (1979) examined the performance of two airfoils over a range of outlet Mach number from 0.7 to 1.4. They varied the incidence from -20° to $+8^\circ$. Their results show the sensitivity of the blade performance to both incidence and the suction side curvature aft of the throat.

In another study, Detemple-Laake (1991) tested a highly-loaded rotor blade at various inlet flow angles and transonic outlet Mach numbers. It was found that both the shock boundary layer interactions and the secondary flow structures depend on these two variables. The passage vortex shifted toward the side wall with increasing Mach number until the cascade choked.

Šťastný et al.'s (1997) study on the rotor blade of a steam turbine is the only study known to present detailed loss measurements for a wide range of both the outlet Mach number and incidence prior to the work performed at Carleton University by Jeffries (2001) and Jouini et al. (2001, 2002). The conclusion that can be drawn

from their data is that turbine blades can sustain much higher positive and negative incidences at high Reynolds number before seeing a significant deterioration in losses as the Mach number increases.

The current study constitutes to a certain extent a follow-up study to that of Jouini (2000). Jouini (2000) performed an experimental investigation on the performance of a baseline and modified transonic turbine cascades at off-design conditions. The baseline and the modified cascade blade geometries differed mainly in their respective leading-edge geometry. The modified profile had its inlet metal angle increased by 10.5° relative to the baseline profile. One of the goals of the study was to identify leading-edge design modifications which would increase the tolerance of the blades to off-design incidence. Jouini's data cover a broad range of incidence and exit Mach numbers. Within the uncertainty of the measurements, both cascades gave very similar loss results at all conditions of exit Mach number and incidence. Therefore, the hoped improvements in positive incidence loss behavior for the redesigned blade did not materialize. In the current study, changes in the blade loading distribution are investigated as a means of reducing the losses.

From the present survey of the open literature, it is clear that turbine loss performance data in transonic regime and at off-design incidences are limited. Therefore, an addition of turbine loss data under these conditions would be very useful if any improvement to existing off-design profile loss correlation is to be expected. Adding to the data available in the open literature is one of the goals of the present research project.

2.7 Profile Loss Correlations for Axial Flow Turbines

2.7.1 Introduction

A survey of the different profile loss correlations published in the literature will be presented in this section. These correlations make use of simple geometric parameters and basic flow condition variables. As an example, a geometric parameter could be the blade thickness to chord ratio (t/C) and a flow condition that could be

used is the outlet Reynolds number (Re_2). The loss correlations are useful to turbine designers in order to get an estimate of the magnitude of the profile losses prior to knowing the detailed blade profile.

2.7.2 Ainley and Mathieson Correlation

In 1951, Ainley and Mathieson proposed a method for estimating the performance of an axial flow turbine. Today, the work of Ainley and Mathieson remains widely used and is the foundation of new or improved correlation methods. The Ainley and Mathieson correlation method consists of a complete set of correlations: one for each loss component. Furthermore, this method can be used to predict losses at both design and off-design conditions. The method is based on experimental cascade test results obtained from several blade profiles.

In terms of profile loss predictions at design conditions, Ainley and Mathieson proposed a system based on two extreme blade designs, namely nozzle blades ($\beta_1 = 0$) and impulse blades ($\beta_1 = \alpha_2$). A definition for the inlet metal angle β_1 and outlet flow angle α_2 can be found in Figure 3.17. Figure 2.18 (nozzle blade) and 2.19 (impulse blade) show the profile loss as a function of the pitch to chord ratio for a family of outlet flow angles. These results were obtained experimentally by Ainley and Mathieson for blades having a maximum thickness to chord ratio of 0.2. The tests were performed at a Reynolds number of 2×10^5 and a Mach number of less than 0.6.

For blades having intermediate values of inlet metal angle between these two extremes, the correlation predicts the profile losses to be weighted average of the values given by the nozzle and impulse blades having the same exit flow angle as the blade in question. This will be explained in detail in a moment.

A nozzle blade with $\beta_1 = 0$ provide the maximum mean acceleration possible to the flow for the outlet angle in question. Therefore, the chance of seeing severe boundary layer separation on the suction surface for this type of blade is very limited for a wide range of incidence.

At the opposite extreme from the nozzle blade stands the impulse blade with $\beta_1 = \alpha_2$, which imparts no net acceleration to the flow. For this type of blade, there

Figure 2.18: Ainley and Mathieson Profile Loss Correlation for Nozzles (Ainley and Mathieson, 1951).

Figure 2.19: Ainley and Mathieson Profile Loss Correlation for Impulse Blades (Ainley and Mathieson, 1951).

Figure 2.20: Variation in Profile Loss With Incidence for Typical Turbine Blades (Adapted from Ainley and Mathieson, 1951).

tend to be stronger adverse pressure gradients on the suction surface. This results in a much reduced range of low loss performance with incidence as the blade boundary layer is subjected to adverse pressure gradient. Figure 2.20 shows the difference in the variation of profile loss with incidence for the two blade types.

For blades having intermediate values of inlet metal angle located between that of a nozzle and an impulse blade, the profile loss coefficient can be estimated using the following interpolation formula:

$$Y_p = \left\{ Y_{p(\beta_1=0)} + \left(\frac{\beta_1}{\alpha_2} \right)^2 [Y_{p(\beta_1=\alpha_2)} - Y_{p(\beta_1=0)}] \right\} \left(\frac{t/C}{0.2} \right)^{\frac{\beta_1}{\alpha_2}}. \quad (2.3)$$

In Equation 2.3 the terms $Y_{p(\beta_1=0)}$ and $Y_{p(\beta_1=\alpha_2)}$ are the profile loss for the nozzle

Figure 2.21: Variation of Profile Loss with Incidence for Typical Turbine Blading (Ainley and Mathieson, 1951).

and the impulse blade respectively read off from Figures 2.18 and 2.19 at the same outlet flow angle and space to chord ratio as the actual blade. The term $\left(\frac{t/C}{0.2}\right)^{\frac{\beta_1}{\alpha_2}}$ is a correction factor to take into account the effect of the maximum thickness-to-chord ratio. This factor is required because as the blade inlet metal angle β_1 approaches α_2 the passage acceleration is reduced and the influence of geometric parameters such as t_m/C is increased. The correction factor for the effect of maximum thickness-to-chord ratio (t_m/C) is only valid for the range 0.15 to 0.25. For values outside this range, the value of t_m/C should be taken as 0.15 or 0.25, whichever is closest.

For the off-design profile loss predictions, Ainley and Mathieson assumed the profile loss ratio $Y_p/Y_{p(i=0)}$ to be defined by a unique function of the incidence ratio i/i_s as shown in Figure 2.21. i_s is the stalling incidence which is defined as the incidence at which the profile loss equals twice the minimum loss or in mathematical terms $Y_p/Y_{p(i=0)} = 2.0$.

In order to use Figure 2.21, one must first determine the stalling incidence of

Figure 2.22: Variation of Stalling Incidence with Inlet Metal Angle and Outlet Flow Angle Ratio (Ainley and Mathieson, 1951).

the blade of interest. To do so, Ainley and Mathieson correlated positive stalling incidence with the pitch to chord ratio (s/C), outlet flow angle α_2 and the ratio of inlet metal angle to outlet flow angle (β_1/α_2) for blades having a $s/C = 0.75$. The results are shown in Figure 2.22.

For blades having space-to-chord ratios differing from 0.75, Ainley and Mathieson provided a means of determining the stalling incidence of an equivalent blade with $s/C = 0.75$ and then that of the actual blade. First, the outlet flow angle of the equivalent blade ($s/C=0.75$) is found using Figure 2.23. Then, the difference in stalling incidence between the actual blade and the equivalent one can be found from Figure 2.24. Finally, the stalling incidence of the equivalent blade ($s/C=0.75$) is read from Figure 2.22 and added to the Δi_s found previously in order to get the stalling incidence of the actual blade.

Knowing the stalling incidence for a blade, one can get the relative profile loss $Y_p/Y_{p(i=0)}$ for any incidence from Figure 2.21.

Figure 2.23: Variation of Outlet Flow Angle With Space to Chord Ratio (Ainley and Mathieson, 1951).

Figure 2.24: Variation of Stalling Incidence Correction With Space to Chord Ratio (Ainley and Mathieson, 1951).

Figure 2.25: Variation of Trailing Edge Correction With Trailing Edge Thickness (Ainley and Mathieson, 1951).

In order to obtain the total midspan losses at design or off-design incidence using the Ainley and Mathieson method, a correction must be applied to the profile losses estimate to take into account the trailing edge losses. This is required since the Ainley and Mathieson data for profile losses were obtained for blades having trailing edge thickness-to-blade-spacing ratio of 0.02 only. However, it is known that losses vary significantly with the trailing edge thickness and modern turbine blades often have greater trailing edges thickness due to the use of cooling. Thus, using Figure 2.25 one can determine the correction factor required for blades having t_{TE}/s ratio different from 0.02.

2.7.3 Dunham and Came Correlation

Nearly 20 years after Ainley and Mathieson (1951) presented their successful loss prediction method, Dunham and Came (1970) suggested improvements to the original scheme. Dunham and Came proposed both Reynolds number and Mach

number corrections. Their improvements are based on detailed comparisons between the predicted and measured performance of 16 turbines of more recent design at the time.

The Reynolds number correction consists in multiplying the original profile loss interpolation, Equation 2.3, by the correction factor $\left(\frac{Re_2}{200000}\right)^{-0.2}$. In this factor, Re_2 is the Reynolds number based on chord and exit flow conditions. As discussed in a previous section, the Reynolds number has a strong influence on the profile losses. However, the original Ainley and Mathieson correlation was limited to a Reynolds number of 2×10^5 . Therefore, attempts to use the Ainley and Mathieson model for loss prediction at much higher and particularly lower Reynolds numbers resulted, for certain blades, in large discrepancies. Dunham and Came's correction factor brought a remedy to this problem.

The second improvement made by Dunham and Came was to implement a compressibility factor so that the Ainley and Mathieson correlation could be used for exit Mach numbers greater than 1.0. A compressibility correction was required since shock losses start to increase as the exit Mach number approaches 1.0 and goes beyond as discussed previously in Section 2.5.1. As for the Reynolds number correction, Dunham and Came proposed to add a multiplying factor to Equation 2.3. The compressibility correction factor K_c has the form $K_c = [1 + 60(M_2 - 1)^2]$. It is valid only for $M_2 > 1.0$. Therefore, the improved Ainley and Mathieson correlation can be written as

$$Y_p = \left\{ Y_{p(\beta_1=0)} + \left(\frac{\beta_1}{\alpha_2}\right)^2 [Y_{p(\beta_1=\alpha_2)} - Y_{p(\beta_1=0)}] \right\} \left(\frac{t/c}{0.2}\right)^{\frac{\beta_1}{\alpha_2}} \left(\frac{Re}{200000}\right)^{-0.2} K_c \quad (2.4)$$

where,

$$K_c = [1 + 60(M_2 - 1)^2]. \quad (2.5)$$

2.7.4 Kacker and Okapuu Correlation

In 1982, Kacker and Okapuu proposed additional improvements to the Ainley and Mathieson and Dunham and Came (AMDC) correlations for turbines operating at design incidence. Kacker and Okapuu (1982) expressed the total midspan loss in a

cascade of blades as the sum of profile and trailing edge losses. The profile losses are corrected for Reynolds number effects. The total midspan losses are estimated using the following equation:

$$Y_t = Y_p f(Re) + Y'_{TET}. \quad (2.6)$$

This expression is seen to differ from that used by Ainley and Mathieson (1951),

$$Y_t = [(Y_p)] K_{TE} \quad (2.7)$$

where K_{TE} is a multiplier used to correct for the effects of the trailing edge thickness. In Equation 2.6, Y_t is the total midspan loss coefficient, Y_p is the profile loss coefficient and Y'_{TET} is trailing-edge loss coefficient. The term $f(Re)$ is the Reynolds number correction. It must be applied to the profile loss coefficient only whenever the Reynolds number differs from 2×10^5 . The correction factor is given by

$$f(Re) = \left(\frac{Re_c}{2 \times 10^5} \right)^{-0.4} \quad \text{for } Re_c \leq 2 \times 10^5, \quad (2.8)$$

$$f(Re) = 1.0 \quad \text{for } 2 \times 10^5 < Re_c < 10^6, \quad (2.9)$$

and by

$$f(Re) = \left(\frac{Re_c}{10^6} \right)^{-0.2} \quad \text{for } Re_c > 10^6. \quad (2.10)$$

The trailing-edge loss coefficient Y'_{TET} is calculated using an interpolation function similar to that used by Ainley and Mathieson to estimate the profile losses. Based on an extensive survey of the literature, Kacker and Okapuu were able to correlate the trailing edge losses expressed as an energy loss coefficient with the trailing edge thickness-to-throat opening ratio t/o for both nozzle and impulse blades. The results are shown in Figure 2.26.

For blades other than the two basic types, the following formula is used to determine the trailing edge losses:

$$\Delta\phi_{TET}^2 = \Delta\phi_{TET(\beta_1=0)}^2 + \left| \frac{\beta_1}{\alpha_2} \right| \left(\frac{\beta_1}{\alpha_2} \right) \left[\Delta\phi_{TET(\beta_1=\alpha_2)}^2 - \Delta\phi_{TET(\beta_1=0)}^2 \right]. \quad (2.11)$$

The conversion from kinetic energy loss coefficient to pressure loss coefficient is

Figure 2.26: Trailing Edge Energy Loss Coefficient Versus the Trailing Edge Thickness to Throat Ratio (Kacker and Okapuu, 1982).

done using the following equation:

$$Y'_{TET} = \frac{\left[1 - \frac{\gamma-1}{2}M_2^2 \left(\frac{1}{1-\Delta\phi_{TET}^2} - 1\right)\right]^{-\frac{\gamma}{\gamma-1}} - 1}{1 - \left(1 + \frac{\gamma-1}{2}M_2^2\right)^{-\frac{\gamma}{\gamma-1}}}. \quad (2.12)$$

It can be shown that for the incompressible flow limit, this expression can be reduced as follows:

$$Y'_{TET} = \frac{1}{1 - \Delta\phi_{TET}^2} - 1. \quad (2.13)$$

For the subsonic regime of operation, Kacker and Okapuu (1982) suggested the following expression for estimating the profile loss coefficient,

$$Y_p = 0.914 \left(\frac{2}{3}Y_{p,AMDC}K_p\right). \quad (2.14)$$

In this equation, the term $Y_{p,AMDC}$ is the profile loss coefficient from AMDC correlation given by

$$Y_p = \left\{Y_{p(\beta_1=0)} + \left|\frac{\beta_1}{\alpha_2}\right| \left(\frac{\beta_1}{\alpha_2}\right) [Y_{p(\beta_1=\alpha_2)} - Y_{p(\beta_1=0)}]\right\} \left(\frac{t/c}{0.2}\right)^{\frac{\beta_1}{\alpha_2}}. \quad (2.15)$$

This interpolation equation is in all aspects the same as to that given by AMDC except for the term $|\frac{\beta_1}{\alpha_2}|$, which was introduced to allow for negative inlet angles.

The factor of 2/3 in Equation 2.14 is applied to the profile loss prediction of AMDC to take into account improvements in turbine design that occurred over the years. The authors obtained this factor by examining the performance data for more modern turbines and comparing their profile losses with that given by Figures 2.18 and 2.19.

The constant 0.914 arises from the fact that Ainley and Mathieson's (1951) loss performance data (Figures 2.18 and 2.19) are for blades having a trailing edge thickness-to-pitch ratio (t/s) of 0.02. However, in the Kacker and Okapuu loss system, the trailing edge losses are treated as a separate additive loss. Therefore, the profile loss values obtained from Figures 2.18 and 2.19 are multiplied by 0.914 which is the correction factor used in AMDC loss system to obtain the profile losses for $t/s = 0.0$.

Finally, the compressibility correction factor K_p was added by the authors to

correct for the effects of exit Mach number and channel acceleration. This correction factor is necessary since the results from Figures 2.18 and 2.19 are derived from cascade tests carried out at low subsonic Mach numbers. According to Kacker and Okapuu (1982), Figures 2.18 and 2.19 tend to overpredict the losses when applied to turbines operating at higher Mach numbers. The correction factor is given by the following equation:

$$K_p = 1 - K_2(1 - K_1) \quad (2.16)$$

where the coefficients K_1 and K_2 are given by:

$$K_1 = 1 - 1.25(M_2 - 0.2) \text{ for } M_2 > 0.2, \quad (2.17)$$

$$K_2 = (M_1/M_2)^2. \quad (2.18)$$

2.7.5 Moustapha et al. Correlation for Off-Design Losses

Moustapha et al. (1990) proposed a new means of predicting losses at off-design incidence. This correlation is intended to be used with the Kacker and Okapuu (1982) loss system to replace the Ainley and Mathieson off-design incidence loss prediction method.

The motivation of Moustapha et al. for revisiting the Ainley and Mathieson correlation for off-design incidence stemmed from one major shortcoming. In the Ainley and Mathieson off-design prediction method, the stalling incidence is independent of the leading edge shape. According to Moustapha et al., this should not be so since stalling of blades results from the boundary layer separation, which is due to peaks in the velocity distribution in the leading edge region. The presence of these peaks is highly dependent on the leading edge geometry. Therefore, in order to capture this effect, Moustapha et al. included the leading edge diameter to pitch ratio d/s in their new correlation.

The Moustapha et al. correlation for off-design incidence takes into account the blade passage acceleration ($\cos\beta_1/\cos\beta_2$) and the incidence through the term

$(\alpha_1 - \alpha_{1_{des}})$. The correlation is given by the following equations:

$$\Delta\phi_p^2 = 0.78 \times 10^{-5}\chi + 0.56 \times 10^{-7}\chi^2 + 0.4 \times 10^{-10}\chi^3 + 2.05 \times 10^{-19}\chi^6 \quad (2.19)$$

for $0 < \chi < 800$,

and by

$$\Delta\phi_p^2 = -5.1734 \times 10^{-6}\chi + 7.69 \times 10^{-9}\chi^2 \quad (2.20)$$

for $-800 < \chi < 0$. In these equations, the incidence parameter χ is given by

$$\chi = \left(\frac{d}{s}\right)^{-1.6} \left(\frac{\cos\beta_1}{\cos\beta_2}\right)^{-2} [\alpha_1 - \alpha_{1_{des}}]. \quad (2.21)$$

This functional relationship was developed using a database of cascade results composed of in-house Pratt & Whitney data as well as data found in the open literature. Figure 2.27 shows a plot of the profile loss correlation for off-design incidences obtained using the database given in Moustapha et al. (1990).

In addition to the profile loss correlation for off-design incidences, Moustapha et al. (1990) also developed a correlation for secondary losses at off-design incidences. However, this correlation will not be described in this review as secondary losses are not an aspect investigated in this study.

2.7.6 Benner et al. Correlation

Following the recommendation of Moustapha et al. (1990), Benner et al. (1997) investigated further the influence of leading edge geometry at off-design incidence. More specifically, they looked at the influence of the leading edge wedge angle and diameter. Based on data collected at Carleton University for two cascades having different leading edge geometries, they were able to revise the Moustapha et al. correlation. The improved correlation of Benner et al. is of the same form as that of Moustapha et al. except that the influence of the leading edge wedge angle was added to the incidence parameter. The new incidence parameter of Benner et al. is written as

$$\chi' = \left(\frac{d}{s}\right)^{-0.05} We^{-0.2} \left(\frac{\cos\beta_1}{\cos\beta_2}\right)^{-2} [\alpha_1 - \alpha_{1_{des}}]. \quad (2.22)$$

Figure 2.27: Moustapha et al. (1990) Profile Loss Correlation for Off-Design Incidences.

Figure 2.28: Profile Incidence Loss Correlation of Benner et al. (Benner et al., 1997).

It can be observed from this equation that Benner et al. found the effect of the leading edge diameter to be much less important than Moustapha et al. had initially proposed. On the other hand, the influence of the leading edge wedge angle is quite significant. These two changes to the initial correlation explain why the Benner et al. correlation captures better the variation of the profile losses with incidence, as shown in Figure 2.28.

In a similar fashion to the Moustapha et al. correlation, Benner et al. developed a function relating the change in energy loss coefficient to their new incidence parameter. The fitting function is given by

$$\Delta\phi_p'^2 = a_8\chi'^8 + a_7\chi'^7 + a_6\chi'^6 + a_5\chi'^5 + a_4\chi'^4 + a_3\chi'^3 + a_2\chi'^2 + a_1\chi' \quad (2.23)$$

where

$$\begin{aligned} a_8 &= +3.711 \times 10^{-7}, a_7 = -5.318 \times 10^{-6}, \\ a_6 &= +1.106 \times 10^{-5}, a_5 = +9.017 \times 10^{-5}, \end{aligned}$$

$$a_4 = -1.542 \times 10^{-4}, a_3 = -2.506 \times 10^{-4},$$

$$a_2 = +1.327 \times 10^{-3}, a_1 = -6.149 \times 10^{-5},$$

for $\chi' \geq 0$,

and by

$$\Delta\phi_p^2 = 1.358 \times 10^{-4}\chi^2 - 8.720 \times 10^{-4}\chi \quad (2.24)$$

for $\chi \leq 0$.

Chapter 3

Experimental Apparatus

3.1 Introduction

The Pratt & Whitney Canada high speed wind tunnel, shown in Figure 3.1, is of the blow-down type. It is used mainly for transonic turbine cascade research. The wind tunnel was originally constructed at McMaster University in Hamilton and was recommissioned at Carleton University a few years ago.

In the current chapter, the different components of the Pratt & Whitney Canada high speed wind tunnel are first described. Then, the instrumentation used to carry out the experiments is presented. Finally, the geometry of the three cascades used for this research is described in detail together with the methodology used to design these airfoils.

3.2 PWC High Speed Wind Tunnel

3.2.1 Compressor

A compressor is used to fill the wind tunnel tanks with air. It is a Broom-Wade model VC500 two-stage intercooled positive displacement compressor. A picture of the compressor is shown in Figure 3.2. The compressor is driven by a 100HP AC, three phase electric motor. The compressor can deliver approximately $14.2\text{m}^3/\text{min}$ (500 S.C.F.M.) of air. This correspond to a mass flow rate of $17\text{kg}/\text{min}$ of air. It takes approximately 20 minutes of pumping time to fill the tanks to a pressure of 690kPa (100psi(g)) when they are initially at atmospheric pressure.

The air exiting the compressor second stage is at a temperature that exceeds 100°C . In order to reduce the delivery temperature of air to the tanks, the air is

Figure 3.1: Pratt & Whitney Canada High Speed Wind Tunnel.

Figure 3.2: Broom-Wade VC500 Compressor.

passed through a reversed flow aftercooler. At the exit from the aftercooler, the air temperature is reduced to about 27°C. The air is then passed through a water separator to remove the excess moisture.

3.2.2 Air Dryer

Before the compressed air is sent to the tanks, its dew point must be reduced in order to prevent the formation of condensation in the test section. As the air is discharged through the test section, the temperature of the air in the tanks, and thus the total temperature T_o of the air in the test section, decreases continuously during the run due to the expansion. Furthermore, the air is accelerated to high velocities through the cascade and hence the static temperature of the air at the cascade outlet can become very low. Therefore, condensation can arise if the static temperature in the test section falls below the dew point of the air. This is not desirable since condensation shocks could form in the blade passages.

The air drying system used for the Pratt & Whitney Canada high speed wind

tunnel is a Comp-Air Kellogg model CDH520 regenerative desiccant dryer. The dryer consists of two tanks as shown in Figure 3.1. The tanks are used alternatively: while one tank is drying the air coming from the compressor the desiccant from the other tank is regenerated. With the use of the dryer, the dew point of the air is reduced to -45°C . This is sufficient to prevent the formation of condensation in the test section up to an outlet Mach number of 1.2.

3.2.3 Air Storage Tanks

The air storage for the blowdown wind tunnel consists of an arrangement of two sets of storage tanks in parallel as shown in Figure 3.3. Each set is composed of two tanks having a volume of about 6.5m^3 . The total combined volume of the four reservoirs is thus 26m^3 . This parallel tank arrangement reduces the pressure losses when the air is discharged. Furthermore, it allows a steady supply of air to both the test section and the ejector system, when the latter is operated. During a run, the control valves are operated under feedback control to obtain the desired blowing pressure in the test section and to the ejectors.

3.2.4 Control and Shutoff Valves

Two control valves are used to operate the wind tunnel. One valve is dedicated to the control of the test section blowing pressure whereas the second one controls the ejector blowing pressure. These two valves can be seen in Figure 3.1.

The test section control valve is a 10.2cm (4in) Neles-Jamesbury model 815L butterfly valve actuated using a Neles-Jamesbury model QP2C actuator.

The ejector-diffuser control valve is a Foxboro model V9000, 4 inch ball valve. The control valve is actuated using a Foxboro model P110 pneumatic actuator.

In combination with the two control valves, two shutoff valves are used. These valves are located just upstream of the control valves. They are fully open before a series of tests and closed when all testings are done. The shutoff valves are Neles-Jamesbury model 815L, 15.2cm (6in) butterfly valves. The shutoff valves are positioned using Neles-Jamesbury model QP3, pneumatic actuators. The air supply to the actuator is simply controlled by a solenoid valve since only two positions are

Figure 3.3: Tanks Arrangement.

required for these valves, namely, open and closed.

3.2.5 Cascade Test Section

Figure 3.4 shows an isometric view of the cascade test section and the ejector-diffuser assembly with the top cover removed. A top view of the cascade test section is also shown schematically in Figure 3.5. The components of the cascade test section include the turntable, the inlet side walls, the sliding walls and the test section side walls.

The turntable is used to vary the incidence of the cascade. The turntable is mounted on a worm and gear so that it can be rotated to yield incidence angles between -20.0° and $+20.0^\circ$ depending on the inlet and outlet metal angle of the cascade used. The inlet flow angle α_1 shown in Figure 3.5 can be set to within $\pm 0.5^\circ$ using a protractor. Furthermore, a protractor mounted on the turntable top cover can also be used to adjust roughly the inlet flow angle when installing a new cascade or changing the incidence angle.

Figure 3.4: Cascade Test Section and Ejector-Diffuser Assembly.

Figure 3.5: Cascade Test Section.

The inlet side walls can be moved sideways as well as back and forth. This is necessary in order to align the inlet side walls with the cascade end blocks (Figure 3.5). Furthermore, the parts of the inlet side walls in contact with the end blocks are spring loaded so that a good seal is obtained between the two parts. The sliding walls are firmly kept in place using pneumatic pistons so that a seal is obtained with the test section side walls. The height of the internal passage, and thus the span of the cascade blades, is fixed at 61mm.

3.2.6 Ejector-Diffuser Section

The Pratt & Whitney Canada high speed wind tunnel is equipped with an ejector-diffuser system as shown in Figures 3.1 and 3.4. The ejector-diffuser is used to control the static pressure downstream of the cascade and thus vary the cascade exit Reynolds number independently of the Mach number. The ejector-diffuser assembly is mounted on casters so that its orientation can be adjusted whenever the incidence is changed. The ejector-diffuser is always aligned with the blade outlet metal angle.

The ejector-diffuser acts like a jet pump. Two supersonic jets of air at a Mach number of 1.5 are supplied to the ejector through convergent-divergent nozzles downstream of the cascade as shown in Figure 3.6. The nozzles are supplied with high-pressure air from the storage tanks. The two supersonic jets run parallel along the ejector wall and mix with the core flow as the flow proceeds further down the mixing tube. The mixing of the supersonic jet with the slow moving core flow entrains it which in turns lower the pressure just downstream of the cascade. As an example, typical static pressures measured downstream of the cascade without the ejector-diffuser system in use are of the order of 75kPa at an outlet Mach number of 1.05. On the other hand, with the use of the ejector-diffuser assembly, the static pressure can be lowered to 60kPa at the cascade exit.

The diffuser part of the ejector-diffuser assembly provides some static pressure recovery. The air at the outlet of the diffuser exhausts to the laboratory at atmospheric pressure.

Figure 3.6: Ejector-Diffuser Assembly.

3.3 Wind Tunnel Instrumentation

3.3.1 Test Section and Ejector-Diffuser Pressure Control Systems

The desired test section and ejector blowing pressures are achieved by controlling the valves using the computer-based control system as shown in Figure 3.7. The pressure control system is composed of the following equipment:

1. a 486 DX2 PC equipped with an HP-IB card
2. an HP 3497A data acquisition and control unit
3. an HP 3437A system voltmeter (test section)
4. an HP 3456A system voltmeter (ejector-diffuser)
5. two Omega pressure transducers model PX613 with a range of 0 to 690kPa (0 to 100psi(g)) to measure the static pressure at the test section and ejector inlet.

6. an HP 44429A dual output digital to analog (D/A) converter
7. two Neles-Jamesbury NE724/U electro-pneumatic positioners
8. a Neles-Jamesbury QP2C control valve pneumatic actuator (test section)
9. a Foxboro model P110 control valve pneumatic actuator (ejector)
10. a Foxboro V9000 10.2cm (4in) ball control valve (ejector)
11. and a Neles-Jamesbury 815L 10.2cm (4in) butterfly valve (test section).

The pressure control is achieved through a feedback loop. The control system transfer function implements Proportional-Integral-Derivative (PID) control. The 486 DX2 PC is used to run the control algorithm. It communicates with the data acquisition and control unit via an HPIB (Hewlett-Packard Interface Bus) interface in order to obtain the required pressure (voltage) readings from the transducers as well as to transmit the required valve positions. The data acquisition unit obtains the transducer voltage readings from an HP 3437A system voltmeter (test section) and an HP 3456A system voltmeter (ejector-diffuser) through an HPIB interface.

The location of the Omega pressure transducer used to control the pressure in the test section is shown in Figure 3.5. It measures the static pressure in the test section at this location and it is referred to as the test section blowing pressure. However, since it is measured ahead of the contraction, the blowing pressure is approximately the total pressure for the flow entering the cascade.

The ejector blowing pressure is also controlled based on the signal from an Omega pressure transducer. It is located just upstream of one of the two ejector nozzles, as shown in Figure 3.6 and it measures the static pressure at this location.

As mentioned previously, the desired valve positions is transmitted from the computer to the data acquisition and control unit. The data acquisition unit is equipped with an HP 44429A dual output digital to analog (D/A) converter. The desired valve position obtained from the computer is converted to a current varying from 4mA (fully closed valve) to 20mA (fully open valve). This current is fed from the D/A converter to a Neles-Jamesbury NE724/U current-to-air positioner. The

positioner then supplies the valve actuators with the required shop air pressure to open the 4 inches ball valves.

This pressure control system yields nearly constant test section and ejector blowing pressure during the runs. This is true for all blowing pressure levels at which the wind tunnel is used, as shown in Figure 3.8. However, the runtime over which the pressure is uniform depends on the blowing pressure, as also seen from the figure.

3.3.2 Data Acquisition System

The use of a transonic blow-down wind tunnel requires a fast and efficient data acquisition system in order to maintain an acceptable level of productivity. The data acquisition system must allow the experimentalist to gather as much data as possible within the very limited run time available.

A Hewlett-Packard data acquisition system is used along with a 48-port Scanivalve to collect the pressure data in the high speed wind tunnel laboratory. In addition, a thermocouple is used to measure the total temperature at the test section inlet. A schematic of the data acquisition system is shown in Figure 3.9 for the cases where downstream static probe measurements are made. Figure 3.10 shows the setup when downstream three-hole probe measurements are made. The data acquisition system is composed of the following elements:

1. a Pentium 90 PC equipped with an HPIB card
2. an HP 3852A data acquisition and control unit
3. an HP 44702B 13 bit high speed voltmeter card
4. a 48 ports Scanivalve system model SSS/CM48-1
5. a Kulite pressure transducer model XCQ-062-25A.
6. and an HP 44708A multiplexer thermocouple compensation card.

The HP 3852A data acquisition unit is controlled using a Pentium 90 PC. The communication between the computer and the data acquisition unit is achieved through the use of a Hewlett-Packard Interface Bus (HPIB) card installed on the

Figure 3.7: Test Section and Ejector-Diffuser Control System.

Figure 3.8: Variation With Time of Cascade Test Section Blowing Pressure.

computer's motherboard. The software used to control the data acquisition unit was developed by Jeffries (1994). The output voltage from the pressure transducer in the Scanivalve is recorded using an HP 44702B 13 bit high speed voltmeter card connected at the back of the data acquisition unit. This high-speed voltmeter is capable of measuring DC voltage at a sample rate up to 100kHz. However, as found by Jeffries (2001), sample rates of only 2kHz were required in order to obtain adequate pressure measurements. This sampling rate was established by analyzing the pressure signal in the frequency domain and applying the Nyquist criteria. At this sampling frequency, the pressure cumulative mean values was found to be reasonably close to the long-term values after approximately a sampling time of 0.1 seconds for all flow conditions. During a run, the recorded voltages must be stored in part to the voltmeter buffer in PACKED format (i.e. 2 bytes per reading) since the time required to download the data to the computer is longer than that required to acquire the data. The voltmeter buffer can hold up to 65536 readings.

The pressure measurements were made using a miniature Kulite pressure trans-

Figure 3.9: Data Acquisition System for Static Probe Traverse.

Figure 3.10: Data Acquisition System for Three-Hole Probe Traverse.

ducer model number XCQ-062-25A. Its useful operating range is from 0 to 172.4kPa (0 to 25psi(a)). The calibration procedure will be described in the following chapter. The pressure transducer was mounted inside the Scanivalve head in order to minimize the length of tubing and thus the volume of air in the system. The Scanivalve model used is the SSS/CM48-1 which has 48 ports. For the current set-up, it was found that a settling time of at least 0.004 seconds was required after stepping the valve for the pressure to settle in the volume contained between the measuring point and the pressure transducer located in the Scanivalve head. A delay of 0.02 seconds, which corresponds to five time constants, was implemented into the data acquisition system to allow the pressure to settle before pressure data was taken. The Scanivalve is controlled using one of the computer's parallel ports.

Total temperature measurements at the inlet of the test section are made using a thermocouple mounted in a United Sensor Cobra probe. The temperature measurements are recorded using an HP 44708A, 20-Channel Relay Multiplexer with Thermocouple Compensation (MUX/TC) combined with an HP 44701A, 5 1/2 Digit Integrating Voltmeter. These two units are connected at the back of the data acquisition unit. The multiplexer card has an isothermal connector block which serves as the reference junction for the type-K thermocouple used.

The software used to operate the data acquisition system was developed by Jeffries (2001) and can be found on the data acquisition system's computer.

3.3.3 Probe Traverse Mechanisms

3.3.3.1 Downstream Pitchwise Traverse Mechanism

Pitchwise downstream traverses were performed using the traversing gear shown in Figure 3.11. The static and three-hole probes used to make the measurements were mounted inside the probe holder located at the end of the probe stem. The probe stem enters the test section through the transition walls downstream of the cascade. The opening in the side wall around the stem is sealed with two TEFLON blocks held in place by plates which are fixed to the transition wall.

The probe holder and probe stem assembly is mounted on the horizontal traversing gear which is made of two guiding rods and one threaded rod. The threaded rod

Figure 3.11: Downstream Probe Traversing Mechanism.

is driven by a Compu-Motor stepper motor which allows for the displacement of the probe in the pitchwise direction. The stepper motor control system was shown in Figures 3.9 and 3.10. The stepper motor movements are controlled using one parallel port of the data acquisition system computer. The parallel port TTL signal must be amplified before being sent to the stepper motor controller which is located in the laboratory at about 15m from the computer. The stepper motor is operated in full-step mode. This results in 200 steps per revolution for the motor and a pitchwise displacement of 1.96mm per revolution for the traversing gear assembly.

The horizontal traversing mechanism is fixed to the transition wall using the extension arms. This ensures that once the transition wall is aligned with the cascade exit metal angle, the probe is set at the same angle.

Provisions were also made to allow for the manual displacement of the traversing mechanism vertically. Therefore, pitchwise downstream probe traverses can be performed at several spanwise location. Among other things, this allows one to verify the extent of two-dimensional flow around the blade midspan section as will be

discussed in Chapter 6. The TEFLON sealing blocks must also be moved manually when the vertical position of the probe support is altered.

3.3.3.2 Upstream Spanwise Traverse Mechanism

A smaller traverse mechanism can also be fixed on the cascade cover to traverse the inlet flow in the spanwise direction 40mm upstream of the cascade inlet. This traverse mechanism is operated using the same stepper motor, controller and software as the downstream pitchwise mechanism just described. This traversing gear was used to document the inlet flow spanwise uniformity which is discussed in Chapter 6. Furthermore, the same mechanism can be fixed to the test section cover to traverse the flow in the spanwise direction 300mm upstream of the cascade.

3.3.4 Pressure Measuring Probes

3.3.4.1 Three-Hole Probe

The three-hole pressure probe designed by Islam (1999) was used to traverse the flow downstream of the cascade. Figure 3.12 shows the dimensions of the probe. The three-hole probe was used to determine the flow total pressure, static pressure and outlet flow angle.

The three-hole probe is made of a 165mm long stainless steel tube and three stainless steel hypodermic tubes with outside diameter of 0.46mm. The probe body has a diameter of 7mm which tapers down to 1.78mm near its head. The three measuring ports are made using three hypodermic tubings soldered side by side. The three tubes extend 5mm outside the probe body. The outer extremity of the left and right tubes is tapered to form a 30° semi-vertex angle.

The width of the probe tip is 1.37mm. This corresponds to about 4.7% of the blade pitch on the baseline cascade HS1A. This is less than the accepted minimum value of 6% suggested for transonic testing (see Kiock et al., 1986, Mee, 1991, and Sieverding, 1993).

Figure 3.12: Three-Hole Probe (Islam, 1999).

3.3.4.2 Static Pressure Probe

The static pressure probe is used to traverse the flow downstream of the cascade in order to determine the static pressure. Although the static pressure is also extracted from the three-hole probe measurements, direct measurement using the static pressure probe were also made. The three-hole and static pressure probes are mounted in the same holder and must therefore be used in separate runs. Figure 3.13 shows the static pressure probe. Like the three-hole probe, it was designed and manufactured by Islam (1999).

The static probe is made of concentric stainless steel hypodermic tubes. The diameter of the tube forming the probe tip is 1.02mm. This yields a probe tip-diameter-to-pitch ratio of 3.5%. This is again below the recommended maximum values. The pressure measuring port is composed of two holes, 0.3mm in diameter and 180° apart. The distance between the conical probe tip and the pressure measuring holes is approximately 12 times the tip tube diameter. The probe tip has a cone angle of 15° to minimize the flow disturbance.

Dimensions are in mm unless otherwise specified

Figure 3.13: Static Pressure Probe (Islam, 1999).

3.3.4.3 Boundary Layer Probe

In order to traverse the cascade inlet flow in the spanwise direction two boundary layer probes had to be made by the present author. The probes are shown in Figure 3.14. Both probes are built in the same manner. The only difference lies in the tip shapes. One of the probe has the tip bent upward to traverse the top wall boundary layer downward. The second probe allows the bottom wall boundary layer to be traversed.

The two probes consist of concentric stainless steel hypodermic tubes glued together using epoxy to provide the required stiffness. The diameters of the tubes are listed in Figure 3.14. The probes are 270mm long and have a maximum diameter of 3.40mm (0.1340in).

The probe tips are made using hypodermic tubes with outside diameter of 0.64mm (0.0253in). The tips are flexible. Therefore, it is possible to rest the probe tip against the wall to accurately determine the measurement locations. This procedure will be explained further in Section 4.5.7.

Figure 3.14: Boundary Layer Pressure Probe.

3.3.5 Miscellaneous Apparatus

3.3.5.1 Probe-Alignment Mechanism

An accurate value for the three-hole probe angle relative to the cascade outlet axial direction is obtained using a probe-alignment mechanism. The apparatus consists of a 30X Pocket Microscope mounted on a pinion and rack mechanism that translate parallel to the cascade outlet metal angle. A system of dowel pins and holes precisely drilled on a guide plate resting against the cascade trailing edge is used to accurately align the traverse mechanism with the blades outlet angle. Figure 3.15 shows a schematic of the probe alignment mechanism.

3.3.5.2 Transducer Calibration Setup

The Kulite pressure transducer used for the pressure measurements was calibrated using a setup built for this purpose. The setup consists of a pressure chamber with two valves, as shown in Figure 3.16. One valve is connected to a vacuum pump

Figure 3.15: Probe Alignment Mechanism (Jouini, 2000).

for calibration at pressures below atmospheric pressure. The other valve is connected to the shop air line for calibration at pressures higher than the ambient pressure.

The Kulite pressure transducer located in the Scanivalve head was connected to the chamber using the Scanivalve pressure manifold usually reserved for the cascade inlet total pressure. The transducer output voltage for a given chamber pressure was recorded using the data acquisition system described previously in Section 3.3.2.

A U-tube mercury manometer was used to record the air chamber gauge pressure. The manometer accuracy was $\pm 0.05 \text{ inHg}$ ($\pm 0.169 \text{ kPa}$). The atmospheric pressure was recorded using an Omega digital barometer with an accuracy of $\pm 0.001 \text{ inHg}$ ($\pm 0.00339 \text{ kPa}$).

Figure 3.16: Transducer Calibration Apparatus (Adapted from Jouini, 2000).

3.3.6 Hot-Wire Anemometry Instrumentation

3.3.6.1 Anemometer Unit

A Thermo-Systems Inc. single channel Constant Temperature (CT) anemometer model number 1054A was used for the turbulence measurements. This anemometer unit was used in conjunction with a variable decade resistor model 1056 which allows change in the overheat ratio to be made easily. Constant temperature anemometers are normally used for measurements in isothermal flows due to their very good frequency response, sometimes in excess of 100kHz. With proper corrections, constant temperature anemometers can be used in non-isothermal flows.

3.3.6.2 Hot-Wire Probe

The probe used for the measurements was a Dantec gold-plated tungsten wire probe, model 55P01. Plated probes have the advantage of minimizing the aerodynamic disturbances caused by the prongs.

3.3.6.3 Digital Data Acquisition

The analog voltage signal from the anemometer unit was digitized using the Hewlett-Packard HP44702B 13-bit high speed voltmeter and HP3852A data acquisition unit. As for the other measurements, the data acquisition unit was controlled using a microcomputer equipped with an HP-IB (HP-Interface Bus) card. Sampling rates as fast as 100kHz per channel were possible for spectral-analysis purposes and time-domain analysis. The data were initially stored in a 64kBytes buffer in the data acquisition unit and then downloaded to the microcomputer for analysis.

3.4 Cascade Geometries

3.4.1 Overview

Three cascade geometries were used for this study. The baseline cascade, designated HS1A, was scaled from the midspan section of an HP turbine from a Pratt & Whitney Canada engine. The two other blades used for this investigation were designed by the author. These latter two blades are essentially derivatives of the baseline cascade but with modified loading distributions. The airfoil and cascade geometries are summarized in Table 3.1. Table 3.1 gives also the design flow parameters in addition to the geometric parameters. Figure 3.17 gives a summary of a blade nomenclature. Finally, Figure 3.18 shows the three profiles superimposed and the corresponding loading distributions. HS1A is regarded as mid-loaded while one of the new profile is aft-loaded (HS1C) and the other is front-loaded (HS1D). The cascades were manufactured with a common axial chord of 37.3mm.

3.4.2 HS1A Turbine Cascade

As mentioned previously, the baseline blade, HS1A (High Speed 1A), is the midspan section of an high pressure turbine from a Pratt & Whitney Canada (PWC) engine. Uncooled HP turbine blade such as this one must be able to sustain very high temperatures as they are located just downstream of the combustor outlet. This explains its relatively thick profile as well as its large leading edge normal axis.

Figure 3.17: Summary of Symbols Used to Describe an Airfoil Geometry.

Figure 3.18: Cascade Blade Profiles and the Corresponding Loading Distribution.

Parameter	HS1A	HS1C	HS1D
Chord Length, C	41.2 mm	40.4 mm	43.3 mm
Axial Chord, C_x	37.3 mm	37.3 mm	37.3 mm
Blade Span, H	61.0 mm	61.0 mm	61.0 mm
Blade Pitch, s	29.14 mm	29.14 mm	29.14 mm
Aspect Ratio, H/C	1.481	1.51	1.409
Inlet Metal Angle, α_1	50.5°	50.5°	50.5°
Leading Edge Wedge Angle, We_{le}	38.0°	15.0°	38.0°
Leading Edge Ellipse Ratio, a/b	1.0	2.1	1.0
Leading Edge Minor Axis, b	1.02 mm	1.02 mm	1.02 mm
Outlet Metal Angle, α_2	59.0°	59.0°	59.0°
Trailing Edge Wedge Angle, We_{te}	6.0°	6.0°	6.0°
Trailing Edge Thickness, t	1.26 mm	1.26 mm	1.26 mm
Stagger Angle, α	25.1°	22.5°	30.5°
Design Incidence, i_{des}	-4.5°	-4.5°	-4.5°
Unguided Turning, α_u	11.5°	14.5°	11.5°
Throat Opening, o	15.9 mm	15.9 mm	15.9 mm
Solidity, s/C	0.707	0.721	0.673
Inlet Mach Number, M_1	0.53	0.53	0.53
Exit Mach Number, M_2	1.04	1.04	1.04
Inlet Flow Angle, α_1	46.0°	46.0°	46.0°
Exit Flow Angle, α_2	58.5°	58.5°	58.5°
Zweifel Coefficient, Z	0.856	0.856	0.856

Table 3.1: Blade Geometric and Design Flow Parameters.

The baseline blade (HS1A) is considered to be a mid-loaded profile. As shown in Figure 3.18, for blade HS1A, the bulk of the loading is concentrated around mid-chord. At this location, a maximum is reached in the Mach number distribution on the suction surface.

HS1A blade profile can be seen in Figure 3.19(a) together with the curvature distribution. The curvature distribution will be referred to later in the discussion of the results.

3.4.3 HS1C Turbine Cascade

Starting from the baseline profile, a second blade was designed by the author using Pratt & Whitney Canada design tools. The objective was to obtain an aft-loaded profile with the same Zweifel coefficients as the baseline profile. In addition, the metal area was kept constant so that the new profile would be structurally compatible with the existing engine airfoils. Finally, it was desired to keep the same metal angles as for HS1A.

The aft-loaded airfoil (HS1C) was obtained from HS1A by modifying several geometric parameters. The stagger angle is the geometric parameter that has the most influence on the loading distribution. The stagger angle was reduced from 25.1° to 22.5° . The uncovered turning angle was increased by 3.0° to 14.5° . The leading edge ellipse ratio and wedge angle were also modified to reduce the velocity overspeeds on both the pressure and suction surfaces in the vicinity of the leading edge. Finally, the suction side curvature was modified to adjust the pressure distribution. As shown in Figure 3.19(b), the suction surface curvature was reduced in the forward part of the blade as well as on the rear part. The resulting Mach number distribution can be seen in Figure 3.18. In this figure, it can be observed that the loading has been reduced slightly on the forward part of the airfoil and increased on the aft part in comparison to the baseline blade.

3.4.4 HS1D Turbine Cascade

The front-loaded airfoil (HS1D) was obtained mainly by increasing the stagger angle to 30.5° . As for HS1C, the Zweifel loading coefficient, the metal area and

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 3.19: Blade Surface Curvature for the Three Blade Profiles.

the metal angles were kept constant. Some minor modifications were also made to the suction-side curvature as shown in Figure 3.19. The suction surface curvature was reduced on the front part of the airfoil as well as on the aft part. The loading distribution for HS1D is also shown in Figure 3.18. The increase in loading on the forward part of the airfoil relative to HS1A can clearly be seen.

3.4.5 Cascade Assembly

In order to form a cascade, the individual blades are mounted on the cascade base plate, as shown in Figure 3.20. This cascade assembly is then inserted in the turntable in the wind tunnel test section. The cascades are composed of 7 blades and 8 complete blade passages. The base plate is instrumented with a row of static taps located at a downstream distance equivalent to 43% of axial chord. The static taps are spaced such that there are five taps per blade passage. The static tap located in the middle of the cascade was used as the cascade downstream reference static pressure and used to monitor the test section operating point during a run.

As seen in Figure 3.20, blades 3 and 5 can be replaced with blades instrumented with static taps. Therefore, surface static pressure measurements can be performed. Blade 3 was instrumented with static taps on the pressure surface and blade 5 with static taps on the suction surface. The static taps on the blade surfaces were staggered in order to have a minimum of 50 diameters spacing between adjacent taps, as recommended by Roach and Turner (1988). The number of static taps for cascade HS1A, HS1C and HS1D is summarized in Table 3.2. The locations of the static taps on the blade surfaces are tabulated in Appendices D, E and F together with the measured surface Mach number and static pressure coefficients.

For cascade HS1A, the blade with the suction surface taps has a base pressure tap at the trailing edge. The static tap has a diameter of 0.25mm (0.010in) which corresponds to 20% of the trailing-edge thickness. The blade with the pressure surface static taps does not have a base pressure tap. Blades HS1C and HS1D have base pressure taps on both instrumented blades. The base pressure tap on the blade with the suction surface taps has a diameter of 0.25mm (0.010in) for both HS1C and HS1D. However, the base pressure tap on the blade with the pressure surface taps has

Figure 3.20: Cascade Assembly.

	HS1A	HS1C	HS1D
Pressure Surface	19	16	16
Suction Surface	22	24	24
Base Pressure	1	2	2

Table 3.2: Blade Static Taps Numbers.

a diameter of 0.51mm (0.02in) for blade HS1C and a diameter of 0.38mm (0.015in) for blade HS1D. These additional base pressure taps were added to verify the effect of tap diameter on the measured base pressure.

Chapter 4

Experimental Procedures

4.1 Introduction

This chapter describes the procedures that were used in the operation of the wind tunnel and the collection of the data.

4.2 Pressure Transducer Calibration

A single Kulite absolute pressure transducer was used for the pressure measurements, as mentioned previously. The pressure transducer was calibrated using the apparatus described in Section 3.3.5.2 over a range of pressures varying between 50kPa and 200kPa, which corresponds to the range of pressures encountered during the tests in the wind tunnel.

For every calibration data point, the transducer output voltage was recorded using the data acquisition system. The reservoir's gauge pressure was also recorded using the U-tube mercury manometer. For each pressure measurement made using the manometer, the corresponding instantaneous atmospheric pressure was recorded using an Omega digital barometer. Thus, the absolute chamber pressure could be calculated for every calibration data point. The calibration data points were then fitted using a linear function in order to obtain the calibration equation.

Figure 4.1 shows selected calibration curves for the Kulite pressure transducer over a one year period of time. It can be observed that the transducer is very stable and that the calibration was practically unchanged over the course of the year.

The absence of hysteresis in the transducer output was also verified. The calibration was performed by first increasing the pressure in steps from 0kPa to about

Figure 4.1: Kulite Pressure Transducer Calibration Over a Year Period.

200kPa. Then, the transducer was calibrated for pressures from 200kPa down to 50kPa. Finally, the pressure was brought back to the ambient value. As seen from the calibration curves shown in Figure 4.1 no observable hysteresis is present.

4.3 Probe Alignment Procedure

The three-hole probe angle relative to the cascade outlet axial direction is determined using the probe alignment mechanism described in Section 3.3.5.1. First, the guide plate is fixed against the cascade. Then, the probe alignment mechanism is fixed to the guide plate and set parallel to the cascade outlet metal angle by inserting the pins in the appropriate precision holes.

Once the alignment mechanism is in place, the 30X Pocket Microscope's cross hair is positioned along one of the probe's edges. The cross hair line having graduations is placed perpendicular to the probe. The microscope is then moved a pre-determined distance using the traverse mechanism. The cross hair offset from the probe edge is recorded by reading the graduations on the microscope lens and the probe misalignment angle can be determined as shown in Figure 4.2

Figure 4.2: Probe Alignment Procedure (Jouini, 2000).

4.4 Three-Hole Probe Calibration

The outlet flow angle measurements and total pressure measurements are obtained from the three-hole probe pitchwise traverses at the cascade exit. In order to extract the flow angles and the total pressure from the three-hole probe measurements, the probe must be calibrated for a range of yaw angles. The three-hole probe was calibrated using a custom designed calibration rig which was inserted into the test section in place of the cascade as shown schematically in Figure 4.3. The calibration rig is simply a convergent nozzle which allows the probe to be calibrated in yaw. The probe was calibrated from time to time for yaw angles varying from -10.0° to $+10.0^\circ$ in increments of 1.0° . The probe was calibrated for a fixed pitch angle of 0.0° in non-nulling mode over a range of Mach numbers varying between 0.6 and 1.0. However, in that range of flow velocities, the effect of Mach number on the calibration coefficients was found to be small. Therefore, the final calibration coefficients used for the data processing correspond to those obtained from curve fits of the data for

Figure 4.3: Schematic Diagram of the Calibration Insert (Top View) (Jouini, 2000).

all the Mach numbers.

The calibration data were processed according to the method outlined by Lewis (1966). The method is explained in detail by Yaras (1991). For this calibration technique, the following coefficients were formed and plotted as a function of the yaw angle ϕ :

$$K_1 = \frac{P_1 - P_s}{P_{o1} - P_s} \quad (4.1)$$

$$K_{12} = \frac{P_1 - P_2}{P_{o1} - P_s} \quad (4.2)$$

$$K_{13} = \frac{P_1 - P_3}{P_{o1} - P_s} \quad (4.3)$$

$$K_{23} = \frac{P_2 - P_3}{P_{o1} - P_s} \quad (4.4)$$

Figure 4.4: Three-Hole Probe Numbering System and Orientation.

$$K_{\phi 2} = \frac{P_1 - P_2}{P_1 - P_3} \quad (4.5)$$

$$K_{\phi 3} = \frac{P_1 - P_3}{P_1 - P_2} \quad (4.6)$$

where P_1 , P_2 and P_3 are the three port pressures. The numbering system used for the pressure ports is defined in Figure 4.4. P_{o1} and P_s are the local total and static pressures respectively. The probe was used into the test section with the same orientation as used during calibration.

The coefficients defined previously allow several quantities such as the total pressure, static pressure and flow angle to be extracted from the three-hole probe measurements taken in an unknown flow. For the current study, only the coefficients K_1 , $K_{\phi 2}$ and $K_{\phi 3}$ were used to calculate the total pressure and flow angle from the three-hole probe measurements downstream of the cascade. Static pressures downstream of the cascade were measured directly using a separate probe, as described below. The variations of the calibration coefficients of interest with yaw angle and Mach number are shown in Figures 4.5 to 4.7. The way the calibration curves were used is described in the next paragraphs.

The total pressure was calculated from the three-hole probe measurements using

Figure 4.5: K_1 Versus Yaw Angle ϕ .

Figure 4.6: $K_{\phi 2}$ Versus Yaw Angle ϕ .

Figure 4.7: $K_{\phi 3}$ Versus Yaw Angle ϕ .

the K_1 coefficient. K_1 represents the local dynamic pressure as measured using the three-hole probe center port and the local static pressure normalized on the actual local dynamic pressure. From Figure 4.5, it can be seen that K_1 is essentially equal to 1.0 over the range of yaw angles and Mach numbers for which the calibration was performed. Therefore, under subsonic conditions, the total pressure downstream of the cascade was assumed equal to the pressure measured on the three-hole probe center port for yaw angles within ± 10.0 degrees relative to the probe centerline. At this time, the calibration of the three-hole probe is limited to exit Mach numbers less than 1.0. Therefore, a normal shock correction for the total pressure was applied to the three-hole probe total pressures for Mach numbers greater than one.

The flow angle relative to the probe centerline was calculated using the $K_{\phi 2}$ and $K_{\phi 3}$ coefficients presented in Figures 4.6 and 4.7. Both coefficients yield an estimate of the flow angle in the range of yaw angles from -10.0° to $+10.0^\circ$. However, as seen from Figure 4.6, the Mach number dependency for the $K_{\phi 2}$ coefficient is much smaller for positive yaw angles, whereas for negative yaw angles the Mach number dependency is less for $K_{\phi 3}$ (Figure 4.7). Therefore, the flow angle determined from

the $K_{\phi 2}$ fit was used whenever the flow angle was found to be positive and the $K_{\phi 3}$ fit was used for negative flow angles.

The uncertainty in the extracted flow angles was estimated by Jeffries (2001) to be $\pm 0.5^\circ$ for yaw angles between -10.0° and $+10.0^\circ$. A more detailed discussion of the uncertainties in the experimental measurements is given in Section 4.7.

4.5 Data Acquisition Procedure

4.5.1 Introduction

The vast majority of the data acquired for this project consists of pressure measurements. Part of the measured data were required for extracting cascade performance data. Some pressure measurements were also made to assess the flow quality in the wind tunnel. In the following sections, the different types of measurements are described.

4.5.2 Downstream Static and Three-Hole Probe Pitchwise Measurements

In order to obtain the cascade performance data, such as the losses, wake traverses downstream of the blade row had to be performed. The measurements were made using the three-hole probe as well as the static probe. These were described in Sections 3.3.4.1 and 3.3.4.2. The measurement plane was located at 43% axial chord downstream of the cascade blade row, as shown in Figure 4.8.

Forty measuring points are used over a distance equivalent to one blade pitch, as indicated in Figure 4.8. The total time required to obtain those measurements is approximately one minute. However, the useful data gathering time available for each run in the blow down wind tunnel varies between 10 and 60 seconds depending on the blowing pressure and on whether or not the ejectors are in use. A downstream traverse over one pitch is normally performed in four runs when the ejectors are not used. Thus, 10 measuring points are recorded per run. With the ejectors running, it was found to be necessary to use a total of eight runs for one pitch traverse. Hence, five measuring points per run were recorded when the ejectors were running.

Figure 4.8: Cascade Blade Row Measurement Locations.

Figure 4.8 shows the downstream measuring plane for the case where four runs are used for one pitch traverse. Each run is composed of ten measuring points equally spaced. However, the probe pitchwise displacement was halved for runs two and three in order to increase the resolution in the wake region.

The three-hole probe and static probe downstream traverses were performed separately. However, the measurements with both probes were performed at the same locations so that the data could be combined during the data processing phase.

For the three-hole probe traverses, six different pressure measurements were recorded at each traverse location, as shown in Figure 3.10. First, the inlet total pressure was measured at the test section inlet using a cobra probe. The location of the probe is shown in Figure 3.5. This total pressure was used as the reference inlet total pressure (P_{o1r}). The inlet static pressure was also recorded using a static tap located at approximately one true chord length upstream of the blade leading edge. This pressure was used as the reference inlet static pressure (P_{1r}). The static tap location is shown in Figure 4.8. The inlet reference static pressure was used in

combination with the inlet total pressure to monitor the inlet isentropic Mach number during each run.

A second reference static pressure was recorded downstream of the cascade. This is the reference downstream static pressure (P_{2r}). The static tap used is located on the cascade base plate in the measuring plane, as seen in Figure 4.8. The reference downstream static pressure was used to monitor the pressure ratio across the cascade.

The last three pressure measurements correspond to the three ports of the three-hole probe. The center hole of the three-hole probe was used to measure the downstream total pressure. The pressure measurements from the two side holes were used to extract the flow angle and static pressure.

For the cases where static probe traverses were made, four pressure measurements were required for each measurement location. The Scanivalve arrangement used for the static probe traverses is shown in Figure 3.9.

The first three measurements made for the static probe traverses are the same three reference pressures mentioned previously for the three-hole probe traverses. The fourth pressure measurement is that of the static probe.

In order to maximize the number of measuring locations during a single wind tunnel run, the six tubes from the pressure probes and taps were each connected to individual pressure manifolds having one inlet port and eight outlet ports. The outlet ports of the six pressure manifolds were then connected sequentially to the 48 ports of the Scanivalve as shown in Figure 3.9 and 3.10. The advantage of this configuration is that the Scanivalve never needs to be homed after a sequence of six pressure readings is taken at each measuring location. Instead the valve is stepped through all 48 ports visiting each of the six measurement pressures repeatedly. This results in a significant time saving and thus increases the number of points that can be collected in a single run since the time required to home the valve is significant.

4.5.3 Reference Inlet Static and Total Pressure Corrections

As mentioned previously, the reference inlet total pressure measurement was conducted at the inlet of the test section, upstream of the contraction. However, some total pressure losses occur between the test section inlet and the cascade

inlet. Furthermore, the inlet flow total pressure is not uniform and therefore the reference probe does not see the same total pressure as what appears on the centerline at the cascade inlet. Therefore, in order to get an accurate value of the total pressure at the cascade inlet additional calibration measurements were performed. For every condition tested, a run was performed where a Pitot-static probe was inserted at a distance upstream of the cascade equivalent to about one true chord, as indicated in Figure 4.8. The total pressure (P_{o1}) and the static pressure (P_1) were measured using the Pitot-static probe together with the reference pressures (P_{o1r}) and (P_{1r}).

From these measurements, the correction factors $\frac{P_{o1}}{P_{o1r}}$ and $\frac{P_1}{P_{1r}}$ are calculated. Typical values for $\frac{P_{o1}}{P_{o1r}}$ and $\frac{P_1}{P_{1r}}$ are 0.997 and 0.995 respectively. When processing the data, these correction factors were applied to the measured reference pressures to get the actual total and static pressure at midspan at the cascade inlet.

4.5.4 Blade Surface Pressure Measurements

Blade surface pressure measurements were performed for all test cases for which downstream traverse data were acquired. The loading measurements were obtained using the instrumented blades described previously in Section 3.4.5.

The inlet reference total pressure (P_{o1r}) as well as the inlet reference static pressure (P_{1r}) and the outlet reference static pressure (P_{2r}) were recorded using respectively port 0, 1 and 2 of the Scanivalve. The surface static pressures for the suction surface were recorded using the Scanivalve, from the leading edge to the trailing edge, including the base pressure tap. Finally, the surface static pressures on the pressure side of the airfoil were measured from the trailing edge to the leading edge using the next available ports on the Scanivalve.

Since, there is no time required to displace the probe using the traversing mechanism when doing loading measurements, the runs are much shorter. Thus, the loading measurements for a given test condition can be performed in a single run for all combinations of wind tunnel test section and ejector blowing pressures used in the current research.

The loading measurement results can be expressed in two forms. First, they can be presented as isentropic Mach number distributions. Thus, from the ratio of

the inlet reference total pressure (P_{o1}) and the surface static pressure (P_s)_x a surface Mach number can be calculated as follows:

$$(M_s)_x = \left(\left(\frac{2}{\gamma - 1} \right) \left(\left(\frac{P_{o1}}{(P_s)_x} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right) \right)^{1/2}. \quad (4.7)$$

Alternatively, the surface static pressure can be expressed as a static pressure coefficient given by

$$C_{ps} = \frac{(P_s)_x - P_1}{q_1}. \quad (4.8)$$

In this equation, the surface static pressure measurement is referenced to the inlet reference static pressure and non-dimensionalized using the inlet dynamic pressure.

For the current work, the loading measurements are mostly presented as surface isentropic Mach number distributions. However, both the surface Mach number and static pressure coefficient distributions are tabulated in Appendices D, E and F for all test cases. Repeatability measurements of blade loadings by Jouini (2000) and the present author have shown that the uncertainty on the Mach number is about ± 0.01 . A typical blade loading is presented in Figure 4.9 for cascade HS1D at design incidence and Mach number.

It can be observed that the repeatability is very good. A small discrepancy can be seen on the pressure surface. Such variation in the blade loadings are usually due to drift in the test section blowing pressure.

4.5.5 Base Pressure Measurements

As mentioned previously, the base pressure measurements were made in conjunction with the loading measurements. Thus, the base pressure was recorded for the same test conditions as the loading measurements.

In the following sections, the base pressure measurements are presented in the form of a base pressure coefficient defined as

$$C_b = \frac{P_b - P_{s2}}{q_2}. \quad (4.9)$$

In this equation, the base pressure P_b is referenced to the mixed-out outlet static

Figure 4.9: Blade Loading for Cascade HS1D at Design Flow Conditions.

pressure. The coefficient is made dimensionless by dividing by the outlet dynamic pressure.

For cascades HS1C and HS1D two base pressure measurements are obtained for each operating point since two trailing-edge taps of different diameters are present. The additional base pressure tap was manufactured for cascade HS1C and HS1D in order to verify if the hole size had any effect on the base pressure measured. The idea of having two base pressure taps of different sizes arise from results published by Sieverding et al. (2003) which showed that the static pressure experiences large variations around the trailing edge in transonic flow regime. Therefore, by having taps of different sizes at the trailing edge, it was possible to verify if the tap size had an effect on the magnitude of the pressure measured.

4.5.6 Pitchwise Inlet Static Pressure Measurements

The inlet flow pitchwise uniformity was assessed by measuring the end wall static pressure across the cascade inlet plane. This was achieved using a row of static

taps drilled one true chord upstream of the cascade inlet. The location of the static taps is shown in Figure 4.8. The static taps cover the whole cascade in the pitchwise direction and are spaced such that there is 5 taps per blade pitch.

Inlet pitchwise static pressure measurements were performed whenever the incidence of the cascade was changed. This was required in order to insure the presence of uniform flow at the cascade inlet. More details on the procedure to obtain good inlet flow uniformity will be given in Section 6.2.

The procedure used to measure the inlet pitchwise static pressure is similar to that performed to record the blade-surface static pressures. The inlet reference total pressure (P_{o1r}) as well as the inlet reference static pressure (P_{1r}) and the outlet reference static pressure (P_{2r}) were recorded using respectively port 0, 1 and 2 of the Scanivalve. The inlet pitchwise static pressure is then recorded sequentially using the following Scanivalve ports from one end of the cascade to the other.

4.5.7 Inlet Spanwise Total Pressure Measurements

Measurements were also made in the spanwise direction at the inlet of the cascade. The location of the measurements corresponds to the position of the Pitot static probe indicated in Figure 4.8. These measurements were required in order to determine the extent of the uniform flow region in the spanwise direction and the thickness of the endwall boundary layers.

The inlet spanwise measurements were performed using the custom-made boundary layer probes described previously in Section 3.3.4.3. The boundary-layer probes were mounted in a small spanwise traverse mechanism and set perpendicular to the endwalls. The traverse mechanism was actuated using the same Compumotor stepper motor used for the downstream traversing gear and thus the same software was used to control it.

The pressure measurements were recorded using the 48 port Scanivalve fitted with the pressure manifold as for the three-hole probe and static probe downstream traverses. Thus, the first pressure manifold was used to record the inlet reference total pressure (P_{o1r}). The second pressure manifold recorded the inlet reference static pressure (P_{1r}) and third manifold the outlet reference static pressure (P_{2r}). The

boundary layer probe readings were recorded using pressure manifold number four.

In order to calculate the boundary layer integral parameters accurately, the probe's position must be located precisely relative to the endwalls. This was achieved by first making sure that the probe tip touched the surface first. Thus, the probe tip was inclined slightly relative to the surface, typically less than 1° . It was shown by Peterson and George (1975) that this has very negligible effect on the measured total pressure.

To find the exact point at which the probe's tip was resting against the wall the following procedure was followed. The probe was first lowered in the test section so that its flexible end rested well against the wall. This resulted in the probe tip bending upward and away from the wall as shown in Figure 4.10. The probe was then moved upward in steps of 0.2mm and the total pressure as measured by the probe was recorded. As shown in Figure 4.10, the total pressure measured by the probe decreases initially indicating that the probe tip is approaching the wall. When a minimum is reached in the total pressure variation, this is an indication that the probe's tip is just touching the wall. This point is considered the starting location for the probe. Finally, as the probe is moved further the pressure is seen increasing again indicating that the probe tip is moving away from the wall. Therefore, by plotting the pressure variation in the vicinity of the wall, the actual probe location can be determined accurately.

4.6 Sampling Rates and Times

An investigation of the sampling rates and times required to obtain accurate mean values of the different pressure measurements was performed by Jeffries (2001). During this study, it was found that the transducer output voltages contained both low and high frequency signals.

The low frequency fluctuations are mainly due to the pressure control system and have a strong peak at a frequency of 0.5Hz. In order to reduce the magnitude of these low frequency oscillations, the pressure control system gain was reduced. However, lowering the gain setting was not enough to completely remove the low frequency pressure signal. Furthermore, reducing the gain lengthened the start-up

Figure 4.10: Boundary Layer Probe Starting Position.

transient which results in shorter useful run times. Hence, a balance must be struck between acceptable pressure fluctuation amplitude and the length of the start-up transient. Thus, there will always be some low frequency pressure fluctuations. Therefore, the measurements will have an uncertainty on the order of the magnitude of the peak-to-peak amplitude of the low frequency fluctuations.

As mentioned, the signal also contains some high frequency fluctuations. These fluctuations are mainly due to the flow turbulence. These pressure fluctuations due to turbulence also contribute to uncertainties in the measurements. These uncertainties are mainly a function of the sampling rates and times used to acquire the data.

By looking at the frequency spectra of the pressure transducer output signal Jeffries (2001) was able to determine that it contained fluctuations up to about 1000Hz. Therefore, applying the Nyquist sampling criteria, which states that the minimum sampling frequency should be at least twice the maximum frequency found in the signal, the data are sampled at a rate of 2000Hz.

The longer the sampling time used the closer the mean sampled value will be to

the long term actual mean. However, as the sampling time for each measurement is increased, the number of data points that can be collected during each run is reduced. Jeffries (2001) found that at a sampling time of 0.1 second provided the best trade-off between the accuracy of the mean values and the wind tunnel productivity.

4.7 Measurement Uncertainties

4.7.1 General Procedure

In this section, an estimate of the uncertainty in the different quantities of interest will be provided. This includes the uncertainty in the pressure measurements as well as in the reduced data such as the Mach numbers, Reynolds numbers, Axial Velocity Density Ratio (AVDR) and loss coefficient.

The uncertainty analysis was performed following the guidelines specified in the AIAA Standard S-071A-1999 titled “Assessment of Experimental Uncertainty With Application to Wind Tunnel Testing”. Furthermore, the method presented by Baines et al. (1991) to perform uncertainty analysis in turbomachinery and cascade testing was followed closely. The method is largely based on the work of Moffat (1982).

The basic analysis of uncertainties and how the individual errors combine into an overall uncertainty can be summarized briefly as follows. Any derived parameter F can be expressed as a function of various fundamental quantities $\phi_1 \dots \phi_N$:

$$F = F(\phi_1, \phi_2, \dots \phi_N). \quad (4.10)$$

Each fundamental quantity ϕ may then be written as

$$\phi_i = \hat{\phi}_i + \delta\phi_i \quad (4.11)$$

where $\hat{\phi}_i$ is the true value and $\delta\phi_i$ is the uncertainty. The contribution to the uncertainty of F due to an uncertainty in ϕ_i is then

$$(\delta F)_i = \left(\frac{\partial F}{\partial \phi_i} \right) \delta\phi_i. \quad (4.12)$$

Therefore, provided that the individual ϕ 's are independent and normally distributed, the resultant uncertainty of F is

$$\delta F = \sqrt{\left(\frac{\partial F}{\partial \phi_1}\right)^2 \delta \phi_1^2 + \left(\frac{\partial F}{\partial \phi_2}\right)^2 \delta \phi_2^2 + \dots + \left(\frac{\partial F}{\partial \phi_N}\right)^2 \delta \phi_N^2}. \quad (4.13)$$

4.7.2 Uncertainties on Pressure Measurements

The uncertainty in the pressure measurements depends on the uncertainty in the pressure transducer calibration and the fluctuation in the measured pressure signal.

For this research project a single absolute pressure transducer was used, as mentioned in Chapter 3. The transducer voltages were converted to pressure using the following relation:

$$P = mV + z. \quad (4.14)$$

In this equation, P is the pressure, m is the transducer calibration curve slope, V is the measured transducer voltage and z is the output voltage at zero absolute pressure.

Using the method described previously, the uncertainty in the pressure becomes

$$\delta P = \sqrt{\left(\frac{\partial P}{\partial m}\right)^2 \delta m^2 + \left(\frac{\partial P}{\partial V}\right)^2 \delta V^2 + \left(\frac{\partial P}{\partial z}\right)^2 \delta z^2}, \quad (4.15)$$

where

δP is the uncertainty in the pressure,

δm is the uncertainty in the slope of the transducer calibration equation,

δV is the uncertainty in the voltage and

δz is the uncertainty in the zero pressure voltage of the calibration equation.

The partial derivatives are simply obtained by differentiating Equation 4.14. Therefore, the uncertainty in the pressure becomes

$$\delta P = \sqrt{(V\delta m)^2 + (m\delta V)^2 + \delta z^2}. \quad (4.16)$$

The values for the slope of the transducer calibration equation and the zero

Figure 4.11: Relative Pressure Uncertainty Versus Pressure Level.

pressure voltage were obtained from linear regressions used to fit straight lines through the pressure transducer calibration data shown in Figure 4.1.

The uncertainty on the voltage signal was obtained from a study performed by Jeffries (2001) on the sampling rate and times. Jeffries (2001) found that the uncertainty in the mean voltage after a sampling time of 0.1 second at a sampling frequency of 2kHz was approximately $\pm 0.25\%$ of the absolute value with a 95 percent confidence limit.

The variation of the relative uncertainty for the absolute pressure measurements is shown in Figure 4.11 for the complete range of the pressure transducer. For the current study, the pressures measured were no lower than 50000kPa (0.5Bars). Therefore, the relative uncertainty in the measured pressures can be assumed approximately constant at $\pm 0.3\%$ of the absolute pressure measured.

4.7.3 Uncertainties on Flow Angles

In this section an estimate in both the inlet and outlet flow angle will be provided.

The cascade inlet flow angle was obtained by two means. The first method to adjust the inlet flow angle consists in using the vernier scale attached to the test section. This vernier scale measures the angle of the turntable, onto which the cascade is mounted, relative to the centerline of the test section. The vernier has a smallest division of 0.1 degree. However, it is not believed that such accuracy in the inlet flow angle can be achieved in setting up the cascade in the test section.

The inlet flow angle is also a function of the orientation of the inlet side walls upstream of the cascade. Therefore, in order to set the inlet flow angle when installing the cascade, the angle between the inlet side walls and the cascade inlet plane can be measured using a protractor. It is believed that setting the inlet flow angle in this manner is preferable. Using this method, inlet flow angles within $\pm 0.5^\circ$ of the desired value can be achieved with high confidence.

The uncertainty in the outlet flow angle is a function of both the uncertainty in the flow angle extracted from the three-hole probe calibration and the uncertainty in the angle of the probe relative to the trailing edge of the blade.

As mentioned previously, the uncertainty in the flow angles extracted from the three-hole probe calibration was found to be $\pm 0.5^\circ$ for angles of misalignment between -10° and $+10^\circ$ relative to the probe centerline.

The angle of the probe relative to the trailing edge of the blade was measured using the alignment procedure described in Section 4.3. The following equation is used to determine the angle of the probe relative to the cascade outlet metal angle:

$$\phi = \arctan \left(\frac{Y_2 - Y_1}{X_2 - X_1} \right) \quad (4.17)$$

where X_1 and X_2 are the longitudinal coordinates of the reference points on the probe body and Y_1 and Y_2 are the lateral coordinates of the reference points on the probe body.

Thus, the uncertainty in the probe angle relative to the cascade outlet metal angle is a function of individual measurements of the longitudinal and lateral co-

ordinates of the reference points on the probe body. Hence, based on the method previously explained, the uncertainty can be expressed as follows

$$\delta\phi = \sqrt{\left(\frac{\partial\phi}{\partial Y_1}\right)^2 \delta Y_1^2 + \left(\frac{\partial\phi}{\partial Y_2}\right)^2 \delta Y_2^2 + \left(\frac{\partial\phi}{\partial X_1}\right)^2 \delta X_1^2 + \left(\frac{\partial\phi}{\partial X_2}\right)^2 \delta X_2^2}. \quad (4.18)$$

The accuracy of the longitudinal measurements was $\pm 0.127\text{mm}$ ($\pm 0.005\text{in}$) and that of the lateral measurements was $\pm 0.0318\text{mm}$ ($\pm 0.00125\text{in}$). From Equation 4.18, the uncertainty on the probe alignment is estimated to be $\pm 0.02^\circ$.

From the above discussion, it can be seen that the uncertainty in the outlet flow angle is dominated by the uncertainty in the angles extracted from the three-hole probe calibration. Therefore, the overall uncertainty in the outlet flow angle is estimated to be $\pm 0.5^\circ$ for angles of misalignment between -10° and $+10^\circ$ relative to the probe centerline.

4.7.4 Uncertainties on Mach Numbers

The isentropic Mach number is given by the equation

$$M = \sqrt{\left(\frac{2}{\gamma - 1}\right) \left(\left(\frac{P_o}{P}\right)^{\frac{\gamma-1}{\gamma}} - 1\right)}. \quad (4.19)$$

The uncertainty on the Mach numbers is thus given by

$$\delta M = \sqrt{\left(\frac{\partial M}{\partial P_o}\right)^2 \delta P_o^2 + \left(\frac{\partial M}{\partial P}\right)^2 \delta P^2}. \quad (4.20)$$

Figure 4.12 shows the variation of the inlet Mach number as a function of the cascade outlet Mach number with the corresponding uncertainties for the baseline cascade HS1A. The uncertainties in Mach number were obtained from Equation 4.20. The uncertainty on the inlet Mach number numbers is ± 0.008 at most over the Mach number range investigated. The exit Mach number uncertainty is less than ± 0.005 for all Mach numbers.

Figure 4.12: Uncertainties for the Cascade Mach Numbers.

4.7.5 Uncertainties on Total Pressure Loss Coefficients

The total pressure loss coefficient is given by

$$Y_t = \frac{P_{o1} - P_{o2}}{\frac{1}{2} P_2 \gamma \left(\frac{2}{\gamma-1} \right) \left(\left(\frac{P_{o2}}{P_2} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right)}. \quad (4.21)$$

Hence, the uncertainty in the total pressure loss coefficient can be written as

$$\delta Y_t = \sqrt{\left(\frac{\partial Y_t}{\partial P_{o1}} \right)^2 \delta P_{o1}^2 + \left(\frac{\partial Y_t}{\partial P_{o2}} \right)^2 \delta P_{o2}^2 + \left(\frac{\partial Y_t}{\partial P_2} \right)^2 \delta P_2^2}. \quad (4.22)$$

The variation of the total-pressure loss coefficient with Mach number together with the uncertainty in the loss coefficients is shown in Figure 4.13 for the baseline cascade at design incidence. The uncertainty in the total-pressure loss coefficient decreases with increasing outlet Mach numbers. At low outlet Mach numbers, the uncertainty on the loss coefficient is ± 0.02 whereas it is ± 0.007 at the highest outlet

Figure 4.13: Uncertainties in Total Pressure Loss Coefficients for the Baseline Cascade at Design Incidence.

Mach number.

4.7.6 Uncertainties in the Axial Velocity Density Ratio

The Axial Velocity Density Ratio (AVDR) can be defined in terms of the measured quantities as

$$\Omega = \sqrt{\frac{\left(\frac{P_{o2}}{P_2}\right)^{\frac{\gamma-1}{\gamma}} \left(\left(\frac{P_{o2}}{P_2}\right)^{\frac{\gamma-1}{\gamma}} - 1\right)}{\left(\frac{P_{o1}}{P_1}\right)^{\frac{\gamma-1}{\gamma}} \left(\left(\frac{P_{o1}}{P_1}\right)^{\frac{\gamma-1}{\gamma}} - 1\right)}} \cdot \left(\frac{P_2}{P_1}\right) \cdot \left(\frac{\cos \alpha_2}{\cos \alpha_1}\right). \quad (4.23)$$

Figure 4.14: Uncertainties in AVDR for the Baseline Cascade HS1A at Design Incidence.

The uncertainty on the AVDR is thus given by

$$\delta\Omega = \sqrt{\left(\frac{\partial\Omega}{\partial P_{o1}}\right)^2 \delta P_{o1}^2 + \left(\frac{\partial\Omega}{\partial P_1}\right)^2 \delta P_1^2 + \left(\frac{\partial\Omega}{\partial P_{o2}}\right)^2 \delta P_{o2}^2 + \left(\frac{\partial\Omega}{\partial P_2}\right)^2 \delta P_2^2 + \left(\frac{\partial\Omega}{\partial \alpha_1}\right)^2 \delta \alpha_1^2 + \left(\frac{\partial\Omega}{\partial \alpha_2}\right)^2 \delta \alpha_2^2}. \quad (4.24)$$

The uncertainties obtained along with the measured values of AVDR for the baseline cascade HS1A are shown in Figure 4.14 for the design incidence. The uncertainty in the AVDR values is less than ± 0.02 for the whole range of Mach numbers investigated.

4.7.7 Uncertainties in the Outlet Reynolds Numbers

The Reynolds numbers based on exit conditions and blade chord C is given by

$$Re_2 = \frac{\rho_2 U_2 C}{\mu_2}. \quad (4.25)$$

The equation can be rewritten in terms of the measured quantities using the compressible flow relations as follows:

$$Re_2 = P_2 \cdot \sqrt{\left(\frac{2\gamma}{\gamma-1}\right) \left(\frac{1}{RT_{o2}}\right) \left(\frac{P_{o2}}{P_2}\right)^{\frac{\gamma-1}{\gamma}} \left[\left(\frac{P_{o2}}{P_2}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \left(\frac{C}{\mu}\right)}. \quad (4.26)$$

The uncertainty in the Reynolds number is then

$$\delta Re_2 = \sqrt{\left(\frac{\partial Re_2}{\partial P_{o2}}\right)^2 \delta P_{o2}^2 + \left(\frac{\partial Re_2}{\partial P_2}\right)^2 \delta P_2^2 + \left(\frac{\partial Re_2}{\partial T_{o2}}\right)^2 \delta T_{o2}^2}. \quad (4.27)$$

Figure 4.15 shows the variation of the average exit Reynolds number with outlet Mach number for the baseline cascade HS1A at design incidence together with the uncertainty associated with the Reynolds number. The uncertainty in the Reynolds number is seen to vary from $\pm 6,000$ at lowest Mach number investigated to $\pm 12,000$ at the highest Mach number.

In addition to the uncertainty in the average Reynolds number, there exists a variation in the Reynolds number related to the fact that the air total temperature in the test section is decreasing throughout a run, as discussed by Jeffries (2001) and Jouini (2000). The effect of this temperature drop on the Reynolds number is shown in Figure 4.15. The line labelled Re_{2min} corresponds to the typical Reynolds numbers at the start of the run. The line labelled Re_{2max} , on the other hand, corresponds to the Reynolds numbers at the end of the run. The variation in Reynolds number due to the temperature variation for low Mach number runs is about $\pm 5,500$ whereas it is $\pm 97,000$ at high Mach numbers.

However, the effect of this Reynolds number variation due to temperature on the loss coefficient is less than the estimated uncertainty in the loss coefficient due to all other sources of error, as shown by Jouini (2000).

Figure 4.15: Uncertainties and Variation in Exit Reynolds number for the Baseline Cascade HS1A at Design Incidence.

Parameter	Range	Uncertainty Range
Pressure	0.5 -1.5 bars	±0.3%
Inlet Flow Angle	36.0° - 56.0°	±1.0°
Outlet Flow Angle	+/-10.0° relative to probe center	±0.5°
Inlet Mach Number	<0.600	±0.008
Oultet Mach Number	0.500 - 1.250	±0.005
Loss Coeffcient	M _{2s} from 0.5 to 1.25	±0.020 – ±0.007
AVDR	M _{2s} from 0.5 to 1.25	±0.02
Reynolds Number	M _{2s} from 0.5 to 1.25	±5,500 – ±97,000

Table 4.1: Summary of the Uncertainties on the Different Parameters of Interest.

4.7.8 Summary of Uncertainties

Table 4.1 presents a summary of the uncertainties on the different parameters of interest.

4.8 Hot-Wire Anemometry Procedure

4.8.1 Location of Measurements

Turbulence measurements were made in the wind tunnel at approximately 300mm upstream of the cascade at the midspan location as shown in Figure 4.16. Access to the flow was through a window located on the top cover of the wind tunnel.

4.8.2 Hot-Wire Response Equation

The form of the hot-wire response equation chosen for the current investigation is King's law. The hot-wire response equation is assumed to be of the form

$$E^2 = A + B(U)^n. \quad (4.28)$$

In this equation, A , B and n are constants which are found by fitting the equation to the calibration data points. E is the voltage required to keep the wire at constant

Figure 4.16: Location of the Hot-Wire Probe in the Test Section.

temperature and U is the flow velocity. In order to optimize the value of the constants, a least-squares curve fitting method was used. Hence, the constants A , B and n were found by minimizing the following equation

$$SES = \sum_{i=1}^N (E_R^2 - E_C^2)^2 \quad (4.29)$$

where E_R is the measured calibration voltage and E_C is the voltage calculated from the power law relationship, Equation 4.28.

In order to evaluate the quality of the fit, the normalized standard deviation was calculated as follows

$$\varepsilon_u = \left(\frac{1}{N} \sum_{i=1}^N (1 - U_R/U_C)^2 \right)^{1/2}. \quad (4.30)$$

Calibration points that were found to have a large deviation were remeasured in order to improve the accuracy of the fit. Typical normalized standard deviations obtained for the fit were approximately 0.15%.

Figure 4.17: Typical Calibration Curve.

4.8.3 Hot-Wire Probe Calibration

To determine the value of the constants in the power law, a calibration of the hot-wire sensor must be performed in a known flow. This calibration procedure yields a set of data points giving the anemometer output voltage for known values of velocity. Typically, a set of 10 to 30 points spaced evenly over the velocity range will yield good results.

For the current investigation, the hot-wire calibration was performed using a DISA type 55D41 calibration apparatus, a small wind-tunnel like calibration unit which provides a uniform, subsonic and incompressible flow with known magnitude and direction. The turbulence intensity of the flow in the calibration unit is about 0.5%.

The relationship between the output voltage and the flow velocity is non-linear as shown in Figure 4.17 for a typical calibration of the hot-wire used in the current investigation.

4.8.4 Turbulent Intensity Calculations

The hot-wire time history records were processed to obtain statistical information such as the mean flow velocity and the Root Mean Square (RMS) of the velocity fluctuations, $\sqrt{u^2}$. The turbulence intensity, Tu , was obtained by dividing the RMS value of the velocity fluctuations by the mean velocity for a time-history record as follows:

$$Tu = \frac{\sqrt{u^2}}{\bar{U}} \quad (4.31)$$

where $\bar{U}$ is the local mean velocity as measured by the probe.

4.8.5 Turbulent Length Scale Calculations

The eddies in a turbulent flow exist in different sizes. The largest eddies tend to persist in the flow since they are not directly affected by the viscous effects. They contain only a small amount of energy. The energy containing eddies receive energy from the mean flow and transfer it to the smaller eddies. The dissipation of the turbulence occurs in the smallest eddies. It is possible to estimate the length-scale of the energy containing eddies. The usual way of defining the size of the energy-containing eddies is by the integral length-scale, L , defined by

$$L = \int_0^\infty f(r_1) d\tau_1 \quad (4.32)$$

where $f(r_1)$ is the spatial correlation coefficient. Expressed in terms of the longitudinal velocity fluctuation, the spatial correlation coefficient is given by

$$f(r_1) = \rho_{11}(r_1, 0, 0) = \frac{\overline{u(0)u(r_1)}}{\overline{u^2}}. \quad (4.33)$$

If the flow is in the r_1 direction, it is very difficult to determine the lateral correlation coefficient. This is due to the thermal wake interference between the two parallel sensors required to make this measurement in the flow direction. Hence, the parallel wire method is generally not used to determine integral length-scale in the mean flow direction.

Instead, Taylor's hypothesis can be used to estimate the length scale from

a single-point measurement. This hypothesis assumes that the eddy is essentially “frozen” as it is convected by the flow over a distance comparable with dimensions of the eddy. Thus measurements made at the same point in space at two different times separated by δt become equivalent to simultaneous measurement made at longitudinal separations of $U\delta t$. It can be stated as

$$\frac{d\bar{A}}{dt} \approx -\bar{U}_1 \frac{d\bar{A}}{dx_1}. \quad (4.34)$$

where a A is a quantity of interest, usually the velocity. In applying this formula, it is assumed that the flow is essentially unidirectional with uniform mean speed. Equation 4.34 relates the longitudinal variation to the temporal variation at a point.

Therefore, the integral length scale can be calculated from a single time-history record of the voltage signal using the autocorrelation coefficient function defined as

$$\rho_u(\tau) = \frac{\overline{u(t)u(t+\tau)}}{\overline{u^2}}, \quad (4.35)$$

where $u(t)$ is the velocity fluctuation at a certain time, t , and τ is the time delay. Since $u(t)$ exists as a digital record, the time shift is easily introduced computationally.

The integral time-scale which is a measure of the time separation over which the signal $u(t)$ and $u(t+\tau)$ are correlated can then be calculated. It is given by

$$T_I = \int_0^\infty \rho_u(\tau) d\tau \quad (4.36)$$

where $\rho_u(\tau)$ is the autocorrelation coefficient function.

The integral time-scale was multiplied by the average velocity as follows to obtain an estimate of the average length-scale of the energy-containing eddies

$$L = \bar{U}_1 T_I. \quad (4.37)$$

4.9 Test Matrices

Measurements were performed on the three cascades described previously, namely HS1A, HS1C and HS1D. A first series of tests was done for the which the ejector-

	HS1A				HS1C				HS1D			
Incidence	-10.0°	0.0°	5.0°	10.0°	-10.0°	0.0°	5.0°	10.0°	-10.0°	0.0°	5.0°	10.0°
Mach Number	0.5 → 1.3				0.5 → 1.3				0.5 → 1.3			
Reynolds Number	450,000 → 1,000,000				450,000 → 1,000,000				450,000 → 1,000,000			
Turbulence Length-Scale	15mm				15mm				15mm			
Turbulence Intensity	4%				4%				4%			

Table 4.2: Baseline Test Matrix.

diffuser was not used. This meant that the cascade exit flow Reynolds number varied with the outlet Mach number. As will be shown later, for the Reynolds number range that resulted from these test cases, the Reynolds number was high enough that the losses were essentially independent of this parameter.

For this first series of tests the Mach number was varied from about 0.5 to 1.3. Four values of incidence were investigated ranging from -10.0° to $+10.0^\circ$. The resulting range of Reynolds numbers for these Mach numbers was from about 450,000 to around 1,000,000. Table 4.2 summarizes the test cases covered in this first series of tests.

A second series of tests was performed to investigate further the effect of Reynolds number on the performance of the cascades. This set of tests involved the use of the ejector-diffuser assembly. This apparatus, described previously, allowed the Reynolds number to be varied independently of the Mach number. Using the ejectors, the static pressure at the cascade exit was lowered to reduce the Reynolds numbers. Similarly, the Reynolds number was increased to higher values while keeping the Mach number constant by partially blocking the ejector outlet.

The test matrix for this second series of tests is given in Table 4.3. The Reynolds number study was performed for the three cascades at four values of Mach number, namely 0.60, 0.85, 0.95 and 1.05, the design Mach number. The range of Reynolds numbers covered for this study varied considerably depending on the test Mach number.

HS1A Mid-loaded blade	M = 0.60	M = 0.85	M = 0.95	M = 1.05
	Re ($\times 10^5$)	Re ($\times 10^5$)	Re ($\times 10^5$)	Re ($\times 10^5$)
Incidence				
-10.0°	3.7 → 8.0	5.8 → 10.3	7.2 → 10.9	7.9 → 11.1
0.0°	3.1 → 8.8	5.8 → 10.3	6.7 → 10.5	7.4 → 11.3
5.0°	2.9 → 7.9	5.7 → 10.6	6.7 → 10.8	8.1 → 11.4
10.0°	2.9 → 7.1	5.5 → 10.4	6.4 → 10.4	8.0 → 11.2

HS1C Aft-loaded blade	M = 0.60	M = 0.85	M = 0.95	M = 1.05
	Re ($\times 10^5$)	Re ($\times 10^5$)	Re ($\times 10^5$)	Re ($\times 10^5$)
Incidence				
-10.0°	3.2 → 8.0	5.7 → 10.6	6.6 → 9.8	7.3 → 10.7
0.0°	3.2 → 8.1	5.9 → 10.6	6.7 → 10.0	7.5 → 10.6
5.0°	3.0 → 7.1	5.6 → 9.8	6.5 → 10.3	7.3 → 10.6
10.0°	2.9 → 7.1	5.4 → 10.2	6.3 → 10.0	7.2 → 10.4

HS1D Front-loaded blade	M = 0.60	M = 0.85	M = 0.95	M = 1.05
	Re ($\times 10^5$)	Re ($\times 10^5$)	Re ($\times 10^5$)	Re ($\times 10^5$)
Incidence				
-10.0°	4.0 → 8.7	6.3 → 11.0	7.2 → 10.8	8.8 → 12.7
0.0°	3.1 → 8.5	5.8 → 11.5	6.6 → 10.6	8.0 → 11.7
5.0°	3.1 → 7.3	5.6 → 10.4	6.5 → 10.5	7.3 → 11.1
10.0°	2.9 → 7.3	5.4 → 9.7	6.3 → 10.1	7.1 → 11.0

Table 4.3: Test Matrix for the Reynolds Number Study.

Chapter 5

Computational Procedure and Results

5.1 Introduction

In order to help in the interpretation of the experimental results, Computational Fluid Dynamic (CFD) simulations were performed using version 6.1 of the commercial code *FLUENT*®. In this chapter, the numerical method selected for the simulation will be briefly described together with the procedure for validating the results.

5.2 Two-Dimensional Compressible Navier-Stokes Solver (*FLUENT*®)

5.2.1 Governing Equations and Discretization Method

The commercial Navier-Stokes solver *FLUENT*® version 6.1 was used for all computations performed in the course of the project.

The two-dimensional steady simulations were performed using the segregated solver of FLUENT. For the particular cascade flow that was solved, the segregated solver was found to yield faster convergence rates over the coupled solver, even for the case where the outlet flow was supersonic. Using this method, the governing equations are solved sequentially as shown in Figure 5.1. The governing equations being non-linear, each discrete governing equation is linearized implicitly with respect to that equation's dependent variable. A point implicit (Gauss-Seidel) linear equation solver is used to solve the resultant scalar system of equations for the dependent variable in each cell. Several iterations must be performed in order to obtain a solution.

FLUENT uses a control volume based technique to convert the governing equations to algebraic equations that can be solved numerically. This control volume

Figure 5.1: Overview of the Segregated Solution Method (FLUENT, 1998).

technique consists of integrating the governing equations about each volume, yielding discrete equations that conserve each quantity on a control-volume basis. FLUENT stores the values of the different quantities it solves for at the cell centers. However, the cell face values are required for the convection terms of the governing equations. Therefore, the cell center values must be interpolated to the cell faces. This is accomplished using an upwind scheme. Several upwinding schemes are available in FLUENT. The second-order upwind scheme was used for the computations performed in the course of this work. A second-order pressure interpolation scheme was used as well in order to obtain the face pressures. The SIMPLE (Semi-Implicit Method for Pressure-Linked Equations) scheme was used to achieve pressure-velocity coupling.

Under-relaxation was used to reduce the changes in the dependent variables from iteration to iteration.

5.2.2 Computational Mesh

The computational grid consists of a single blade passage as shown in Figure 5.2. The flow field was meshed using a hybrid grid. The main flow field was meshed using an unstructured grid composed of triangular elements. A total of approximately 18000 nodes was used to mesh this region of the grid. The boundary-layer region consists of a structured grid of quadrilateral elements, as shown in the enlargement in Figure 5.2. This part of the grid is composed of about 2000 nodes. The size of the boundary-layer mesh was adjusted such that the dimensionless distance of the first element from the wall, y^+ , was about 1.0 everywhere around the blade. More details about the geometry of the the boundary-layer mesh are given in Figure 5.2.

Grid independence tests were performed and grids composed of 20000 nodes or more were found to yield essentially constant losses (Figure 5.3) as well as identical pressure distributions (Figure 5.4) for a given set of boundary conditions. The blade passage was initially meshed using a fine grid composed of approximately 60000 nodes. However, to speed up the computations, the grid for the main flow field was coarsened while maintaining the same parameters for the boundary-layer mesh. This explains the large discontinuity in the cells size observed in Figure 5.2 between the boundary-layer and main flow field meshes. Such a discontinuity in cell size is known to be the source of additional truncation errors. For the current grid, the depth of the boundary-layer mesh was such that it generally encompassed completely the predicted boundary-layer region, thus minimizing the gradients at the interface between the two grid regions. On the other hand, for regions where boundary-layer separation was present, it is conceivable that the extent of flow separation might not have been capture adequately. This should be kept in mind when looking at the computational results.

5.2.3 Boundary Conditions

The boundary conditions used for the simulations are shown in Figure 5.2. An inlet total pressure boundary condition was used for the flow domain. The total pressure measured experimentally was specified for the inlet pressure. In addition to the total pressure, the flow direction was specified at the inlet. At the outlet of the

Figure 5.2: Unstructured Mesh Used for the Computations.

Figure 5.3: Total Pressure Loss Coefficient Versus Number of Nodes Used for Meshing. Results Presented for the Baseline Cascade HS1A at Design Conditions.

Figure 5.4: Loading Distribution for a 20000 Nodes Mesh and a 40000 Nodes Mesh. Results Presented for the Baseline Cascade HS1A at Design Conditions.

domain, a uniform pressure was specified. The magnitude of the pressure specified was equivalent to the mixed-out outlet static pressure measured experimentally for the corresponding test case. Periodic boundary conditions were used at mid passage on either side of the blade in order to simulate an infinite blade row. A turbulence intensity of 4% was specified as measured at the inlet of the cascade test section.

5.2.4 Turbulence Model

The simulations were performed as fully turbulent cases. No laminar-turbulent transition model was implemented. The Spalart and Allmaras (1994) turbulence model was used to simulate the turbulent characteristics of the flow. The Spalart and Allmaras turbulence model is a relatively simple one-equation model that solves a transport equation for the turbulent kinematic eddy viscosity. The Spalart and Allmaras turbulence model was designed for wall-bounded flows and has been shown to give good results for boundary layers subjected to adverse pressure gradients (FLUENT, 1998).

5.3 Validation of Numerical Results

5.3.1 Convergence Criteria

To judge whether or not a solution had converged, three convergence criteria were used. First, the scaled residuals of the governing equations were monitored. Convergence was assumed to have occurred when the magnitude of the residuals had reduced by six orders of magnitude for the energy equation and by three orders of magnitude for the other equations. Figure 5.5 shows a typical convergence history for the scaled residuals. Typically, 4000 iterations were required to get a converged solution independently of the outlet Mach number.

As another indication of convergence, the mass imbalance between the inlet and outlet of the flow domain was monitored as shown in Figure 5.6.

Finally, the lift on the airfoil was monitored for convergence as shown in Figure 5.7.

Cases not initialized with a pre-converged solution were first converged laminar

Figure 5.5: Typical Convergence History for the Scaled Residuals.

Figure 5.6: Typical Convergence History for the Mass Imbalance.

Figure 5.7: Typical Convergence History for the Lift.

first-order, than turbulent first-order and finally second-order turbulent in order to get proper and rapid convergence. This explains the three peaks and plateaus seen in Figures 5.5 and 5.6.

5.3.2 Losses

Comparison between the total-pressure loss coefficients determined experimentally and those calculated numerically are shown in Figure 5.8 for the three cascades at design incidence. The predicted losses are very similar to the ones measured experimentally. The main features in the variation of losses with Mach number observed experimentally, such as the rise in losses at a Mach number of about 0.85, can be seen in the numerical results as well. Perhaps, the most significant difference between the numerical and the experimental results resides in the fact that the losses predicted numerically for the aft-loaded blade, HS1C, do not level-off close to design Mach number as observed experimentally. The lack of a transition model might explain some of the differences. In any event, the predicted losses for the aft-loaded blade were also lower than for the baseline blade at the design Mach number, as in the experiment.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 5.8: Comparison Between Experimental and Numerically Determined Mixed-Out Total Pressure Loss Coefficients for the Three Blade Profiles at Design Incidence.

5.3.3 Outlet Flow Angle

Comparisons of the outlet flow angles measured experimentally with those obtained using FLUENT at design incidence show good agreement at high outlet Mach numbers, as shown in Figure 5.9. The small discrepancies observed at lower Mach numbers might be the result of a low Reynolds number effect on the experimental results. At low Mach numbers, the suction surface boundary layer might have been partially laminar or transitional whereas the computations were performed assuming a fully turbulent boundary layer.

5.3.4 Loading Distributions

Comparisons of the loading distributions measured experimentally with those obtained numerically at design incidence and Mach number reveal very good agreement, as shown in Figure 5.10. The location of the shock impingement points are captured very well. The only slight discrepancy observable for the three profiles is the slightly lower Mach numbers predicted on the forward part of the airfoils' suction surfaces.

5.3.5 Wake Profiles

In Figure 5.11, a comparison between the experimentally and computationally determined variations in pitchwise total-pressure loss coefficients at the design incidence and Mach number are shown. Figure 5.11 shows that the depths of the wakes predicted computationally are larger than that measured experimentally for all three airfoils. The presence of deeper wakes as predicted computationally is an indication of higher trailing edge and mixing losses.

The results for the baseline mid-loaded cascade, HS1A, also show lower (less negative) predicted loss coefficient outside the wake region. This suggests lower blade mid-passage losses. The presence of a weaker trailing shock system could explain this difference, however, comparison of the loading distributions in Figure 5.10(a) does not readily support this explanation. The widths of the wakes for blade HS1A, as determined experimentally and computationally are nominally similar indicating

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 5.9: Comparison Between Experimental and Numerically Determined Outlet Flow Angles for the Three Blade Profiles at Design Incidence.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 5.10: Comparison Between Experimental and Numerically Determined Loading Distributions for the Three Blade Profiles at Design Incidence.

comparable surface boundary layer thicknesses. The higher trailing edge losses predicted numerically seem to be compensated by the lower free-stream losses, thus yielding predictions of the total pressure loss coefficient close to the experimentally determined value, as shown in Figure 5.8(a).

For the aft-loaded blade, HS1C, the predicted mid-passage losses are in better agreement with the experimental measurements. However, the difference between the depth of the wake predicted numerically and measured experimentally is the greatest for HS1C. This large difference accounts for the poor prediction of the total pressure loss coefficient for blade HS1C at a Mach number of 1.10 seen in Figure 5.8(b). The wake appears slightly larger on the pressure surface for the experimental results. This might be due to a slightly thicker boundary layer on the pressure surface of blade HS1C for the experimental case.

The experimental and computational results for the front-loaded blade, HS1D, are very similar to those for blade HS1A. That is, the mid-passage losses measured experimentally are greater but the wake is not as deep with the result that the overall loss coefficient compares well, as seen in Figure 5.8(c).

Comparisons between the experimentally and numerically determined pitchwise static pressure variations are shown in Figure 5.12 for the three airfoils. The computational predictions agree very well with the experimental results. The rise in static pressure due to the presence of the trailing edge shock and the subsequent reduction in pressure due to an expansion fan are captured accurately by the numerical simulations for all three airfoils.

Figure 5.13 shows the comparisons between the pitchwise variation in outlet flow angle predicted computationally and measured experimentally. For the three airfoils, there are some noticeable differences. The most significant difference is the presence of two peaks in the outlet flow angle variation seen experimentally and not reproduced numerically. The presence of two successive peaks in the experimental results may be partly explained by the difficulty in determining accurately with the three-hole probe the flow angularity in flows where significant flow non-uniformities are present. The probe calibration is performed in a uniform flow where the flow velocity and pressures are uniform in the vicinity of the probe tip. In an actual flow where strong pressure gradients and shear layers are present, such as across a

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 5.11: Experimentally and Computationally Determined Pitchwise Total Pressure Loss Coefficient Variation at Design Incidence.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 5.12: Experimentally and Computationally Determined Pitchwise Outlet Static Pressure Variation at Design Incidence.

shockwave or a wake, the two side holes of the three-hole probe might be exposed to very different flow conditions. In processing the measurements, differences in the side hole pressures are interpreted as flow angularity whereas the differences might be due to shear or static pressure gradient in the flow. As shown previously in Figure 5.11, the wake extends from a pitchwise location of about 0.25 to around 0.75 for the three airfoils. It is suspected that the first peak in the experimental pitchwise outlet flow angle variation is due to the fact that the three-hole probe crosses the wake flow. It is not believed to be genuine but rather due to the effect of the wake on the three-hole probe measurements. The second peak and the subsequent drop in outlet flow angle is due to the presence of the trailing edge shock (labelled “e” in Figure 2.7). This latter reduction in outlet flow angle, which is observed numerically as well, is believed to be genuine. It is attributed to the presence of the oblique reflected trailing edge shock which turns the flow towards the axial direction.

In order to offer a clearer picture, a comparison between the pitchwise variation in outlet flow angle predicted computationally and measured experimentally for HS1A at a subsonic Mach number of 0.94 is presented in Figure 5.14. At this outlet Mach number, shock waves are not present in the wake and therefore the interpretation of the outlet flow angle wake profile is made easier.

The variation in the pitchwise outlet flow angle at this lower Mach number is more consistent with the expected trends. That is, the outlet flow angle is greater than the freestream value on the suction side of the wake and smaller on the pressure side. However, as for the data at the design Mach number, the computational predictions do not match the experimental result very well, for reasons explained previously.

Baines et al. (1986) present wake profile data for an HP turbine blade at an outlet Mach number of 0.91 as measured in four European wind tunnels. Comparisons of the pitchwise variation in outlet flow angle shows that it varies greatly between the four wind tunnels. This demonstrates the great difficulty in accurately determining the outlet flow angle in high-speed cascade measurements. However, the authors of that study note the consistency of the outlet flow angle profiles from the Braunschweig (DFVLR) and Goettingen (DFVLR) wind tunnels. Comparisons of the trends in the outlet flow angle profiles shown in Figure 5.14 with those presented by the Goettingen laboratory show good similarity. This provides support for the validity of the wake

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 5.13: Experimentally and Computationally Determined Pitchwise Outlet Flow Angle Variation at Design Incidence.

Figure 5.14: Experimentally and Computationally Determined Pitchwise Outlet Flow Angle Variation at an Outlet Mach Number of 0.94 for Blade HS1A.

traverse data presented in this study.

Chapter 6

Wind Tunnel Flow Quality Assessment

6.1 Introduction

Cascade testing has proved to be a valuable tool over the years for gaining a better understanding of the flow behaviour in turbine blade passages and thus for improving the performance of gas turbine engines. Today, with the advent of numerical tools to predict turbomachinery flows, cascade testing has gained an additional use in the validation of these Computational Fluid Dynamics (CFD) codes. However, for the cascade results to be useful, great care must be exercised to ensure that the flow simulates as accurately as possible the conditions in the “infinite” blade row being modelled. Among the flow quantities that must be considered are the inlet boundary layer thickness, the inlet flow uniformity, the outlet flow periodicity and the Axial Velocity Density Ratio (AVDR). However, relatively little guidance is provided in the literature as to what constitutes acceptable flow quality for transonic cascade measurements. This is especially true for off-design incidence.

6.2 Cascade Inlet Flow

6.2.1 Inlet Boundary Layer

In nominally two-dimensional cascade studies, it is necessary that the inlet flow should exhibit a significant region of uniform flow around midspan in the spanwise direction. However, very little guidance exists as to what constitutes acceptable inlet boundary layer characteristics. In the open literature, data concerning the inlet boundary layer to actual turbines is scarce. However, the inlet boundary layer characteristics for a model turbine are given by Hunter (1982) and those for a cascade

by Hodson and Dominy (1987) and Langston et al. (1977) . Marchal and Sieverding (1977) performed a cascade study on secondary flows. They tested a turbine rotor blade in a cascade wind tunnel having thick and thin inlet boundary layers. It was found that the increase in inlet boundary layer thickness had little effect on the downstream midspan losses and the secondary losses.

Uniformity of the flow in the spanwise direction is an important parameter to consider. Before reaching the inlet of the cascade in the Pratt & Whitney Canada high speed wind tunnel, the flow must travel a significant distance ($\simeq 2\text{m}$) in the test section. Over such a distance, the endwall boundary layers can grow such that its thickness can become non-negligible relative to the blade span.

The inlet flow was traversed in the spanwise direction with a boundary layer probe at the location shown in Figure 4.8. Figures 6.1 and 6.2 show the results for four different outlet Mach numbers at design incidence and at $+10.0^\circ$ of incidence. These results were obtained with cascade HS1A inserted in the test section. In these figures, the inlet flow is seen to be uniform over approximately 60 percent of the inlet passage height. The corresponding endwall boundary layers thus have a thickness equivalent to about 20 percent of the blade span. As expected, for exit Mach numbers greater than 1.0, that is when the cascade flow is choked, the inlet flow becomes invariant, as seen in Figures 6.1 and 6.2

The integral parameters of the boundary layers are summarized in Table 6.1 for the design incidence. Essentially the same boundary layer parameter values were obtain at an incidence of $+10.0^\circ$. They were obtained using the compressible form of the appropriate equations. These equations are summarized as follows:

$$\delta^* = \int_0^\infty \left(1 - \frac{\rho}{\rho_e} \frac{u}{U_e}\right) dy \quad (6.1)$$

$$\theta = \int_0^\infty \frac{\rho}{\rho_e} \frac{u}{U_e} \left(1 - \frac{u}{U_e}\right) dy \quad (6.2)$$

and

$$H = \frac{\delta^*}{\theta}. \quad (6.3)$$

Figure 6.1: Spanwise Inlet Flow Uniformity for Cascade HS1A at Design Incidence.

Figure 6.2: Spanwise Inlet Flow Uniformity for Cascade HS1A at +10.0° of Incidence.

Site		Site	Site
		δ^*	δ^*
Midspan		0.39	0.52
Tip	δ^*	2.3mm	2.0mm
Root	θ	1.6mm	1.4mm
Tip		1.4	1.4
Root	$\delta_{99\%}$	17.7mm	17.7mm

Table 6.1: Inlet Boundary Layer Parameters.

The results show that the inlet boundary layer is turbulent with shape factors, H , of about 1.4 for both choked and unchoked inlet flow. This shape factor is typical of that found in other cascade wind tunnels. Marchal and Sieverding's (1977) study of secondary flows within turbine blade passages compared the results obtained with thin and thick inlet boundary layers. The thick boundary layer used in their study is very similar to the one observed in the present experimental facility in terms of the ratio of boundary-layer thickness to blade span. Marchal and Sieverding found that the midspan losses for a rotor blade of low aspect ratio (0.79) were essentially the same for the thin and thick inlet boundary layers tested at design incidence. Therefore, it appears that the inlet boundary layer thickness has only a limited effect on the midspan results as long as a significant region of two-dimensional flow is present at midspan. However, where two-dimensional midspan results are the goal, the thickness of the inlet boundary layer should nevertheless be kept to a minimum. This issue will be discussed further in a later section.

In order to possibly extend the region of uniform inlet flow in the spanwise direction, blow-off of the endwall boundary layer was tentatively implemented. Porous metal plates were installed 300mm upstream of the cascade inlet, as shown in Figure 3.5. The porous plate thicknesses were 3.18mm (0.125in) and the pore size was $100\mu\text{m}$. During the runs, no attempt was made to control the amount of blow off. Thus, the quantity of air bleed increased with the test section blowing pressure.

Inlet spanwise flow traverses were performed with cascade HS1A installed in the test section at an incidence of $+10.0^\circ$. The boundary layer probe traverses were

Figure 6.3: Spanwise Inlet Flow Uniformity With and Without Boundary Layer Blow-Off for Cascade HS1A at $+10.0^\circ$ of Incidence.

done from the top endwall down to midspan for several Mach numbers.

The traverse results are shown in Figure 6.3 together with those obtained when boundary layer blow-off was not implemented for an outlet Mach number of 1.07. It can be observed that the blow-off did indeed improve the uniformity of the flow in the spanwise direction.

Hence, it appears that the blowing-off of the inlet endwall boundary layers could be useful in the present facility. This may turn out to be required if it is found that the inlet boundary layers contribute to generate unacceptable level of secondary flow downstream of the cascade. It would then be necessary to develop a means of controlling the amount air blown-off.

For the current research project, it was decided that the inlet core flow extent was sufficient.

6.2.2 Inlet Flow Pitchwise Uniformity

Two-dimensional continuously-running cascade wind tunnels are limited in size due in part to limitations on the power available to drive the wind tunnels. In the case of blow-down wind tunnels the restriction in size comes from the need to achieve practical run times. These limitations in wind tunnel sizes therefore impose a limit also on the number of blades that are used in a cascade to get periodic flow. As opposed to an ideal cascade with an infinite number of blades, finite size cascades are bounded by side walls. The presence of these side walls influences the flow field and thus the results. Therefore, it is important to design and adjust these side walls in order to obtain good inlet flow uniformity as in an infinite cascade.

In theory, using frictionless side walls having the shape of the stagnation streamlines would solve the problem. However, the shape of the streamlines are unknown prior to the experiment and frictionless walls are only hypothetical. In practice, experience has shown that it is not necessary to reproduce the stagnation streamlines exactly for the side walls (Starken and Lichtfuss, 1975). The essential condition that must be fulfilled to obtain good inlet-flow uniformity is that the stagnation streamlines of the outer blades be identical.

Methods for obtaining good flow quality in terms of inlet flow uniformity have been discussed by a few authors. Gostelow (1984) briefly discussed in his book on cascade aerodynamics the question of good flow quality in cascade wind tunnels. Starken and Lichtfuss (1975) presented a more detailed discussion on cascade flow adjustments. They describe procedures for obtaining good flow uniformity for different flow conditions (e.g., compressible-incompressible, subsonic-supersonic) at the inlet of a blade passage. However, the authors do not present any results to demonstrate what is considered good flow quality. Furthermore, the effect of inlet flow non-uniformity on the results is not discussed. Starken and Lichtfuss suggest that it is sufficient to measure the static pressure by means of wall static taps in a plane upstream of the cascade and to verify the flow angle at one point to assess the uniformity of the inlet flow. This procedure appears to be well accepted as it is used by several other research groups. For example, Kiock et al. (1986) presented turbine cascade measurements made in four European wind tunnels with the same blade row

geometry. Inlet flow uniformity checks were made at the cascade inlet by measuring the static pressure over several pitches. The inlet flow angle was also measured in some of the facilities.

In order to obtain good inlet flow uniformity and downstream flow periodicity, the wind tunnel control surfaces must be adjusted such that the mass flow rate of air in each passage is equal.

A review of the literature has shown that different research groups in turbine aerodynamics use different means for adjusting the flow in their wind tunnels. Langston (1989) uses adjustable bleed at the two extremities of his cascade together with adjustable tailboards downstream of the cascade. Other researchers, such as Mee et al. (1991), use adjustable shutters to control the amount of air flowing through the end passages.

In the current facility, inlet flow adjustment is achieved by moving the test section side walls, as shown in Figure 6.4. No provision is made for bleeding the inlet side wall boundary layers. Furthermore, no tailboards are used downstream of the cascade. This setup is similar to that used at the Von Karman Institute (see Kiock et al., 1986). Through experimentation, it was found that good inlet flow uniformity could be achieved by setting the side walls parallel to each other and by aligning the test section side walls with the leading edge of the cascade end blocks as shown in Figure 6.5. The required inlet duct width, or design inlet flow area, can be calculated as follows. For a cascade composed of blades having a pitch (s) and operating at an incidence α_1 , the distance (w) between the bounding streamlines for the streamtube entering a particular passage is given by

$$w = s \cos \alpha_1. \quad (6.4)$$

Thus, for an arbitrary cascade composed of N blades and $N+1$ passages, the required design inlet duct width (W) is given by

$$W = (N + 1)w \quad (6.5)$$

and the design inlet flow area is obtained by multiplying the duct width W by the

Figure 6.4: Test Section Wall Displacement.

blade span H . This calculation neglects the blockage due to the displacement effect of the sidewall boundary layers. Given that the flow over much of the sidewall length is highly accelerated, the sidewall boundary layers are expected to be very thin.

Wall static pressure measurements in the pitchwise direction were made at the inlet of the cascade for three different values of incidence and for different upstream passage widths. The aim of these measurements was to determine the effect of the side wall alignment on the uniformity of the inlet flow.

The variation of the static pressure in the pitchwise direction is presented in Figure 6.6 in the form of an inlet static pressure coefficient. The results are for roughly the design outlet Mach number of the cascade.

The results show that for the cases for which the side walls were aligned with the end passages (design inlet flow area), the inlet uniformity was very good for the five centermost passages. Non-uniformity in the inlet flow is only present for the outermost passages, due to the test section sidewall effects. On the other hand, for the case where the inlet flow area was increased by moving the right side wall

Figure 6.5: Test Section Wall Alignment for Good Inlet Flow Uniformity.

(looking downstream) outward as shown in Figure 6.7, a pressure gradient can be clearly observed across the cascade inlet. This pressure gradient is an indication that the inlet streamlines are curved. If the streamlines are curved then the inlet flow is not aligned with the wind tunnel axis, giving rise to erroneous and probably varying values of incidence across the cascade. The impact of this on the results will be discussed in the following sections.

Similar results were obtained for the cases where the right side wall was moved inward. However, as seen from Figure 6.8, the pressure gradient across the cascade inlet was reversed when the flow area was reduced relative to the design value. The pressure gradient for the case where the right side wall was moved inward is not very steep. This is explained by the limited distance by which the wall could be moved due to physical limitations.

Figure 6.6: Variation of the Static Pressure Coefficient Across the Cascade Inlet Plane.

Figure 6.7: Test Section Wall Alignment for Poor Inlet Flow Uniformity.

6.2.3 Inlet Flow Turbulence Intensity and Length Scale

The turbulence measurements were made in the high speed wind tunnel at blowing pressures of 105kPa (1.05Bar) and 110kPa (1.1Bar) with cascade HS2 in the test section. Details about this profile can be obtained from Jeffries (2001). For these blowing pressures, the corresponding inlet Mach numbers are 0.15 and 0.21. For the cases where the blowing pressure was at 105kPa, the turbulence intensity was measured to be 3.5%. At a blowing pressure of 110kPa the turbulence intensity was found to be 3.8%. Taking into account the different sources of experimental errors, such as the probe orientation and the temperature variation in the wind tunnel, an error band of ± 0.005 or 0.5% is estimated for the turbulence intensity. These results seem to suggest that the turbulence intensity increases as the blowing pressure is increased. Therefore, it is recommended that additional turbulence measurements be performed to assess the level of turbulence in the wind tunnel at higher blowing pressures.

The length-scale of the energy containing eddies was made. Figure 6.9 shows a

Figure 6.8: Variation of the Static Pressure Coefficient Across the Cascade Inlet Plane for Reduced Inlet Area (-3%) and Increased Inlet Flow Area (+9.2%).

Figure 6.9: Variation of the Autocorrelation Coefficient With Time Delay at a Blowing Pressure of 105kPa.

plot of the variation of the autocorrelation coefficient with time delay. As seen from the figure, the autocorrelation coefficient decays rapidly initially. As the time delay is further increased the coefficient decays to a value close to zero. This trend is typical of what can be found in the literature (see for example Comte-Bellot and Corrsin, 1971).

The length-scale measurements of the energy containing eddies were made at a blowing pressure of 105kPa. On average, it was found that the length-scale of a typical energy containing eddy was 1.5×10^{-2} m or 1.5cm. This result appears to be plausible taking into consideration there is a butterfly valve of 10cm in diameter located just upstream of the test section. One would expect that eddies generated within the valve are propagated downstream with a length-scale similar to that of the valve.

6.3 Cascade Outlet Flow

6.3.1 Outlet Flow Periodicity

Periodicity refers to having the same flow conditions from blade to blade in a cascade. That is, the losses associated with each of the blades and the mass flow rate through each passage should be the same. Furthermore, the exit Mach number and flow angle distribution along one blade pitch should be the same for all the blades.

On an actual turbine disk, provided the incoming flow is periodic, outlet flow periodicity is usually achieved due to the circumferential arrangement of the blades. However, for a linear cascade to reproduce the same effect, an infinite cascade would be required. Therefore, for cascades with a limited number of blades perfect periodicity is not possible. On the other hand, by properly setting the control surfaces it is possible to obtain reasonable periodicity for the central blade passages.

Once good inlet flow uniformity has been established, outlet flow periodicity is usually achieved easily in subsonic flow. Therefore, very little information is found in the literature concerning methods of obtaining outlet flow periodicity. However, if supersonic outlet flow is present, periodicity can be harder to achieve due to the presence of reflected shocks, as mentioned by Sieverding (1993). Sieverding discusses the different types of boundaries that can be used at the extremities of the cascade to achieve periodic flow. Three types of boundaries most commonly used at the extremities of cascade exits to improve the periodicity are discussed. First, free exhaust or dump diffusion can be applied whereby the area is abruptly increased immediately downstream of the cascade. The exit flow can develop as a free jet and the flow angle adjusts based on the downstream back pressure. This technique is widely used for transonic cascade testing.

Another method used to achieve outlet flow periodicity consists of having solid tailboards hinged to the trailing edge of the cascade end blades. Langston (1989) used this type of arrangement. It is used as well in the low speed cascade wind tunnel at Carleton University. However, if the outlet flow is transonic or supersonic, solid tailboards can cause problems with shock reflections.

Finally, perforated tailboards can be used in supersonic flows to avoid shock

reflections which can lead to erroneous wake traverse measurements. The porosity of these tailboards must be carefully tailored to achieve the shock cancellation.

In order to assess the periodicity, Starken and Lichtfuss (1975) suggest using the same procedure as for the inlet flow: that is, measuring the endwall pressures along the cascade outlet. This procedure was followed by Mee et al. (1991). However, a review of the literature reveals that other methods have also been used to assess periodicity. Midspan wake traverses over several pitches are often performed to verify periodicity and this would seem to be more reliable than just relying on endwall static pressures. This was the procedure followed by Kiock et al. (1986). Additionally, comparing the loading distributions for adjacent blades in the cascade is often used. This approach was used by Langston (1989) and Langston et al. (1977) in his low-speed cascades.

The cascade outlet flow periodicity is somewhat weakly linked to the inlet flow uniformity, particularly near design incidence. It should be noted that the current experimental facility has no control surfaces available downstream of the cascade to adjust the flow: the cascade outlet flow discharges into a plenum of the same height but considerably wider dimensions compared to the cascade. Figure 6.10 shows the variation of the midspan outlet flow Mach number and the total pressure loss coefficients together with the inlet static pressure coefficients for the full cascade width at an incidence of $+10.0^\circ$. The results are shown for good (design inlet flow area) and poor (design inlet flow area $+8\%$) inlet flow uniformity. It is noticeable that the non-uniformity in the inlet flow is not readily detectable in the cascade outlet flow, which appears to have good periodicity for both cases. The clear implication is that the cascade outlet flow cannot be used reliably to detect problems with the cascade inlet flow. The most significant difference between the two cases is the depth of the wakes. The reasons for this will be discussed later.

Once good inlet flow uniformity was established for each new cascade, downstream periodicity measurements were performed. Periodicity check measurements had to be performed whenever a new cascade was installed to verify that the blades were properly manufactured and mounted on the baseplate. In the following three figures (6.11, 6.12, 6.13), periodicity checks for cascade HS1A, HS1C and HS1D are shown. The downstream probe traverses cover seven blade pitches and are made at a downstream distance equivalent to 40% axial chord. These results were obtained at

Figure 6.10: Effect of Inlet Flow Uniformity on Outlet Flow Periodicity. (a) Inlet Pitchwise Static Pressure Coefficient (b) Outlet Pitchwise Mach Number Variation (c) Outlet Pitchwise Total Pressure Loss Coefficient.

design incidence and for an outlet Mach number close to the design point. Results are presented for the outlet total pressure coefficient, outlet flow angle and outlet Mach number.

As seen from the variation of the total pressure loss coefficient as a function of the pitchwise location, good outlet flow periodicity is obtained for the centermost passages for all three cascades. It can be observed that for these passages the peak to peak variations in total pressure losses and the width of the wakes are very similar. Only the outermost passage flows appear to be disturbed by the presence of the inlet side wall boundary layer.

In terms of the outlet flow angle, periodicity is a little more difficult to achieve in the transonic range of Mach numbers. This is probably due in part to the presence of a shock system at the cascade outlet that is not completely steady under these conditions. From Figures 6.11, 6.12 and 6.13 one can see that the blade-to-blade variation of the outlet flow angle is not as uniform as for the total pressure coefficient. Acceptable periodicity is only attained for the three centermost passages. For the outermost passages, the pitchwise flow angle distributions differ significantly.

The isentropic outlet Mach number exhibits good outlet flow periodicity for the three cascades tested. For the centermost wakes, the peak-to-peak Mach numbers are very similar. Again, slight variations can be observed for the outermost passages due to the presence of the inlet side wall boundary layer.

The blade-to-blade variations in the mixed-out values of the cascade exit Mach number (M_2), total pressure loss coefficient (Y_t), outlet flow angle (α_2) and mass flow rate per unit area ($\dot{m}/A$) are summarized in Table 6.2 for the three cascades at operating points close to the design Mach number. The values are shown for the seven blades. The mean values and the standard deviation of the different parameters are calculated for the three centermost blades.

The blade-to-blade variations observed in the present study compare very favourably with those observed in a study conducted in four European transonic wind tunnels (Kiock et al., 1986). The standard deviations quoted for (M_2), (Y_t) and (α_2) fall within the range of values quoted by Kiock et al.. Hence, it appears that periodicity for the three cascades tested is in line with what can be achieved in other high-quality transonic cascade wind tunnels.

Figure 6.11: Cascade HS1A Outlet Flow Periodicity in Terms of Total Pressure Coefficient, Outlet Flow Angle and Outlet Mach Number.

Figure 6.12: Cascade HS1C Outlet Flow Periodicity in Terms of Total Pressure Coefficient, Outlet Flow Angle and Outlet Mach Number.

Figure 6.13: Cascade HS1D Outlet Flow Periodicity in Terms of Total Pressure Coefficient, Outlet Flow Angle and Outlet Mach Number.

Cascade	HS1A				HS1C				HS1D			
Outlet Mach Number	$M_2=1.03$				$M_2=1.03$				$M_2=1.05$			
Blade	M_2	Y_1	β_2	$\dot{m}/A^*$	M_2	Y_1	β_2	$\dot{m}/A^*$	M_2	Y_1	β_2	$\dot{m}/A^*$
1	1.094	0.085	56.29	383.9	1.109	0.113	56.27	383.6	1.116	0.129	55.3	382.5
2	1.033	0.095	56.73	366.6	1.113	0.084	56.25	389.2	1.097	0.095	55.71	383.2
3	1.046	0.089	56.53	370.1	1.095	0.076	56.52	385.7	1.059	0.09	56.09	374.3
4	1.033	0.093	56.69	365.7	1.084	0.077	56.78	382.3	1.056	0.091	56.07	373.1
5	1.044	0.092	56.83	368.4	1.077	0.082	56.67	380.1	1.059	0.097	55.93	372.7
6	1.013	0.100	56.95	358.8	1.065	0.087	56.9	375.9	1.059	0.104	56.25	371.6
7	1.011	0.087	56.79	360.0	1.061	0.078	56.95	376.5	1.045	0.091	56.24	369.8
Mean	1.041	0.091	56.69	368.0	1.086	0.079	56.66	382.7	1.058	0.093	56.03	373.4
Std. Deviation	0.007	0.002	0.15	2.2	0.009	0.003	0.13	2.8	0.002	0.004	0.09	0.9

* unit of $\text{kg}/(\text{s m}^2)$

Table 6.2: Blade-to-Blade Variations of the Cascades Outlet Flow Aerodynamic Parameters.

6.3.2 Flow Two-Dimensionality

Cascade measurements are often conducted to determine the two-dimensional aerodynamic behaviour of a particular blade profile. The two-dimensionality of a cascade flow can be assessed in several ways. A region of constant profile losses near midspan is a necessary condition for two-dimensionality of the flow. Sieverding (1993) recommended a minimum aspect ratio for reliable two-dimensional behaviour in a turbine cascade as a function of the velocity ratio across the cascade. A minimum aspect ratio is required to obtain reasonably two-dimensional midspan flow for typical values of inlet endwall boundary layer thickness. Based on Sieverding recommendations, the current cascade would require an aspect ratio of about 1.8 at design incidence and Mach number, given that the velocity ratio equals 0.53 at these conditions. This is somewhat higher than the present cascade aspect ratio of 1.53. Figure 6.14 shows that the current cascade nevertheless does have a region of a two-dimensional flow at design incidence. However, the upper part of the figure shows that for $+12.0^\circ$ of incidence there is essentially no region of two-dimensional flow at midspan, even though the endwall boundary layer had essentially the same thickness at the cascade inlet for both cases.

The inlet endwall boundary layer is the source of the secondary vortex that

Figure 6.14: Spanwise Distribution of Total Pressure Losses.

wraps around the blade leading edge. This vortex is augmented by the cross-channel flow inside the passage. By controlling the inlet boundary layer thickness, one can limit to some degree the extent of the secondary flow in the cascade.

The extent of the secondary flow is also a function of the flow acceleration through the cascade. High acceleration tends to stretch the secondary flow vortex and thus limit its spanwise dimension. In a typical turbine cascade the streamwise acceleration in the passage decreases as the incidence increases. This explains the reduction in the spanwise extent of uniform midspan flow shown in Figure 6.14. Thus, it appears that aspect ratios that are adequate for studies at design incidence may be too low for measurements at positive, off-design values of incidence.

6.4 Axial Velocity Density Ratio (AVDR)

The increased extent of the region of three-dimensional flow for higher values of incidence calls for a closer examination of the two-dimensionality of the flow through the cascade. A necessary but not sufficient condition for two-dimensional midspan flow is that the AVDR be equal to 1.0 at midspan.

The influence of AVDR on turbine cascade results under low-speed conditions has been examined by Rodger et al. (1992). They found that AVDR had a significant effect on the losses at higher incidence where boundary layer separation was present. However, the effect of AVDR on transonic cascade results is still somewhat unclear. Kiock et al. (1986) concluded that the AVDR had no effect on the losses or deviation in the range of 0.9 to 1.0. Sharma and Graziani (1983) concluded that the AVDR in itself could not explain the effect of the endwall flow on the aerodynamics and heat transfer on midspan blade surface. More recently, Jouini et al. (2001), (2002) observed that the AVDR increased with incidence for measurements made in a transonic cascade. They also concluded that the losses are affected by the varying AVDR.

AVDR is a function of the aspect ratio of the blades, among other things. For three-dimensional effects to be kept as small as possible the aspect ratio of the cascade blades must be high. However this requirement must be balanced with the need for sufficient chord lengths, to give measurable wakes for measurement of loss and to allow the blades to be instrumented with static taps. As mentioned in Section 6.3.2,

Sieverding (1993) recommended a minimum aspect ratio for reliable two-dimensional performance in a turbine cascade as a function of the velocity ratio across the cascade.

Measurements of AVDR were made in the present study at 0.0° , -10.0° and $+10.0^\circ$ of incidence for different outlet Mach numbers. The results are presented in Figure 6.15 for uniform and non-uniform inlet flow. It can be observed that for the design incidence the AVDR is equal to 1.0 over the whole range of Mach numbers tested when the inlet flow is uniform. On the other hand, for an incidence of $+10.0^\circ$ the AVDR is about 1.1. This is mainly the result of the increased blockage caused by the enlarged secondary flows. As noted earlier, the larger secondary flows are expected due to the reduced channel acceleration at positive values of incidence.

There is also greater uncertainty in the values of AVDR for positive incidence due to the greater uncertainties in the outlet flow angle measurements at these values of incidence. At $+10.0^\circ$, significant boundary layer separation occurs and therefore the deviation angles are much higher. The flow separation together with the presence of secondary flow vortices at $+10.0^\circ$ contribute to the increase in uncertainty for the flow angle measurements. Baines et al. (1991) have shown that the inferred values of AVDR are very sensitive to the inlet and outlet flow angle measurements. It can be shown that a 1° variation in the outlet angle results in a change in the value of AVDR of the order of 0.04.

The non-uniformity of the inlet flow also has an influence on the inferred values of AVDR, as shown in Figure 6.15. The differences between the cases where the flow is uniform versus non-uniform are due to the fact that the AVDR is a function of the inlet flow angle. When calculating the AVDR, the inlet flow is assumed to be aligned with the wind tunnel axis. As noted earlier, in the cases where the inlet flow is not aligned a pressure gradient exists across the cascade inlet. This pressure gradient is associated with streamline curvature that alters the effective inlet flow angle at the cascade. The precise change in inlet angle seen by the cascade is not known. However, the changes in the direction of the inlet angle implied by the direction of the observed pressure gradients are consistent with the differences in AVDR shown in Figure 6.15.

Figure 6.15: Effects of Incidence and Mach Number on the Axial Velocity Density Ratio (AVDR). Filled Symbols $\bullet$: Uniform Inlet Flow; Open Symbols $\circ$: Non-Uniform Inlet Flow.

6.5 Effect of Inlet Flow Pitchwise Non-Uniformity on Results

6.5.1 Introduction

The changes in inlet flow angle alter the effective incidence seen by the blades. For example, the variations of the static pressure across the inlet for the cases shown in Figure 6.6 are consistent with a reduction in effective incidence. Such changes in incidence can have significant effects on the aerodynamic performance of the cascade. These effects are examined next.

6.5.2 Profile Losses

Figure 6.16 shows the variation of profile losses with Mach number at -10.0° , $+0.0^\circ$ and $+10.0^\circ$ of incidence. The results are presented for both uniform and non-uniform inlet flows. At design and -10.0° incidence, it is seen that the influence of the inlet flow uniformity on the losses is limited. Both curves follow the same trend

Figure 6.16: Effects of Incidence and Mach Number on the Total Pressure Losses. Filled Symbols $\bullet$: Uniform Inlet Flow; Open Symbols $\circ$: Non-Uniform Inlet Flow.

and the losses are similar. The variations in losses with Mach number also resemble those observed by other research groups, such as Mee et al. (1992).

However, at an incidence of $+10.0^\circ$ the influence of the inlet flow uniformity is clearly seen, especially at lower Mach numbers. The high losses observed at $+10.0^\circ$ for uniform inlet flow case are due to a flow separation on the blade suction surface as well as the presence of secondary flow at midspan. The losses for non-uniform inlet flow are much closer to those observed at design incidence, suggesting a reduction in the extent of the separation and secondary losses. This is consistent with the expected reduction in inlet flow incidence associated with the streamline curvature that occurs due to the presence of the pressure gradient.

6.5.3 Deviation

The variation of exit flow angle with Mach number and incidence is shown in Figure 6.17 for uniform and non-uniform inlet flow. As for the losses, the influence of the inlet flow uniformity is primarily seen at high positive values of incidence. As

Figure 6.17: Effects of Incidence and Mach Number on the Exit Flow Angle. Filled Symbols $\bullet$: Uniform Inlet Flow; Open Symbols $\circ$: Non-Uniform Inlet Flow.

mentioned previously, at $+10.0^\circ$ of incidence boundary layer separation occurs on the suction surface of the blade. This results in smaller exit flow angles (less turning) for both inlet flows, as shown in Figure 6.17. However, for the case where the inlet flow is non-uniform, the outlet flow angles are closer to those observed at design incidence. This is again an indication that the boundary layer separation occurring at this condition may be considerably less severe. Again, this is in agreement with a reduced incidence resulting from the streamline curvature at the cascade inlet.

6.5.4 Blade Loading and Base Pressure

Loading measurements were also performed for uniform and non-uniform inlet flows. For the design incidence, no significant effect of the inlet flow uniformity can be observed on the blade's pressure distribution. However, as with the losses and deviation, the uniformity of the inlet flow had an influence on the loading when the blade was at incidence, particularly at high positive values of incidence. The results are shown in Figure 6.18 for an incidence value of $+10.0^\circ$ and in Figure 6.19

Figure 6.18: Effect of Inlet Flow Uniformity on the Blade Loading at $M_{2s} = 1.03$ and an Incidence of $+10.0^\circ$.

for -10.0° , both for an exit Mach number of 1.03. The isentropic Mach number is calculated from the ratio of the measured surface static pressure to the upstream total pressure. For the positive incidence, the pressure distribution is not affected on the pressure surface by the non-uniformity of the inlet flow. However, on the suction surface, it can be observed that the pressure distributions are significantly different. The suction surface pressure distribution is somewhat flatter on the rear part of the blade for the case where the inlet flow is uniform. This is an indication that flow separation occurs at this location, which is consistent with the higher losses and smaller outlet flow angles noted previously for this case. At -10° of incidence, mild discrepancies between the pressure distributions for uniform and non-uniform inlet flows are observed, as shown in Figure 6.19. The differences are confined to the forward part of the blade but affect both the pressure and suction surfaces. The differences result from the increased negative incidence induced by the non-uniformity of the inlet flow. However, the differences in the pressure distributions had little impact on the losses, as shown previously. This is presumably because they are quite localized and there seem to be no separations associated with them.

Figure 6.19: Effect of Inlet Flow Uniformity on the Blade Loading at $M_{2s} = 1.03$ and an Incidence of -10.0° .

The base pressure is known to have a strong influence on the losses. Therefore, if the changes in inlet flow uniformity are reflected in the base pressures this would provide part of the explanation for the differences in losses noted earlier. In Figure 6.20 the base pressure coefficient is plotted against the outlet Mach number for incidence values of $+10.0^\circ$ and -10.0° . The lines connecting the data are included mainly to guide the eye. The results are shown for both uniform and non-uniform inlet flow. Results at design incidence are omitted for clarity. At design incidence, the inlet flow uniformity had very little influence on the base pressure. The variation of the base pressure at design incidence was similar to that observed at -10.0° for uniform inlet flow.

At an incidence of $+10.0^\circ$, a significant difference exists between the values of the base pressure coefficient for uniform and non-uniform flow. This is especially true at lower Mach numbers. Again, this can be explained by the fact that at $+10.0^\circ$ of incidence with uniform flow, the boundary layer on the suction surface of the blade was separated, whereas the degree of separation was much reduced for the non-uniform flow case. Both the thickness of the blade surface boundary layers at the trailing edge

Figure 6.20: Effects of Incidence and Mach Number on the Base Pressure Coefficient. Filled Symbols $\bullet$: Uniform Inlet Flow; Open Symbols $\circ$: Non-Uniform Inlet Flow.

and the presence of trailing edge separation are known to influence the base pressure. For an incidence of -10.0° , the base pressure appears to be less influenced by the inlet flow uniformity, with increasing difference also at the low Mach numbers.

6.6 Conclusions

Results obtained from detailed measurements performed on a turbine cascade at transonic Mach number for both design and off-design condition were presented. The aim of these experiments was to investigate the effect of the overall flow quality in the wind tunnel on the measured aerodynamic performance of the turbine cascade. Data were obtained at three incidences, namely, -10.0° , 0.0° and $+10.0^\circ$ as well as for a wide range of transonic Mach numbers.

The results show the importance of carefully monitoring the quality of the flow both upstream and downstream of the cascade. The results presented in this chapter have shown that the lack of inlet flow uniformity cannot be readily detected

by measurements made downstream of the cascade. Such measurements should only be used to assess the flow periodicity in the cascade.

Lack of good inlet flow uniformity yields erroneous results. This is especially true at positive off-design incidence for which the cascade becomes sensitive to changes in the effective incidence. Detailed pitchwise measurements of wall static pressure made close to the leading edge of the cascade are essential for verifying the uniformity of the inlet flow.

Finally, Sieverding's recommendations concerning the required blade aspect ratio for making two-dimensional cascade measurements appear to be satisfactory, and perhaps even a little conservative for design incidence. However, it also appears that considerably higher aspect ratios may be needed to obtain a sufficient width of uniform flow near midspan for studies conducted at large positive values of incidence.

Chapter 7

Results and Discussion of the Cascades' Performance at Design Incidence

7.1 Introduction

Three transonic turbine cascades were studied experimentally at both design and off-design incidences. Cascade HS1A constitutes the baseline cascade and is considered mid-loaded. Cascades HS1C and HS1D were generated starting from the baseline profile as described previously. Cascade HS1C is aft-loaded and cascade HS1D is front-loaded. The objective of the research consists in determining the impact of loading distribution on the performance of a blade row at both design and off-design conditions.

The purpose of this chapter is to present the experimental results and discuss the interpretation of the flow physics associated with each of the three cascades at design incidence.

Initially, the performance of the baseline mid-loaded cascade (HS1A) is examined at design incidence and Mach number for high Reynolds numbers. The flow physics associated with the baseline cascade is examined in detail and comparisons are then made between the three profiles.

The analysis of the cascades at design conditions then provides a basis for the description of the flow physics at off-design Mach numbers and lower Reynolds numbers. Once the flow mechanisms associated with the three cascades at design incidence have been established, the flow is then examined for off-design incidence in the following chapter.

The mixed-out loss data as well as several other parameters of interest for cascades HS1A, HS1C and HS1D are tabulated in Appendices D, E and F.

Figure 7.1: Isentropic Mach Number Distribution for Cascade HS1A at Design Mach Number and Incidence.

7.2 Cascade Performance at Design Mach Number (High Re)

The blade surface Mach number distribution is shown in Figure 7.1 for the baseline cascade HS1A at design incidence and Mach number. The corresponding Mach number distribution obtained from the numerical simulation is also shown as a solid line. These results were obtained for an outlet Reynolds number of 10.4×10^5 and an inlet turbulence intensity of 4%. The Mach number distribution shows that at these conditions, the flow accelerates initially on the suction surface in the vicinity of the leading edge. The flow then settles at a constant Mach number just below 1.0 until it crosses the sonic line at $x/C_x = 0.37$. The flow then continues to accelerate until it reaches a maximum Mach number at $x/C_x = 0.58$. The fact that the flow accelerates from the leading edge up to this point on the suction surface suggests that the boundary layer is probably laminar over this region. However, considering the relatively high Reynolds number and turbulence intensity, the boundary layer probably transitions to a turbulent state upon encountering the small adverse pressure

Figure 7.2: Turbine Cascade Trailing-Edge Shock System.

gradient at $x/C_x = 0.6$. Following the small region of deceleration, the flow re-accelerates due to the presence of expansion Mach waves emanating from the pressure surface trailing region of the adjacent blade, as shown schematically in Figure 7.2. These Mach waves form due to the rapid expansion of the flow on the pressure side in the vicinity of the trailing edge, as seen in Figure 7.1.

The impingement of the trailing edge shock from the adjacent blade follows right after the expansion. The shock boundary layer interaction can clearly be seen from the numerical simulations. Figure 7.3 shows the contour plot obtained at design conditions. The incident shock and the reflected shock can be observed. Closer inspection of the velocity vectors near the blade surface showed that the computations predict a small separation zone right at the shock impingement point, at $x/C_x = 0.83$. The turbulent boundary layer, benefitting from a favorable pressure gradient up to the trailing edge, remains attached.

On the pressure surface, the Mach number distribution shows that the flow is accelerating from the leading edge to the trailing edge. Therefore, no boundary layer

Figure 7.3: Mach Number Contour Plot for Cascade HS1A at Design Mach Number and Incidence Obtained Numerically.

separation is present. Hence, the contribution of the pressure surface flow to the overall total pressure loss is expected to be small at design operating conditions.

The sonic condition is reached in the vicinity of the trailing edge on the pressure surface. As mentioned previously, the sonic line is crossed at $x/C_x = 0.37$ on the suction surface. Therefore, for the design flow conditions, the cascade is choked and the sonic line is curved as shown in Figure 7.4(a).

The nominal surface Mach number at $x/C_x = 1.0$ was obtained from the base pressure measurement. As mentioned previously, the base pressure is the static pressure on the trailing edge. The sketch shown in Figure 7.5 shows the expected structure of supersonic trailing edge flow. Close to the trailing edge, the incoming flow expands through a series of Mach waves (a) before separating from the blade suction and pressure surfaces. The flow separation produces the separation shocks (b). The separated surface boundary layers form shear layers (c) which converge to form the wake (d). Compression Mach waves and shocks, (e), (f), form in order to provide the required directional adjustment to the external flow brought about by the convergence of the two shear layers. The triangular shaped region (g) immediately behind the trailing edge and bounded by the shear layer is a stagnant flow region called the base region. The pressure in this region is called the base pressure.

The base Mach number for blade HS1A at design conditions is 1.30. This corresponds to a base pressure to inlet total pressure ratio, P_b/P_{o1} , of 0.36. Comparisons with Sieverding's et al. (1980) correlation indicates that the base pressure for HS1A at design conditions ($P_2/P_{o1} = 0.5$) is lower than that predicted by the correlation, as shown in Figure 7.6. More discussion on Sieverding's et al. (1980) correlation will be given in the following section.

The magnitude of the base pressure has a strong influence on the magnitude of the losses and hence the performance of an airfoil at a given operating point. Without elaborating on the explanation for the lower base pressure at this point, the consequences for the performance of the blade are noted. The presence of a region of low pressure at the trailing edge results in the creation of a strong trailing edge shock system. Furthermore, the low pressure at the trailing edge results in an increasing the drag on the blade. These two phenomena contribute to an increase of the losses.

The main objective when designing the aft-loaded blade (HS1C) was to shift

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 7.4: Sonic Line for the Three Blade Profiles at Design Conditions Obtained Numerically.

Figure 7.5: Turbine Cascade Trailing Edge Shock Structure.

Figure 7.6: Comparison of Measured Base Pressure with Sieverding's Correlation for Cascade HS1A at Design Incidence.

Figure 7.7: Surface Mach Number Distribution for the Three Cascades at Design Mach Number and Incidence.

the blade loading to the rear of the airfoil with the objective of possibly reducing the losses. Comparing the Mach number distributions for blades HS1A and HS1C in Figure 7.7 it can be observed that the objective (aft-loading) was achieved in terms of modifying the loading.

The HS1C pressure distributions obtained experimentally and numerically at design Mach number and incidence are shown in Figure 7.8. At the leading edge, a short velocity overshoot can be observed on the suction surface. This feature is not desirable as the adverse pressure gradient that ensues will contribute to thickening the boundary layer and possibly causing transition to a turbulent state thus increasing the losses. Profile HS1A described previously had only a very small overshoot on suction surface as seen from Figure 7.1.

Similar to HS1A, the HS1C suction surface velocity increases on the front part of the airfoil up to an axial position of $x/C_x = 0.59$. The Mach number reaches sonic conditions at $x/C_x = 0.38$, which is practically the same location as that on HS1A.

Following a short region of deceleration, the flow is re-accelerated due to the presence of expansion waves originating from the adjacent blade. These expansion

Figure 7.8: Surface Isentropic Mach Number Distribution for Cascade HS1C at Design Mach Number and Incidence.

waves are evident from the contour plots obtained from computations and shown in Figure 7.9. The subsequent trailing edge shock from the adjacent blade impinges on the surface at an axial distance $x/C_x = 0.80$. This shock impingement location is very similar to that of cascade HS1A under these conditions.

However, there are notable differences between HS1A and HS1C in the apparent strength of the impinging shock, and the corresponding drop in Mach number. For HS1A at $M_2 = 1.07$ (Figure 7.1) the pressure rise begins at about $0.75C_x$ and the decrease in Mach number is about 0.20. On the other hand, for HS1C (Figure 7.8) the decrease in Mach number is only about 0.12 at an exit Mach number of 1.05. This indicates that near the design $M_2 = 1.05$ the impinging shock is weaker for the aft-loaded airfoil. This in turn will favorably affect the losses in two ways: through the lower direct total pressure losses through the shock wave and through its milder effect on the suction surface boundary layer. As will be shown later, the measured losses for HS1C are significantly lower than those for HS1A at design conditions. The difference in the impinging shock strength account for part of this observed difference. After the shock, the flow re-accelerates up to the trailing edge.

Figure 7.9: Mach Number Contour Plot for Cascade HS1C at Design Mach Number of 1.05 and Design Incidence Obtained Numerically.

As for cascade HS1A, on the pressure surface of HS1C the Mach number distribution shows that the flow is accelerating from the leading edge to the trailing edge. Therefore, no boundary layer separation will be present, except perhaps a short bubble at the overspeed near the leading edge. Hence, the contribution of the pressure surface flow to the overall total pressure loss is not expected to be significant at design operating conditions except perhaps for the slightly stronger velocity overshoot that occurs close to the leading edge on the pressure surface.

The sonic condition is reached close to the trailing edge on the pressure surface. As mentioned previously, the sonic line is crossed at $x/C_x = 0.38$ on the suction surface. Therefore, for the design flow conditions, the cascade is choked and the sonic line is curved as shown in Figure 7.4(b).

The surface Mach number at $x/C_x = 1.0$ obtained from the base pressure measurement is equal to 1.08. This corresponds to a ratio of base pressure to inlet total pressure of 0.47. Comparison with other cascade tests performed under similar conditions, such as those of Sieverding et al. (1980), indicates that this value of pressure ratio is relatively high. As seen from Figure 7.7, the base Mach number for the aft-loaded cascade, HS1C, is much lower when compared with that of the mid-loaded cascade and the front-loaded cascades. The reasons for this will be discussed in Section 7.3.4. However, at this point it is worth mentioning that a higher base pressure for HS1C is certainly beneficial as it is associated with a weaker trailing edge shock system and a reduced amount of pressure drag.

The pressure distribution for the front-loaded blade HS1D is shown in Figure 7.10 for the design Mach number and incidence. Starting from the leading edge, the flow accelerates rapidly on the suction surface to cross the sonic line at $x/C_x = 0.06$. By comparison, the flows on the suction surface of the mid-loaded blade HS1A and the aft-loaded blade HS1C crossed the sonic line at about $x/C_x = 0.38$. The larger loading on the forward part of the blade is evident compared to those of the previous two blades. Once the sonic line is crossed, the flow on the suction surface of HS1D accelerates up to $x/C_x = 0.16$. Starting from that point, the flow then faces a mild adverse pressure gradient to $x/C_x = 0.63$ where sonic conditions are reached. The presence of an adverse pressure gradient over almost 50% of the suction surface is not a desirable feature due to enhanced boundary layer growth and losses.

Figure 7.10: Surface Isentropic Mach Number Distribution for Cascade HS1D at Design Mach Number and Incidence.

If adverse pressure gradients persist for extended lengths, boundary layer separation can occur with its very detrimental effect on losses. However, for HS1D, there does not appear to be any flow separation triggered by the adverse pressure gradient. This conclusion is supported by the numerically predicted Mach number distribution on the blade suction surface, which is also shown on Figure 7.10.

Following the mild adverse pressure gradient, the flow is re-accelerated up to $x/C_x = 0.80$. The flow acceleration is caused by the expansion fan from the adjacent blade as for blades HS1A and HS1C. However, for blade HS1D, the expansion is stronger as evidenced by the magnitude of the change in Mach number. The expansion is then followed by a reduction in Mach number resulting from the impingement of the adjacent blade's trailing edge shock on the blade surface. However, the impinging shock is weak and the rise in pressure that ensues is relatively small. Hence, there does not appear to be any boundary layer separation at this location. This conclusion was supported by the predicted velocity field. Past the region of interaction with the shock wave, the flow experiences a favorable pressure gradient which lasts until the trailing edge is reached.

On the pressure surface, the Mach number distribution shows that the flow is accelerating from the leading edge to the trailing edge. Therefore, no boundary layer separation is present and thus the contribution of the pressure surface flow to the overall total pressure loss should be small at design operating conditions. However, the flow acceleration is significantly higher on the pressure surface of the front-loaded cascade, HS1D, when compared to the other two blades. Hence, the resulting loading at the rear of the blade is much lower than that observed for the other two blades.

The sonic condition is reached close to the trailing edge on the pressure surface. As mentioned previously, the sonic line is crossed at $x/C_x = 0.16$ on the suction surface. Therefore, for the design flow conditions, the cascade is choked and the sonic line is curved, as shown in Figure 7.4(c).

The surface Mach number at $x/C_x = 1.0$ is equal to 1.22. This corresponds to a base pressure to inlet total pressure ratio, P_2/P_{o1} , of 0.42, which is in line with observations made on similar blades.

7.3 Performance at Off-Design Mach Number (High Re)

7.3.1 Profile Losses

The effect on the total pressure loss coefficient of varying the pressure ratio across the cascade for cascade HS1A is shown in Figure 7.11 for both experimentally and numerically obtained results.

The trends in the variation of the experimental total pressure loss coefficients with Mach number at design incidence agree very well with what is usually found in the literature (e.g. Mee et al., 1992). At low Mach number, the losses are relatively independent of the Mach number. The boundary layer losses and the mixing losses in the wake are the sole contributors to the total losses at these low Mach numbers. As the Mach number increases, the total pressure loss coefficient reaches a minimum at about $M_{2s} = 0.85$. The increase in losses as the exit Mach number is reduced from 0.85 to 0.60 is partly the result of a Reynolds number effect. This is evidenced by the comparing the losses measured for blade HS1A at the highest Reynolds number

Figure 7.11: Effect of Mach Number on Losses for Cascade HS1A at Design Incidence.

investigated for outlet Mach numbers of 0.60 and 0.85 as shown in Figure 7.28(a). In this figure, it can clearly be seen that the losses at the highest Reynolds number investigated, where the losses are independent of this parameter, are essentially the same for exit Mach numbers of 0.60 and 0.85. Starting at an outlet Mach number of 0.85, the losses start increasing sharply. This sharp increase in losses is due to a significant increase in the mixing losses as the base pressure is reduced at the trailing edge as seen from Figure 7.23 in Section 7.3.4. The change in losses is also attributable to a lesser extent to the formation of shock waves. The losses then level off briefly in the vicinity of the design Mach number. Again, this phenomenon is related to the behaviour of the base pressure which decreases and re-increases rapidly at these Mach numbers. The influence of the base pressure will be discussed in more detail in Section 7.3.4. Finally, as the Mach number is increased further, losses begin to rise rapidly once more as the shock strength increases.

Comparison of the numerically predicted losses with the experimental results indicates that the numerical model is able to simulate the most salient features in the variation of the losses with Mach numbers. However, except at the lowest Mach

Figure 7.12: Effect of Mach Number on Losses (High Re).

numbers, the computations slightly over-predict the losses. One possible explanation, for the discrepancies between the numerical and experimental results is the inability of the Spalart-Almaras turbulence model to predict transition from a laminar to a turbulent boundary layer.

Figure 7.12 shows the variation of the profile total-pressure-loss coefficient with outlet Mach number for the three cascades. All the results presented were obtained at Reynolds numbers of 5.0×10^5 or higher, for which the results were found to be independent of Reynolds number. The influence of Reynolds number on the losses is examined in Section 7.4. The mixed-out loss data presented in this section are tabulated along with other data in Appendices D, E and F.

For low subsonic outlet Mach numbers ($M_2 < 0.75$), losses are essentially constant for the three blades. However, small differences can be seen in the magnitude of the losses between the three blades. The aft-loaded blade, HS1C, has the lowest losses followed by the baseline, mid-loaded blade, HS1A. The front-loaded blade, HS1D, shows the poorest performance at low Mach numbers. The reasons for these differences in performance will be discussed in connection with the loading distribu-

tions in Section 7.3.3.

A minimum is reached in the losses for blades HS1A and HS1D at an outlet Mach number of about 0.85. Blade HS1C does not exhibit a minimum in losses around $M_2 = 0.85$, but this may be because no measurements were performed for blade HS1C close to $M_2 = 0.85$.

Losses start to increase for all three blades for outlet Mach numbers above about 0.85. In the range of Mach numbers from 1.0 to approximately 1.15, which includes the design value of 1.05, the losses level-off for both the mid-loaded and aft-loaded airfoils, HS1A and HS1C. However, the losses are noticeably lower, by roughly 20%, for the aft-loaded airfoil (HS1C) compared with the baseline. The reasons for this will be discussed in connection with the loading distributions and base pressures. Above an outlet Mach number of about 1.15, the losses for HS1C begin to rise rapidly and its performance becomes significantly poorer than that of the baseline airfoil. The loss performance of the forward-loaded airfoil, HS1D, is inferior to, or at best equal to, that of at least one of the two other airfoils at all operating conditions.

Comparison of the numerically predicted losses with the experimental measurements for cascades HS1C and HS1D are shown in Figures 7.13 and 7.14, respectively. The loss predictions are very similar to the ones measured experimentally. The main features in the variation of losses with Mach number observed experimentally, such as the rise in losses at a Mach number of about 0.85, can be seen in the numerical results as well. Perhaps, the most significant difference between the numerical and the experimental results resides in the fact that the losses predicted numerically for the aft-loaded blade HS1C do not level-off close to design Mach number as observed experimentally. The lack of a transition model might explain some of the differences. Furthermore, as shown previously in Figure 5.11(b) the numerically predicted wake is much deeper than the one measured experimentally at a Mach number of 1.10. This is an indication that the numerical simulation predicts much higher trailing edge and mixing losses. On the other hand, the predicted losses for the aft-loaded blade were also lower than that predicted for the baseline blade at the design Mach number, as in the experiment.

Figure 7.13: Effect of Mach number on Losses for Cascade HS1C at Design Incidence.

Figure 7.14: Effect of Mach number on Losses for Cascade HS1D at Design Incidence.

Figure 7.15: Effect of Mach number on Outlet Flow Angle for Cascade HS1A at Design Incidence.

7.3.2 Outlet Flow Angle

Outlet flow angle measurements were obtained using the three-hole probe. Figure 7.15 shows the variation of exit flow angle with Mach number for high Reynolds number values as measured experimentally and predicted numerically for the mid-loaded baseline cascade HS1A. As mentioned previously, all these results are tabulated in Appendices D, E and F. The exit metal angle is indicated for reference. The outlet flow angle reaches a maximum at an exit Mach number close to 1.0. As expected, as the outlet flow becomes supersonic the flow direction must rotate towards the axial direction in order to increase the flow area and the flow turning is reduced.

Comparison between the numerically predicted and experimental outlet flow angles shows good agreement between the two, except at the lowest Mach numbers. The difference is attributed to the difference between the predicted extent of the laminar boundary layer on the suction surface and the actual one. Thicker or thinner boundary layers at the trailing edge result in more or less flow turning.

Figure 7.16 shows the variation of exit flow angle with Mach number at high

Figure 7.16: Measured Effect of Mach Number on Exit Flow Angle (High Re).

Reynolds number for the three cascades. The trends in outlet flow angle with Mach number are very similar for the three cascades. The forward-loaded airfoil HS1D is seen to give the lowest flow turning, particularly at outlet Mach numbers close to 1.0. This suggests the presence of a thicker suction-side boundary layer for this airfoil and perhaps even a mild flow separation near the trailing edge, which would be consistent with the higher profile losses measured for HS1D.

The aft-loaded blade HS1C exhibits slightly lower outlet flow angles than the baseline cascade, HS1A, at both low Mach numbers and close to the design Mach number. This is counter-intuitive as the losses for HS1C were lower for these outlet Mach numbers. A similar trend is observed from the numerical results for HS1A and HS1C as shown in Figure 7.17. No reasonable explanation for that could be found.

7.3.3 Blade Loading

Loading measurements were performed on the mid-loaded cascade, HS1A, for outlet Mach numbers ranging from 0.5 to 1.2. These measurements were made at high enough Reynolds numbers such that this parameter had no effect on the results.

Figure 7.17: Numerical Predictions of the Effect of Mach Number on Exit Flow Angle.

The effect of varying the Mach number on the loading distribution, as represented by the isentropic Mach number, for HS1A is shown in Figure 7.18(a). At low Mach numbers, HS1A exhibits an adverse pressure gradient starting at about $x/C_x = 0.6$ followed by a region of near-constant pressure on the rear part of the suction surface. This might normally be seen as evidence of boundary layer separation at this location. However, the losses shown in Figure 7.12 and the outlet flow angle results shown in Figure 7.16 do not support this conclusion: as observed, the losses at low Mach number are relatively low and the exit flow angles do not decrease significantly at lower exit Mach numbers. The loading distributions are simply those resulting at low exit Mach numbers for airfoils designed for use at much higher Mach numbers.

For exit Mach numbers less than about 0.95 the flow through the cascade is not choked. This is evident by the fact that the Mach numbers on the pressure surface and on the forward part of the suction surface continue to change for these subsonic Mach numbers. In addition, as shown in Figure 7.19, the inlet Mach number is increasing for subsonic outlet Mach numbers indicating that the passage is not choked.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 7.18: Effects of Exit Mach Number on Blade Loading for the Three Blade Profiles.

Figure 7.19: Variations in Cascade Inlet Mach Number with Outlet Mach Number and Theoretical 1-D Isentropic Choking Mach Number Based on Throat Width.

As the exit Mach number nears 0.8, a local region of supersonic flow develops on the suction side of the airfoil. A clear example of that can be seen in Figure 7.18(a) for the case where the exit Mach number is 0.83. At this Mach number, on the suction surface of the blade, there is a supersonic region extending from $x/C_x = 0.43$ to $x/C_x = 0.62$. The flow returns to subsonic velocity through a weak shock.

For exit Mach numbers greater than about 0.96, the suction-side loading distributions remain constant from the leading edge up to an axial distance of about $x/C_x = 0.56$ as the outlet exit Mach number increases. This is due to choking of the flow in the blade passage.

A very weak, nearly normal shock seems to appear in the blade passages at a Mach number of 0.96. The shock runs from the base region of one blade and impinges on the suction side of the adjacent blade at an axial distance of $x/C_x = 0.70$.

With further increases in Mach number, the normal shock observed at $M_2 = 0.96$ becomes more oblique and the impingement point migrates towards the trailing edge. Furthermore, the shock becomes stronger. However, as also supported by nu-

merical predictions, the presence of a stronger impinging shock with increasing Mach number does not appear to trigger boundary layer separation. The only boundary layer separation being predicted is the very small separation bubble mentioned at design operating conditions in Section 7.2.

The effect of varying the outlet Mach number on the loading distribution for the aft-loaded cascade, HS1C, is shown in Figure 7.18(b). At low Mach numbers, the Mach number distribution for blade HS1C is very similar to that observed for the mid-loaded blade, HS1A. As the Mach number increases, supersonic regions develop on the suction surface of the blade as seen from the Mach number distributions beginning at $M_2 = 0.84$.

At an outlet Mach number of 0.93, the blade passage is nearly choked for cascade HS1C, as evidenced by the fact that the Mach number distribution on the forward part of the suction surface is almost identical to that for higher exit Mach numbers. However, on the pressure surface it can be seen that the sonic line is not crossed and thus the cascade is not completely choked at this outlet Mach number.

At outlet Mach numbers higher than 0.93 HS1C chokes and an oblique shock originating at the trailing edge of the adjacent airfoil impinges on the rear of the suction surface of the blade. As mentioned previously at the design Mach number of 1.05 the oblique shock impinges at an axial distance $x/C_x = 0.80$. With further increases in Mach number, the impinging shock wave becomes more oblique and stronger, as for HS1A. However, whereas the impinging shock strength appeared to be lower for the aft-loaded blade HS1C at the design condition compared to HS1A, at higher supersonic Mach numbers the opposite is true. Comparing Figures 7.18(a) and 7.18(b) at M_2 of about 1.18, the rise in pressure at the shock impingement is noticeably higher for HS1C than for HS1A. This contributes to the poorer loss performance of HS1C at higher Mach numbers.

The loading distributions for the front-loaded blade, HS1D, are shown in Figure 7.18(c). As for blades HS1A and HS1C, the surface Mach number distribution at low Mach numbers is unusual since the blade was designed to operate in a transonic regime. However, even at these low Mach numbers, the forward loading of the blade can be observed.

A comparison of the loading distribution for the three blades at the lowest Mach

Figure 7.20: Loading Distribution for the Three Cascades at Low Exit Mach Numbers.

number tested is shown in Figure 7.20. At this low Mach number, the aft-loaded blade HS1C is seen to exhibit a favorable pressure gradient on the forward half of the suction surface. On the other hand, a favorable pressure gradient is only present close to the leading edge for blades HS1A and HS1D. On the second half of blades HS1A and HS1C suction surface the flow experiences very similar adverse pressure gradient whereas it appears a little stronger for blade HS1D. The stronger adverse pressure gradient for HS1D can be explained in part by the fact that the loading distribution shown for that blade is for an outlet Mach number of 0.56 which is slightly higher than for the other two blades. Hence, the aft-loaded blade HS1C having a more favorable pressure gradient on the forward part of the suction surface and an adverse pressure gradient similar to the other two blades on the rear suction surface has probably a much thinner boundary layer. This is consistent with blade HS1C having the lowest losses at low Mach numbers as shown previously in Figure 7.12. The base pressure has probably no role to play in the relative performance of the three blades at low Mach numbers as it is essentially equal to the outlet value of static pressure.

At a Mach number of 0.88, a supersonic region forms on the forward part of the

blade HS1D suction surface. The region extends from an axial position of about 0.06 down to about $x/C_x = 0.50$. When comparing the extent of the supersonic regions to those for the mid and aft-loaded cascades at similar outlet Mach numbers, one can see that it is much longer for HS1D and that it starts much closer to the leading edge.

The forward loaded cascade HS1D appears to be choked at an outlet Mach number of 0.98. At this Mach number, a nearly normal shock is impinging on the suction surface at an axial distance $x/C_x = 0.64$.

For higher Mach numbers, Figure 7.18(c) shows that the strength of the impinging shock wave for the forward-loaded airfoil, HS1D, is similar to that for HS1C at most conditions. However, the HS1D airfoil experiences the onset of mild deceleration on the suction surface at about $0.3C_x$ whereas it is delayed until about $0.6C_x$ for the other two airfoils. As a result, the suction side boundary layers will tend to be thicker for HS1D at all conditions and at the supersonic outlet conditions the layer will be more sensitive to the rise in pressure due the impinging shock waves.

Figure 7.21 shows a comparison of the loading distributions for the three blade profiles at an outlet Mach number near 1.20. Clearly, the shock impinging on the rear suction surface is much stronger for blades HS1C and HS1D than it is for the baseline cascade HS1A for this outlet Mach number. This is further evidenced by comparing the wake traverses for the three cascades. Figure 7.27 in Section 7.3.4 presents the wake traverses for the three airfoils at an outlet Mach number near 1.20. The figure shows the variation of the total pressure loss coefficient with the pitchwise location. The mid-passage losses are higher for the aft-loaded blade, HS1C, and slightly higher for the front-loaded blade, HS1D, compared with HS1A. This indicates the presence of weaker trailing edge shock for blade HS1A compared with the other two. Following the drop in Mach number resulting from the shock impingement, the flow re-accelerates on the suction surface of blade HS1A. However, for blades HS1C and HS1D the flow appears to be moving at near constant Mach number following the shock impingement region. Based on experimental investigations on shock boundary-layer interactions such as that of Graham and Kost (1979), it would appear that the region of flow moving at constant Mach number is probably a separation bubble. The cause of the separation bubble is the interaction of the adjacent blade's trailing edge shock with the suction surface boundary layer. A typical shock boundary layer interaction

Figure 7.21: Loading Distributions for the Three Cascades at High Exit Mach Number.

is shown schematically in Figure 7.22. This diagram shows the compression waves upstream of the incident shock which combine to form the separation shock. The compression waves and the separation shock are required in order to deflect the flow around the separation bubble. After passing through the separation shock the flow encounters the incident shock wave followed by its reflected expansion waves. As the flow reattaches, a second series of compression waves forms and coalesces to generate the reattachment shock. The reattachment shock is needed in order to realign the flow with the suction surface after the separation bubble.

The effect of the shock wave interaction with the boundary layer is to increase its thickness. The presence of a stronger impinging shock wave which leads to flow separation on the rear suction surface of both blades HS1C and HS1D certainly contributes to explain their higher losses at higher Mach numbers when compared to the baseline cascade HS1A.

Figure 7.22: Schematic of Shock-Wave Boundary Layer Interaction (Adapted from Japikse and Baines, 1994).

7.3.4 Base Pressure

The base pressure is a measure of the static pressure at the center of the blade's trailing edge. All three cascades included one airfoil instrumented with a static tap at the trailing edge for measuring the base pressure. As mentioned previously in Section 7.2, the surface Mach numbers at $x/C_x = 1.0$ in Figure 7.18 are calculated from the static pressure measured on the blade's trailing edge.

In Section 3.4.5, it was mentioned that the base pressure tap of the three cascades has a diameter which corresponds to 20% of the trailing-edge thickness. This represents a significant fraction on the trailing edge. Studies, such as that of Sieverding et al. (2003), have shown that the base pressure varies around the trailing edge. Therefore, the base pressures measured in the course of this study represent an average of the pressure in the central portion of the trailing edge.

The variations of the base pressure coefficient with Mach number for the three cascades are shown in Figure 7.23.

The base pressure coefficient is defined as

$$C_b = \frac{(P_b - P_2)}{q_2} \quad (7.1)$$

Figure 7.23: Variation of Base Pressure Coefficient With Mach Number.

where P_b is the base pressure. The trends in the variation of the base pressure with Mach number are significantly different for the three profiles.

In order to assist in the interpretation of the base pressure measurements, the Sieverding et al. (1980) correlation is useful. Sieverding et al. correlated the base pressure and the downstream static pressure together with the trailing edge wedge angle and the uncovered turning angle. For blades having a converging passage, curves of P_b/P_{o1} versus P_2/P_{o1} are given for different values of $(We_{te} + \theta_u)/2$, the average of the wedge angle, We , and the uncovered turning angle, θ_u . Using the correlation, the trends in the base pressure variation can be explained more easily in terms of the trailing edge shock system development.

Figure 7.24 shows the base pressure measurements for the the three cascades together with Sieverding's correlation. The parameter $(We_{te} + \theta_u)/2$ for blades HS1A and HS1D has a value of 8.75. For the aft-loaded cascade, HS1C, $(We_{te} + \theta_u)/2$ has a value of 10.25 since its unguided turning angle is 3.0° higher.

Using Figure 7.23 and Figure 7.24, the relative cascades performance can be explained in more detail. The base pressure is known to have a strong influence on

Figure 7.24: Comparison of Measured Base Pressure with Sieverding's Correlation for the Three Cascades at Design Incidence.

the profile losses. For example, Denton (1993) has shown using a simplified control-volume analysis that the trailing edge losses for incompressible flow can be related to the boundary-layer parameters and base pressure coefficient as follows:

$$Y_{te} = -\frac{C_b t}{o} + \frac{2\theta}{o} + \left(\frac{\delta^* + t}{o} \right)^2 \quad (7.2)$$

where C_b is the base pressure coefficient. The relative effect of C_b is increased in compressible flow. The base pressure is known to be sensitive to both the thickness of the blade surface boundary layers at the trailing edge and the presence of trailing edge separation. Thus, base pressure measurements can be very helpful in understanding the sources of losses in cascade flows.

Returning to Figure 7.23, it can be seen that the mid-loaded blade HS1A exhibits positive or near zero values of the base pressure coefficient for subsonic Mach numbers. A sharp drop in the base pressure then occurs just before $M_2=1.0$ which corresponds to a significant rise in losses (see Figure 7.12). A minimum C_p is then reached at an exit Mach number close to 1.02. Comparison of the base pressure measurements for blade HS1A with Sieverding et al. correlation (Figure 7.24) shows that they are in good agreement with the correlation for values of the pressure ratio P_2/P_{o1} greater than 0.55. This corresponds to an isentropic outlet Mach number of 0.97. However, as the pressure ratio is reduced below that which is required for supersonic outlet flow, significant discrepancies appear between the measured and the Sieverding et al. correlated base pressures. For pressure ratios of 0.52 and 0.49, which correspond to Mach numbers close to the design value, the measured base pressure is seen to fall significantly below what is predicted by the correlation for geometrically similar blades such as HS1D.

The presence of a low base pressure at the trailing edge contributes to the creation of strong trailing edge shocks. This can be seen by comparing the loading distributions of blades HS1A and HS1D at $M=1.07$ and blade HS1C at $M=1.05$ in Figure 7.7. For these conditions the shock impinging on the rear suction surface of blade HS1A is stronger, as evidenced by the greater amount of diffusion. The presence of a low base pressure also increases the losses. Both the shocks and the increased trailing edge loss contribute to increase the overall losses. This might explain why

the mid-loaded cascade (HS1A) exhibits higher losses close to design Mach number compared to the aft-loaded cascade (HS1C).

In order to fully understand why the losses for the mid-loaded blade HS1A are higher compared to that of the aft-loaded blade HS1C close to the design Mach number, the presence of lower than expected base pressure for HS1A must be explained. To do so, the factors influencing the base pressure must be examined. The parameters having an effect on the base pressure can be listed as follows:

- 1. Outlet Mach number**
- 2. Outlet Reynolds number based on outlet velocity**
- 3. Trailing edge wedge angle**
- 4. Blade rear surface Mach number**
- 5. Boundary layer thickness**

At design conditions, the outlet Mach number, the outlet Reynolds number and the wedge angle are essentially the same for the three cascades. Thus, the only parameters that can explain the differences in base pressure between the three airfoil are the boundary layer thickness and the blade rear surface Mach numbers.

It is generally believed that a thicker suction side boundary layer upstream of the trailing edge results in higher base pressure (e.g. Xu and Denton (1988)). This was also noticed for the three cascades investigated during this research project. When the blades were operating at positive incidence, higher base pressures were recorded as shown in Figure 8.11. It is well known that blades operating at incidence tend to have thicker suction side boundary layers.

The influence of the boundary layer shape upstream of the trailing edge on the base pressure is explained by Xu and Denton (1987). The shear stress in the boundary layer after it has separated from the trailing edge entrains the fluid trapped behind the trailing edge into the shear layer. The higher the shear stress, the higher the entrainment rate and the lower the base pressure. Boundary layers with lower shape factors (H) have higher shear stress in the inner layer which thus results in a higher entrainment rate and lower base pressure.

The effect of the rear blade surface velocity on the base pressure is a little harder to demonstrate. The Sieverding et al. correlation indicates that blades having convergent-divergent passage should exhibit much lower base pressure. One noticeable difference between convergent-divergent blade passages and the convergent passages, used in the current research project, is that the velocities on the rear pressure surface of the former are much higher. Thus, it appears that increasing the flow velocity on the blade surface prior to the trailing edge contributes to reducing the base pressure.

The mechanism by which higher blade surface velocities lower the base pressure is essentially the same as that of the boundary layer thickness. That is, by increasing the flow velocity at the trailing edge, the average fluid momentum is increased. This in turn results in higher entrainment power for the shear layer flow which yields lower base pressure.

The thickness of the rear suction surface boundary layer cannot be used as an argument to explain the low base pressure for the mid-loaded baseline cascade, HS1A, at the design conditions as it is expected to be very similar to that of the aft-loaded cascade, HS1C. This can be seen by comparing the width of the wake for blades HS1A and HS1C in Figure 7.25. In this figure, it is seen that the wake for the mid-loaded blade HS1A and the aft-loaded blade HS1C are very similar. This tends to confirm that the blade surface losses are essentially the same for HS1A and HS1C. Thus, it appears that suction surface boundary layer thickness is not a major factor in determining the base pressure for blade HS1A. The boundary layer on the pressure surface is not expected to play a major role either as it is believed to be very similar for both cascades HS1A and HS1C since their respective pressure surface Mach number distributions are very similar.

If the surface boundary layer thickness cannot explain the lower base pressure for cascade HS1A at the design conditions then the surface velocities or Mach numbers may be responsible. The surface Mach number distributions shown in Figure 7.7 for the three cascades reveal that the flow Mach number increases rapidly on the rear suction surface of the mid-loaded blade HS1A up to the trailing edge. On the other hand, the flow on the rear suction surface of blades HS1C and HS1D does not accelerate as fast as that of HS1A. The flow even seems to be decelerating prior to the trailing edge for these two airfoils. Thus, even though the surface Mach numbers

Figure 7.25: Variation of the Total Pressure Loss Coefficient with Pitchwise Location at Design Mach Number of 1.05.

are lower for HS1A on most of the rear suction surface, close to the trailing edge it surpasses that of HS1C and HS1D. Hence, the rear suction surface flow of the mid-loaded blade HS1A has probably more momentum and is thus more capable of entraining the base region fluid, as explained previously. This may explain the relatively low base pressure for HS1A.

In order to understand why the flow is accelerating more on the suction surface of the baseline cascade, HS1A, the blade curvature must be examined. As seen in Figure 7.26, the curvature on the rear suction surface of blade HS1A increases rapidly and in the vicinity of the trailing edge it is higher than that of blades HS1C and HS1D. Sieverding (1983) mentioned that the advantage of having low rear suction side curvature is that it limits the velocity difference across the trailing edge. Hence, it appears that the relatively high rear suction surface curvature of blade HS1A contributes to increasing the flow velocity towards the trailing edge, which in turns contributes to reducing the base pressure. The presence of a low base pressure results in a stronger trailing edge shock system for the baseline cascade HS1A. This shock system increases the losses which explains the poorer performance of HS1A at

design conditions when compared to the aft-loaded cascade HS1C.

At higher Mach numbers, the base pressure coefficient for the mid-loaded blade, HS1A, rises sharply as shown in Figure 7.23. Comparison with the Sieverding et al. (1980) correlation in Figure 7.24 indicates that the base pressure measurements for HS1A at high Mach numbers are in line with the correlation once again. This is consistent with the fact that at higher Mach numbers the performance of blade HS1A is superior to that of the aft-loaded blade, HS1C, and the front-loaded blade, HS1D.

The presence of higher base pressure at high Mach numbers for the baseline cascade HS1A can be explained as follows. For higher outlet Mach numbers, the adjacent blade trailing edge shock impingement point shifts to a location closer to the trailing edge on the suction surface, as seen from Figure 7.18(a). Thus, at higher Mach numbers, the flow cannot re-accelerate as much following the region of diffusion, at the shock impingement point, before reaching the trailing edge. Hence, the fluid momentum in the shear layer is reduced and its entrainment power on the base region fluid is consequently reduced. This results in higher base pressure at the trailing edge.

As mentioned previously, the increase in base pressure for HS1A at high Mach numbers resulted in a relative improvement of its performance when compared to HS1C and HS1D. As the base pressure increased, the strength of the trailing edge shocks was reduced. This can be seen by comparing the loading distributions of HS1A at exit Mach numbers of 1.07 and 1.17 in Figure 7.18(a). As observed from this figure, the amount of diffusion induced on the rear suction surface of the blade by the impinging shock is reduced slightly at the highest outlet Mach number, even though it corresponds to a significantly higher Mach number than 1.07.

Another indication of the presence of a relatively weaker trailing edge shock for cascade HS1A at the higher outlet Mach number of 1.17 can be obtained by comparing the wake traverses for the three cascades. Figure 7.25 and Figure 7.27 presents the wake traverses for the three airfoils at outlet Mach numbers of 1.07 and 1.17 respectively. The two figures show the variation of the total pressure loss coefficient with the pitchwise location. At outlet Mach numbers close to design, it can be seen that the mid-passage losses are greater for the mid-loaded cascade, HS1A. This is due to the presence of stronger trailing edge shocks resulting from the low base pressure. However, at outlet Mach numbers close to 1.17 the situation is reversed.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 7.26: Blade Surface Curvature for the Three Blade Profiles.

Figure 7.27: Variation of the Total Pressure Loss Coefficient with Pitchwise Location at High Mach Numbers.

The mid-passage losses are now higher for the aft-loaded blade HS1C, and slightly higher for the front-loaded blade, HS1D, compared with HS1A. This indicates the presence of weaker trailing edge shocks for blade HS1A compared with the other two.

The relatively lower mid-passage losses for HS1A at higher Mach numbers can also be explained in part by the location of the impinging shock on the suction surface. As seen in Figure 7.18(a) the shock impingement point on the suction surface of the blade at $M_{2s} = 1.07$ is at about an $x/C_x = 0.80$ whereas it is located at about $x/C_x = 0.88$ for an outlet Mach number of 1.17. From the suction surface curvature distribution for HS1A shown in Figure 7.26(a), it can be seen that at an axial location of 0.80 the curvature reaches a value of nearly zero. On the other hand, at higher axial locations such as $x/C_x = 0.88$, the curvature has non-zero values. The impingement of a shock wave on a more convex surface results in a weaker reflected shock and thus lower losses.

Unlike for the other two airfoils, the base pressure coefficient for the aft-loaded blade (HS1C) is positive or near zero over most of the Mach number range investi-

gated. Such a variation of the base pressure coefficient with Mach number is typical of blades having significant rear suction side curvature. Sieverding et al. (1980) suggested that higher values of unguided or uncovered turning result in higher base pressures. The uncovered turning angle for HS1C is 3.0° higher than that of blades HS1A and HS1D. A priori, higher base pressure coefficients may appear to be favorable for losses, as indicated by Equation 7.2 and observed from Figure 7.12 where the losses for HS1C are lower than those of HS1A at the design Mach number. However, blades with strong rear suction side curvature sometimes experience a recompression around the trailing edge when the base pressure is higher than the suction side pressure before the separation point. Such diffusion close to the trailing can be seen from the Mach number distribution of HS1C. From Figure 7.18(b), it can be seen that for exit Mach numbers of 1.14 and 1.18, HS1C, unlike HS1A, is experiencing increasingly strong diffusion just before the trailing edge. The presence of the suction surface adverse pressure gradient thickens the boundary layer which probably contributes to the higher base pressure. This would help to explain why the base pressure coefficient at high Mach number for HS1C does not diminish significantly. Furthermore, the presence of a thick or separated boundary layer on the suction surface contributes directly to the increase in the losses at higher Mach numbers seen in Fig. 7.12. This can be seen also in Figure 7.27 where the wakes for HS1C are larger indicating higher blade surface loss than for HS1A.

For HS1D the base pressure coefficient was negative for all values of exit Mach number. The base pressure coefficient remains constant from a Mach number of 0.5 to about 1.0. At this point, the C_b diminishes rapidly. Comparison with Sieverding et al. (1980) data shows that the variation of the base pressure coefficient with Mach number for HS1D is similar to that usually observed for blades with similar unguided turning angle and trailing edge wedge angle.

7.4 Influence of Reynolds Number on Performance

7.4.1 Profile Losses

The effect of Reynolds number on the losses is shown in Fig. 7.28 for the three airfoils. Measurements were made at Mach numbers of 0.60, 0.85, 0.95 and 1.05. As mentioned previously, the Reynolds number was lowered at the cascade exit by using the ejector-diffuser assembly and raised by partially blocking the diffuser outlet.

As seen from Figure 7.28, the losses increase significantly at low Reynolds numbers for an exit Mach number of 0.60. For Mach numbers of 0.85 and 0.95 the losses are seen to increase slightly for the lowest Reynolds numbers achieved. For an outlet Mach number of 1.05, the Reynolds number could not be lowered enough to detect its effect on the losses. Interestingly, the aft-loaded airfoil (HS1C) showed the lowest sensitivity to reduced Reynolds numbers, in addition to its superior performance at high Reynolds numbers near the design Mach number.

The deterioration in performance as the Reynolds number is reduced is explained by the changes from turbulent to laminar and from attached to separated states of the suction surface boundary layer. This is described in detail by Hourmouziadis (1989). At high Reynolds numbers, transition occurs early on the suction surface. Any turbulent separation that might occur in the vicinity of the trailing edge results in mixing losses which are added to the blade surface losses. As the Reynolds number is reduced, the turbulent separation disappears and the transition point moves towards the trailing edge. This flow pattern corresponds to a fully attached flow. The losses associated with such suction surface flow pattern are the lowest. Referring back Figure 7.28, it appears that this corresponds to the flow condition obtained for the tests performed at high Reynolds numbers. It is seen from Figure 7.28 that the losses reach a constant minimum value at high Reynolds numbers. This is also corroborated by the fact that little or no flow separation was seen from the loading distributions at high Reynolds numbers presented in the previous section.

With further reductions in Reynolds number, transition moves forward of the laminar separation point and a separation bubble forms on the suction surface. As

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 7.28: Reynolds Number Effect on Losses for Several Mach Numbers.

the Reynolds number is decreased, the separation extends towards the leading edge and eventually a separated laminar boundary layer is obtained. The sharp increases in losses seen with the reduction in outlet flow Reynolds number occur when the transition point has moved so far downstream that the turbulent free shear layer cannot reattach before the trailing edge. At this point most of the losses are produced in the wake due to the mixing out of the separated flow.

In Figure 7.28, it can be observed that the different curves converge to the same loss value for the lower subsonic Mach numbers at high Reynolds numbers. This is due to the fact that the losses are independent of the Mach number at high Reynolds numbers and for subsonic Mach numbers, as seen in Figure 7.12.

Interestingly, the aft-loaded airfoil (HS1C) shows the lowest sensitivity to reduced Reynolds numbers, in addition to its superior performance at high Reynolds numbers near the design Mach number. Comparing blade HS1C with blades HS1A and HS1D at exit Mach numbers of 0.85 and 0.95, it can be observed that the rise in losses is not as pronounced for the aft-loaded blade, HS1C, as the Reynolds number is decreased.

7.4.2 Outlet Flow Angle

Figure 7.29 shows the variation of the outlet flow angle with the Reynolds number for the three cascades. As for the losses shown in Fig. 7.28, the Reynolds number effects are strongest at the lowest Mach number, namely 0.60. Both the lower flow turning and higher losses are presumably associated with the presence of a boundary layer separation on suction surface at lower Reynolds numbers.

7.4.3 Blade Loading

The effect of varying the Reynolds number on the loading distribution, while maintaining the outlet Mach number constant, is shown in Figures 7.30 to 7.33 for the mid-loaded baseline cascade, HS1A. Similar results are presented in Figures 7.34 to 7.41 for the aft-loaded cascade, HS1C, and the front-loaded cascade, HS1D, respectively.

Given the limited range of outlet Reynolds numbers that could be achieved in

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 7.29: Reynolds Number Effect on Exit Flow Angle for Several Mach Numbers.

Figure 7.30: Reynolds Number Effect on the Loading Distribution of blade HS1A at Design Incidence and at an Outlet Mach Number of 0.60.

Figure 7.31: Reynolds Number Effect on the Loading Distribution of blade HS1A at Design Incidence and at an Outlet Mach Number of 0.85.

Figure 7.32: Reynolds Number Effect on the Loading Distribution of blade HS1A at Design Incidence and at an Outlet Mach Number of 0.95.

Figure 7.33: Reynolds Number Effect on the Loading Distribution of blade HS1A at Design Incidence and at an Outlet Mach Number of 1.05.

Figure 7.34: Reynolds Number Effect on the Loading Distribution of blade HS1C at Design Incidence and at an Outlet Mach Number of 0.60.

Figure 7.35: Reynolds Number Effect on the Loading Distribution of blade HS1C at Design Incidence and at an Outlet Mach Number of 0.85.

Figure 7.36: Reynolds Number Effect on the Loading Distribution of blade HS1C at Design Incidence and at an Outlet Mach Number of 0.95.

Figure 7.37: Reynolds Number Effect on the Loading Distribution of blade HS1C at Design Incidence and at an Outlet Mach Number of 1.05.

Figure 7.38: Reynolds Number Effect on the Loading Distribution of blade HS1D at Design Incidence and at an Outlet Mach Number of 0.60.

Figure 7.39: Reynolds Number Effect on the Loading Distribution of blade HS1D at Design Incidence and at an Outlet Mach Number of 0.85.

Figure 7.40: Reynolds Number Effect on the Loading Distribution of blade HS1D at Design Incidence and at an Outlet Mach Number of 0.95.

Figure 7.41: Reynolds Number Effect on the Loading Distribution of blade HS1D at Design Incidence and at an Outlet Mach Number of 1.05.

the wind tunnel for a given outlet Mach number, the Reynolds number effect on the loading distribution is limited. Furthermore, the fact that the Reynolds numbers that can be achieved are relatively high also limits the changes observed in the loading distributions since the changes in the surface boundary layers occur more slowly for high Reynolds numbers.

Chapter 8

Results and Discussion of the Cascades' Performance at Off-Design Incidence

8.1 Introduction

The previous chapter dealt with the performance of a family of three HP turbine blades at design incidence. It was seen that by shifting the loading to the rear of the airfoil, the design-point loss performance could be greatly improved. In this chapter, the measurements are extended to off-design incidence.

Turbine blades for use in aircraft engines should be designed such that they can operate at off-design incidence without incurring prohibitive loss penalties. In order to verify that this is indeed the case for the newly designed blade HS1C and HS1D, the study was extended to include tests at both positive and negative incidence angles.

The three cascades were tested at three values of off-design incidence, namely, -10.0, +5.0 and +10.0 degrees relative to design. Initially, the off-design incidence performance of the three airfoils is discussed for the design Mach number in Section 8.2. This then provides the basis for the examination of the flow physics at off-design Mach numbers in Section 8.3 and lower Reynolds numbers in Section 8.4. It should be noted that in both Section 8.2 and 8.3, the key results (losses, outlet flow angles, blade loadings and base pressures) are first presented with minimal discussion. Detailed explanations for the observed behaviour are given once all the quantitative results have been presented.

This study also represents an extension of the off-design low speed turbine cascade research of Benner et al. (1997) into the transonic Mach number range. The present data may allow the correlation for off-design profile losses of Benner et al. (1997) to be extended in the future to include Mach number influence.

The mixed-out loss data as well as several other parameters of interest for the three cascades are tabulated in Appendices D, E and F. Furthermore, in Appendix G, some of the data gathered at off-design Mach numbers and incidences, and presented in this chapter, are cross-plotted in order to offer some other perspectives on the results.

8.2 Performance at Design Mach Number

8.2.1 Profile Losses

The variation of the total pressure loss coefficients with incidence at the design Mach number of 1.05 is shown in Figure 8.1. The losses are seen to be reasonably constant from an incidence of -10.0° to about $+5.0^\circ$. The only exception is the aft-loaded blade which exhibits a slight reduction in losses in going from an incidence of -10.0° to 0.0° . Losses start to increase much more rapidly for incidences above about $+5.0^\circ$ for the front loaded blade HS1D and the aft-loaded blade HS1C. The rise in losses for the baseline cascade HS1A appears not as steep at higher incidences.

Comparison of the three airfoils at the design Mach number shows that the superior performance of the aft-loaded blade previously observed at the design Mach number and incidence extends over a range of inlet flow angles. Losses for blade HS1C are lower than those for the other two blades from an incidence of about -5.0° up to an incidence of about $+5.0^\circ$. The loss performance of the baseline mid-loaded profile is inferior or equal to that of the other two blades for incidences less than about $+5.0^\circ$. However, for the highest incidence tested, the baseline cascade exhibits much lower losses when compared to the other two profiles. The front-loaded blade HS1D has a loss variation with incidence very similar to that of the baseline cascade up to an incidence of $+5.0^\circ$. For higher inlet flow angles, the losses increase much more rapidly for the front-loaded blade.

A comparison of the total pressure loss coefficient variation with incidence measured experimentally and determined computationally using the CFD code FLUENT is shown in Figure 8.2. The numerical predictions follow the general trend in the experimental variation of losses with incidence for the three blade profiles. However,

Figure 8.1: Effect of Incidence on the Total Pressure Loss Coefficient at Design Mach Number.

Figure 8.2: Effect of Incidence on the Total Pressure Loss Coefficient at the Design Mach Number of 1.05 Obtained Experimentally and Computationally.

some important differences can be seen, especially at $+10.0^\circ$ of incidence. This might be related to the presence of increasingly three-dimensional flow at midspan at $+10.0^\circ$ of incidence. This possibility will be examined in more detail at the end of Section 8.2 and in conjunction with the results at off-design Mach numbers.

The reasons for the different trends in the loss variation for the three profiles will be discussed in connection with loading distributions and the base pressures.

8.2.2 Outlet Flow Angle

The variation of the outlet flow angle with incidence is shown in Figure 8.3 for the design outlet Mach number of 1.05. For the three airfoils the highest levels of flow turning are achieved for incidences between -10.0° and 0.0° . The trends in the variation of the outlet flow angle are very similar for both the baseline mid-loaded airfoil HS1A and the aft-loaded airfoil HS1C. That is, the outlet flow angle is almost constant and the highest for values of incidence between -10.0° and 0.0° . As the incidence becomes positive, the amount of flow turning is gradually reduced, due, possibly, to the appearance of some boundary layer separation on the aft suction surface of the airfoils.

For the front-loaded blade, HS1D, the exit flow angle appears to be decreasing for incidences between -10.0° and 0.0° . However, for positive incidence angles, the outlet flow angle for HS1D does not seem to decrease as much as for the other two airfoils.

8.2.3 Blade Loading

Blade loading measurements were made at the design Mach number of 1.05 for the three test cascades at the four incidences investigated. Figure 8.4 shows the results. The small variation in the exit Mach number from the exact target value of 1.05 results from the difficulty in precisely setting the wind tunnel operating point. Repeatability measurements showed that the uncertainty in the surface Mach number is about ± 0.01 .

Reducing the incidence to -10.0° clearly reduces the initial flow acceleration on the suction surface for all three blade designs. The flow velocities on the suction

Figure 8.3: Effect of Incidence on the Outlet Flow Angle at Design Mach Number.

surface only reach those observed at design incidence at about an axial distance x/C_x of 0.4. From this point onwards, very few differences between the loading distribution at -10.0° and design incidence can be observed. For the baseline mid-loaded profile, HS1A, and the aft-loaded profile, HS1C, the only noticeable differences in loading at the design incidence and -10.0° are the locations of the shock impingement points on the aft part of the blade. The shock appears to be stronger and the impingement point slightly shifted to the rear at -10.0° of incidence. For the front-loaded blade, the loading distribution remains practically unchanged from aft of $x/C_x = 0.4$ when the incidence was reduced to -10.0° . The small differences present can be attributed to the fact that the exit Mach number varied slightly.

On the pressure surface, there are essentially no differences in the loading as the incidence is reduced from design to -10.0° .

The fact that a reduction of 10 degrees in incidence has little impact on the rear suction surface loading distribution of the three profiles is consistent with the loss and outlet flow angle measurements presented earlier. As seen previously in Figure 8.1, the losses remain essentially constant for negative incidences except for HS1C which

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.4: Effects of Incidence on Blade Loading for the Three Blade Profiles at Design Mach Number of 1.05.

exhibits a slight increase in losses.

For a positive incidence angle of $+5.0^\circ$, a velocity peak forms on the suction surface near the leading edge, as seen in Figure 8.4. The peak in velocity is followed by a region of adverse pressure gradient, which will tend to thicken the boundary layer. This region of adverse pressure gradient appears somewhat shorter for the aft-loaded profile than for the other two blades. This might explain in part why the loss performance of HS1C is still superior to that of the other two blades at an incidence of $+5.0^\circ$. This observed behaviour is consistent with the argument of Benner et al. (1997) that a larger discontinuity in curvature at the blend points makes a turbine airfoil more sensitive to off-design incidence. The presence of a lower suction surface velocity overspeed close to the leading edge for the aft-loaded blade, HS1C, can be explained by the use of an elliptically-shaped leading edge for this profile. A standard leading edge circle was used for the baseline mid-loaded profile, HS1A, as well as for the front-loaded profile, HS1D. As a result, there is a smaller jump in the surface curvature at the leading-edge blend points for HS1C, compared with HS1A and HS1D. The main reason for designing blade HS1C with an elliptical leading edge had to do precisely with the objective of reducing the overspeeds predicted at the leading edge during design.

As for the negative incidence, the flow reaches the design incidence loading at an axial distance of about $0.4C_x$ at an incidence of $+5.0^\circ$. The differences in the aft part of the loading distribution are still limited when compared to the loading at design incidence.

At an incidence of $+10.0^\circ$, the effect of incidence on the loading distribution is stronger on the forward part and affects a larger extent of the suction surface. The overspeed and the subsequent diffusion are much more severe than at $+5.0^\circ$ of incidence. The velocity reaches approximately the design incidence value at an x/C_x of 0.4 for the three profiles as observed previously at the lower values of incidence. However, for profiles HS1A and HS1C, the loading at an incidence of $+10.0^\circ$ aft of $x/C_x = 0.4$ is slightly lower for some axial distance.

The presence of a region of large overspeed and diffusion at the leading edge can result in the formation of lambda shocks as shown in Figure 8.5. The presence of such shocks together with the adverse pressure gradient is likely to contribute greatly

Figure 8.5: Schlieren Picture of a Transonic Turbine Blade Operating at Incidence (Detemple-Laake, 1989).

to increasing the boundary layer thickness. In addition to representing increased loss, an increase in boundary layer thickness also affects the base pressure. The effect of incidence on base pressure will be discussed in more detail in the following section. Another indication of the presence of thicker surface boundary layers at an incidence of $+10.0^\circ$ can be obtained from the width of the wakes as measured downstream of the airfoils. Figure 8.6 shows the variation of the total pressure loss coefficient in the pitchwise direction for the three blades at all incidences investigated. It can clearly be seen that the width of the wake increases significantly with incidence, particularly at $+10.0^\circ$ where the surface boundary layers are much thicker. The increased thickness also makes the boundary layer more prone to separation.

Whereas increasing the incidence results in greater boundary layer losses, it appears that at the same time the shock losses decrease. This can be seen for the baseline mid-loaded cascade HS1A and to a lesser extent for the aft-loaded and front-loaded profiles. From Figure 8.6(a), it can be observed that the mid-passage total pressure loss coefficient for blade HS1A is lower (less negative) for an incidence of $+10.0^\circ$. This suggests the presence of weaker shocks across the blade passages.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.6: Effect of Incidence on the Variation of the Total Pressure Loss Coefficient with Pitchwise Location.

Figure 8.7: Comparison of the Blade Loading for the Three Cascades at Design Incidence and for an Outlet Mach Number of 1.05.

A comparison of the loading distributions for the three profiles at the design exit Mach number of 1.05 and for each of the four incidence values is presented in Figures 8.7 to 8.10. These figures are a re-plotting of the results from Figure 8.4 for easier comparison of the effects of airfoil geometry at a given incidence value. The results for design incidence from Chapter 7 are also repeated in Figure 8.7 for completeness.

Apart from the distinctive loading distributions for each of the three airfoils, several other differences are observed at design incidence (Figure 8.7). For example, the presence of a weaker impinging shock on the rear suction surface is evident for the aft-loaded blade, HS1C, when compared to HS1A, as well as a lower base Mach number. For the front-loaded blade, HS1D, a mild adverse pressure gradient was present over about 50% of the forward part of the suction surface.

The loading distributions for the three airfoils at an incidence of -10.0° are shown in Figure 8.8. As the incidence was reduced from $+0.0^\circ$ to -10.0° , the extent of the diffusion region on the suction surface of the front-loaded blade, HS1D, was reduced significantly. At an incidence of -10.0° , the appearance of deceleration on

Figure 8.8: Comparison of the Blade Loading for the Three Cascades at an Incidence of -10.0° and for an Outlet Mach Number of 1.05.

Figure 8.9: Comparison of the Blade Loading for the Three Cascades at an Incidence of $+5.0^\circ$ and for an Outlet Mach Number of 1.05.

Figure 8.10: Comparison of the Blade Loading for the Three Cascades at an Incidence of $+10.0^\circ$ and for an Outlet Mach Number of 1.05.

the suction surface is delayed until an axial location x/C_x of about 0.4 whereas it started at about x/C_x of 0.16 at design incidence.

At an incidence of $+5.0^\circ$ (Figure 8.9), a velocity overspeed can clearly be seen for the three blades on the suction surface near the leading edge. The velocity overspeed is weakest for HS1C and is not followed by a significant length of diffusion, unlike for the other two blades. At an axial distance x/C_x of 0.6, the suction surface of the mid-loaded baseline cascade (HS1A) experiences an adverse pressure gradient that extends up to an axial distance x/C_x of 0.8. The corresponding drop in Mach number is almost 0.25. For the aft-loaded blade, HS1C, the adverse pressure gradient observed at x/C_x of 0.6 is weaker than that for HS1A and of shorter extent.

Comparison of the loading distributions for the three profiles at an incidence of $+10.0^\circ$ is shown in Figure 8.10. This figure together with Figure 8.9 helps to explain the differences in the loss trends. Whereas the aft-loaded profile, HS1C, was shown to have a moderate overspeed on the suction surface at an incidence of $+5.0^\circ$ when compared to the other two profiles, at an incidence of $+10.0^\circ$ it exhibits the largest overspeed and the most rapid rate of diffusion at the leading edge. This is consistent

Figure 8.11: Effect of Incidence on the Base Pressure at Design Mach Number, $M_2 = 1.05$.

with the large rise in the losses for HS1C, seen in Figure 8.1, as the incidence increased from $+5.0^\circ$ to $+10.0^\circ$.

8.2.4 Base Pressure

As discussed in the previous chapter, the base pressure is influenced by both the thickness of the boundary layer and the velocity at the trailing edge. Measurements of the base pressure were performed for all test cases. Figure 8.11 presents these results for the three blade profiles as a function of incidence.

From this figure, the rise in base pressure with incidence can clearly be seen for the baseline mid-loaded blade, HS1A, and to a lesser extent for the other two profiles. This is as expected due to the thickening of the boundary layer on the suction surface with increasing incidence. Also noticeable in Figure 8.11, the base pressure coefficient is higher for the aft-loaded blade, HS1C, over the whole range of incidence. As discussed in the previous chapter in connection with the design incidence, this can be attributed to the higher unguided turning of the aft-loaded

blade, HS1C.

The reduction in losses by about 0.015 observed in Figure 8.1 for the aft-loaded profile, HS1C, as the incidence was increased from -10.0° to 0.0° is consistent with the rise in the base pressure coefficient seen in Figure 8.11. Using the control volume analysis of Denton (1993) (Equation 2.1), the increase in base pressure can be estimated to have contributed about one third of the reduction in losses, as follows:

$$\Delta Y_{te} = -\Delta C_b \left(\frac{t}{o} \right) = -0.066 \left(\frac{1.26mm}{15.9mm} \right) = -0.005. \quad (8.1)$$

The presence of a relatively lower base pressure at an incidence of -10.0° for HS1C, compared to that observed at design incidence, also contributes to the formation of a stronger trailing edge shock. The stronger shock impinging on the rear suction surface at -10.0° of incidence, compared to design incidence, can be seen by comparing the Mach number contour plots obtained numerically for both cases in Figures 8.12 and 7.9 respectively.

The stronger rear suction surface impinging shock for HS1C at -10.0° can also be seen by comparing the loading distribution of that blade with that obtained at design incidence, as shown in Figure 8.13. The results are presented for both experimentally measured and numerically calculated Mach number distributions. The stronger rear suction surface impinging shock can be seen from both the measured and computed results. Just as the lower base pressure observed at -10.0° of incidence is believed to contribute to higher losses for blade HS1C, the presence of a stronger rear suction surface impinging shock should also increase the losses. Comparison of wake traverse for blade HS1C at -10.0° and design incidence, in Figure 8.6(b), shows that the free-stream total pressure loss coefficient is lower (less negative) by about 0.02 when blade HS1C is operating at design incidence. Therefore, the overall free-stream shock losses contribute to at least 0.005 to the difference in losses observed between -10.0° of incidence and design incidence.

The improved performance of blade HS1C at design incidence relative to -10.0° of incidence is probably also due in part to the lower boundary layer losses at design incidence. This is inferred from Figure 8.6(b) where it can be seen that the wake for blade HS1C operating at -10.0° of incidence is slightly wider than that for the design

Figure 8.12: Mach Number Contour Plot for Cascade HS1C at Design Mach Number and at an Incidence of -10.0° .

Figure 8.13: Comparison of the Mach Number Distribution for the Aft-Loaded Blade HS1C at -10.0° and Design Incidences Obtained Both Experimentally and Numerically at Design Mach Number.

incidence.

As mentioned previously at the beginning of this chapter and explained in detail in Chapter 7, the loss performance of the aft-loaded blade, HS1C, is superior to that of the other two profiles at design incidence. It was also shown in Figure 8.1 that the superior loss performance of HS1C is also present at an incidence of $+5.0^\circ$. This can be explained by both the slightly weaker rear suction surface impinging shock and the lower flow velocity on the front suction surface for the aft-loaded blade, HS1C.

On the front part of the suction surface, both blades HS1A and HS1D experience a very large velocity overspeed whereas it is much weaker for the aft-loaded blade, HS1C, for reasons explained previously.

The presence of a very large velocity overspeed contributes in two ways to making the losses for HS1A and HS1D larger than those for HS1C. First, a thicker boundary layer results from the subsequent strong adverse pressure gradient. Furthermore, the presence of high velocities at the surface of a blade contributes to the generation of a significant amount of entropy in the boundary layer since the entropy production

rate is proportional to the cube of the velocity at the edge of the boundary layer, as discussed by Denton (1993). Denton showed that the total entropy generation up to a point in a blade surface boundary layer is given by

$$\dot{S} = \frac{\rho V_\delta^3 \delta_s}{T_\delta} = \int_0^x \frac{\rho V_\delta^3 C_d}{T_\delta} dx, \quad (8.2)$$

where, δ_s is the entropy thickness of the boundary layer and C_d is a dimensionless dissipation coefficient. Denton and Cumpsty (1987) suggest that for many turbomachine blades, where the average value of Re_θ is of the order of 1000, a reasonable approximation is simply to take

$$C_d = 0.002 = \text{constant} \quad (8.3)$$

for turbulent boundary layers.

Since blades HS1A and HS1D have significantly higher flow velocity on the suction surface in the vicinity of the leading edge compared to HS1C, the losses generated in the boundary layer in this area are expected to be higher. The aft-loaded blade, HS1C, has slightly higher flow velocity at the rear of the suction surface compared to HS1A and HS1D. However, the velocity differences are not large enough to cause a significant difference in the loss production in this region for the three blades.

Between an incidence of $+5.0^\circ$ and $+10.0^\circ$, it was shown previously in Figure 8.1 that the losses increase much faster for the aft-loaded blade, HS1C, and the front-loaded blade, HS1D. Furthermore, at an incidence of $+10.0^\circ$, the losses for HS1C and HS1D are much higher than that of HS1A.

The milder rate of increase of losses for HS1A is due in part to the rise in base pressure this blade is experiencing as the incidence increases from $+0.0^\circ$ and $+10.0^\circ$. As seen in Figure 8.11, HS1A shows an increase in base pressure coefficient of about 0.2 as the incidence increases from $+0.0^\circ$ to $+10.0^\circ$. According to Denton's (1993) control volume analysis (Equation 2.1), this increase in base pressure coefficient would decrease the base pressure contribution to the total loss coefficient by about 0.016. This partly offsets the other factors tending to increase the losses as the incidence

increases.

For blades HS1C and HS1D, the base pressure was shown to be essentially constant or vary only slightly from $+0.0^\circ$ to $+10.0^\circ$ of incidence. Therefore, the base pressure does not appear to play a major role in the rapid increase of the losses for HS1C and HS1D over this range of incidence.

Another advantage of blade HS1A over HS1C is that the direct shock losses appear to be decreasing with increasing incidence. This is evidenced by looking at the wake traverses in Figure 8.6(a). It can be seen that outside the wake the total pressure loss coefficient is lower at $+10.0^\circ$ of incidence than at any other inlet angle, indicating the presence of weaker shocks across the passage.

Furthermore, the presence of weaker trailing edge shocks at high positive incidences is indicated by the CFD calculations. Figure 8.14 shows the Mach number contour plot for the baseline mid-loaded blade, HS1A, at an incidence of $+10.0^\circ$. Comparing Figure 8.14 with Figure 7.3, which shows the equivalent contour plot for HS1A at design incidence, the weaker impinging trailing edge shock can readily be seen for $+10.0^\circ$.

Another evidence of weaker shocks for HS1A at $+10.0^\circ$ is present in the Mach number distribution for that blade at various values of incidences, as shown previously in Figure 8.4(a). In this figure, it can be clearly seen that the strength of the shock impinging on the rear suction surface diminishes with increasing incidence. At an incidence of $+10.0^\circ$, the location of the impinging shock cannot even be deduced from the Mach number distribution. For the aft-loaded blade, HS1C, and to a lesser extent for the front-loaded blade, HS1D, the strength of the rear suction surface impinging shock appears to remain more or less the same when the incidence is increased from $+5.0^\circ$ to $+10.0^\circ$.

Another contributing factor to the better performance of blade HS1A at high positive incidence is its apparently relatively thin boundary layers. Comparing the width of the wake in Figure 8.6 for the three blades at an incidence of $+10.0^\circ$, it can be seen that the wake for the mid-loaded profile, HS1A, is significantly thinner than that of the other two blades, indicating much thinner surface boundary layers.

Finally, the magnitude of the velocity on the blade surfaces certainly plays a role in determining the level of losses for the three blades. As mentioned previously,

Figure 8.14: Mach Number Contour Plot for Cascade HS1A at Design Mach Number and at an Incidence of $+10.0^\circ$.

Figure 8.15: Effect of Incidence on the Axial Velocity Density Ratio, AVDR, at Design Mach Number.

the losses in the boundary layer are proportional to the velocity cubed. The velocity on the suction surface of the front-loaded blade, HS1D, is higher than that of blade HS1A following the velocity overspeed at the leading edge. Furthermore, the velocity on the pressure surface of blade HS1D is higher than that of the other two blades, starting at an axial location x/C_x of about 0.4 up to the trailing edge.

Similarly, the aft-loaded blade, HS1C, exhibits higher flow velocity on its rear suction surface when compared to the mid-loaded blade, HS1A. This is yet another factor contributing to the poorer loss performance of the aft-loaded profile relative baseline mid-loaded blade at $+10.0^\circ$ of incidence.

Finally, the possible presence of three-dimensional flow at midspan for $+10.0^\circ$ of incidence may be affecting the losses. This is evidenced by the non-unity values of AVDR measured at $+10.0^\circ$ as shown in Figure 8.15. The axial velocity density ratio, AVDR, at the mid-span location varied between 0.98 and 1.02 ± 0.02 for incidences between -10.0° and $+5.0^\circ$ as shown in Fig. 8.15. Therefore, the cascade mid-span flow is taken as essentially two-dimensional for these values of incidence, within the experimental uncertainties in the measurements. However, at an incidence of $+10.0^\circ$,

the AVDR for the three blade profiles is about 1.08 ± 0.02 and therefore the flow is starting to deviate noticeably from two-dimensionality. It is possible that the flow convergence implied by the rise in AVDR may also mean that some secondary losses are present at midspan. Although the effect does not appear to be very large, nevertheless the data presented for $+10.0^\circ$ should probably be treated with some caution.

This would also explain the somewhat large discrepancies at $+10.0^\circ$ of incidence observed between the experimental loss measurements and the two-dimensional numerical predictions shown previously in Figure 8.2. This possibility will be examined in more detail in conjunction with the results at off-design Mach numbers where three-dimensional flow effects are much more severe.

8.3 Performance at Off-Design Mach Numbers

8.3.1 Profile Losses

In the previous section results were presented for the performance of the three cascades at off-design incidence and at design Mach number. In this section, the discussion of the results is extended to the effect of off-design Mach numbers.

In Figures 8.16(a), (b) and (c), the variation of the total pressure loss coefficient with Mach number is shown for the three blade profiles. Results are presented for incidence values of -10.0° , 0.0° , $+5.0^\circ$ and $+10.0^\circ$.

Starting with the baseline mid-loaded cascade, HS1A, several differences can be seen in the variation of the losses with Mach number for the different values of incidence, as shown in Figure 8.16(a). At incidence values of -10.0° , 0.0° and $+5.0^\circ$, the trends in the variation of the losses with Mach number are very similar to one another and are consistent with similar measurements made by Mee et al. (1992) for example, as discussed in Section 2.4. That is, the losses are essentially constant at low Mach numbers. At transonic Mach numbers, the losses start increasing due to the appearance of shock waves in the blade passages. Among the differences that can be observed at these three incidences, are slight increases in losses at low outlet Mach numbers as the incidence increases from -10.0° to $+5.0^\circ$. This can also be seen in

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.16: Effect of Mach Number on the Variation of the Total Pressure Loss Coefficient for Various Incidences.

Appendix G (e.g. Figure G.1(a)) where the variation of the losses with incidence are shown for constant values of outlet Mach number.

Similar trends are observed for the aft-loaded blade, HS1C, and the front-loaded blade, HS1D, for incidence values between -10.0° and $+5.0^\circ$.

At an incidence of $+10.0^\circ$, the highest incidence for which measurements were made, the trend in the variation of the losses with Mach number is very different. For the lowest Mach numbers, the losses are high compared to those observed at the lower values of incidence. Such high levels of loss would usually be attributed to extensive boundary layer separation on the suction surface. However, as shown in Chapter 5, at high values of incidence and low Mach numbers there is essentially no region of two-dimensional flow at midspan. Therefore, the high losses observed at $+10.0^\circ$ and low Mach numbers may be due to some extent to the presence at midspan of secondary losses. The origin of the high losses observed at $+10.0^\circ$ will be discussed further in conjunction with the discussion for the loading measurements and base pressure measurements.

The results just presented showed the effect of incidence on the variation of the losses with Mach number for the three blade profile individually. In order to facilitate the comparison between the three blades, and thus see the influence of the loading distribution on the losses at off-design incidence and Mach number, the data were re-plotted as shown next.

A comparison of the effect of the outlet Mach number on the losses for the three cascades is presented in Figures 8.17 to 8.20 for the four values of incidence investigated.

At an incidence of -10.0° (Figure 8.17), the variation of the profile losses with Mach number is very similar for the three blades except at high Mach numbers where the baseline mid-loaded blade has significantly better loss performance. For outlet Mach numbers in the range from 0.80 to about 1.05 (the design Mach number), the aft-loaded blade, HS1C, and the front-loaded blade, HS1D, appear to have slightly lower losses compared to the baseline mid-loaded blade, HS1A. This appears to be due to the fact that the rise in losses due to the appearance of shock waves at transonic Mach numbers ($M_{2s} \approx 0.85$) occurs slightly earlier for HS1A.

As the the incidence is increased to its design value (Figure 8.18), the perfor-

Figure 8.17: Effect of Mach Number on Losses for the Three Cascade at -10.0° Incidence.

Figure 8.18: Effect of Mach Number on Losses for the Three Cascade at Design Incidence.

Figure 8.19: Effect of Mach Number on Losses for the Three Cascade at +5.0° Incidence.

Figure 8.20: Effect of Mach Number on Losses for the Three Cascade at +10.0° Incidence.

mance of the aft-loaded profile becomes superior to that of the other two profiles in the vicinity of the design Mach number, as discussed earlier in Chapter 7. However, for outlet Mach numbers greater than about 1.10 for HS1C and 1.15 for HS1D, the losses start increasing rapidly and the loss performance of these two blades deteriorates much faster than that of HS1A.

For an incidence of $+5.0^\circ$, the plot of the variation of losses with outlet Mach number (Figure 8.19) shows that the losses for the aft-loaded blade, HS1C, are much higher at subsonic Mach numbers than that of HS1A and HS1D. However, as mentioned previously, the losses for the aft-loaded blade appear slightly lower near the design Mach number of 1.05.

At an incidence of $+10.0^\circ$, the variation of losses with Mach number is very different from that observed at lower values of incidence. For operation at this incidence, the baseline mid-loaded cascade offers the lowest profile losses of the three airfoils at practically all outlet Mach numbers. The aft-loaded blade, HS1C, which showed surprisingly good loss performance at design incidence, has the poorest loss performance over almost the entire Mach number range at an incidence of $+10.0^\circ$.

Comparisons between the experimental loss measurements and the numerically predicted losses for the three blade profiles are presented for the four values of incidence in Figures 8.21 to 8.24. As described in Chapter 5, the numerical simulations were performed using the commercial CFD code FLUENT.

Comparing the measured and numerical results in Figure 8.21 and 8.22, it can be seen that for -10.0° of incidence and design incidence the numerical calculations predict the same trend in the losses with Mach number as were observed experimentally. The magnitude of the predicted losses is also very close to the experimentally measured losses excepted for few cases. At $+5.0^\circ$ of incidence (Figure 8.23), the loss predictions depart slightly from the measurements for outlet Mach numbers lower than about 1.0. However, for outlet Mach numbers greater than 1.0, the predictions are still very good. The poor agreement between the experimental and numerical results at subsonic Mach numbers will be shown at the end of this section to be due to three-dimensional flow at midspan in the cascade test section at these operating conditions.

Interestingly, the trend in loss variation with Mach number and the magnitude

Figure 8.21: Comparison of the Effect of Mach Number on Losses Measured Experimentally and Predicted Numerically for the Three Cascade at -10.0° Incidence.

Figure 8.22: Comparison of the Effect of Mach Number on Losses Measured Experimentally and Predicted Numerically for the Three Cascade at Design Incidence.

Figure 8.23: Comparison of the Effect of Mach Number on Losses Measured Experimentally and Predicted Numerically for the Three Cascade at $+5.0^\circ$ Incidence.

Figure 8.24: Comparison of the Effect of Mach Number on Losses Measured Experimentally and Predicted Numerically for the Three Cascade at $+10.0^\circ$ Incidence.

of the losses could not be reproduced numerically at an incidence of $+10.0^\circ$. Again, the large discrepancies between the losses measurements and the predictions at $+10.0^\circ$ of incidence are explained at the end of Section 8.3 by the fact that the experimental loss measurements for these conditions include some secondary losses.

Notwithstanding the discrepancies between the experimentally measured losses and those obtained computationally at low Mach numbers and at positive incidences, the computational results agree reasonably well with the experimental measurements. Specifically, the general trends in the variation of losses with Mach number are predicted satisfactorily. As an example, the rise in losses due to the appearance of shock waves around a Mach number of about 0.85 is predicted well for most cases. Furthermore, the magnitude of the predicted losses is generally in line with those measured experimentally.

Based on the comparisons made previously, it appears that state-of-art CFD codes are now capable of predicting reasonably well the flow through transonic turbine cascades operating at high Reynolds numbers and at off-design conditions. Areas where improvements are still needed in terms of prediction capabilities are for cascade flows at low Reynolds numbers where the flow is transitional over the airfoils. Adequate predictions of such cascade flows required transition models developed specifically to handle the effects of pressure gradient on transition. As it stands now, most of these transition models are still in development or are highly specialized for a particular class of flows. Therefore, transition models are not readily available as a standard option in commercially available codes. However, given the interest that these transition models generate in turbine aerodynamics at the moment, as evidence by the number of paper published on the subject (see for example Praisner and Clark (2004), Menter et al. (2004) or Roberts and Yaras (2004)), it is foreseeable in the near future that these transition models will be a standard feature in commercially available CFD codes.

In order to offer another perspective on the profile loss results, the data presented in Figure 8.16 were cross-plotted as a function of effective incidence ($i_{effective} = i - i_{des}$). The resulting plots are presented in Appendix G, Figure G.1(a), (b) and (c). The results are shown for four constant values of Mach number, namely, 0.65, 0.85, 1.05 and 1.25.

Similarly, the data presented Figures 8.17 to 8.20 were cross-plotted as a function of effective incidence ($i_{effective} = i - i_{des}$). The results are shown in Figures G.2 to G.5 where the variation of profile losses with incidence is shown for four constant Mach numbers for the three blade profiles.

8.3.2 Outlet Flow Angle

The effect of incidence on the variation of the outlet flow angle with Mach number is shown in Figure 8.25(a), (b) and (c) for the three blade profiles. As expected, the amount of deviation is increased with increasing positive incidences. This is in agreement with the increase in losses with incidence observed previously in Figure 8.16.

A comparison of the effect of outlet Mach number on the outlet flow angle for the three blades at different incidences is presented in Figures 8.26 to 8.29. In Figure 8.26 very little difference can be seen in the variation of outlet flow angle with Mach number except perhaps for the highest Mach numbers for which the flow turning is a little higher for the baseline cascade, HS1A. This should not be surprising given that the losses presented previously in Figure 8.17 for the same conditions did not vary significantly between the three cascades except for the highest Mach numbers where the losses were lower for HS1A. Therefore, the trends in the variation of the outlet deviation appear consistent with the variation of the losses with the outlet Mach number. This is as expected given that higher losses can be associated with thicker blade surface's boundary layers which result in higher deviation.

As the incidence angle becomes positive, differences in the outlet flow angle between the three cascades can be observed over a wider range of Mach numbers. For example, at an incidence of $+5.0^\circ$, the outlet flow angle is slightly lower for the aft-loaded blade, HS1C, for subsonic outlet Mach numbers. This corresponds to the higher losses measured for HS1C at these conditions and presented in Figure 8.19.

Finally, at an incidence of $+10.0^\circ$ significant differences are present between the three profiles in the variation of the outlet flow angle with Mach number. First, for the aft-loaded profile, HS1C, the outlet flow angle is much lower than that achieved by the other two blades. This is as expected given the higher losses for HS1C over

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.25: Effect of Mach Number on the Variation of the Outlet Flow Angle for Various Incidences.

Figure 8.26: Effect of Mach Number on Outlet Flow Angle for the Three Cascades at -10.0° Incidence.

almost the entire Mach number range at this incidence, as shown in Figure 8.20.

When comparing the outlet flow angle variation for the front-loaded blade, HS1D, with that of the baseline mid-loaded blade, HS1A, at an incidence of $+10.0^\circ$ something unexpected can be seen. For Mach numbers greater than about 0.80, the front-loaded profile is achieving greater flow turning despite the fact that it is producing much greater losses for these conditions, as shown in Figure 8.20. As will be shown at the end of Section 8.3, there are strong evidences that point out to the presence of a significant amount of secondary flow a midspan for blade HS1D at an incidence of $+10.0^\circ$. The presence of secondary flow at midspan can certainly contribute to modify greatly the outlet flow angle.

The variation of the outlet flow angle with incidence is presented in Figure G.6 in Appendix G. As for the profile losses versus incidence plots presented in the previous section, the data used for these figures were obtained by interpolation from Figure 8.25.

In Figures G.7 to G.10, a comparison of the outlet flow angle variation with incidence is made between the three airfoils at four different Mach numbers.

Figure 8.27: Effect of Mach Number on Outlet Flow Angle for the Three Cascades at Design Incidence.

Figure 8.28: Effect of Mach Number on Outlet Flow Angle for the Three Cascades at +5.0° Incidence.

Figure 8.29: Effect of Mach Number on Outlet Flow Angle for the Three Cascades at $+10.0^\circ$ Incidence.

8.3.3 Blade Loading

As mentioned previously, blade loading measurements were performed for all test conditions. Figures 8.30, 8.31, 8.32, 8.33 show the effects of the exit Mach number on the surface isentropic Mach number distributions for all flow conditions that were tested with the baseline mid-loaded cascade HS1A.

At low Mach numbers, the loading distributions show a region of nearly constant Mach number on the suction surface for all incidences. This could be associated with the presence of boundary layer separation on the suction surface. For example, this would be consistent with the much higher losses observed at an incidence of $+10.0^\circ$. However, given that the flatness in the Mach number distributions at low Mach numbers is observed for all values of incidence, the unusual loading distributions are probably simply those resulting at low Mach numbers for airfoils designed for use at much higher Mach numbers.

For exit Mach numbers greater than about 1.0, the suction-side loading distributions remain constant from the leading edge up to an axial distance of about

Figure 8.30: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at an Incidence of -10.0° .

Figure 8.31: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at Design Incidence.

Figure 8.32: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at an Incidence of $+5.0^\circ$.

Figure 8.33: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at an Incidence of $+10.0^\circ$.

Figure 8.34: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Aft-Loaded Cascade HS1C at an Incidence of -10.0° .

$x/C_x = 0.75$ as the outlet exit Mach number increases. This is due to the choking of the flow in the blade passage. At all incidences, for the highest outlet Mach numbers investigated, an oblique shock originating at the trailing edge of the adjacent airfoil impinges on the rear of the suction surface. However, there are notable variations in the apparent strength of this shock, and the corresponding drop in Mach number. These differences will be used to explain the variation in loss performance between the three airfoils in the next section.

Similarly, Figures 8.34, 8.35, 8.36, 8.37 show the effects of exit Mach number on the surface Mach number distributions for the aft-loaded blade, HS1C, for incidence values of -10.0° , 0.0° , $+5.0^\circ$ and $+10.0^\circ$ respectively.

The effects of the outlet Mach number on the loading distribution of the front-loaded blade, HS1D, are shown in Figures 8.38, 8.39, 8.40, 8.41 for the different incidences investigated.

The effects of incidence on the surface Mach number distribution of the baseline mid-loaded profile, HS1A, is shown in Figures 8.42, 8.43, 8.44 and 8.45. This series

Figure 8.35: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Aft-Loaded Cascade HS1C at Design Incidence.

Figure 8.36: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Aft-Loaded Cascade HS1C at an Incidence of $+5.0^\circ$.

Figure 8.37: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Aft-Loaded Cascade HS1C at an Incidence of $+10.0^\circ$.

Figure 8.38: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Front-Loaded Cascade HS1D at an Incidence of -10.0° .

Figure 8.39: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Front-Loaded Cascade HS1D at Design Incidence.

Figure 8.40: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Front-Loaded Cascade HS1D at an Incidence of $+5.0^\circ$.

Figure 8.41: Effect of Outlet Flow Mach Number on the Surface Mach Number Distribution of the Front-Loaded Cascade HS1D at an Incidence of $+10.0^\circ$.

of plots presents the results for four distinct outlet Mach numbers.

At an outlet Mach number of 0.65, the effect of incidence on the loading distribution of blade HS1A is mainly concentrated on the suction surface of the blade as shown in Figure 8.42. Increasing the incidence from -10.0° to $+10.0^\circ$ results in the formation of a slight velocity overspeed at the leading edge on the suction surface for incidence of $+5.0^\circ$ and $+10.0^\circ$. At an incidence of $+5.0^\circ$, the velocity overspeed on the suction surface is very moderate and hence the subsequent adverse pressure gradient is very mild. Therefore, the effect of the velocity overspeed is limited essentially to the leading edge region of the blade at an incidence of $+5.0^\circ$. Further aft on the suction surface, the loading distribution at $+5.0^\circ$ of incidence is very similar to what was observed at incidence of -10.0° and 0.0° . At $+10.0^\circ$ of incidence, a slight reduction in loading can be observed on the suction surface between an x/C_x of 0.3 and 0.7. Similar reductions in loading can be observed at higher outlet Mach number at an incidence of $+10.0^\circ$. The reasons for that will be discussed in the following section.

Figure 8.42: Effect of Incidence on the Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at an Outlet Mach Number of 0.65.

Figure 8.43: Effect of Incidence on the Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at an Outlet Mach Number of 0.85.

Figure 8.44: Effect of Incidence on the Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at an Outlet Mach Number of 1.05.

Figure 8.45: Effect of Incidence on the Mach Number Distribution of the Baseline Mid-Loaded Cascade HS1A at an Outlet Mach Number of 1.15.

Similar loading distributions at incidences between -10.0° and $+5.0^\circ$ can be observed at an outlet Mach number of 0.85 for the baseline mid-loaded cascade as shown in Figure 8.43. The limited effect of incidence on the loading distribution for incidence between -10.0° and $+5.0^\circ$ is consistent with the variation of losses with Mach number shown previously in Figure 8.16(a). In this figure, it can be observed that for outlet Mach number below about 0.85, variations in incidence have limited effect on the losses up to an incidence of $+5.0^\circ$. As the incidence is increased to $+10.0^\circ$, the velocity overspeed at the leading edge on the suction surface becomes much more important. The strong adverse pressure gradient that ensues usually results in a boundary layer separation. Such a region of flow separation can be seen, for example, in Figure 8.14 which shows the Mach number contour plot for blade HS1A at the design Mach number of 1.05.

Loading distributions for the design outlet flow Mach number of 1.05 (Figure 8.44) were discussed in the previous section. The trends observed in the loading distributions at this Mach number are similar to those at an outlet Mach number of 1.15.

The loading distributions for the baseline mid-loaded cascade, HS1A, at high outlet Mach numbers of about 1.15 are presented in Figure 8.45. At these high Mach numbers, the influence of incidence on the loading is still seen mainly on the suction surface. However, for positive incidence, the velocity overspeed at the leading edge is much more significant as is the resulting flow deceleration.

The effect of incidence on the Mach number distribution of the aft-loaded blade, HS1C, is in some aspects very similar to that of the baseline mid-loaded cascade, HS1A. That is, the effect of incidence is felt mostly on the suction surface in the form of a velocity overspeed close to the leading edge.

The effects of incidence on the surface Mach number distributions for the aft-loaded profile, HS1C, are shown in Figures 8.46, 8.47, 8.48 and 8.49. This series of plots presents the results for four distinct outlet Mach numbers.

The effect of incidence on the Mach number distributions for the front-loaded blade, HS1D, can be seen by looking at the graphs presented in Figures 8.50, 8.51, 8.52 and 8.53. As for blades HS1A and HS1C, a velocity overspeed develops on the suction surface of this blade close to the leading edge at high incidences. However, contrary

Figure 8.46: Effect of Incidence on the Mach Number Distribution of the Aft-Loaded Cascade HS1C at an Outlet Mach Number of 0.65.

Figure 8.47: Effect of Incidence on the Mach Number Distribution of the Aft-Loaded Cascade HS1C at an Outlet Mach Number of 0.85.

Figure 8.48: Effect of Incidence on the Mach Number Distribution of the Aft-Loaded Cascade HS1C at an Outlet Mach Number of 1.05.

Figure 8.49: Effect of Incidence on the Mach Number Distribution of the Aft-Loaded Cascade HS1C at an Outlet Mach Number of 1.15.

Figure 8.50: Effect of Incidence on the Mach Number Distribution of the Front-Loaded Cascade HS1D at an Outlet Mach Number of 0.65.

to the other blades the adverse pressure gradient that follows the overspeed extends much further downstream, to an axial distance x/C_x of about 0.6. This results in a separation region on the suction surface of blade HS1D that extends almost to the mid-chord region as shown in Figure 8.54.

Figure 8.55 presents a comparison between the Mach number distribution measured experimentally and the one obtained computationally close to the design Mach number and at an incidence of $+10.0^\circ$ for HS1D. From this figure, it can be seen that the Mach number distribution is predicted well even though boundary layer separation is present.

Comparison of the Mach number distribution for the three blade profiles at all incidences investigated and at outlet Mach numbers of 0.65, 0.85, 1.05 and 1.15 are presented in Figures G.11 to G.26 in Appendix G.

The loading distributions will be used in the following section along with the other measurements to discuss the observed loss behaviours for the three airfoils.

Figure 8.51: Effect of Incidence on the Mach Number Distribution of the Front-Loaded Cascade HS1D at an Outlet Mach Number of 0.85.

Figure 8.52: Effect of Incidence on the Mach Number Distribution of the Front-Loaded Cascade HS1D at an Outlet Mach Number of 1.05.

Figure 8.53: Effect of Incidence on the Mach Number Distribution of the Front-Loaded Cascade HS1D at an Outlet Mach Number of 1.15.

8.3.4 Base Pressure

Measurements of the base pressure were performed for all test cases. For completeness, the base pressure measured at various outlet Mach numbers and values of incidence for the three blade profiles will be presented. Then explanations for the trends observed in the variation of losses with Mach number and incidence will be given based on the outlet flow angle measurements, Mach number distributions and the base pressure measurements. Comparison with the numerical results will be made whenever useful in order to explain the results.

The effect of incidence on the variation of the base pressure coefficient with Mach number is shown in Figure 8.56(a), (b) and (c) for the three blade profiles. A comparison of the effect of outlet Mach number on the base pressure coefficient for the three blades at different values of incidence is presented in Figures 8.57 to 8.60. Cross-plots of the variation of the base pressure coefficient with incidence are also presented in Appendix G (Figure G.27). The data used for these figures were obtained by interpolation from Figure 8.56. Finally, in Figures G.28 to G.31, a

Figure 8.54: Mach Number Contour Plot for Cascade HS1D at Design Mach Number and at an Incidence of $+10.0^\circ$.

Figure 8.55: Comparison of the Mach Number Distribution Obtained Experimentally and Numerically at +10.0° of Incidence and an Outlet Mach Number of 1.07 for HS1D.

comparison of the base pressure coefficient variation with incidence is made between the three airfoils at four different Mach numbers.

For all incidences, the mid-loaded blade, HS1A, experiences a sharp drop in base pressure for outlet Mach numbers starting at about 0.90, and a subsequent rise in base pressure beginning at about the design Mach number of 1.05, as shown in Figure 8.56(a). The drop in base pressure coincides with the point where the losses start to increase at incidences of -10.0° , $+0.0^\circ$ and $+5.0^\circ$, as shown in Figure 8.16(a). Denton (1993) has shown using a control volume analysis for incompressible flow that the trailing edge losses could be related to the base pressure coefficient and the boundary layer parameter as follows:

$$Y_{te} = -\frac{C_{bt}}{o} + \frac{2\theta}{o} + \left(\frac{\delta^* + t}{o}\right)^2 \quad (8.4)$$

As mentioned previously, for blade HS1A, the parameter (t/o) equals 0.08 and thus from Equation 8.4, the reduction in base pressure of about 0.25 in Figure 8.56(a)

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.56: Effect of Mach Number on the Variation of the Base Pressure Coefficient for Various Incidences.

Figure 8.57: Effect of Mach Number on Base Pressure Coefficient for the Three Cascade at -10.0° Incidence.

Figure 8.58: Effect of Mach Number on Base Pressure Coefficient for the Three Cascade at Design Incidence.

Figure 8.59: Effect of Mach Number on Base Pressure Coefficient for the Three Cascade at +5.0° Incidence.

Figure 8.60: Effect of Mach Number on Base Pressure Coefficient for the Three Cascade at +10.0° Incidence.

contributes to more or less 0.02 to the increase in Y_{te} . This account for roughly a third of the losses observed close to the sonic condition for HS1A, for incidences between -10.0° and $+5.0^\circ$. The remainder of the rise in losses is probably mainly due to the direct losses occurring through the shock waves that begins forming at outlet Mach numbers greater than about 0.85.

Above the design Mach number of 1.05, the losses briefly level-off for the mid-loaded blade. This levelling-off of the losses coincides with the rising base pressure observed in Figure 8.56(a) in that range of Mach numbers. As the exit Mach number is increased further, the losses start rising again due to the presence of much higher flow velocity on the rear suction surface of the blade, as seen from the loadings in Figures 8.30, 8.31 and 8.32. Furthermore, stronger trailing edge shocks are present at high exit Mach numbers, thus contributing to the increased losses.

At an incidence of $+10.0^\circ$, the loss variation with Mach number is very much different from that at lower incidences for the mid-loaded blade, HS1A, as seen in Figure 8.16(a). The losses are highest at the lowest Mach number investigated and decrease up to a Mach number of 0.95 where they level-off. The presence of high losses at low Mach numbers can have a number of possible origins.

First, the high losses at low Mach numbers could be a Reynolds number effect. As mentioned previously, when the ejector-diffuser is not used in the Pratt & Whitney Canada high speed wind tunnel, the Reynolds number decreases with Mach number. When operating at low Mach numbers, the Reynolds number reaches values of the order of 4×10^5 . At low Reynolds number, the transition location on the blade can be delayed significantly, therefore yielding a greater extent of laminar boundary layer on the blade surface. Since laminar boundary layers are more prone to separation in the presence of adverse pressure gradient, losses can be increased significantly through separation. However, the high losses at low Mach numbers seen in Figure 8.16(a) at an incidence of $+10.0^\circ$ do not appear to be due to a Reynolds number effect. This can be seen by comparing the magnitude of the losses at an outlet Mach number of 0.60 in Figure 8.16(a) with those observed in Figure 8.69(a). In these figures, it can be seen that the magnitude of the losses at a Mach number of 0.60 in Figure 8.16(a) corresponds essentially to the range where the losses are independent of Reynolds number in Figure 8.69(a).

The other possible explanation for the high losses observed at low Mach number could be the presence of secondary flow at midspan. As shown previously in Figure 6.14, the secondary flow through the cascade passage grows in importance with increasing positive incidence. Furthermore, the secondary flow was shown to reach the passage midspan location at lower Mach numbers first. It was shown in Figure 6.14 that at an incidence of $+12.0^\circ$ and an outlet Mach number of 0.80 there was essentially no region of two-dimensional flow at midspan. Therefore, it is quite conceivable that the secondary flow had reached midspan even at an incidence of $+10.0^\circ$ for subsonic Mach numbers. This seems the most likely explanation for the increased losses measured at midspan for these conditions.

Another indication of the presence of three-dimensional flow at midspan is non-unity AVDR values. As mentioned previously, a value of AVDR of 1.0 is a necessary (but not sufficient) condition in order to have two-dimensional flow at midspan. Figure 8.61(a) shows the variation of the AVDR with outlet Mach number for the baseline mid-loaded blade, HS1A, at the various incidences investigated.

From Figure 8.61(a), it can be seen that the AVDR for HS1A at $+10.0^\circ$ is significantly higher than that observed at lower values of incidence, at which the values were closer to 1.0. The AVDR for HS1A at $+10.0^\circ$ of incidence is about 1.12 for Mach numbers below 1.0, indicating the presence of some degree of flow convergence at midspan. Therefore, it appears that the relatively high losses at an incidence of $+10.0^\circ$ and Mach numbers below 1.0 for the mid-loaded profile are due to the presence of three-dimensional flow at the midspan location.

Further evidence of the presence of three-dimensional flow at midspan for HS1A at the aforementioned operating conditions is obtained by comparing the losses measured experimentally with those calculated numerically, as shown in Figure 8.62(a). As described in Chapter 5, the numerical computations are strictly two-dimensional.

Comparison of the loss predictions with the experimental results reveals that the losses calculated numerically are significantly lower than those measured experimentally for the mid-loaded blade at an incidence of $+10.0^\circ$ and outlet Mach numbers below 1.0. This is explained by the fact that the numerical simulations were two-dimensional and thus the losses obtained numerically only account for the midspan

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.61: Effect of Mach Number on the Variation of the Axial Velocity Density Ratio for Various Incidences.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.62: Comparison of Loss Variation With Mach Number for Experimental and Numerical Results at $+10.0^\circ$ of Incidence.

Figure 8.63: Comparison of the Mach Number Distribution Obtained Experimentally and Numerically at +10.0° of Incidence and an Outlet Mach Number of 0.53 for HS1A.

losses as opposed to the losses determined experimentally which seem to include some three-dimensional effects.

In Figure 8.63, a comparison between the Mach number distribution measured experimentally and that determined numerically for the lowest outlet Mach number investigated at an incidence of +10.0° is presented for HS1A. In this figure, it can be seen that the predicted loading on the suction surface of HS1A is higher over the entire surface. This difference can be explained by the presence of flows differing significantly experimentally and numerically on the blade's surface at these conditions. The presence of three-dimensional flow at midspan would certainly contribute to modify the loading distribution over blade HS1A. In Figure 8.64, similar results are plotted but this time at design incidence for which it was shown previously that the flow is two-dimensional. It can be seen that the numerically predicted loading distribution is in good agreement with the experimental measurements.

For Mach numbers greater than 1.0, the numerical loss predictions compare favorably with the experimental results as shown in Figure 8.62(a). This is indicating

Figure 8.64: Comparison of the Mach Number Distribution Obtained Experimentally and Numerically at Design Incidence and an Outlet Mach Number of 0.51 for HS1A.

possibly a lesser amount of three-dimensional flow at midspan. This is corroborated by the fact that with increasing Mach number the AVDR for HS1A decreases and approaches 1.0, as shown in Figure 8.61(a). Furthermore, comparing the Mach number distributions obtained experimentally and numerically for HS1A at $+10.0^\circ$ of incidence and at the highest Mach number (Figure 8.65), it can be seen the the numerical prediction matched the experimental results very well except at the leading edge on the suction surface where a strong overspeed is predicted compared with the measured. Given that the numerical simulation was two-dimensional, this tends to confirm that the midspan flow at higher Mach numbers was closer to being two-dimensional.

The largest differences between the loss variations with Mach number at off-design incidences for HS1C, HS1D, and HS1A occur at high outlet Mach numbers, except at $+10.0^\circ$ of incidence where the three-dimensional flows were shown to play a role.

It appears that the reason why the losses are much higher at high outlet Mach

Figure 8.65: Comparison of the Mach Number Distribution Obtained Experimentally and Numerically at +10.0° of Incidence and an Outlet Mach Number of 1.20 for HS1A.

numbers for HS1C and HS1D compared to HS1A is the fact that the losses do not level-off for these two blades in the range of Mach numbers from 1.05 to 1.15. The only exception to that being for blade HS1C at design incidence for reasons discussed in Chapter 7. This can be seen clearly in Figure 8.17 to Figure 8.19. In these figures, it can be observed that the losses increase at approximately the same rate for the three profiles starting at about an outlet Mach number of 0.85. The fact that the losses do not level-off for HS1C and HS1D is consistent with the fact that, unlike for blade HS1A, the base pressure does not rise sharply for HS1C and HS1D at Mach numbers greater than about 1.05, as seen in Figure 8.56(b) and (c). As discussed previously, the rapid rise in base pressure at high Mach number for HS1A was shown to contribute to the levelling-off of the losses through a direct reduction in the trailing edge losses. The rapid rise in base pressure for HS1A thus contributes to delay the rapid increase in losses whereas for blade HS1C and HS1D the base pressure varies only slightly at high outlet Mach number.

The rapid rise in base pressure for HS1A at high outlet Mach numbers can be

explained as follows. For higher outlet Mach numbers, the adjacent blade trailing edge shock impingement point shifts to a location closer to the trailing edge on the suction surface, as seen from Figures 8.30 to 8.32. Thus, at higher Mach numbers, the flow cannot re-accelerate as much following the region of diffusion, at the shock impingement point, before reaching the trailing edge. Hence, the fluid momentum in the shear layer is reduced and its entrainment power on the base region fluid is consequently reduced. This results in higher base pressure at the trailing edge.

As mentioned previously, the increase in base pressure for HS1A at high Mach numbers resulted in a relative improvement of its performance when compared to HS1C and HS1D for values of incidence between -10.0° and $+5.0^\circ$. As the base pressure increased, the strength of the trailing edge shocks was reduced. This can be seen by comparing the loading distributions of HS1A at exit Mach numbers close to the design Mach number of 1.05 with that at the higher Mach number investigated in Figures 8.30 to 8.32. As observed from these figures, the amount of diffusion induced on the rear suction surface of the blade by the impinging shock is reduced slightly at the highest outlet Mach number, even though it corresponds to a significantly higher Mach number than the design Mach number of 1.05.

At an incidence of $+10.0^\circ$, the secondary flow appears to have a significant influence on the midspan loss performance of blades HS1C and HS1D just as was the case for HS1A. In Figure 8.16, it can be observed that the losses are significantly higher at an incidence of $+10.0^\circ$ for all three blade profiles. Comparison of the experimentally determined losses with the two-dimensional numerical calculations in Figure 8.62(b) and (c) shows that for Mach numbers lower than approximately 1.0 significant discrepancies exist just as for HS1A due to the presence of three-dimensional flow at the blades' midspan section. Again, this is evidenced by the fact that the AVDR values for HS1C and HS1D at $+10.0^\circ$ are greater than 1.0 (Figure 8.61(b) and (c)).

For the aft-loaded blade, HS1C, it even appears that the three-dimensional flow effect on losses starts being significant at an incidence as low as $+5.0^\circ$ for the lowest Mach numbers investigated. This can be seen from Figure 8.23 which shows a comparison between the losses measured experimentally and those determined numerically at an incidence of $+5.0^\circ$ for the three blades.

As noted previously, that the losses for HS1C and HS1D are significantly higher

at an incidence of $+10.0^\circ$ is believed to be due to the presence of three-dimensional flow at midspan, as was the case for HS1A. However, the magnitude of the losses for HS1C are much higher than those of HS1A over the whole range of outlet Mach numbers. Losses for HS1D are slightly lower than those of HS1A at the lowest Mach number investigated. However, as the Mach number is increased, the loss performance of HS1D deteriorates rapidly relative to HS1A.

The higher losses for HS1C and HS1D at an incidence of $+10.0^\circ$ and for subsonic Mach numbers appear to be the result of greater three-dimensional flow effects for these two profiles. This conclusion can be reached by looking at the comparison between the numerical loss predictions and the losses measured experimentally in Figure 8.24. In this figure, it can be seen that the two-dimensional loss predictions for subsonic Mach numbers do not exhibit as large a variation between the three blades as the experimentally determined losses. Therefore, if the numerical predictions are accurate, one can conclude that the higher experimentally measured losses seen for blades HS1C and HS1D in Figure 8.24 are the result of higher midspan three-dimensional flow effects.

An objective of the current research was to look at the use of alternative loading distributions as a possible mean of reducing the losses for HP transonic turbine blades. The results presented previously in Chapter 7 and in the current chapter have shown that indeed a loss reduction could be achieved by modifying the loading distribution of the baseline mid-loaded blade. In the range of angle of attack varying between -10.0° and $+5.0^\circ$ at the design Mach number of 1.05, shifting the blade loading towards the rear of the airfoil (aft-loading) resulted in a noticeable reduction in losses. On the other hand, displacing the loading to the front of the airfoil (front-loading) did not result in significant improvements for these operating conditions.

At $+10.0^\circ$ of incidence and design Mach number, the picture is not as clear due to possibly the presence of secondary flow at midspan for blades HS1C and HS1D, as explain previously. Looking at the experimental results alone in Figure 8.24, one would be tempted to conclude that aft-loading and front-loading contributed to increase the losses significantly compared to the baseline mid-loaded blade. However, the two-dimensional computational results in that same figure seem to suggest that aft-loading and front-loading did not have a major effect on the losses for these con-

ditions.

At off-design Mach number, the loss performance of the aft-loaded (HS1C) and front-loaded (HS1D) airfoils is similar to that of the baseline mid-loaded airfoil (HS1A) for subsonic outlet Mach numbers if the three-dimensional flow effect discussed previously are neglected. The main shortcomings of the aft-loaded blade (HS1C) and the front-loaded blade (HS1D) begin to appear for outlet Mach numbers greater than about 1.10.

At high Mach numbers, the aft-loaded blade, HS1C, has a much higher rear suction peak Mach number, followed by a large region of diffusion at all values of incidence, compared to the baseline mid-loaded blade, HS1A. Reasons for the presence of these adverse pressure gradients were discussed previously in this chapter and Chapter 7. In order to improve the performance of HS1C at high Mach numbers, it is this rear suction surface flow deceleration that would have to be reduced. Reducing the rear suction surface curvature of HS1C would be one possible solution to reducing the flow acceleration and subsequent deceleration at this location.

Based on the observations made at design and off-design flow conditions, it appears that aft-loading can be a suitable design philosophy to follow in order to reduce the losses, provided the rearward deceleration is carefully limited.

The front-loaded blade, HS1D, also exhibits significant deceleration on the rear suction surface at high Mach number. This is in addition to the adverse pressure gradient that extends up to about an x/C_x of 0.6 for most flow conditions on the suction surface. The loss performance of the front-loaded blade, HS1D, is inferior or at best equal to that of the other two blades for all operating conditions. As such, it appears that there is no advantage in shifting the loading to the front of the airfoil for transonic high-pressure turbine blades. A similar conclusion was reached by Hoheisel et al. (1987) for their low-pressure turbine blades.

8.4 Influence of Reynolds Number on Performance

8.4.1 Profile Losses

As previously presented in Chapter 7 for design incidence, results will now be presented for the off-design performance at various Reynolds numbers. The measurements were performed for outlet flow Mach numbers of 0.60, 0.85, 0.95 and 1.05.

Figure 8.66 shows the effect of Reynolds number on the losses for the three blade profiles at an incidence of -10.0° . Similar graphs are presented in Figures 8.67, 8.68 and 8.69 for 0.0° , $+5.0^\circ$ and $+10.0^\circ$ of incidence.

As discussed previously in Chapter 7, the Reynolds number could only be lowered enough at a Mach number of 0.60 to be able to see its real effect on the losses.

Whereas the losses showed the lowest sensitivity with Reynolds number for the aft-loaded blade, HS1C, at design incidence (Figure 8.67), it appears that for an incidence of -10.0° the mid-loaded baseline blade is superior. This can be seen in Figure 8.66 at an outlet Mach number of 0.60.

It can also be observed from Figure 8.66 that the high Reynolds number losses for the aft-loaded blade, HS1C, are significantly lower than those of the baseline mid-loaded cascade, HS1A, at design Mach number of 1.05 and an incidence of -10.0° . This was not readily observable from Figure 8.1 possibly due to the uncertainty in the measurements.

At an incidence of $+5.0^\circ$ (Figure 8.68), the Reynolds number sensitivity of the baseline mid-loaded blade is much more significant than that of the other two profiles at an outlet Mach number of 0.60. In fact, for blades HS1C and HS1D the Reynolds number could not be lowered enough to see the losses increase.

An interesting point to note from Figure 8.68(b) is that the losses for blade HS1C at an outlet Mach number of 0.60 are higher than what is observed at higher Mach numbers for this blade. This is contrary to what was observed at lower incidence and for the other two profiles where the losses increase with Mach number. As explained previously, this is probably the result of the presence of three-dimensional flow effects at midspan for these conditions. Such trend is only observed at an incidence of $+10.0^\circ$

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.66: Reynolds Number Effect on Losses for Several Mach Numbers at an Incidence of -10.0° .

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.67: Reynolds Number Effect on Losses for Several Mach Numbers at Design Incidence.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.68: Reynolds Number Effect on Losses for Several Mach Numbers at an Incidence of $+5.0^\circ$.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.69: Reynolds Number Effect on Losses for Several Mach Numbers at an Incidence of $+10.0^\circ$.

for the other two profiles, as shown in Figure 8.69.

8.4.2 Outlet Flow Angle

The plots of outlet flow angle versus exit Reynolds number are shown in Figures 8.70, 8.71, 8.72 and 8.73. As for the losses, the effect of Reynolds number is seen mainly at an outlet flow Mach number of 0.60.

At an incidence of -10.0° for an outlet Mach number of 0.60, the outlet flow angle is not reduced significantly at the lowest Reynolds number investigated even though the losses increased significantly for these conditions. Therefore, it appears that under these flow conditions, the rise in losses is not associated with separation of the boundary layer at the trailing edge. The rise in losses might be the result of some localized regions of boundary layer separation on the blade surface. However, the Mach number distributions presented in the following section do not show evidence of such separations. If the increase in losses is not attributable to the presence of boundary layer separation, they must be due to attached flow effects. For example, the surface flow velocity for blade HS1A at an incidence of -10.0° and an outlet Mach number of 0.60 is higher on both the suction surface and the pressure at low Reynolds numbers. This can be clearly seen from Figure G.32(a) in Appendix G. As mentioned previously, the presence of higher surface flow velocity contributes to higher losses since the losses generated in the boundary layer are proportional to the velocity cubed. Increased loss generation in the boundary layers is reflected also in larger boundary layer displacement thickness and momentum thickness which both contribute to increase the trailing edge losses.

For a positive incidence of $+5.0^\circ$, it was observed earlier that the aft-loaded blade, HS1C, exhibited higher losses than the other two blades at higher values of Reynolds number at an outlet Mach number of 0.60. This is reflected in the outlet flow angles, with HS1C exhibiting somewhat lower flow turning for these conditions. In this case, therefore the higher losses do appear to be related to the presence of some separation in the trailing-edge region.

At an incidence of $+10.0^\circ$, the aft-loaded blade HS1C achieves the lowest flow turning at an outlet Mach number of 0.60, although all three airfoils are showing sig-

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.70: Reynolds Number Effect on Outlet Flow Angle for Several Mach Numbers at an Incidence of -10.0° .

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.71: Reynolds Number Effect on Outlet Flow Angle for Several Mach Numbers at Design Incidence.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.72: Reynolds Number Effect on Outlet Flow Angle for Several Mach Numbers at an Incidence of +5.0°.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure 8.73: Reynolds Number Effect on Outlet Flow Angle for Several Mach Numbers at an Incidence of +10.0°.

nificant deviation at this condition. This suggests that increasingly extensive trailing-edge separation is present for all three blades.

8.4.3 Blade Loading

Loading measurements were performed for all test conditions of the Reynolds number study. That is, for the three cascades at outlet Mach numbers of 0.60, 0.85, 0.95 and 1.05 and for incidences of -10.0° , 0.0° , $+5.0^\circ$ and $+10.0^\circ$. However, given the limited range of Reynolds numbers that could be achieved, the variation in the loading distribution due to the changes in Reynolds numbers was also somewhat limited. Consequently, the results are not discussed further here. The results are presented in Appendix G for completeness.

Figures G.32, G.33, G.34 and G.35 show the blade loading distributions at an incidence of -10.0° and outlet Mach numbers of 0.60, 0.85, 0.95, 1.05. For each Mach number and incidence, the loading distributions are presented for two or three outlet Reynolds numbers.

Figures G.36, G.37, G.38 and G.39 present the blade loading distributions at design incidence. These results were previously presented in Chapter 7 but are repeated here to facilitate comparisons.

Figures G.40, G.41, G.42 and G.43 present the blade loading distributions for $+5.0^\circ$ and Figures G.44, G.45, G.46 and G.47 those for $+10.0^\circ$.

Chapter 9

Conclusions and Recommendations

9.1 Conclusions

Detailed experimental and numerical results were presented for the midspan aerodynamic performance of a family of three transonic HP turbine blades at both design and off-design incidences. The three profiles used consist of the baseline mid-loaded blade, HS1A, taken from an actual engine together with two derivatives, an aft-loaded profile, HS1C, and a front-loaded profile, HS1D, obtained from the baseline profile.

The modified geometries were intended to permit the study of the effect of the loading distribution on the performance of transonic turbine blades at both design and off-design conditions.

The data cover a broad range of incidences, exit Mach numbers and Reynolds numbers. The incidence was varied from -10.0° to $+10.0^\circ$. The exit Mach number ranged from 0.5 to 1.3. Outlet Reynolds numbers as low as 2.0×10^5 based on the blade chord were achieved at low Mach numbers. For the highest Mach numbers investigated, Reynolds numbers of the order of 12.0×10^5 were obtained. In addition to helping improve our understanding of HP turbine blade aerodynamics, these data provide a substantial contribution to the database on the performance of transonic turbine blades available in the open literature. Hence, they can ultimately be used to develop improved loss correlations.

The results presented in this thesis have shown that a loss reduction could be achieved by modifying the loading distribution of the baseline mid-loaded blade. At design incidence and the design Mach number of 1.05, shifting the blade loading towards the rear of the airfoil (aft-loading) resulted in a noticeable reduction in losses.

Based on the analysis of wake traverse data, blade loading measurements and base pressure measurements combined with the numerical results, it was found that the better loss performance of the aft-loaded profile compared to the baseline mid-loaded blade at design incidence and Mach number could be traced to the latter's higher rear suction side curvature. The presence of higher rear suction surface curvature resulted in higher flow velocity in that region. Higher flow velocity at the trailing edge apparently contributed to reducing the base pressure. The lower base pressure at the trailing edge resulted in a stronger trailing edge shock system for the mid-loaded blade. This shock system increased the losses for the mid-loaded baseline profile when compared to the aft-loaded profile.

At design incidence and Mach number, front-loading did not result in any significant change in losses. For all test conditions, the performance of the front-loaded blade was at best equal to that of at least one of the other two profiles.

Reducing the outlet Mach number below the design value at design incidence did not reveal any advantage in terms of reduced losses for any of the three profiles. For Mach numbers greater than about 1.10 the baseline mid-loaded blade, HS1A, has a definite advantage over the other two profiles in terms of losses. This is explained by the rapid rise in base pressure for blade HS1A at high Mach numbers that results in lower trailing edge shocks.

At off-design incidence, the superior loss performance of the aft-loaded airfoil HS1C at design Mach number can be observed from incidence varying from about -5.0° to $+5.0^\circ$. The better loss performance of the aft-loaded blade at mild positive incidences appears to be due to the fact that it has a much less severe overspeed region on the suction surface close to the leading edge when compared to the other two profiles. This results in a milder pressure gradient and lower boundary layer losses.

At off-design Mach number and incidence, the loss performance of the aft-loaded (HS1C) and front-loaded (HS1D) airfoils is similar to that of the baseline mid-loaded airfoil (HS1A) for subsonic outlet Mach numbers, omitting those cases where three-dimensional flow effects are believed to be present. The main weakness of the aft-loaded blade (HS1C) and the front-loaded blade (HS1D), as they are currently designed, is for outlet Mach numbers greater than about 1.10. For all off-design

incidences investigated, the baseline mid-loaded blade, HS1A, had the lowest losses at high outlet Mach numbers. This is similar to what was observed at design incidence.

At high Mach number, the aft-loaded blade, HS1C, has a much higher rear suction peak Mach number, which is followed by a large region of diffusion at all values of incidence investigated compared to the baseline mid-loaded blade, HS1A. In order to improve the performance of HS1C at high Mach numbers, this rear suction surface flow deceleration would have to be reduced. Reducing the rear suction surface curvature of HS1C would be one possible solution.

Based on the observations made at design and off-design flow conditions, it appears that aft-loading can be a viable design philosophy to employ in order to reduce the losses within a blade row provided the rearward deceleration is carefully limited.

The front-loaded blade, HS1D, also exhibits significant deceleration on the rear suction surface at high Mach number. This is in addition to the adverse pressure gradient that extends up to about $x/C_x \simeq 0.6$ on the suction surface for most flow conditions. The loss performance of the front-loaded blade, HS1D, is inferior or at best equal to that of the other two blades for all operating conditions. As such, it appears that there is no advantage in shifting the loading to the front of the airfoil for transonic high-pressure turbine blade.

The effect of Reynolds number on the performance of the three airfoils was examined for outlet Mach numbers of 0.60, 0.85, 0.95 and 1.05. Data were collected at incidences varying between -10.0° and $+10.0^\circ$. However, for most cases, the Reynolds number could only be lowered enough to affect the losses for the lowest outlet Mach number studied. Therefore, the conclusions that can be drawn from the Reynolds number study are limited. Interestingly, the data at an incidence of $+5.0^\circ$ indicate that at low outlet Mach number, the sensitivity of the aft-loaded blade, HS1C, to low Reynolds number might be lower compared with the baseline mid-loaded blade, HS1A. However, this should be investigated further as results published by other researchers indicates that an aft-loaded blade should not be advantageous at low Reynolds numbers.

In order to help in the interpretation of the results, simulations were performed with the commercial CFD code FLUENT. Comparisons of computationally predicted

losses with those determined experimentally showed that the predictions generally agree well with the measurements. Specifically, the general trends in the variation of losses with Mach number are predicted satisfactorily. As an example, the rise in losses due to the appearance of shock waves around a Mach number of about 0.85 is predicted well for most cases. Furthermore, the magnitude of the predicted losses is generally in line with those measured experimentally. Comparisons of measured and computed blade surface Mach number distributions showed that the predictions were fairly good. The presence of impinging shocks on the blades' suction surface was usually captured very well.

Based on the comparisons made in the previous chapters, it appears that state-of-art CFD codes are now capable of predicting reasonably well the flow through transonic turbine cascades operating at high Reynolds numbers and at off-design conditions. Areas where improvements might still be needed in terms of prediction capabilities are for cascade flows at low Reynolds numbers where the flow is transitional over the airfoils and where severely separated flow might be present.

9.2 Recommendations

The Pratt & Whitney Canada high speed wind tunnel is a fairly new facility. As such, there exists room for improvement of the facility and the procedures used for the acquisition of the data.

As discussed in Chapter 6, the two-dimensionality of the flow is of the utmost importance in cascade work. At design incidence, it was shown that a region of two-dimensional flow existed around the midspan region of the blades. Furthermore, measurements of Axial Velocity Density Ratio (AVDR) indicate that it is essentially equal to 1.0 at design incidence for the cascades used for this project. However, at $+12.0^\circ$ of incidence, it was shown that there was essentially no region of two-dimensional flow at midspan and that the AVDR was as high as 1.1. Furthermore, the presence of three-dimensional flow at the midspan section of the cascades was shown to increase the losses significantly at $+10.0^\circ$ of incidence.

In order to improve the two-dimensionality of the flow at the cascade midspan section at high positive incidence two possible solutions could be implemented.

First, the use of blades having higher aspect ratio (H/C) should improve the two-dimensionality of the flow at incidence. For example, based on Sieverding's recommendations, the cascades used for this project should have an aspect ratio of at least 1.8 whereas 1.53 was used.

The second solution would consist of blowing off the endwall boundary layer upstream of the cascade. The inlet endwall boundary is the source of secondary flow in cascade and therefore reducing the thickness of this boundary layer should contribute to the improvement of the two-dimensionality of the flow. Two boundary layer blowoff slots are already available upstream of the cascade test section and it has been demonstrated that the blowoff significantly thins the incoming endwall boundary layer.

Implementation of a flow visualization technique would be highly desirable in the Pratt & Whitney Canada high speed wind tunnel. One of the cascade endwalls consists of clear Lexan window. Therefore, visual access to the cascade is already incorporated in the test section. For transonic wind tunnels, the shadowgraph and Schlieren techniques can be helpful in the interpretation of the results. This would allow the determination of the shock wave patterns as well as the location of shock boundary layer interactions.

The current project includes a study of the effect of Reynolds number on the performance of transonic HP turbine blades. As shown in Chapters 7 and 8, the Reynolds number can have a significant impact on the performance of turbine blade, especially for the low values encountered in smaller flight engines at cruise altitude. The very low Reynolds numbers that represents actual engine flight conditions can not be achieved with the current facility. Therefore, the effect of low Reynolds number on the performance of the three blade profiles could not be fully investigated. It would provide extremely interesting data if a study could be performed on the three cascades at relevant Reynolds numbers. In order to achieve this the ejector-diffuser system would have to be modified to improve its efficiency. An alternative would be to connect an exhaustor to the outlet of the diffuser section of the wind tunnel in order to lower the static pressure at the outlet of the cascade test section to values low enough to achieve the desired Reynolds numbers. As an alternative, two of the tanks could perhaps be converted to vacuum tanks, which would then be connected

to the test section outlet, to allow operation at low static pressures throughout the system.

A revision of the off-design loss prediction correlations such as that of Benner et al. (1997) should be performed in order to include the compressibility effects. The large body of data gathered during this study provides a good basis for such a revision.

Finally, an attempt should be made to devise a satisfactory diffusion factor to predict diffusion limits on turbine blades. The fact that this study generated data for blades having very different loading characteristics provides a good database of cases to initiate the development of such a correlation.

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Appendix A

HS1A, HS1B and HS1D Blade Coordinates

The coordinates in these tables correspond exactly to those of the actual engine blade section. In order to get the blade coordinates for one of the test cascades, the data in these tables must be divided by the axial chord C_x of the engine blade (last x-coordinate of a given table minus first x-coordinate of that table) and multiplied by the cascade axial chord of 37.3mm.

Suction Surface		Pressure Surface	
x (inches)	y (inches)	x (inches)	y (inches)
-0.26634	-0.01086	-0.26634	-0.01086
-0.26339	0.00050	-0.26339	-0.02130
-0.26044	0.00748	-0.26044	-0.02504
-0.25749	0.01379	-0.25749	-0.02746
-0.25454	0.01958	-0.25454	-0.02910
-0.25159	0.02494	-0.25159	-0.03015
-0.23684	0.04702	-0.23684	-0.02846
-0.22209	0.06397	-0.22209	-0.02021
-0.20734	0.07760	-0.20734	-0.01345
-0.19259	0.08881	-0.19259	-0.00798
-0.17784	0.09809	-0.17784	-0.00365
-0.16309	0.10576	-0.16309	-0.00030
-0.14834	0.11203	-0.14834	0.00208
-0.13359	0.11702	-0.13359	0.00362
-0.11884	0.12082	-0.11884	0.00434
-0.10409	0.12351	-0.10409	0.00428
-0.08934	0.12510	-0.08934	0.00346
-0.07459	0.12563	-0.07459	0.00190
-0.05984	0.12510	-0.05984	-0.00040
-0.04509	0.12350	-0.04509	-0.00336
-0.03034	0.12080	-0.03034	-0.00705
-0.01559	0.11697	-0.01559	-0.01142
-0.00080	0.11194	-0.00080	-0.01649
0.01391	0.10565	0.01391	-0.02224
0.02866	0.09800	0.02866	-0.02869
0.04341	0.08886	0.04341	-0.03583
0.05816	0.07810	0.05816	-0.04367

Table A.1: Blade HS1A Coordinates.

Suction Surface		Pressure Surface	
x (inches)	y (inches)	x (inches)	y (inches)
0.07291	0.06559	0.07291	-0.05222
0.08766	0.05121	0.08766	-0.06147
0.10241	0.03500	0.10241	-0.07145
0.11716	0.01724	0.11716	-0.08216
0.13191	-0.00154	0.13191	-0.09361
0.14666	-0.02097	0.14666	-0.10582
0.16141	-0.04084	0.16141	-0.11881
0.17617	-0.06099	0.17617	-0.13260
0.19091	-0.08128	0.19091	-0.14720
0.20566	-0.10169	0.20566	-0.16263
0.22042	-0.12223	0.22042	-0.17891
0.23516	-0.14294	0.23516	-0.19606
0.24991	-0.16394	0.24991	-0.21407
0.26466	-0.18538	0.26466	-0.23296
0.27941	-0.23059	0.27941	-0.25276
0.27941	-0.20749	0.27941	-0.25276
0.29416	-0.23059	0.29416	-0.27346
0.30006	-0.24020	0.30006	-0.28200
0.30301	-0.24510	0.30301	-0.28633
0.30596	-0.25007	0.30596	-0.29060
0.30891	-0.25512	0.30891	-0.29302
0.31187	-0.26026	0.31187	-0.29406
0.31481	-0.26546	0.31481	-0.29416
0.31776	-0.27078	0.31776	-0.29334
0.32071	-0.27620	0.32071	-0.29131
0.32366	-0.28422	0.32366	-0.28422

Table A.2: Blade HS1A Coordinates (Continued).

Suction Surface		Pressure Surface	
x (inches)	y (inches)	x (inches)	y (inches)
-0.29070	0.02913	-0.29070	0.02913
-0.28979	0.03483	-0.28979	0.02204
-0.28770	0.03868	-0.28770	0.01528
-0.28347	0.04158	-0.28347	0.00654
-0.27805	0.04234	-0.27805	-0.00158
-0.27172	0.04120	-0.27172	-0.00949
-0.26509	0.03838	-0.26509	-0.01750
-0.25756	0.03365	-0.25756	-0.02617
-0.24551	0.02639	-0.24551	-0.03907
-0.23045	0.01872	-0.23045	-0.05364
-0.21538	0.01243	-0.21538	-0.06656
-0.20032	0.00737	-0.20032	-0.07794
-0.18526	0.00341	-0.18526	-0.08792
-0.17019	0.00045	-0.17019	-0.09661
-0.15513	-0.00159	-0.15513	-0.10406
-0.14007	-0.00277	-0.14007	-0.11034
-0.12501	-0.00314	-0.12501	-0.11546
-0.10995	-0.00274	-0.10995	-0.11946
-0.09488	-0.00160	-0.09488	-0.12236
-0.07982	0.00027	-0.07982	-0.12415
-0.06475	0.00284	-0.06475	-0.12483
-0.04969	0.00611	-0.04969	-0.12440
-0.03463	0.01006	-0.03463	-0.12281
-0.01957	0.01469	-0.01957	-0.12002
-0.00450	0.01998	-0.00450	-0.11596
0.01056	0.02592	0.01056	-0.11052
0.02562	0.03250	0.02562	-0.10358

Table A.3: Blade HS1C Coordinates.

Suction Surface		Pressure Surface	
x (inches)	y (inches)	x (inches)	y (inches)
0.04069	0.03970	0.04069	-0.09500
0.05575	0.04752	0.05575	-0.08471
0.07081	0.05595	0.07081	-0.07268
0.08588	0.06499	0.08588	-0.05898
0.10094	0.07464	0.10094	-0.04387
0.11600	0.08491	0.11600	-0.02771
0.13106	0.09581	0.13106	-0.01062
0.14613	0.10736	0.14613	0.00731
0.16119	0.11961	0.16119	0.02599
0.17626	0.13259	0.17626	0.04538
0.19132	0.14636	0.19132	0.06546
0.20638	0.16098	0.20638	0.08634
0.22144	0.17652	0.22144	0.10807
0.23651	0.19307	0.23651	0.13072
0.25157	0.21071	0.25157	0.15431
0.26663	0.22956	0.26663	0.17890
0.27748	0.24395	0.27748	0.19723
0.28682	0.25694	0.28682	0.21344
0.29465	0.26808	0.29465	0.22736
0.30068	0.27120	0.30068	0.23826
0.30429	0.27099	0.30429	0.24489
0.30730	0.26969	0.30730	0.25046
0.30972	0.26749	0.30972	0.25496
0.31122	0.26480	0.31122	0.25797
0.31182	0.26137	0.31182	0.26137

Table A.4: Blade HS1C Coordinates (Continued).

Suction Surface		Pressure Surface	
x (inches)	y (inches)	x (inches)	y (inches)
-0.27410	-0.01921	-0.27410	-0.01921
-0.27320	-0.01326	-0.27320	-0.02513
-0.27109	-0.00866	-0.27109	-0.03065
-0.26686	-0.00382	-0.26686	-0.04019
-0.26144	-0.00061	-0.26144	-0.05070
-0.25511	0.00075	-0.25511	-0.06120
-0.24848	0.00004	-0.24848	-0.07074
-0.24094	-0.00377	-0.24094	-0.08017
-0.22888	-0.00978	-0.22888	-0.09294
-0.21381	-0.01528	-0.21381	-0.10583
-0.19874	-0.01893	-0.19874	-0.11613
-0.18366	-0.02103	-0.18366	-0.12432
-0.16859	-0.02186	-0.16859	-0.13073
-0.15351	-0.02160	-0.15351	-0.13556
-0.13844	-0.02041	-0.13844	-0.13896
-0.12337	-0.01839	-0.12337	-0.14101
-0.10830	-0.01561	-0.10830	-0.14175
-0.09322	-0.01209	-0.09322	-0.14121
-0.07814	-0.00788	-0.07814	-0.13939
-0.06307	-0.00299	-0.06307	-0.13625
-0.04800	0.00254	-0.04800	-0.13178
-0.03292	0.00870	-0.03292	-0.12595
-0.01785	0.01547	-0.01785	-0.11875
-0.00278	0.02283	-0.00278	-0.11022
0.01230	0.03076	0.01230	-0.10038
0.02737	0.03927	0.02737	-0.08929
0.04244	0.04834	0.04244	-0.07697

Table A.5: Blade HS1D Coordinates.

Suction Surface		Pressure Surface	
x (inches)	y (inches)	x (inches)	y (inches)
0.05752	0.05796	0.05752	-0.06346
0.07259	0.06814	0.07259	-0.04879
0.08767	0.07886	0.08767	-0.03304
0.10274	0.09014	0.10274	-0.01636
0.11781	0.10197	0.11781	0.00108
0.13289	0.11436	0.13289	0.01910
0.14796	0.12733	0.14796	0.03761
0.16303	0.14089	0.16303	0.05653
0.17811	0.15505	0.17811	0.07579
0.19318	0.16986	0.19318	0.09543
0.20826	0.18534	0.20826	0.11551
0.22333	0.20153	0.22333	0.13612
0.23840	0.21847	0.23840	0.15737
0.25348	0.23622	0.25348	0.17935
0.26855	0.25485	0.26855	0.20216
0.28362	0.27443	0.28362	0.22591
0.29448	0.28916	0.29448	0.24368
0.30382	0.30232	0.30382	0.25945
0.31166	0.31352	0.31166	0.27308
0.31770	0.31664	0.31770	0.28384
0.32131	0.31643	0.32131	0.29039
0.32432	0.31514	0.32432	0.29593
0.32673	0.31295	0.32673	0.30041
0.32824	0.31025	0.32824	0.30344
0.32884	0.30683	0.32884	0.30683

Table A.6: Blade HS1D Coordinates (Continued).

Appendix B

Cascade HS1C Drawings

GENERAL NOTES

1. The tolerance on the dimensions in mm is (+/-) 0.1 mm.
2. Wherever indicated the symbol (p) designates a tolerance of (+/-) 0.0005.
3. The tolerance on the location of holes is (+/-) 0.0005.
4. The tolerance on the blade coordinates is (+/-) 0.0005.
5. (en) designates that this tolerance is also for the pressure and suction surfaces.

TITLE	BASIC CASCADE BLADE		SCALE	N/A	QUANTITY	11	DRAWN BY	D. CORRIEVEAU	DATE	31/11/01
PROJECT	CASCADE TEST SECTION		DRAWING NO.	HS1C3	MATERIAL	ALUMINUM	CHECKED BY		DATE	
AUTOCAD FILENAME	C:\DANIEL\DRAWING\HS1C3.DWG		DIMENSIONS	DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED			APPROVED BY		DATE	

Figure B.1: Blade HS1C Overall Dimensions.

GENERAL NOTES

1. POSITIVE ROTATION ANGLE IS IN THE CCW DIRECTION
2. NEGATIVE ROTATION ANGLE IS IN THE CW DIRECTION
3. THE PRESSURE TAP HOLES TO BE DRILLED ON THE SUCTION SURFACE ARE 0.010" IN DIA.
4. THE HOLES ON THE TOP SURFACE TO BE DRILLED FOR THE HYPERDERMIC TUBING ARE 1 MM IN DIA.
5. THE TOLERANCE ON THE DIMENSIONS IN INCHES IS (+/-) 0.0005".
6. THE TOLERANCE ON THE DIMENSIONS IN METRIC IS (+/-) 0.1 MM.

HOLE #	POSITION OF THE HOLES ON THE TOP SURFACE (IN TUBING)			LOCATION OF THE PRESSURE TAP HOLES ON THE SUCTION SURFACE		
	X (")	Y (")	DEPTH (mm)	ANGLE OF HOLES OF HOLES (°)	Y (")	Z (mm)
1	0.2039	0.4854	19	70.92	0.3513	18
2	0.1584	0.4454	23	50.51	0.4055	22
3	0.1107	0.3998	21	47.45	0.3950	20
4	0.0518	0.3403	19	41.80	0.2882	18
5	-0.0025	0.2689	23	34.53	0.2201	22
6	-0.0466	0.1972	21	26.89	0.1548	20
7	-0.0794	0.1216	19	18.39	0.0904	18
8	-0.0975	0.0379	23	8.44	0.0231	22
9	-0.1037	-0.0391	21	-1.34	-0.0367	20
10	-0.0932	-0.1125	19	-11.44	-0.0928	18
11	-0.0721	-0.1844	23	-21.72	-0.1416	22
12	-0.0416	-0.2486	21	-31.22	-0.1911	20
13	0.0022	-0.3085	19	-39.15	-0.2406	18
14	0.0522	-0.3639	23	-44.49	-0.2962	22
15	0.1039	-0.4160	21	-47.32	-0.3584	20
16	0.1560	-0.4631	19	-49.29	-0.4203	18
17	0.2169	-0.5060	23	-50.91	-0.4874	22
18	0.2699	-0.5540	21	-52.18	-0.5529	20
19	0.3372	-0.6045	19	-53.58	-0.6302	18
20	0.4104	-0.6567	23	-55.15	-0.7121	22
21	0.4871	-0.7088	21	-56.71	-0.7962	20
22	0.5465	-0.7568	19	-57.93	-0.8649	18
23	0.6121	-0.8033	23	-59.13	-0.9376	22
24	0.6790	-0.8484	21	-60.25	-1.0105	20
25	0.7491	-0.8928	19	-61.40	-1.0891	18

TOP SURFACE

TITLE		SCALE		QUANTITY		DRAWN BY		DATE	
HASIC CASCADE BLADE		N/A		1		D. CORRIVEAU		11/27/01	
PROJECT	CASCADE TEST SECTION	DRAWING NO.		MATERIAL		CHECKED BY		DATE	
		H51C1PSS		ALUMINUM					
AUTOCAD FILENAME		DIMENSIONS		DIMENSIONS IN INCHES		APPROVED BY		DATE	
C:\DRAVING\H51C1PSS.DWG		UNLESS OTHERWISE SPECIFIED							

Figure B.2: Blade HS1C Suction Surface Static Taps Location.

- GENERAL NOTES**
1. POSITIVE ROTATION ANGLE IS IN THE CCW DIRECTION.
  2. NEGATIVE ROTATION ANGLE IS IN THE CW DIRECTION.
  3. THE PRESSURE TAP HOLES TO BE DRILLED ON THE SUCTION SURFACE ARE 0.010" IN DIA EXCEPTED HOLE 17, 0.020" IN DIA.
  4. THE HOLES ON THE TOP SURFACE TO BE DRILLED FOR THE HYPERDERMIC TUBING ARE 1 mm IN DIA.
  5. THE TOLERANCE ON THE DIMENSIONS IN METRIC IS (+/-) 0.1 mm.
  6. THE TOLERANCE ON THE DIMENSIONS IN INCHES IS (+/-) 0.005.

HOLE #	POSITION OF THE HOLES ON THE TOP SURFACE OR TUBING			POSITION OF THE PRESSURE TAP HOLES ON THE SUCTION SURFACE		
	X (")	Y (")	DEPTH (mm)	ANGLE OF HOLES OF HOLES	Y (")	Z (mm)
1	0.1831	0.4595	19	31.44	0.4876	18
2	0.1376	0.3766	21	22.15	0.4006	20
3	0.1094	0.2798	23	18.82	0.2971	22
4	0.0943	0.1925	19	5.36	0.2005	18
5	0.0924	0.0887	21	-2.54	0.0845	20
6	0.1083	-0.0113	23	-9.51	-0.0290	22
7	0.1261	-0.1098	19	-13.79	-0.1399	18
8	0.1561	-0.2012	21	-21.14	-0.2439	20
9	0.1925	-0.2885	23	-25.69	-0.3434	22
10	0.2415	-0.3769	19	-29.97	-0.4471	18
11	0.2878	-0.4533	21	-33.35	-0.5369	20
12	0.3475	-0.5371	23	-36.89	-0.6381	22
13	0.4083	-0.6141	19	-40.20	-0.7356	18
14	0.4798	-0.6930	21	-43.74	-0.8324	20
15	0.5672	-0.7664	23	-47.45	-0.9362	22
16	0.6536	-0.8292	19	-50.80	-1.0395	18
17	0.7484	-0.8928	21	32.73	-0.3470	20

TITLE	HASIC CASCADE BLADE	SCALE	N/A	QUANTITY	1	DRAWN BY	D. CORRIE	DATE	02/27/02
PROJECT	CASCADE TEST SECTION	DRAWING NO.	HSICTAPPS	MATERIAL	ALUMINUM	CHECKED BY		DATE	
AUTOCAD FILENAME	C:\DANIEL\DRAWING\HSICTAPPS-C.DWG	DIMENSIONS	DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED	APPROVED BY		DATE			

Figure B.3: Blade HS1C Pressure Surface Static Taps Location.

GENERAL NOTES

1. The tolerance on the dimensions in mm is (+/-) 0.1 mm.
2. Wherever indicated the symbol (s) designates a tolerance of (+/-) 0.0005'.
3. The tolerance of location are in inches.
4. The tolerance on the blade coordinates is (+/-) 0.0005'.
5. (as) designates that this tolerance is also for the pressure and suction surfaces.

TITLE		HASIC CASCADE BLADE			SCALE	N/A	QUANTITY	1	DRAWN BY	D. CORRIVEAU	DATE	11/27/01
PROJECT		CASCADE TEST SECTION			DRAWING NO.	HA5C1APB.LASS	MATERIAL	ALUMINUM	CHECKED BY		DATE	
AUTOCAD FILENAME		C:\DRAWING\HA5C1APB.LASS.DWG			DIMENSIONS IN INCHES		UNLESS OTHERWISE SPECIFIED		APPROVED BY		DATE	

Figure B.4: Blade HS1C Suction Surface Static Taps Location.

- GENERAL NOTES**
1. The tolerance on the dimensions in mm is (1/-) 0.1 mm.
  2. Wherever indicated the symbol (s) designates a tolerance of (1/-) 0.0005".
  3. The tolerance on the location are in inches.
  4. The tolerance on the blade coordinates is (1/-) 0.0005".
  5. (ss) designates that this tolerance is also for the pressure and suction surfaces.

TITLE	BASIC CASCADE BLADE		SCALE	N/A	QUANTITY	1	DRAWN BY	D. CORRIVEAU	DATE	02/18/02
PROJECT	CASCADE TEST SECTION		DRAWING NO.	HS1C/BLAPS-B	MATERIAL	ALUMINUM	CHECKED BY		DATE	
AUTOCAD FILENAME	C:\DANIEL\DRAWING\HS1C/BLAPS-B.DWG		DIMENSIONS	DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED			APPROVED BY		DATE	

Figure B.5: Blade HS1C Pressure Surface Static Taps Location.

Figure B.6: Blade HSIIC Pressure Surface Side Block.

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Figure B.7: Blade HS1C Suction Surface Side Block.

Figure B.8: Cascade HS1C Lexan Window.

Figure B.9: Cascade HS1C Lexan Window.

Figure B.10: Cascade HS1C Lexan Window.

NOTE:
DRILL 43 HOLES 0.052" IN DIA. EACH. THE LOCATION OF THE HOLES ARE
TABULATED BELOW FOR BOTH THE PRESSURE AND THE SUCTION
SURFACES.

HOLE #	SUCTION SURFACE		PRESSURE SURFACE	
	X (")	Y (")	X (")	Y (")
1	-0.3488	1.1411	-0.3829	-1.1748
2	-0.3088	1.0937	-0.2400	-1.2203
3	-0.2632	1.0480	-0.1432	-1.2485
4	-0.2037	0.9890	-0.0559	-1.2636
5	-0.1323	0.9347	0.0479	-1.2655
6	-0.0606	0.8907	0.1479	-1.2496
7	0.0150	0.8578	0.2463	-1.2318
8	0.0987	0.8397	0.3378	-1.2018
9	0.1757	0.8335	0.4251	-1.1654
10	0.2501	0.8441	0.5135	-1.1164
11	0.3210	0.8652	0.5899	-1.0701
12	0.3852	0.8957	0.6736	-1.0104
13	0.4451	0.9394	0.7506	-0.9495
14	0.5005	0.9895	0.8295	-0.8781
15	0.5526	1.0411	0.9030	-0.7906
16	0.5997	1.0932	0.9658	-0.7043
17	0.6426	1.1542	1.0184	-0.6292
18	0.6906	1.2071	1.0454	-0.5840
19	0.7411	1.2744		
20	0.7933	1.3476		
21	0.8453	1.4243		
22	0.8934	1.4837		
23	0.9399	1.5494		
24	0.9850	1.6162		
25	1.0294	1.6864		

TITLE		SCALE		QUANTITY		DRAWN BY		DATE	
LEXAN WINDOW		N/A		1		D. CORREVEAU		02/18/02	
PROJECT		DRAWING NO.		MATERIAL		CHECKED BY		DATE	
AUTOCAD FILENAME		HS1C/IN4-B		LEXAN		APPROVED BY		DATE	
C:\DANIEL\DRAWING\HS1C\IN4-B.DWG		DIMENSIONS		DIMENSIONS IN INCHES					
		UNLESS OTHERWISE SPECIFIED							

Figure B.11: Cascade HS1C Lexan Window.

Appendix C

Cascade HS1D Drawings

TITLE		H4SID CASCADE BLADE				SCALE	N/A	QUANTITY	11	DRAWN BY	D. CORRIYEAU	DATE	12/06/01
PROJECT		CASCADE TEST SECTION				DRAWING NO.	H5ID2	MATERIAL	ALUMINUM	CHECKED BY		DATE	
AUTOCAD FILENAME		C:\DANIEL\DRAWING\H5ID2.DWG		DIMENSIONS		DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED							
						APPROVED BY							
						DATE							

GENERAL NOTES

1. POSITIVE ROTATION ANGLE IS IN THE CCW DIRECTION.
2. NEGATIVE ROTATION ANGLE IS IN THE CW DIRECTION.
3. THE PRESSURE TAP HOLES TO BE DRILLED ON THE SUCTION SURFACE ARE 0.010" IN DIA.
4. THE HOLES ON THE TOP SURFACE TO BE DRILLED FOR THE HYPERDERMIC TUBING ARE 1 mm IN DIA.
5. THE TOLERANCE ON THE DIMENSIONS IN INCHES IS (+/-) 0.0005".
6. THE TOLERANCE ON THE DIMENSIONS IN METRIC IS (+/-) 0.1 mm.

HOLE #	POSITION OF THE HOLES ON THE TOP SURFACE (IN. TUBING)			LOCATION OF THE PRESSURE TAP HOLES ON THE SUCTION SURFACE		
	X (")	Y (")	DEPTH (mm)	ANGLE OF HOLES (°)	TOOL DIA (mm)	Z (mm)
1	0.1038	0.4054	19	60.26	0.2762	18
2	0.0468	0.4454	23	56.90	0.2702	22
3	-0.0113	0.3998	21	48.04	0.2589	20
4	-0.0636	0.3403	19	36.03	0.2378	18
5	-0.1041	0.2689	23	23.43	0.2053	22
6	-0.1271	0.1972	21	11.98	0.1665	20
7	-0.1398	0.1216	19	0.44	0.1206	18
8	-0.1273	0.0379	23	-12.10	0.0637	22
9	-0.1021	-0.0391	21	-23.05	0.0040	20
10	-0.0636	-0.1135	19	-31.85	-0.0627	18
11	-0.0195	-0.1844	23	-38.42	-0.1361	22
12	0.0416	-0.2486	21	-43.14	-0.2098	20
13	0.1006	-0.3085	19	-46.62	-0.2850	18
14	0.1638	-0.3639	23	-48.94	-0.3626	22
15	0.2253	-0.4160	21	-50.26	-0.4392	20
16	0.2751	-0.4631	19	-51.13	-0.5048	18
17	0.3359	-0.5060	23	-51.78	-0.5769	22
18	0.3975	-0.5540	21	-52.50	-0.6526	20
19	0.4631	-0.6045	19	-53.34	-0.7324	18
20	0.5310	-0.6567	23	-54.45	-0.8139	22
21	0.5960	-0.7088	21	-55.61	-0.8922	20
22	0.6599	-0.7568	19	-56.89	-0.9663	18
23	0.7220	-0.8033	23	-58.09	-1.0375	22
24	0.7958	-0.8484	21	-59.54	-1.1160	20
25	0.8645	-0.8928	19	-146.96	0.2764	18

TOP SURFACE

HYPERDERMIC TUBING TO BE GLUED ONE IN EACH HOLE (1 mm DIA) 25 TUBES IN TOTAL

TITLE	HASID CASCADE BLADE		SCALE	N/A	QUANTITY	1	DRAWN BY	D. CORRIVEAU	DATE	12/08/01
PROJECT	CASCADE TEST SECTION		DRAWING NO.	HS1DAPSS	MATERIAL	ALUMINUM	CHECKED BY		DATE	
AUTOCAD FILENAME	C:\DANIEL\DRAWING\HS1DAPSS.DWG		DIMENSIONS	DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED			APPROVED BY		DATE	

Figure C.2: Blade HS1D Suction Surface Static Taps Location.

Figure C.3: Blade HSID Pressure Surface Static Taps Location.

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Figure C.4: Blade HS1D Suction Surface Static Taps Location.

Figure C.5: Blade HS1D Pressure Surface Static Taps Location.

GENERAL NOTES

1. THE TOLERANCE ON THE BLADE COORDINATES IS (+/-) 0.0005".
2. WHEREVER INDICATED THE (+) DESIGNATES A TOLERANCE OF (+/-) 0.0005".
3. THE TOLERANCE ON THE DIMENSIONS IN METRIC IS (+/-) 0.1 mm.
4. ALL THE TOLERANCES OF LOCATION ARE IN METRIC.
5. THIS DIMENSION IS USED FOR REFERENCE ONLY.

TITLE		BASE PLATE LEFT END MOUNTING BLOCK			SCALE	N/A	QUANTITY	1	DRAWN BY	D. CORRIVEAU	DATE	12/06/2001
PROJECT		CASCADE TEST SECTION			DRAWING NO.	HS1DPS	MATERIAL	ALUMINUM	CHECKED BY		DATE	
AUTOCAD FILENAME		C:\DANIEL\DRAWING\HS1D\BPS.DWG		DIMENSIONS	DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED			APPROVED BY		DATE		

Figure C.6: Blade HS1D Pressure Surface Side Block.

Figure C.7: Blade HS1D Suction Surface Side Block.

Figure C.8: Cascade HS1D Lexan Window.

Figure C.9: Cascade HS1D Lexan Window.

Figure C.10: Cascade HS1D Lexan Window.

NOTE:
DRILL 42 HOLES 0.050" IN DIA. EACH. THE LOCATION OF THE HOLES ARE
TABULATED BELOW FOR BOTH THE PRESSURE AND THE SUCTION
SURFACES.

SUCTION SURFACE			PRESSURE SURFACE		
HOLE #	X (")	Y (")	X (")	Y (")	
1	-0.3488	1.0411	-0.3229	-1.2579	
2	-0.3088	0.9841	-0.2400	-1.2899	
3	-0.2632	0.9259	-0.1432	-1.3023	
4	-0.2037	0.8736	-0.0559	-1.2992	
5	-0.1323	0.8332	0.0479	-1.2829	
6	-0.0606	0.8102	0.1479	-1.2496	
7	0.0150	0.8021	0.2463	-1.2136	
8	0.0987	0.8100	0.3378	-1.1678	
9	0.1757	0.8351	0.4251	-1.1140	
10	0.2501	0.8735	0.5135	-1.0538	
11	0.3210	0.9237	0.5899	-0.9965	
12	0.3852	0.9788	0.6736	-0.9265	
13	0.4451	1.0378	0.7506	-0.8561	
14	0.5005	1.1011	0.8295	-0.7760	
15	0.5526	1.1626	0.9030	-0.6846	
16	0.5997	1.2123	0.9658	-0.5911	
17	0.6426	1.2731	1.0294	-0.4947	
18	0.6906	1.3347			
19	0.7411	1.4003			
20	0.7933	1.4683			
21	0.8453	1.5333			
22	0.8934	1.5971			
23	0.9399	1.6593			
24	0.9850	1.7330			
25	1.0294	1.8017			

TITLE		SCALE		QUANTITY		DRAWN BY		DATE	
LEXAN WINDOW		N/A		1		D. CORRIVEAU		12/12/01	
PROJECT		DRAWING NO.		MATERIAL		CHECKED BY		DATE	
AUTOCAD FILENAME		HS1DWIN4		DIMENSIONS IN INCHES		APPROVED BY		DATE	
C:\DANIEL\DRAWING\HS1DWIN4.DWG		DIMENSIONS UNLESS OTHERWISE SPECIFIED							

Figure C.11: Cascade HS1D Lexan Window.

Appendix D

HS1A Cascade Results

M_{2MO}	M_1	$Re_2 (\times 10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.496	0.317	465	0.98	0.042	56.4	0.051	0.048
0.698	0.390	648	1.00	0.031	56.9	0.045	0.040
0.843	0.420	778	0.99	0.032	57.5	0.052	0.044
0.974	0.435	882	0.98	0.043	57.4	0.082	0.066
1.086	0.434	954	0.98	0.043	57.2	0.093	0.071
1.165	0.431	1026	0.97	0.039	57.3	0.093	0.068
1.282	0.430	1083	0.98	0.039	55.6	0.109	0.075

Table D.1: HS1A Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = -10.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.484	-0.091	0.834	0.852
0.686	-0.068	0.706	0.730
0.812	-0.099	0.610	0.649
0.925	-0.137	0.514	0.575
1.026	-0.376	0.357	0.512
1.111	-0.259	0.346	0.462
1.209	-0.259	0.286	0.407

Table D.2: HS1A Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = -10.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^3)$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.60	0.36	379	0.98	56.4	0.072	0.013
0.60	0.60	0.36	432	0.99	56.7	0.053	0.010
0.60	0.59	0.36	477	0.98	56.5	0.058	0.011
0.60	0.59	0.36	529	0.99	56.6	0.054	0.010
0.60	0.60	0.36	550	0.99	56.7	0.047	0.009
0.60	0.60	0.36	565	1.00	56.7	0.047	0.009
0.60	0.59	0.35	595	0.99	56.8	0.052	0.010
0.60	0.60	0.35	623	1.00	56.7	0.044	0.008
0.60	0.60	0.36	667	0.99	56.8	0.051	0.010
0.60	0.60	0.35	757	0.99	56.8	0.046	0.009
0.60	0.60	0.35	787	1.00	56.7	0.046	0.009
0.85	0.85	0.43	663	0.98	57.6	0.059	0.018
0.85	0.84	0.42	713	0.99	57.5	0.052	0.016
0.85	0.89	0.42	789	1.00	57.5	0.050	0.016
0.85	0.84	0.42	869	0.98	57.6	0.053	0.016
0.85	0.85	0.42	950	1.00	57.6	0.045	0.014
0.85	0.85	0.42	1057	1.01	56.6	0.039	0.012
0.95	0.95	0.44	744	0.97	57.7	0.069	0.024
0.95	0.95	0.43	818	0.98	57.7	0.068	0.023
0.95	0.95	0.43	849	0.98	57.8	0.067	0.023
0.95	0.95	0.43	907	0.98	57.7	0.063	0.022
0.95	0.94	0.43	1036	0.99	57.7	0.055	0.019
0.95	0.96	0.43	1119	0.99	57.6	0.064	0.022
1.05	1.04	0.43	819	0.98	57.2	0.098	0.036
1.05	1.04	0.44	897	0.98	57.3	0.100	0.037
1.05	1.06	0.43	925	0.98	57.3	0.096	0.036
1.05	1.05	0.43	1015	0.99	57.2	0.092	0.035
1.05	1.05	0.43	1146	0.99	57.2	0.091	0.034

Table D.3: HS1A Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = -10.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.372	-0.301	0.066	0.313	0.067
0.050	0.461	-0.954	0.113	0.213	0.573
0.087	0.514	-1.382	0.157	0.201	0.621
0.136	0.555	-1.727	0.235	0.196	0.640
0.190	0.583	-1.970	0.283	0.199	0.627
0.252	0.604	-2.166	0.340	0.207	0.598
0.310	0.623	-2.334	0.393	0.218	0.551
0.371	0.642	-2.508	0.445	0.227	0.511
0.434	0.662	-2.691	0.504	0.238	0.463
0.493	0.673	-2.792	0.553	0.252	0.394
0.545	0.669	-2.759	0.601	0.269	0.312
0.581	0.655	-2.626	0.645	0.283	0.235
0.624	0.630	-2.395	0.695	0.305	0.114
0.662	0.579	-1.942	0.736	0.324	0.003
0.699	0.555	-1.732	0.785	0.354	-0.183
0.736	0.539	-1.589	0.828	0.376	-0.331
0.774	0.534	-1.546	0.862	0.386	-0.397
0.812	0.531	-1.527	0.901	0.426	-0.683
0.847	0.533	-1.543	0.941	0.464	-0.979
0.882	0.540	-1.601			
0.917	0.544	-1.634			
0.953	0.545	-1.639			
1.000	0.516	-1.401			

Table D.4: HS1A Cascade Blade Loading at $M_2 = 0.48$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.455	-0.272	0.066	0.410	-0.041
0.050	0.582	-0.993	0.113	0.276	0.528
0.087	0.662	-1.486	0.157	0.265	0.569
0.136	0.720	-1.851	0.235	0.258	0.592
0.190	0.762	-2.123	0.283	0.259	0.588
0.252	0.798	-2.353	0.340	0.262	0.580
0.310	0.832	-2.574	0.393	0.271	0.547
0.371	0.862	-2.764	0.445	0.282	0.510
0.434	0.899	-3.006	0.504	0.298	0.447
0.493	0.936	-3.236	0.553	0.316	0.379
0.545	0.946	-3.299	0.601	0.338	0.290
0.581	0.925	-3.170	0.645	0.358	0.200
0.624	0.890	-2.946	0.695	0.390	0.056
0.662	0.826	-2.536	0.736	0.417	-0.075
0.699	0.769	-2.168	0.785	0.465	-0.322
0.736	0.744	-2.009	0.828	0.497	-0.500
0.774	0.743	-1.998	0.862	0.512	-0.581
0.812	0.743	-2.000	0.901	0.576	-0.954
0.847	0.749	-2.039	0.941	0.637	-1.326
0.882	0.762	-2.125			
0.917	0.774	-2.202			
0.953	0.781	-2.243			
1.000	0.723	-1.871			

Table D.5: HS1A Cascade Blade Loading at $M_2 = 0.69$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.481	-0.257	0.066	0.443	-0.072
0.050	0.623	-0.998	0.113	0.294	0.530
0.087	0.712	-1.502	0.157	0.277	0.586
0.136	0.779	-1.887	0.235	0.261	0.635
0.190	0.829	-2.174	0.283	0.266	0.621
0.252	0.869	-2.406	0.340	0.267	0.617
0.310	0.908	-2.629	0.393	0.281	0.574
0.371	0.942	-2.824	0.445	0.290	0.543
0.434	1.000	-3.144	0.504	0.311	0.469
0.493	1.068	-3.508	0.553	0.338	0.369
0.545	1.111	-3.733	0.601	0.360	0.286
0.581	1.098	-3.666	0.645	0.385	0.182
0.624	1.054	-3.434	0.695	0.420	0.030
0.662	1.021	-3.257	0.736	0.453	-0.119
0.699	0.908	-2.630	0.785	0.511	-0.402
0.736	0.862	-2.370	0.828	0.553	-0.621
0.774	0.867	-2.394	0.862	0.572	-0.719
0.812	0.876	-2.447	0.901	0.651	-1.154
0.847	0.894	-2.553	0.941	0.734	-1.629
0.882	0.926	-2.732			
0.917	0.955	-2.895			
0.953	0.976	-3.013			
1.000	0.872	-2.422			

Table D.6: HS1A Cascade Blade Loading at $M_2 = 0.81$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.497	-0.281	0.066	0.475	-0.174
0.050	0.637	-0.995	0.113	0.317	0.474
0.087	0.732	-1.513	0.157	0.298	0.538
0.136	0.805	-1.923	0.235	0.283	0.586
0.190	0.855	-2.198	0.283	0.280	0.597
0.252	0.897	-2.432	0.340	0.282	0.590
0.310	0.941	-2.670	0.393	0.306	0.511
0.371	0.982	-2.887	0.445	0.332	0.422
0.434	1.042	-3.205	0.504	0.350	0.353
0.493	1.121	-3.600	0.553	0.370	0.278
0.545	1.191	-3.934	0.601	0.386	0.212
0.581	1.216	-4.050	0.645	0.405	0.135
0.624	1.166	-3.818	0.695	0.438	-0.008
0.662	1.144	-3.710	0.736	0.468	-0.143
0.699	1.087	-3.431	0.785	0.529	-0.434
0.736	0.951	-2.725	0.828	0.566	-0.620
0.774	0.934	-2.633	0.862	0.590	-0.745
0.812	0.955	-2.744	0.901	0.698	-1.325
0.847	0.985	-2.907	0.941	0.796	-1.868
0.882	1.030	-3.139			
0.917	1.085	-3.420			
0.953	1.157	-3.774			
1.000	1.023	-3.103			

Table D.7: HS1A Cascade Blade Loading at $M_2 = 0.93$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.506	-0.295	0.066	0.316	0.489
0.050	0.649	-1.019	0.113	0.308	0.516
0.087	0.744	-1.532	0.157	0.292	0.566
0.136	0.814	-1.915	0.235	0.288	0.579
0.190	0.865	-2.192	0.283	0.293	0.563
0.252	0.905	-2.409	0.340	0.303	0.533
0.310	0.945	-2.621	0.393	0.310	0.508
0.371	0.985	-2.830	0.445	0.320	0.473
0.434	1.052	-3.173	0.504	0.337	0.414
0.493	1.130	-3.556	0.553	0.359	0.333
0.545	1.204	-3.899	0.601	0.386	0.232
0.581	1.245	-4.078	0.645	0.409	0.136
0.624	1.238	-4.049	0.695	0.446	-0.022
0.662	1.203	-3.894	0.736	0.482	-0.181
0.699	1.178	-3.779	0.785	0.545	-0.486
0.736	1.189	-3.829	0.828	0.586	-0.690
0.774	1.213	-3.940	0.862	0.606	-0.794
0.812	1.089	-3.357	0.901	0.701	-1.298
0.847	1.024	-3.034	0.941	0.804	-1.860
0.882	1.047	-3.149			
0.917	1.100	-3.412			
0.953	1.179	-3.784			
1.000	1.307	-4.342			

Table D.8: HS1A Cascade Blade Loading at $M_2 = 1.03$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.497	-0.282	0.066	0.457	-0.094
0.050	0.642	-1.025	0.113	0.306	0.509
0.087	0.735	-1.539	0.157	0.291	0.560
0.136	0.813	-1.969	0.235	0.280	0.593
0.190	0.860	-2.232	0.283	0.282	0.589
0.252	0.904	-2.476	0.340	0.290	0.561
0.310	0.949	-2.721	0.393	0.300	0.528
0.371	0.984	-2.908	0.445	0.310	0.497
0.434	1.047	-3.240	0.504	0.333	0.416
0.493	1.124	-3.626	0.553	0.357	0.327
0.545	1.199	-3.982	0.601	0.383	0.225
0.581	1.242	-4.177	0.645	0.410	0.111
0.624	1.236	-4.148	0.695	0.444	-0.036
0.662	1.199	-3.982	0.736	0.475	-0.180
0.699	1.173	-3.861	0.785	0.539	-0.490
0.736	1.190	-3.940	0.828	0.578	-0.687
0.774	1.256	-4.237	0.862	0.625	-0.935
0.812	1.340	-4.590	0.901	0.699	-1.334
0.847	1.346	-4.614	0.941	0.803	-1.913
0.882	1.216	-4.060			
0.917	1.154	-3.772			
0.953	1.201	-3.988			
1.000	1.331	-4.556			

Table D.9: HS1A Cascade Blade Loading at $M_2 = 1.11$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.488	-0.231	0.066	0.437	0.001
0.050	0.636	-0.980	0.113	0.295	0.551
0.087	0.734	-1.515	0.157	0.281	0.594
0.136	0.806	-1.914	0.235	0.273	0.618
0.190	0.861	-2.216	0.283	0.276	0.611
0.252	0.904	-2.454	0.340	0.280	0.599
0.310	0.946	-2.680	0.393	0.291	0.562
0.371	0.982	-2.871	0.445	0.303	0.525
0.434	1.044	-3.192	0.504	0.324	0.451
0.493	1.123	-3.590	0.553	0.353	0.347
0.545	1.194	-3.927	0.601	0.374	0.266
0.581	1.240	-4.130	0.645	0.395	0.181
0.624	1.235	-4.111	0.695	0.432	0.022
0.662	1.194	-3.924	0.736	0.468	-0.137
0.699	1.168	-3.804	0.785	0.536	-0.460
0.736	1.185	-3.881	0.828	0.576	-0.667
0.774	1.251	-4.181	0.862	0.623	-0.910
0.812	1.339	-4.548	0.901	0.693	-1.285
0.847	1.423	-4.872	0.941	0.798	-1.869
0.882	1.443	-4.943			
0.917	1.268	-4.255			
0.953	1.338	-4.546			
1.000	1.467	-5.031			

Table D.10: HS1A Cascade Blade Loading at $M_2 = 1.21$, and $i - i_{des} = -10.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.570	0.399	511	1.01	0.049	56.3	0.064	0.060
0.749	0.478	669	1.00	0.038	56.9	0.057	0.050
0.834	0.490	770	1.01	0.031	57.2	0.051	0.043
0.964	0.512	870	0.99	0.040	57.6	0.075	0.060
1.059	0.511	954	1.00	0.045	57.0	0.094	0.072
1.172	0.512	1020	0.98	0.040	57.0	0.096	0.070
1.277	0.512	1070	0.99	0.044	55.0	0.121	0.084

Table D.11: HS1A Cascade Performance Data at Design Incidence, $i - i_{des} = 0.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.506	0.080	0.840	0.840
0.693	-0.002	0.714	0.726
0.834	-0.017	0.619	0.634
0.962	0.008	0.540	0.552
1.017	-0.312	0.389	0.518
1.066	-0.292	0.361	0.488
1.170	-0.127	0.358	0.407

Table D.12: HS1A Cascade Base Pressure Data at Design Incidence, $i - i_{des} = 0.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^{-3})$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.59	0.40	320	1.04	54.5	0.108	0.019
0.60	0.60	0.40	370	1.03	55.4	0.088	0.016
0.60	0.59	0.40	425	1.02	56.1	0.066	0.012
0.60	0.60	0.41	456	1.01	56.0	0.074	0.014
0.60	0.59	0.41	497	1.01	56.2	0.069	0.012
0.60	0.61	0.42	568	1.01	56.1	0.054	0.010
0.60	0.60	0.42	571	1.00	56.1	0.059	0.011
0.60	0.59	0.41	604	1.02	56.2	0.046	0.009
0.60	0.60	0.42	674	1.00	56.3	0.056	0.010
0.60	0.60	0.41	825	1.01	56.2	0.056	0.010
0.60	0.60	0.41	876	1.01	56.1	0.058	0.011
0.60	0.60	0.41	904	1.01	56.0	0.054	0.010
0.85	0.85	0.52	614	0.98	56.8	0.077	0.023
0.85	0.85	0.51	685	0.98	57.0	0.071	0.022
0.85	0.85	0.50	741	1.00	56.9	0.063	0.019
0.85	0.85	0.50	778	1.00	57.1	0.057	0.017
0.85	0.85	0.50	929	1.00	57.1	0.059	0.018
0.85	0.85	0.50	1001	1.00	57.1	0.054	0.017
0.85	0.85	0.50	1092	1.00	57.1	0.056	0.017
0.95	0.95	0.53	709	0.97	57.3	0.075	0.025
0.95	0.95	0.52	777	0.99	57.4	0.068	0.023
0.95	0.95	0.51	833	0.99	57.4	0.063	0.021
0.95	0.95	0.52	862	0.99	57.5	0.071	0.024
0.95	0.95	0.52	908	0.99	57.5	0.065	0.022
0.95	0.95	0.51	1037	0.99	57.4	0.063	0.021
0.95	0.96	0.52	1110	0.99	57.3	0.069	0.023
1.05	1.05	0.52	786	0.99	56.7	0.102	0.038
1.05	1.06	0.52	862	0.99	56.8	0.096	0.036
1.05	1.05	0.52	924	0.99	56.7	0.109	0.041
1.05	1.05	0.52	977	0.99	56.7	0.103	0.039
1.05	1.05	0.52	1050	0.99	56.8	0.102	0.038
1.05	1.05	0.52	1198	0.99	56.8	0.101	0.038

Table D.13: HS1A Cascade Performance Data at Design Incidence at Design Incidence, $i - i_{des} = 0.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.508	-0.861	0.066	0.206	0.692
0.050	0.575	-1.326	0.113	0.191	0.740
0.087	0.611	-1.579	0.157	0.181	0.770
0.136	0.627	-1.698	0.235	0.187	0.752
0.190	0.633	-1.742	0.283	0.207	0.690
0.252	0.635	-1.758	0.340	0.191	0.740
0.310	0.637	-1.774	0.393	0.200	0.710
0.371	0.648	-1.850	0.445	0.210	0.679
0.434	0.657	-1.919	0.504	0.220	0.645
0.493	0.658	-1.924	0.553	0.239	0.577
0.545	0.658	-1.929	0.601	0.250	0.535
0.581	0.645	-1.828	0.645	0.269	0.460
0.624	0.610	-1.571	0.695	0.290	0.368
0.662	0.571	-1.294	0.736	0.310	0.278
0.699	0.548	-1.133	0.785	0.337	0.146
0.736	0.529	-1.002	0.828	0.363	0.015
0.774	0.524	-0.968	0.862	0.382	-0.092
0.812	0.524	-0.966	0.901	0.409	-0.246
0.847	0.522	-0.956	0.941	0.444	-0.449
0.882	0.528	-0.998			
0.917	0.532	-1.022			
0.953	0.531	-1.018			
1.000	0.505	-0.841			

Table D.14: HS1A Cascade Blade Loading at $M_2 = 0.51$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.653	-0.863	0.066	0.266	0.684
0.050	0.758	-1.384	0.113	0.237	0.760
0.087	0.817	-1.681	0.157	0.231	0.775
0.136	0.841	-1.802	0.235	0.241	0.750
0.190	0.844	-1.815	0.283	0.261	0.699
0.252	0.844	-1.815	0.340	0.253	0.720
0.310	0.848	-1.838	0.393	0.262	0.696
0.371	0.862	-1.908	0.445	0.269	0.676
0.434	0.887	-2.031	0.504	0.286	0.628
0.493	0.903	-2.110	0.553	0.308	0.563
0.545	0.918	-2.185	0.601	0.321	0.524
0.581	0.900	-2.093	0.645	0.340	0.461
0.624	0.864	-1.918	0.695	0.375	0.344
0.662	0.806	-1.624	0.736	0.401	0.247
0.699	0.759	-1.391	0.785	0.440	0.095
0.736	0.733	-1.262	0.828	0.482	-0.077
0.774	0.730	-1.244	0.862	0.514	-0.218
0.812	0.729	-1.240	0.901	0.558	-0.413
0.847	0.735	-1.272	0.941	0.607	-0.643
0.882	0.748	-1.335			
0.917	0.757	-1.380			
0.953	0.758	-1.383			
1.000	0.710	-1.147			

Table D.15: HS1A Cascade Blade Loading at $M_2 = 0.69$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.706	-0.834	0.066	0.293	0.676
0.050	0.837	-1.406	0.113	0.252	0.775
0.087	0.914	-1.739	0.157	0.246	0.788
0.136	0.940	-1.848	0.235	0.254	0.771
0.190	0.940	-1.851	0.283	0.276	0.719
0.252	0.937	-1.837	0.340	0.268	0.737
0.310	0.942	-1.859	0.393	0.283	0.703
0.371	0.960	-1.933	0.445	0.298	0.665
0.434	1.005	-2.124	0.504	0.315	0.618
0.493	1.046	-2.288	0.553	0.336	0.560
0.545	1.089	-2.460	0.601	0.361	0.486
0.581	1.073	-2.398	0.645	0.382	0.421
0.624	1.026	-2.207	0.695	0.421	0.295
0.662	0.988	-2.052	0.736	0.452	0.186
0.699	0.908	-1.715	0.785	0.495	0.030
0.736	0.867	-1.538	0.828	0.546	-0.165
0.774	0.867	-1.535	0.862	0.586	-0.326
0.812	0.873	-1.562	0.901	0.640	-0.551
0.847	0.887	-1.623	0.941	0.707	-0.837
0.882	0.915	-1.744			
0.917	0.935	-1.827			
0.953	0.943	-1.862			
1.000	0.857	-1.492			

Table D.16: HS1A Cascade Blade Loading at $M_2 = 0.83$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.731	-0.826	0.066	0.321	0.634
0.050	0.868	-1.393	0.113	0.279	0.737
0.087	0.955	-1.743	0.157	0.271	0.753
0.136	0.982	-1.850	0.235	0.275	0.746
0.190	0.981	-1.846	0.283	0.293	0.703
0.252	0.977	-1.831	0.340	0.281	0.732
0.310	0.986	-1.868	0.393	0.298	0.691
0.371	1.003	-1.932	0.445	0.314	0.650
0.434	1.050	-2.114	0.504	0.338	0.587
0.493	1.112	-2.346	0.553	0.364	0.514
0.545	1.180	-2.588	0.601	0.388	0.445
0.581	1.187	-2.612	0.645	0.411	0.374
0.624	1.132	-2.417	0.695	0.446	0.262
0.662	1.113	-2.350	0.736	0.479	0.149
0.699	1.033	-2.050	0.785	0.524	-0.011
0.736	0.959	-1.760	0.828	0.575	-0.203
0.774	0.946	-1.708	0.862	0.623	-0.388
0.812	0.963	-1.775	0.901	0.692	-0.666
0.847	0.995	-1.903	0.941	0.773	-1.001
0.882	1.036	-2.061			
0.917	1.086	-2.248			
0.953	1.143	-2.456			
1.000	0.981	-1.847			

Table D.17: HS1A Cascade Blade Loading at $M_2 = 0.96$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.734	-0.833	0.066	0.309	0.664
0.050	0.873	-1.404	0.113	0.270	0.757
0.087	0.955	-1.736	0.157	0.266	0.767
0.136	0.985	-1.854	0.235	0.272	0.753
0.190	0.985	-1.852	0.283	0.294	0.702
0.252	0.977	-1.823	0.340	0.284	0.725
0.310	0.987	-1.862	0.393	0.302	0.682
0.371	1.007	-1.938	0.445	0.314	0.651
0.434	1.059	-2.138	0.504	0.330	0.611
0.493	1.128	-2.392	0.553	0.353	0.548
0.545	1.202	-2.651	0.601	0.381	0.467
0.581	1.236	-2.763	0.645	0.409	0.384
0.624	1.206	-2.663	0.695	0.448	0.258
0.662	1.169	-2.537	0.736	0.479	0.151
0.699	1.152	-2.476	0.785	0.523	-0.004
0.736	1.156	-2.493	0.828	0.576	-0.204
0.774	1.080	-2.217	0.862	0.625	-0.392
0.812	0.995	-1.890	0.901	0.696	-0.678
0.847	1.020	-1.988	0.941	0.797	-1.094
0.882	1.061	-2.144			
0.917	1.123	-2.375			
0.953	1.204	-2.658			
1.000	1.245	-2.793			

Table D.18: HS1A Cascade Blade Loading at $M_2 = 1.02$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.737	-0.835	0.066	0.311	0.662
0.050	0.876	-1.406	0.113	0.273	0.753
0.087	0.965	-1.763	0.157	0.264	0.773
0.136	0.988	-1.851	0.235	0.271	0.758
0.190	0.987	-1.848	0.283	0.293	0.706
0.252	0.978	-1.812	0.340	0.283	0.730
0.310	0.983	-1.833	0.393	0.301	0.688
0.371	1.003	-1.910	0.445	0.315	0.653
0.434	1.058	-2.121	0.504	0.333	0.604
0.493	1.127	-2.374	0.553	0.357	0.539
0.545	1.200	-2.628	0.601	0.381	0.471
0.581	1.239	-2.757	0.645	0.408	0.389
0.624	1.212	-2.668	0.695	0.446	0.269
0.662	1.172	-2.534	0.736	0.481	0.150
0.699	1.154	-2.467	0.785	0.529	-0.021
0.736	1.173	-2.534	0.828	0.582	-0.219
0.774	1.221	-2.697	0.862	0.617	-0.355
0.812	1.074	-2.182	0.901	0.689	-0.640
0.847	1.038	-2.043	0.941	0.810	-1.135
0.882	1.065	-2.148			
0.917	1.132	-2.391			
0.953	1.219	-2.693			
1.000	1.300	-2.951			

Table D.19: HS1A Cascade Blade Loading at $M_2 = 1.07$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.737	-0.860	0.066	0.303	0.675
0.050	0.876	-1.434	0.113	0.258	0.780
0.087	0.962	-1.785	0.157	0.247	0.804
0.136	0.984	-1.873	0.235	0.254	0.790
0.190	0.982	-1.865	0.283	0.275	0.742
0.252	0.973	-1.829	0.340	0.269	0.757
0.310	0.977	-1.845	0.393	0.285	0.719
0.371	0.999	-1.932	0.445	0.298	0.689
0.434	1.052	-2.137	0.504	0.322	0.627
0.493	1.122	-2.399	0.553	0.346	0.562
0.545	1.196	-2.660	0.601	0.362	0.517
0.581	1.234	-2.789	0.645	0.389	0.439
0.624	1.214	-2.721	0.695	0.433	0.301
0.662	1.170	-2.568	0.736	0.472	0.168
0.699	1.148	-2.492	0.785	0.525	-0.020
0.736	1.171	-2.571	0.828	0.577	-0.218
0.774	1.239	-2.805	0.862	0.625	-0.404
0.812	1.323	-3.071	0.901	0.692	-0.674
0.847	1.409	-3.324	0.941	0.815	-1.182
0.882	1.384	-3.253			
0.917	1.256	-2.862			
0.953	1.280	-2.937			
1.000	1.307	-3.023			

Table D.20: HS1A Cascade Blade Loading at $M_2 = 1.17$, and $i - i_{des} = 0.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.491	0.392	454	1.06	0.089	54.3	0.089	0.084
0.693	0.515	643	1.05	0.054	55.0	0.078	0.069
0.840	0.575	778	1.04	0.043	55.7	0.072	0.061
0.971	0.595	884	1.02	0.042	56.2	0.079	0.063
1.078	0.595	954	1.02	0.050	55.9	0.109	0.083
1.176	0.592	1034	1.00	0.043	56.0	0.103	0.075
1.283	0.592	1082	1.00	0.045	54.7	0.125	0.087

Table D.21: HS1A Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = +5.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.474	-0.026	0.844	0.857
0.669	-0.007	0.726	0.741
0.814	-0.003	0.632	0.647
0.920	-0.018	0.559	0.579
1.034	-0.150	0.432	0.507
1.075	-0.224	0.379	0.483
1.184	-0.135	0.346	0.421

Table D.22: HS1A Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = +5.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^{-3})$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.59	0.46	313	1.01	53.8	0.225	0.037
0.60	0.60	0.47	373	1.03	53.5	0.170	0.030
0.60	0.60	0.46	487	1.06	54.3	0.096	0.017
0.60	0.61	0.47	561	1.04	54.7	0.090	0.017
0.60	0.60	0.47	585	1.04	54.7	0.083	0.015
0.60	0.60	0.47	684	1.04	54.7	0.086	0.016
0.60	0.60	0.46	815	1.06	54.4	0.084	0.015
0.60	0.60	0.47	835	1.05	54.3	0.088	0.016
0.85	0.85	0.58	606	1.04	54.9	0.093	0.027
0.85	0.85	0.58	679	1.04	55.3	0.083	0.025
0.85	0.85	0.58	747	1.05	55.3	0.075	0.023
0.85	0.85	0.58	767	1.03	55.4	0.074	0.021
0.85	0.85	0.57	949	1.04	55.5	0.070	0.021
0.85	0.85	0.58	1000	1.04	55.5	0.074	0.022
0.85	0.85	0.57	1070	1.04	55.5	0.076	0.022
0.85	0.85	0.57	1123	1.02	55.4	0.076	0.021
0.95	0.95	0.60	710	1.04	55.6	0.086	0.029
0.95	0.95	0.59	779	1.03	55.8	0.076	0.026
0.95	0.95	0.59	843	1.04	55.8	0.069	0.024
0.95	0.95	0.59	890	1.03	55.9	0.077	0.026
0.95	0.95	0.59	960	1.04	55.9	0.069	0.024
0.95	0.95	0.59	1081	1.04	55.9	0.069	0.024
0.95	0.95	0.59	1146	1.04	55.9	0.070	0.024
1.05	1.05	0.60	857	1.03	55.6	0.107	0.040
1.05	1.05	0.60	934	1.02	55.6	0.115	0.043
1.05	1.05	0.60	977	1.03	55.6	0.110	0.042
1.05	1.05	0.60	1057	1.03	55.6	0.111	0.041
1.05	1.05	0.59	1213	1.03	55.5	0.106	0.040

Table D.23: HS1A Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = +5.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.608	-1.203	0.066	0.167	0.844
0.050	0.649	-1.462	0.113	0.166	0.847
0.087	0.665	-1.566	0.157	0.171	0.836
0.136	0.662	-1.543	0.235	0.181	0.810
0.190	0.653	-1.490	0.283	0.184	0.802
0.252	0.642	-1.417	0.340	0.192	0.782
0.310	0.636	-1.380	0.393	0.206	0.743
0.371	0.639	-1.396	0.445	0.211	0.731
0.434	0.644	-1.430	0.504	0.228	0.680
0.493	0.647	-1.447	0.553	0.244	0.628
0.545	0.637	-1.387	0.601	0.256	0.589
0.581	0.618	-1.267	0.645	0.269	0.541
0.624	0.581	-1.033	0.695	0.293	0.454
0.662	0.554	-0.871	0.736	0.307	0.396
0.699	0.534	-0.748	0.785	0.332	0.292
0.736	0.517	-0.654	0.828	0.360	0.169
0.774	0.512	-0.622	0.862	0.378	0.089
0.812	0.509	-0.607	0.901	0.404	-0.041
0.847	0.510	-0.612	0.941	0.443	-0.240
0.882	0.512	-0.623			
0.917	0.514	-0.636			
0.953	0.515	-0.641			
1.000	0.498	-0.544			

Table D.24: HS1A Cascade Blade Loading at $M_2 = 0.47$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.815	-1.184	0.066	0.211	0.875
0.050	0.899	-1.530	0.113	0.210	0.876
0.087	0.942	-1.702	0.157	0.210	0.876
0.136	0.924	-1.630	0.235	0.227	0.845
0.190	0.895	-1.511	0.283	0.228	0.843
0.252	0.867	-1.398	0.340	0.241	0.816
0.310	0.854	-1.345	0.393	0.252	0.795
0.371	0.858	-1.362	0.445	0.266	0.763
0.434	0.879	-1.448	0.504	0.281	0.730
0.493	0.897	-1.521	0.553	0.301	0.682
0.545	0.898	-1.524	0.601	0.316	0.643
0.581	0.873	-1.421	0.645	0.333	0.598
0.624	0.808	-1.155	0.695	0.366	0.507
0.662	0.768	-0.988	0.736	0.392	0.430
0.699	0.738	-0.865	0.785	0.430	0.308
0.736	0.715	-0.768	0.828	0.470	0.175
0.774	0.713	-0.759	0.862	0.502	0.064
0.812	0.710	-0.747	0.901	0.541	-0.081
0.847	0.713	-0.760	0.941	0.604	-0.321
0.882	0.718	-0.782			
0.917	0.723	-0.802			
0.953	0.723	-0.801			
1.000	0.692	-0.673			

Table D.25: HS1A Cascade Blade Loading at $M_2 = 0.67$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.917	-1.144	0.066	0.253	0.856
0.050	1.049	-1.582	0.113	0.246	0.868
0.087	1.113	-1.780	0.157	0.246	0.869
0.136	1.081	-1.682	0.235	0.258	0.847
0.190	1.032	-1.525	0.283	0.262	0.840
0.252	0.971	-1.325	0.340	0.273	0.818
0.310	0.956	-1.275	0.393	0.290	0.785
0.371	0.966	-1.308	0.445	0.303	0.759
0.434	1.008	-1.447	0.504	0.321	0.721
0.493	1.057	-1.607	0.553	0.346	0.664
0.545	1.088	-1.702	0.601	0.368	0.612
0.581	1.064	-1.627	0.645	0.389	0.561
0.624	0.978	-1.350	0.695	0.425	0.466
0.662	0.932	-1.196	0.736	0.452	0.391
0.699	0.895	-1.068	0.785	0.496	0.264
0.736	0.865	-0.966	0.828	0.541	0.126
0.774	0.863	-0.959	0.862	0.582	-0.004
0.812	0.862	-0.958	0.901	0.636	-0.180
0.847	0.872	-0.991	0.941	0.715	-0.447
0.882	0.888	-1.047			
0.917	0.899	-1.085			
0.953	0.896	-1.074			
1.000	0.837	-0.871			

Table D.26: HS1A Cascade Blade Loading at $M_2 = 0.81$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.947	-1.129	0.066	0.259	0.862
0.050	1.109	-1.627	0.113	0.252	0.874
0.087	1.199	-1.882	0.157	0.251	0.876
0.136	1.151	-1.749	0.235	0.268	0.846
0.190	1.095	-1.588	0.283	0.272	0.839
0.252	0.987	-1.257	0.340	0.276	0.831
0.310	0.991	-1.269	0.393	0.297	0.793
0.371	1.008	-1.323	0.445	0.306	0.774
0.434	1.054	-1.465	0.504	0.327	0.731
0.493	1.127	-1.680	0.553	0.350	0.682
0.545	1.188	-1.852	0.601	0.375	0.625
0.581	1.208	-1.909	0.645	0.401	0.562
0.624	1.113	-1.639	0.695	0.439	0.466
0.662	1.036	-1.411	0.736	0.474	0.371
0.699	0.997	-1.288	0.785	0.520	0.241
0.736	0.966	-1.190	0.828	0.571	0.087
0.774	0.964	-1.184	0.862	0.616	-0.051
0.812	0.973	-1.212	0.901	0.683	-0.268
0.847	0.991	-1.271	0.941	0.776	-0.572
0.882	1.027	-1.383			
0.917	1.063	-1.492			
0.953	1.079	-1.539			
1.000	0.951	-1.145			

Table D.27: HS1A Cascade Blade Loading at $M_2 = 0.92$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.952	-1.105	0.066	0.268	0.853
0.050	1.120	-1.608	0.113	0.266	0.856
0.087	1.225	-1.897	0.157	0.266	0.856
0.136	1.175	-1.762	0.235	0.272	0.846
0.190	1.102	-1.558	0.283	0.276	0.839
0.252	0.991	-1.225	0.340	0.287	0.818
0.310	1.001	-1.257	0.393	0.301	0.793
0.371	1.018	-1.307	0.445	0.314	0.767
0.434	1.070	-1.462	0.504	0.330	0.734
0.493	1.138	-1.660	0.553	0.354	0.682
0.545	1.205	-1.844	0.601	0.381	0.620
0.581	1.244	-1.945	0.645	0.409	0.555
0.624	1.226	-1.899	0.695	0.450	0.449
0.662	1.166	-1.736	0.736	0.484	0.358
0.699	1.131	-1.640	0.785	0.532	0.223
0.736	1.111	-1.582	0.828	0.588	0.056
0.774	1.041	-1.378	0.862	0.632	-0.078
0.812	1.010	-1.284	0.901	0.699	-0.290
0.847	1.033	-1.354	0.941	0.802	-0.624
0.882	1.075	-1.478			
0.917	1.131	-1.639			
0.953	1.204	-1.839			
1.000	1.164	-1.731			

Table D.28: HS1A Cascade Blade Loading at $M_2 = 1.03$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.954	-1.113	0.066	0.256	0.874
0.050	1.123	-1.617	0.113	0.247	0.888
0.087	1.226	-1.900	0.157	0.248	0.887
0.136	1.175	-1.763	0.235	0.257	0.871
0.190	1.100	-1.553	0.283	0.261	0.866
0.252	0.990	-1.222	0.340	0.272	0.845
0.310	0.999	-1.251	0.393	0.291	0.810
0.371	1.017	-1.305	0.445	0.304	0.786
0.434	1.067	-1.457	0.504	0.325	0.743
0.493	1.137	-1.658	0.553	0.354	0.681
0.545	1.205	-1.844	0.601	0.378	0.628
0.581	1.247	-1.954	0.645	0.404	0.567
0.624	1.225	-1.898	0.695	0.441	0.471
0.662	1.166	-1.738	0.736	0.472	0.390
0.699	1.134	-1.650	0.785	0.524	0.244
0.736	1.164	-1.734	0.828	0.581	0.077
0.774	1.228	-1.906	0.862	0.627	-0.064
0.812	1.152	-1.699	0.901	0.697	-0.284
0.847	1.058	-1.428	0.941	0.799	-0.614
0.882	1.074	-1.476			
0.917	1.131	-1.640			
0.953	1.215	-1.871			
1.000	1.264	-1.997			

Table D.29: HS1A Cascade Blade Loading at $M_2 = 1.08$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.957	-1.133	0.066	0.267	0.852
0.050	1.122	-1.630	0.113	0.251	0.880
0.087	1.228	-1.922	0.157	0.248	0.886
0.136	1.177	-1.786	0.235	0.265	0.856
0.190	1.078	-1.504	0.283	0.267	0.852
0.252	0.992	-1.242	0.340	0.278	0.834
0.310	0.999	-1.265	0.393	0.291	0.808
0.371	1.018	-1.322	0.445	0.305	0.781
0.434	1.064	-1.460	0.504	0.325	0.742
0.493	1.138	-1.677	0.553	0.348	0.692
0.545	1.201	-1.850	0.601	0.375	0.631
0.581	1.243	-1.960	0.645	0.398	0.577
0.624	1.225	-1.915	0.695	0.439	0.474
0.662	1.164	-1.749	0.736	0.476	0.376
0.699	1.134	-1.666	0.785	0.526	0.235
0.736	1.166	-1.756	0.828	0.579	0.077
0.774	1.240	-1.952	0.862	0.624	-0.061
0.812	1.329	-2.176	0.901	0.692	-0.278
0.847	1.413	-2.366	0.941	0.796	-0.616
0.882	1.469	-2.486			
0.917	1.318	-2.150			
0.953	1.332	-2.183			
1.000	1.331	-2.181			

Table D.30: HS1A Cascade Blade Loading at $M_2 = 1.18$, and $i - i_{des} = +5.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.533	0.455	458	1.12	0.142	51.9	0.198	0.187
0.712	0.580	617	1.12	0.112	52.4	0.174	0.156
0.847	0.644	746	1.11	0.087	53.3	0.150	0.128
0.963	0.677	852	1.10	0.068	53.9	0.130	0.105
1.048	0.678	929	1.08	0.063	54.5	0.133	0.104
1.124	0.680	966	1.06	0.058	54.7	0.133	0.100
1.200	0.677	1067	1.03	0.055	55.2	0.138	0.100

Table D.31: HS1A Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = +10.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.450	-0.003	0.848	0.870
0.640	0.003	0.732	0.759
0.776	0.001	0.645	0.672
0.896	-0.007	0.567	0.594
0.992	-0.005	0.507	0.533
1.072	-0.131	0.412	0.485
1.153	0.001	0.415	0.438

Table D.32: HS1A Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = +10.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (\times 10^{-3})$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.60	0.52	338	1.07	50.8	0.308	0.049
0.60	0.60	0.51	385	1.07	51.0	0.294	0.047
0.60	0.60	0.51	436	1.08	51.0	0.277	0.045
0.60	0.60	0.51	506	1.08	51.6	0.245	0.040
0.60	0.60	0.51	522	1.08	51.4	0.254	0.042
0.60	0.60	0.52	609	1.08	51.4	0.261	0.043
0.60	0.60	0.51	700	1.08	51.2	0.270	0.044
0.60	0.60	0.51	729	1.08	51.2	0.273	0.045
0.85	0.86	0.66	569	1.11	52.1	0.214	0.064
0.85	0.85	0.66	629	1.11	52.4	0.203	0.060
0.85	0.85	0.65	686	1.10	52.6	0.197	0.059
0.85	0.85	0.65	746	1.11	52.7	0.184	0.055
0.85	0.85	0.65	878	1.11	52.6	0.185	0.055
0.85	0.85	0.65	914	1.10	52.6	0.193	0.057
0.85	0.85	0.65	979	1.11	52.6	0.190	0.057
0.85	0.85	0.65	1072	1.12	52.4	0.192	0.057
0.95	0.95	0.68	663	1.10	53.1	0.176	0.060
0.95	0.95	0.68	721	1.09	53.5	0.167	0.056
0.95	0.95	0.68	788	1.09	53.5	0.156	0.053
0.95	0.95	0.68	831	1.09	53.5	0.164	0.055
0.95	0.95	0.68	908	1.10	53.5	0.159	0.054
0.95	0.95	0.67	1029	1.10	53.5	0.153	0.052
0.95	0.95	0.67	1073	1.10	53.5	0.154	0.052
1.05	1.05	0.69	821	1.07	54.3	0.147	0.055
1.05	1.05	0.69	881	1.07	54.4	0.152	0.056
1.05	1.05	0.69	943	1.07	54.4	0.148	0.055
1.05	1.05	0.69	1015	1.07	54.4	0.152	0.056
1.05	1.05	0.68	1158	1.07	54.4	0.148	0.055

Table D.33: HS1A Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = +10.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.685	-1.396	0.066	0.156	0.892
0.050	0.698	-1.472	0.113	0.162	0.880
0.087	0.703	-1.501	0.157	0.164	0.876
0.136	0.686	-1.402	0.235	0.175	0.852
0.190	0.660	-1.255	0.283	0.183	0.836
0.252	0.637	-1.122	0.340	0.185	0.831
0.310	0.622	-1.039	0.393	0.196	0.804
0.371	0.613	-0.987	0.445	0.207	0.779
0.434	0.614	-0.992	0.504	0.219	0.747
0.493	0.595	-0.888	0.553	0.236	0.700
0.545	0.596	-0.890	0.601	0.247	0.669
0.581	0.570	-0.748	0.645	0.264	0.618
0.624	0.541	-0.592	0.695	0.286	0.545
0.662	0.526	-0.514	0.736	0.297	0.507
0.699	0.513	-0.446	0.785	0.325	0.404
0.736	0.507	-0.417	0.828	0.345	0.327
0.774	0.499	-0.372	0.862	0.362	0.260
0.812	0.496	-0.360	0.901	0.390	0.141
0.847	0.498	-0.368	0.941	0.431	-0.040
0.882	0.493	-0.343			
0.917	0.496	-0.358			
0.953	0.507	-0.414			
1.000	0.491	-0.332			

Table D.34: HS1A Cascade Blade Loading at $M_2 = 0.45$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	0.977	-1.473	0.066	0.203	0.924
0.050	1.006	-1.575	0.113	0.205	0.921
0.087	1.021	-1.627	0.157	0.207	0.917
0.136	0.958	-1.409	0.235	0.224	0.889
0.190	0.890	-1.168	0.283	0.230	0.880
0.252	0.848	-1.016	0.340	0.240	0.862
0.310	0.833	-0.962	0.393	0.251	0.842
0.371	0.832	-0.958	0.445	0.262	0.822
0.434	0.838	-0.977	0.504	0.275	0.796
0.493	0.827	-0.937	0.553	0.295	0.755
0.545	0.836	-0.973	0.601	0.312	0.716
0.581	0.795	-0.824	0.645	0.330	0.676
0.624	0.749	-0.655	0.695	0.361	0.601
0.662	0.729	-0.581	0.736	0.389	0.527
0.699	0.711	-0.519	0.785	0.431	0.411
0.736	0.704	-0.493	0.828	0.465	0.313
0.774	0.694	-0.456	0.862	0.492	0.229
0.812	0.695	-0.460	0.901	0.529	0.112
0.847	0.700	-0.479	0.941	0.592	-0.097
0.882	0.695	-0.460			
0.917	0.699	-0.473			
0.953	0.710	-0.516			
1.000	0.682	-0.415			

Table D.35: HS1A Cascade Blade Loading at $M_2 = 0.64$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	1.203	-1.642	0.066	0.226	0.942
0.050	1.199	-1.630	0.113	0.224	0.944
0.087	1.287	-1.844	0.157	0.224	0.944
0.136	1.180	-1.582	0.235	0.245	0.914
0.190	0.971	-1.009	0.283	0.243	0.917
0.252	0.940	-0.918	0.340	0.255	0.899
0.310	0.929	-0.886	0.393	0.272	0.872
0.371	0.933	-0.900	0.445	0.279	0.859
0.434	0.958	-0.972	0.504	0.297	0.829
0.493	0.965	-0.991	0.553	0.319	0.787
0.545	0.992	-1.070	0.601	0.343	0.741
0.581	0.948	-0.943	0.645	0.363	0.699
0.624	0.895	-0.784	0.695	0.405	0.608
0.662	0.868	-0.705	0.736	0.435	0.536
0.699	0.845	-0.633	0.785	0.484	0.414
0.736	0.838	-0.613	0.828	0.527	0.300
0.774	0.829	-0.587	0.862	0.565	0.198
0.812	0.833	-0.597	0.901	0.618	0.048
0.847	0.846	-0.639	0.941	0.697	-0.186
0.882	0.843	-0.627			
0.917	0.854	-0.661			
0.953	0.868	-0.704			
1.000	0.817	-0.550			

Table D.36: HS1A Cascade Blade Loading at $M_2 = 0.78$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	1.269	-1.516	0.066	0.253	0.936
0.050	1.285	-1.549	0.113	0.242	0.951
0.087	1.374	-1.729	0.157	0.241	0.953
0.136	1.335	-1.652	0.235	0.259	0.928
0.190	1.137	-1.217	0.283	0.269	0.912
0.252	0.985	-0.834	0.340	0.280	0.896
0.310	0.964	-0.779	0.393	0.296	0.870
0.371	0.970	-0.794	0.445	0.305	0.857
0.434	1.006	-0.890	0.504	0.325	0.822
0.493	1.037	-0.968	0.553	0.354	0.770
0.545	1.095	-1.114	0.601	0.373	0.733
0.581	1.071	-1.055	0.645	0.398	0.683
0.624	1.010	-0.900	0.695	0.441	0.591
0.662	0.988	-0.842	0.736	0.474	0.517
0.699	0.959	-0.766	0.785	0.527	0.392
0.736	0.943	-0.724	0.828	0.568	0.291
0.774	0.943	-0.723	0.862	0.609	0.183
0.812	0.949	-0.739	0.901	0.675	0.009
0.847	0.972	-0.801	0.941	0.767	-0.243
0.882	0.980	-0.822			
0.917	0.998	-0.868			
0.953	1.010	-0.900			
1.000	0.938	-0.709			

Table D.37: HS1A Cascade Blade Loading at $M_2 = 0.90$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	1.276	-1.491	0.066	0.266	0.923
0.050	1.306	-1.553	0.113	0.256	0.937
0.087	1.378	-1.695	0.157	0.258	0.934
0.136	1.356	-1.651	0.235	0.276	0.908
0.190	1.195	-1.316	0.283	0.281	0.901
0.252	1.032	-0.924	0.340	0.291	0.885
0.310	0.978	-0.786	0.393	0.308	0.857
0.371	0.975	-0.777	0.445	0.319	0.840
0.434	1.015	-0.883	0.504	0.332	0.816
0.493	1.062	-0.999	0.553	0.357	0.771
0.545	1.127	-1.158	0.601	0.380	0.728
0.581	1.130	-1.164	0.645	0.406	0.676
0.624	1.085	-1.057	0.695	0.446	0.592
0.662	1.068	-1.015	0.736	0.479	0.518
0.699	1.043	-0.953	0.785	0.537	0.380
0.736	1.012	-0.874	0.828	0.584	0.264
0.774	1.011	-0.872	0.862	0.628	0.149
0.812	1.020	-0.894	0.901	0.695	-0.027
0.847	1.051	-0.971	0.941	0.794	-0.295
0.882	1.079	-1.042			
0.917	1.126	-1.154			
0.953	1.172	-1.263			
1.000	1.035	-0.933			

Table D.38: HS1A Cascade Blade Loading at $M_2 = 0.99$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	1.295	-1.516	0.066	0.262	0.930
0.050	1.312	-1.549	0.113	0.256	0.939
0.087	1.386	-1.693	0.157	0.255	0.941
0.136	1.363	-1.649	0.235	0.268	0.922
0.190	1.206	-1.325	0.283	0.269	0.921
0.252	1.047	-0.949	0.340	0.283	0.900
0.310	0.981	-0.783	0.393	0.298	0.877
0.371	0.978	-0.776	0.445	0.314	0.851
0.434	1.015	-0.871	0.504	0.333	0.817
0.493	1.068	-1.003	0.553	0.362	0.766
0.545	1.136	-1.166	0.601	0.383	0.725
0.581	1.148	-1.194	0.645	0.407	0.677
0.624	1.118	-1.122	0.695	0.446	0.595
0.662	1.116	-1.119	0.736	0.479	0.520
0.699	1.119	-1.127	0.785	0.541	0.375
0.736	1.117	-1.120	0.828	0.583	0.271
0.774	1.107	-1.098	0.862	0.629	0.154
0.812	1.048	-0.954	0.901	0.697	-0.026
0.847	1.069	-1.005	0.941	0.801	-0.305
0.882	1.101	-1.082			
0.917	1.161	-1.224			
0.953	1.245	-1.411			
1.000	1.200	-1.313			

Table D.39: HS1A Cascade Blade Loading at $M_2 = 1.07$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.016	1.293	-1.527	0.066	0.256	0.937
0.050	1.307	-1.555	0.113	0.249	0.947
0.087	1.382	-1.702	0.157	0.251	0.944
0.136	1.364	-1.669	0.235	0.265	0.924
0.190	1.203	-1.334	0.283	0.273	0.913
0.252	1.045	-0.959	0.340	0.282	0.899
0.310	0.986	-0.808	0.393	0.298	0.874
0.371	0.980	-0.792	0.445	0.310	0.854
0.434	1.018	-0.890	0.504	0.332	0.817
0.493	1.068	-1.015	0.553	0.355	0.775
0.545	1.132	-1.170	0.601	0.377	0.734
0.581	1.146	-1.203	0.645	0.402	0.684
0.624	1.120	-1.142	0.695	0.446	0.591
0.662	1.115	-1.129	0.736	0.482	0.511
0.699	1.130	-1.164	0.785	0.537	0.381
0.736	1.161	-1.238	0.828	0.578	0.277
0.774	1.236	-1.407	0.862	0.626	0.156
0.812	1.315	-1.572	0.901	0.693	-0.022
0.847	1.386	-1.710	0.941	0.794	-0.294
0.882	1.302	-1.546			
0.917	1.253	-1.443			
0.953	1.301	-1.542			
1.000	1.195	-1.316			

Table D.40: HS1A Cascade Blade Loading at $M_2 = 1.15$, and $i - i_{des} = +10.0^\circ$.

Appendix E

HS1C Cascade Results

M_{2MO}	M_1	$Re_2 (x10^3)$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.531	0.338	475	0.98	0.040	56.2	0.050	0.008
0.728	0.408	644	0.98	0.035	56.9	0.052	0.045
0.872	0.431	764	0.98	0.031	57.6	0.053	0.044
0.988	0.441	856	0.97	0.039	57.7	0.075	0.060
1.113	0.440	928	0.97	0.048	56.7	0.108	0.081
1.195	0.439	984	0.97	0.054	56.0	0.134	0.097
1.311	0.436	1024	0.99	0.059	53.7	0.173	0.118

Table E.1: HS1C Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = -10.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.494	-0.031	0.836	0.846
0.705	-0.014	0.705	0.717
0.845	0.002	0.617	0.627
0.962	0.040	0.552	0.552
1.065	-0.096	0.434	0.489
1.150	-0.078	0.387	0.440
1.254	-0.088	0.323	0.384

Table E.2: HS1C Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = -10.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^{-3})$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.60	0.37	332	0.96	56.0	0.075	0.014
0.60	0.60	0.37	484	0.97	56.2	0.063	0.012
0.60	0.60	0.37	533	0.98	56.4	0.051	0.010
0.60	0.60	0.36	667	0.99	56.5	0.040	0.008
0.60	0.60	0.36	831	1.00	56.4	0.043	0.008
0.85	0.85	0.43	596	0.97	57.2	0.054	0.016
0.85	0.85	0.43	664	0.97	57.4	0.055	0.017
0.85	0.85	0.43	757	0.98	57.4	0.052	0.016
0.85	0.85	0.43	895	0.98	57.4	0.053	0.016
0.85	0.85	0.42	1104	0.99	57.3	0.045	0.014
0.95	0.95	0.44	686	0.96	57.8	0.061	0.020
0.95	0.95	0.44	835	0.97	57.9	0.057	0.019
0.95	0.95	0.43	1024	0.97	58.0	0.054	0.018
1.05	1.04	0.44	762	0.97	57.6	0.074	0.026
1.05	1.05	0.44	887	0.98	57.6	0.069	0.024
1.05	1.04	0.43	1112	0.98	57.5	0.072	0.025

Table E.3: HS1C Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = -10.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.345	-0.073	0.063	0.286	0.255
0.031	0.412	-0.514	0.115	0.210	0.605
0.061	0.428	-0.623	0.175	0.193	0.672
0.104	0.468	-0.921	0.231	0.197	0.655
0.155	0.514	-1.273	0.296	0.202	0.637
0.209	0.547	-1.544	0.361	0.209	0.610
0.265	0.576	-1.781	0.424	0.223	0.554
0.327	0.614	-2.108	0.483	0.232	0.514
0.386	0.648	-2.405	0.540	0.245	0.455
0.443	0.681	-2.702	0.598	0.269	0.344
0.497	0.697	-2.843	0.648	0.288	0.244
0.546	0.697	-2.843	0.704	0.310	0.128
0.591	0.673	-2.624	0.754	0.327	0.031
0.631	0.652	-2.438	0.807	0.346	-0.083
0.667	0.611	-2.082	0.859	0.374	-0.256
0.699	0.586	-1.870	0.906	0.407	-0.480
0.733	0.578	-1.796	1.000	0.518	-1.305
0.763	0.571	-1.740			
0.798	0.567	-1.707			
0.835	0.566	-1.701			
0.871	0.561	-1.656			
0.899	0.554	-1.598			
0.928	0.546	-1.534			
0.956	0.538	-1.472			
1.000	0.513	-1.267			

Table E.4: HS1C Cascade Blade Loading at $M_2 = 0.49$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.426	-0.052	0.063	0.373	0.186
0.031	0.520	-0.531	0.115	0.277	0.557
0.061	0.544	-0.661	0.175	0.267	0.590
0.104	0.599	-0.968	0.231	0.275	0.565
0.155	0.661	-1.335	0.296	0.280	0.549
0.209	0.706	-1.601	0.361	0.288	0.522
0.265	0.754	-1.894	0.424	0.301	0.474
0.327	0.812	-2.248	0.483	0.312	0.433
0.386	0.869	-2.592	0.540	0.325	0.384
0.443	0.926	-2.934	0.598	0.350	0.284
0.497	0.966	-3.173	0.648	0.374	0.182
0.546	0.983	-3.268	0.704	0.402	0.058
0.591	0.954	-3.099	0.754	0.432	-0.084
0.631	0.930	-2.960	0.807	0.461	-0.226
0.667	0.875	-2.628	0.859	0.503	-0.443
0.699	0.826	-2.333	0.906	0.554	-0.718
0.733	0.813	-2.254	1.000	0.736	-1.782
0.763	0.811	-2.242			
0.798	0.814	-2.256			
0.835	0.814	-2.257			
0.871	0.807	-2.214			
0.899	0.797	-2.153			
0.928	0.784	-2.078			
0.956	0.771	-1.999			
1.000	0.725	-1.717			

Table E.5: HS1C Cascade Blade Loading at $M_2 = 0.71$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.435	-0.004	0.063	0.390	0.189
0.031	0.548	-0.544	0.115	0.267	0.631
0.061	0.577	-0.698	0.175	0.243	0.700
0.104	0.631	-0.985	0.231	0.240	0.710
0.155	0.701	-1.366	0.296	0.235	0.724
0.209	0.756	-1.674	0.361	0.242	0.702
0.265	0.810	-1.979	0.424	0.286	0.572
0.327	0.881	-2.375	0.483	0.318	0.463
0.386	0.946	-2.731	0.540	0.343	0.372
0.443	1.023	-3.145	0.598	0.378	0.237
0.497	1.090	-3.491	0.648	0.403	0.136
0.546	1.147	-3.776	0.704	0.432	0.008
0.591	1.113	-3.608	0.754	0.464	-0.137
0.631	1.089	-3.484	0.807	0.505	-0.332
0.667	1.046	-3.269	0.859	0.559	-0.601
0.699	0.995	-2.999	0.906	0.619	-0.918
0.733	0.954	-2.775	1.000	0.871	-2.320
0.763	0.956	-2.790			
0.798	0.974	-2.885			
0.835	0.994	-2.991			
0.871	0.999	-3.018			
0.899	0.986	-2.951			
0.928	0.967	-2.846			
0.956	0.938	-2.690			
1.000	0.860	-2.257			

Table E.6: HS1C Cascade Blade Loading at $M_2 = 0.85$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.449	0.015	0.063	0.411	0.172
0.031	0.562	-0.492	0.115	0.305	0.552
0.061	0.588	-0.618	0.175	0.288	0.603
0.104	0.655	-0.953	0.231	0.298	0.573
0.155	0.723	-1.303	0.296	0.301	0.563
0.209	0.781	-1.606	0.361	0.311	0.531
0.265	0.840	-1.913	0.424	0.329	0.473
0.327	0.908	-2.262	0.483	0.345	0.418
0.386	0.979	-2.623	0.540	0.365	0.348
0.443	1.058	-3.012	0.598	0.393	0.245
0.497	1.133	-3.360	0.648	0.421	0.132
0.546	1.213	-3.712	0.704	0.460	-0.029
0.591	1.207	-3.688	0.754	0.495	-0.185
0.631	1.168	-3.516	0.807	0.537	-0.376
0.667	1.135	-3.368	0.859	0.586	-0.608
0.699	1.108	-3.246	0.906	0.653	-0.946
0.733	1.087	-3.149	1.000	0.982	-2.637
0.763	1.052	-2.982			
0.798	1.055	-2.994			
0.835	1.084	-3.135			
0.871	1.131	-3.351			
0.899	1.164	-3.499			
0.928	1.206	-3.682			
0.956	1.236	-3.809			
1.000	0.962	-2.540			

Table E.7: HS1C Cascade Blade Loading at $M_2 = 0.96$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.456	0.014	0.063	0.419	0.164
0.031	0.563	-0.464	0.115	0.311	0.547
0.061	0.596	-0.621	0.175	0.298	0.586
0.104	0.659	-0.927	0.231	0.300	0.581
0.155	0.730	-1.283	0.296	0.305	0.565
0.209	0.787	-1.573	0.361	0.314	0.536
0.265	0.842	-1.858	0.424	0.329	0.487
0.327	0.915	-2.224	0.483	0.347	0.427
0.386	0.984	-2.566	0.540	0.370	0.347
0.443	1.068	-2.963	0.598	0.399	0.241
0.497	1.142	-3.302	0.648	0.425	0.140
0.546	1.227	-3.663	0.704	0.459	-0.001
0.591	1.248	-3.750	0.754	0.496	-0.158
0.631	1.207	-3.582	0.807	0.536	-0.338
0.667	1.169	-3.419	0.859	0.596	-0.618
0.699	1.158	-3.373	0.906	0.665	-0.957
0.733	1.190	-3.507	1.000	1.180	-3.467
0.763	1.239	-3.713			
0.798	1.337	-4.094			
0.835	1.287	-3.905			
0.871	1.206	-3.576			
0.899	1.231	-3.678			
0.928	1.270	-3.838			
0.956	1.297	-3.942			
1.000	1.160	-3.381			

Table E.8: HS1C Cascade Blade Loading at $M_2 = 1.07$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.455	-0.017	0.063	0.420	0.130
0.031	0.569	-0.537	0.115	0.317	0.511
0.061	0.599	-0.683	0.175	0.301	0.561
0.104	0.658	-0.980	0.231	0.303	0.555
0.155	0.727	-1.339	0.296	0.305	0.547
0.209	0.786	-1.648	0.361	0.313	0.524
0.265	0.846	-1.966	0.424	0.326	0.478
0.327	0.914	-2.316	0.483	0.344	0.417
0.386	0.986	-2.680	0.540	0.369	0.328
0.443	1.068	-3.082	0.598	0.405	0.189
0.497	1.141	-3.424	0.648	0.432	0.082
0.546	1.229	-3.812	0.704	0.461	-0.042
0.591	1.258	-3.931	0.754	0.497	-0.200
0.631	1.204	-3.703	0.807	0.542	-0.410
0.667	1.161	-3.515	0.859	0.600	-0.688
0.699	1.153	-3.480	0.906	0.670	-1.042
0.733	1.188	-3.636	1.000	1.258	-3.932
0.763	1.241	-3.859			
0.798	1.340	-4.255			
0.835	1.461	-4.688			
0.871	1.596	-5.103			
0.899	1.513	-4.855			
0.928	1.386	-4.429			
0.956	1.395	-4.458			
1.000	1.247	-3.888			

Table E.9: HS1C Cascade Blade Loading at $M_2 = 1.15$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.450	-0.005	0.063	0.411	0.158
0.031	0.564	-0.530	0.115	0.299	0.563
0.061	0.597	-0.690	0.175	0.291	0.588
0.104	0.655	-0.985	0.231	0.296	0.572
0.155	0.727	-1.361	0.296	0.297	0.568
0.209	0.783	-1.660	0.361	0.301	0.554
0.265	0.838	-1.949	0.424	0.322	0.488
0.327	0.908	-2.317	0.483	0.345	0.409
0.386	0.980	-2.686	0.540	0.369	0.322
0.443	1.062	-3.095	0.598	0.394	0.226
0.497	1.139	-3.458	0.648	0.419	0.123
0.546	1.226	-3.845	0.704	0.453	-0.017
0.591	1.253	-3.957	0.754	0.490	-0.183
0.631	1.196	-3.713	0.807	0.531	-0.373
0.667	1.150	-3.507	0.859	0.587	-0.641
0.699	1.149	-3.503	0.906	0.659	-1.009
0.733	1.181	-3.646	1.000	1.453	-4.715
0.763	1.235	-3.884			
0.798	1.334	-4.284			
0.835	1.460	-4.738			
0.871	1.592	-5.151			
0.899	1.699	-5.437			
0.928	1.726	-5.502			
0.956	1.621	-5.233			
1.000	1.381	-4.460			

Table E.10: HS1C Cascade Blade Loading at $M_2 = 1.25$, and $i - i_{des} = -10.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.534	0.394	477	1.02	0.050	55.6	0.064	0.060
0.729	0.495	652	1.00	0.038	56.3	0.057	0.050
0.879	0.531	776	0.99	0.033	57.0	0.057	0.047
1.003	0.545	868	0.98	0.040	57.2	0.079	0.062
1.103	0.545	952	0.98	0.037	56.7	0.082	0.062
1.195	0.549	995	0.96	0.055	55.8	0.138	0.100
1.296	0.548	1039	0.97	0.063	54.1	0.181	0.125

Table E.11: HS1C Cascade Performance Data at Design Incidence, $i - i_{des} = 0.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.497	0.115	0.851	0.845
0.699	0.042	0.722	0.722
0.835	0.026	0.631	0.633
0.933	0.031	0.569	0.571
1.049	-0.011	0.481	0.499
1.142	-0.002	0.420	0.444
1.182	-0.049	0.374	0.422

Table E.12: HS1C Cascade Base Pressure Data at Design Incidence, $i - i_{des} = 0.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^3)$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.60	0.43	324	1.01	54.6	0.114	0.020
0.60	0.60	0.43	377	1.02	55.2	0.086	0.016
0.60	0.60	0.43	483	1.02	55.5	0.061	0.011
0.60	0.60	0.42	535	1.02	55.7	0.054	0.010
0.60	0.60	0.43	547	1.01	55.7	0.062	0.011
0.60	0.60	0.43	666	1.01	55.9	0.053	0.010
0.60	0.60	0.42	780	1.03	55.6	0.053	0.010
0.60	0.60	0.42	813	1.02	55.8	0.055	0.010
0.85	0.85	0.52	602	1.02	56.6	0.059	0.018
0.85	0.85	0.52	661	1.00	56.8	0.062	0.019
0.85	0.85	0.51	722	1.01	56.7	0.056	0.017
0.85	0.85	0.52	755	1.01	56.9	0.053	0.016
0.85	0.85	0.51	874	1.02	56.9	0.050	0.015
0.85	0.85	0.51	926	1.02	56.8	0.047	0.014
0.85	0.85	0.51	962	1.01	56.8	0.054	0.016
0.85	0.85	0.51	1080	1.03	56.8	0.046	0.014
0.95	0.95	0.53	687	1.00	57.2	0.058	0.020
0.95	0.95	0.53	751	1.00	57.2	0.054	0.018
0.95	0.95	0.53	823	1.00	57.3	0.058	0.019
0.95	0.95	0.53	866	1.00	57.3	0.053	0.018
0.95	0.95	0.52	962	1.01	57.3	0.051	0.017
0.95	0.95	0.52	1023	1.01	57.3	0.053	0.018
1.05	1.05	0.53	762	0.99	57.3	0.075	0.026
1.05	1.05	0.53	825	1.00	57.3	0.072	0.026
1.05	1.05	0.53	885	0.99	57.3	0.075	0.026
1.05	1.05	0.53	912	1.00	57.3	0.067	0.024
1.05	1.05	0.53	1085	1.00	57.2	0.074	0.026

Table E.13: HS1C Cascade Performance Data at Design Incidence at Design Incidence, $i - i_{des} = 0.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.523	-0.997	0.063	0.209	0.674
0.031	0.529	-1.043	0.115	0.185	0.750
0.061	0.518	-0.964	0.175	0.176	0.776
0.104	0.534	-1.074	0.231	0.186	0.749
0.155	0.557	-1.236	0.296	0.191	0.734
0.209	0.574	-1.360	0.361	0.199	0.709
0.265	0.588	-1.461	0.424	0.212	0.665
0.327	0.612	-1.638	0.483	0.217	0.647
0.386	0.633	-1.792	0.540	0.233	0.592
0.443	0.650	-1.920	0.598	0.259	0.490
0.497	0.665	-2.030	0.648	0.272	0.435
0.546	0.661	-2.004	0.704	0.294	0.339
0.591	0.631	-1.778	0.754	0.315	0.240
0.631	0.609	-1.610	0.807	0.333	0.149
0.667	0.574	-1.358	0.859	0.356	0.031
0.699	0.554	-1.217	0.906	0.385	-0.125
0.733	0.545	-1.154	1.000	0.485	-0.739
0.763	0.538	-1.103			
0.798	0.532	-1.059			
0.835	0.531	-1.055			
0.871	0.522	-0.994			
0.899	0.517	-0.956			
0.928	0.509	-0.905			
0.956	0.498	-0.828			
1.000	0.486	-0.751			

Table E.14: HS1C Cascade Blade Loading at $M_2 = 0.50$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.683	-0.875	0.063	0.274	0.692
0.031	0.710	-1.002	0.115	0.242	0.771
0.061	0.690	-0.908	0.175	0.230	0.798
0.104	0.713	-1.013	0.231	0.237	0.782
0.155	0.748	-1.176	0.296	0.241	0.774
0.209	0.771	-1.287	0.361	0.251	0.749
0.265	0.793	-1.389	0.424	0.271	0.701
0.327	0.835	-1.587	0.483	0.284	0.665
0.386	0.871	-1.757	0.540	0.307	0.604
0.443	0.908	-1.925	0.598	0.336	0.516
0.497	0.940	-2.072	0.648	0.362	0.434
0.546	0.943	-2.087	0.704	0.389	0.343
0.591	0.909	-1.932	0.754	0.421	0.230
0.631	0.874	-1.767	0.807	0.453	0.113
0.667	0.824	-1.533	0.859	0.491	-0.038
0.699	0.793	-1.392	0.906	0.534	-0.215
0.733	0.782	-1.336	1.000	0.700	-0.954
0.763	0.776	-1.309			
0.798	0.772	-1.290			
0.835	0.771	-1.289			
0.871	0.761	-1.241			
0.899	0.749	-1.182			
0.928	0.743	-1.154			
0.956	0.727	-1.079			
1.000	0.699	-0.949			

Table E.15: HS1C Cascade Blade Loading at $M_2 = 0.70$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.747	-0.893	0.063	0.304	0.675
0.031	0.786	-1.057	0.115	0.260	0.777
0.061	0.766	-0.972	0.175	0.256	0.787
0.104	0.785	-1.054	0.231	0.264	0.769
0.155	0.826	-1.222	0.296	0.267	0.763
0.209	0.850	-1.319	0.361	0.279	0.735
0.265	0.874	-1.420	0.424	0.308	0.664
0.327	0.920	-1.607	0.483	0.320	0.634
0.386	0.965	-1.787	0.540	0.347	0.563
0.443	1.027	-2.028	0.598	0.377	0.475
0.497	1.081	-2.234	0.648	0.400	0.407
0.546	1.121	-2.381	0.704	0.433	0.303
0.591	1.087	-2.256	0.754	0.465	0.194
0.631	1.049	-2.115	0.807	0.501	0.069
0.667	0.992	-1.893	0.859	0.553	-0.121
0.699	0.951	-1.730	0.906	0.614	-0.355
0.733	0.938	-1.678	1.000	0.842	-1.288
0.763	0.938	-1.680			
0.798	0.941	-1.690			
0.835	0.951	-1.731			
0.871	0.942	-1.694			
0.899	0.929	-1.643			
0.928	0.920	-1.605			
0.956	0.893	-1.496			
1.000	0.839	-1.275			

Table E.16: HS1C Cascade Blade Loading at $M_2 = 0.84$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.752	-0.808	0.063	0.325	0.650
0.031	0.811	-1.037	0.115	0.277	0.762
0.061	0.781	-0.918	0.175	0.266	0.786
0.104	0.811	-1.037	0.231	0.285	0.743
0.155	0.856	-1.212	0.296	0.290	0.733
0.209	0.878	-1.297	0.361	0.296	0.720
0.265	0.902	-1.391	0.424	0.316	0.671
0.327	0.950	-1.575	0.483	0.331	0.634
0.386	1.001	-1.767	0.540	0.357	0.568
0.443	1.070	-2.018	0.598	0.388	0.481
0.497	1.130	-2.226	0.648	0.415	0.402
0.546	1.207	-2.481	0.704	0.449	0.296
0.591	1.185	-2.411	0.754	0.490	0.163
0.631	1.143	-2.271	0.807	0.528	0.033
0.667	1.093	-2.098	0.859	0.579	-0.147
0.699	1.059	-1.977	0.906	0.648	-0.406
0.733	1.038	-1.903	1.000	0.940	-1.538
0.763	1.037	-1.898			
0.798	1.055	-1.965			
0.835	1.089	-2.086			
0.871	1.127	-2.217			
0.899	1.146	-2.279			
0.928	1.173	-2.372			
0.956	1.069	-2.013			
1.000	0.935	-1.517			

Table E.17: HS1C Cascade Blade Loading at $M_2 = 0.93$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.763	-0.824	0.063	0.333	0.637
0.031	0.819	-1.042	0.115	0.292	0.734
0.061	0.797	-0.956	0.175	0.282	0.756
0.104	0.821	-1.049	0.231	0.292	0.734
0.155	0.866	-1.223	0.296	0.297	0.723
0.209	0.888	-1.308	0.361	0.307	0.701
0.265	0.917	-1.418	0.424	0.328	0.650
0.327	0.963	-1.591	0.483	0.346	0.604
0.386	1.019	-1.795	0.540	0.368	0.545
0.443	1.085	-2.031	0.598	0.405	0.440
0.497	1.151	-2.256	0.648	0.427	0.375
0.546	1.230	-2.511	0.704	0.462	0.266
0.591	1.242	-2.548	0.754	0.501	0.141
0.631	1.189	-2.381	0.807	0.540	0.006
0.667	1.138	-2.211	0.859	0.596	-0.193
0.699	1.138	-2.212	0.906	0.665	-0.450
0.733	1.177	-2.340	1.000	1.093	-2.058
0.763	1.223	-2.489			
0.798	1.249	-2.568			
0.835	1.139	-2.216			
0.871	1.153	-2.261			
0.899	1.190	-2.383			
0.928	1.233	-2.519			
0.956	1.262	-2.608			
1.000	1.078	-2.008			

Table E.18: HS1C Cascade Blade Loading at $M_2 = 1.05$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.768	-0.853	0.063	0.321	0.665
0.031	0.820	-1.055	0.115	0.278	0.763
0.061	0.787	-0.928	0.175	0.268	0.784
0.104	0.819	-1.050	0.231	0.278	0.763
0.155	0.864	-1.224	0.296	0.280	0.760
0.209	0.887	-1.313	0.361	0.288	0.741
0.265	0.909	-1.398	0.424	0.311	0.687
0.327	0.957	-1.580	0.483	0.337	0.625
0.386	1.008	-1.767	0.540	0.362	0.557
0.443	1.079	-2.023	0.598	0.396	0.464
0.497	1.144	-2.246	0.648	0.421	0.391
0.546	1.227	-2.515	0.704	0.453	0.290
0.591	1.250	-2.589	0.754	0.492	0.165
0.631	1.189	-2.396	0.807	0.538	0.010
0.667	1.126	-2.186	0.859	0.589	-0.174
0.699	1.131	-2.205	0.906	0.658	-0.430
0.733	1.177	-2.358	1.000	1.168	-2.326
0.763	1.235	-2.542			
0.798	1.330	-2.825			
0.835	1.447	-3.137			
0.871	1.429	-3.092			
0.899	1.252	-2.595			
0.928	1.284	-2.692			
0.956	1.298	-2.731			
1.000	1.186	-2.386			

Table E.19: HS1C Cascade Blade Loading at $M_2 = 1.14$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.769	-0.884	0.063	0.314	0.675
0.031	0.819	-1.082	0.115	0.270	0.774
0.061	0.787	-0.954	0.175	0.262	0.792
0.104	0.815	-1.064	0.231	0.276	0.761
0.155	0.860	-1.242	0.296	0.287	0.738
0.209	0.887	-1.347	0.361	0.293	0.724
0.265	0.909	-1.430	0.424	0.315	0.671
0.327	0.956	-1.613	0.483	0.335	0.621
0.386	1.007	-1.806	0.540	0.357	0.565
0.443	1.078	-2.061	0.598	0.386	0.483
0.497	1.144	-2.291	0.648	0.410	0.414
0.546	1.226	-2.561	0.704	0.444	0.307
0.591	1.243	-2.617	0.754	0.487	0.168
0.631	1.178	-2.408	0.807	0.532	0.013
0.667	1.117	-2.199	0.859	0.590	-0.195
0.699	1.135	-2.261	0.906	0.662	-0.466
0.733	1.175	-2.397	1.000	1.266	-2.687
0.763	1.235	-2.591			
0.798	1.327	-2.871			
0.835	1.452	-3.207			
0.871	1.580	-3.506			
0.899	1.651	-3.653			
0.928	1.466	-3.243			
0.956	1.457	-3.220			
1.000	1.274	-2.712			

Table E.20: HS1C Cascade Blade Loading at $M_2 = 1.18$, and $i - i_{des} = 0.0^\circ$.

M_{2MO}	M_1	$Re_2 (\times 10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.534	0.428	457	1.04	0.108	53.5	0.146	0.137
0.724	0.537	623	1.04	0.075	54.5	0.114	0.101
0.870	0.587	748	1.03	0.052	55.4	0.090	0.075
0.992	0.599	850	1.02	0.043	56.1	0.084	0.067
1.101	0.601	930	1.01	0.044	56.1	0.097	0.074
1.199	0.599	978	0.99	0.056	55.5	0.140	0.101
1.299	0.599	1012	1.00	0.070	53.2	0.203	0.140

Table E.21: HS1C Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = +5.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.490	-0.025	0.828	0.849
0.688	-0.028	0.702	0.729
0.830	-0.016	0.615	0.636
0.935	-0.018	0.549	0.569
1.058	-0.003	0.475	0.493
1.150	0.002	0.417	0.440
1.191	-0.050	0.365	0.417

Table E.22: HS1C Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = +5.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (\times 10^{-3})$	AVDR	α_2	Y_1	Y_{pwc}
0.60	0.60	0.45	305	1.09	52.6	0.122	0.020
0.60	0.60	0.46	353	1.07	53.0	0.125	0.021
0.60	0.60	0.46	446	1.06	53.5	0.122	0.021
0.60	0.60	0.47	509	1.04	53.8	0.135	0.023
0.60	0.60	0.47	615	1.04	53.8	0.131	0.023
0.85	0.85	0.58	630	1.04	55.1	0.099	0.029
0.85	0.85	0.58	685	1.04	55.1	0.096	0.028
0.85	0.85	0.58	731	1.03	55.2	0.095	0.028
0.85	0.85	0.58	845	1.03	55.3	0.097	0.028
0.85	0.85	0.58	908	1.04	55.3	0.096	0.028
0.85	0.85	0.57	990	1.04	55.3	0.086	0.025
0.95	0.95	0.60	720	1.03	55.7	0.088	0.029
0.95	0.95	0.60	804	1.03	55.7	0.084	0.028
0.95	0.95	0.59	865	1.03	55.8	0.085	0.028
0.95	0.95	0.59	978	1.03	55.8	0.080	0.026
0.95	0.95	0.59	1041	1.04	55.8	0.070	0.023
1.05	1.05	0.60	735	1.02	56.0	0.092	0.032
1.05	1.05	0.60	791	1.02	56.0	0.083	0.029
1.05	1.04	0.60	855	1.02	56.1	0.084	0.029
1.05	1.04	0.60	931	1.02	56.2	0.083	0.029
1.05	1.05	0.59	1067	1.03	56.1	0.078	0.027

Table E.23: HS1C Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = +5.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.687	-1.415	0.063	0.193	0.811
0.031	0.669	-1.309	0.115	0.183	0.835
0.061	0.618	-1.017	0.175	0.180	0.843
0.104	0.613	-0.993	0.231	0.195	0.808
0.155	0.630	-1.086	0.296	0.196	0.804
0.209	0.635	-1.116	0.361	0.200	0.795
0.265	0.641	-1.150	0.424	0.220	0.745
0.327	0.665	-1.286	0.483	0.231	0.714
0.386	0.679	-1.370	0.540	0.248	0.665
0.443	0.696	-1.467	0.598	0.273	0.585
0.497	0.705	-1.520	0.648	0.289	0.535
0.546	0.693	-1.452	0.704	0.308	0.466
0.591	0.652	-1.213	0.754	0.336	0.360
0.631	0.626	-1.062	0.807	0.351	0.302
0.667	0.601	-0.924	0.859	0.376	0.202
0.699	0.586	-0.841	0.906	0.411	0.049
0.733	0.578	-0.797	1.000	0.523	-0.501
0.763	0.572	-0.764			
0.798	0.567	-0.733			
0.835	0.559	-0.692			
0.871	0.554	-0.665			
0.899	0.555	-0.669			
0.928	0.539	-0.586			
0.956	0.536	-0.568			
1.000	0.526	-0.515			

Table E.24: HS1C Cascade Blade Loading at $M_2 = 0.49$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.900	-1.389	0.063	0.274	0.767
0.031	0.908	-1.420	0.115	0.262	0.793
0.061	0.813	-1.050	0.175	0.254	0.810
0.104	0.807	-1.028	0.231	0.262	0.792
0.155	0.829	-1.113	0.296	0.263	0.791
0.209	0.833	-1.130	0.361	0.257	0.804
0.265	0.843	-1.167	0.424	0.271	0.774
0.327	0.866	-1.258	0.483	0.279	0.756
0.386	0.890	-1.352	0.540	0.296	0.720
0.443	0.930	-1.505	0.598	0.323	0.654
0.497	0.958	-1.610	0.648	0.342	0.606
0.546	0.959	-1.614	0.704	0.372	0.526
0.591	0.910	-1.426	0.754	0.410	0.415
0.631	0.870	-1.273	0.807	0.440	0.325
0.667	0.828	-1.110	0.859	0.486	0.174
0.699	0.809	-1.033	0.906	0.541	-0.013
0.733	0.801	-1.004	1.000	0.725	-0.703
0.763	0.792	-0.969			
0.798	0.788	-0.954			
0.835	0.777	-0.908			
0.871	0.771	-0.885			
0.899	0.775	-0.902			
0.928	0.754	-0.820			
0.956	0.745	-0.783			
1.000	0.729	-0.720			

Table E.25: HS1C Cascade Blade Loading at $M_2 = 0.69$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.984	-1.330	0.063	0.271	0.829
0.031	1.034	-1.489	0.115	0.252	0.862
0.061	0.893	-1.030	0.175	0.256	0.855
0.104	0.890	-1.018	0.231	0.268	0.835
0.155	0.917	-1.110	0.296	0.276	0.818
0.209	0.920	-1.118	0.361	0.280	0.811
0.265	0.925	-1.135	0.424	0.308	0.756
0.327	0.951	-1.223	0.483	0.320	0.729
0.386	0.990	-1.349	0.540	0.345	0.674
0.443	1.043	-1.520	0.598	0.379	0.594
0.497	1.098	-1.689	0.648	0.402	0.537
0.546	1.123	-1.764	0.704	0.435	0.452
0.591	1.082	-1.640	0.754	0.472	0.349
0.631	1.033	-1.486	0.807	0.507	0.245
0.667	0.983	-1.326	0.859	0.557	0.093
0.699	0.955	-1.236	0.906	0.617	-0.099
0.733	0.956	-1.238	1.000	0.856	-0.902
0.763	0.951	-1.221			
0.798	0.948	-1.212			
0.835	0.949	-1.214			
0.871	0.939	-1.181			
0.899	0.937	-1.175			
0.928	0.913	-1.095			
0.956	0.895	-1.035			
1.000	0.863	-0.926			

Table E.26: HS1C Cascade Blade Loading at $M_2 = 0.83$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.995	-1.261	0.063	0.293	0.804
0.031	1.083	-1.529	0.115	0.274	0.839
0.061	0.916	-1.012	0.175	0.268	0.850
0.104	0.920	-1.024	0.231	0.284	0.820
0.155	0.949	-1.116	0.296	0.289	0.811
0.209	0.948	-1.115	0.361	0.298	0.794
0.265	0.958	-1.145	0.424	0.326	0.738
0.327	0.988	-1.238	0.483	0.339	0.711
0.386	1.025	-1.355	0.540	0.363	0.656
0.443	1.086	-1.537	0.598	0.398	0.574
0.497	1.146	-1.709	0.648	0.418	0.524
0.546	1.206	-1.874	0.704	0.455	0.428
0.591	1.178	-1.799	0.754	0.499	0.309
0.631	1.132	-1.672	0.807	0.537	0.197
0.667	1.074	-1.500	0.859	0.586	0.050
0.699	1.058	-1.454	0.906	0.657	-0.170
0.733	1.065	-1.475	1.000	0.969	-1.181
0.763	1.063	-1.469			
0.798	1.078	-1.513			
0.835	1.111	-1.609			
0.871	1.129	-1.663			
0.899	1.132	-1.671			
0.928	1.120	-1.637			
0.956	1.033	-1.377			
1.000	0.967	-1.173			

Table E.27: HS1C Cascade Blade Loading at $M_2 = 0.94$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.999	-1.265	0.063	0.285	0.820
0.031	1.098	-1.563	0.115	0.253	0.876
0.061	0.906	-0.973	0.175	0.238	0.901
0.104	0.926	-1.034	0.231	0.244	0.892
0.155	0.953	-1.122	0.296	0.232	0.911
0.209	0.946	-1.100	0.361	0.232	0.910
0.265	0.955	-1.128	0.424	0.249	0.884
0.327	0.991	-1.240	0.483	0.295	0.801
0.386	1.034	-1.372	0.540	0.362	0.661
0.443	1.093	-1.548	0.598	0.395	0.584
0.497	1.156	-1.727	0.648	0.420	0.522
0.546	1.227	-1.920	0.704	0.458	0.423
0.591	1.221	-1.904	0.754	0.503	0.300
0.631	1.172	-1.773	0.807	0.545	0.178
0.667	1.114	-1.608	0.859	0.597	0.021
0.699	1.120	-1.627	0.906	0.664	-0.187
0.733	1.164	-1.751	1.000	1.090	-1.538
0.763	1.210	-1.875			
0.798	1.229	-1.926			
0.835	1.174	-1.777			
0.871	1.176	-1.784			
0.899	1.211	-1.878			
0.928	1.242	-1.958			
0.956	1.249	-1.978			
1.000	1.089	-1.537			

Table E.28: HS1C Cascade Blade Loading at $M_2 = 1.06$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.991	-1.236	0.063	0.297	0.798
0.031	1.102	-1.570	0.115	0.270	0.848
0.061	0.893	-0.925	0.175	0.260	0.865
0.104	0.927	-1.036	0.231	0.277	0.835
0.155	0.957	-1.130	0.296	0.282	0.826
0.209	0.953	-1.119	0.361	0.298	0.797
0.265	0.957	-1.131	0.424	0.325	0.742
0.327	0.987	-1.224	0.483	0.341	0.708
0.386	1.030	-1.356	0.540	0.362	0.663
0.443	1.094	-1.546	0.598	0.399	0.575
0.497	1.157	-1.726	0.648	0.424	0.514
0.546	1.230	-1.923	0.704	0.454	0.435
0.591	1.225	-1.911	0.754	0.495	0.325
0.631	1.167	-1.755	0.807	0.535	0.208
0.667	1.113	-1.601	0.859	0.588	0.052
0.699	1.127	-1.642	0.906	0.662	-0.181
0.733	1.179	-1.788	1.000	1.170	-1.761
0.763	1.236	-1.939			
0.798	1.326	-2.163			
0.835	1.433	-2.407			
0.871	1.483	-2.511			
0.899	1.297	-2.094			
0.928	1.293	-2.083			
0.956	1.304	-2.110			
1.000	1.191	-1.820			

Table E.29: HS1C Cascade Blade Loading at $M_2 = 1.15$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	1.002	-1.252	0.063	0.285	0.824
0.031	1.109	-1.570	0.115	0.269	0.852
0.061	0.877	-0.862	0.175	0.269	0.852
0.104	0.924	-1.011	0.231	0.282	0.830
0.155	0.955	-1.107	0.296	0.278	0.836
0.209	0.949	-1.088	0.361	0.290	0.814
0.265	0.959	-1.121	0.424	0.322	0.751
0.327	0.989	-1.214	0.483	0.337	0.720
0.386	1.031	-1.341	0.540	0.361	0.669
0.443	1.094	-1.529	0.598	0.391	0.599
0.497	1.155	-1.700	0.648	0.416	0.539
0.546	1.226	-1.893	0.704	0.452	0.447
0.591	1.219	-1.874	0.754	0.496	0.328
0.631	1.161	-1.718	0.807	0.538	0.209
0.667	1.116	-1.592	0.859	0.593	0.043
0.699	1.134	-1.642	0.906	0.663	-0.174
0.733	1.178	-1.764	1.000	1.296	-2.069
0.763	1.232	-1.909			
0.798	1.324	-2.137			
0.835	1.448	-2.413			
0.871	1.569	-2.648			
0.899	1.650	-2.786			
0.928	1.606	-2.713			
0.956	1.476	-2.471			
1.000	1.292	-2.059			

Table E.30: HS1C Cascade Blade Loading at $M_2 = 1.19$, and $i - i_{des} = +5.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.533	0.468	430	1.07	0.219	50.3	0.335	0.317
0.715	0.596	583	1.09	0.175	50.9	0.292	0.264
0.855	0.663	712	1.09	0.128	52.1	0.231	0.198
0.983	0.696	828	1.08	0.091	53.2	0.181	0.147
1.067	0.702	908	1.07	0.078	53.7	0.169	0.131
1.157	0.702	980	1.05	0.069	54.0	0.167	0.124
1.223	0.700	1021	1.03	0.083	53.5	0.220	0.159

Table E.31: HS1C Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = +10.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.472	-0.124	0.806	0.858
0.645	-0.082	0.694	0.756
0.782	-0.053	0.612	0.667
0.917	-0.012	0.543	0.581
1.007	-0.006	0.492	0.524
1.091	-0.020	0.437	0.474
1.161	-0.007	0.395	0.434

Table E.32: HS1C Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = +10.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^{-3})$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.61	0.53	292	1.07	49.2	0.445	0.068
0.60	0.60	0.53	340	1.08	49.4	0.406	0.063
0.60	0.60	0.52	431	1.09	49.9	0.355	0.056
0.60	0.60	0.52	496	1.09	50.4	0.310	0.050
0.60	0.60	0.52	505	1.09	50.3	0.307	0.049
0.60	0.60	0.52	585	1.11	50.2	0.297	0.048
0.60	0.60	0.52	674	1.10	50.3	0.297	0.048
0.60	0.60	0.52	721	1.10	50.2	0.309	0.050
0.85	0.85	0.67	541	1.08	51.4	0.287	0.076
0.85	0.85	0.67	599	1.08	51.6	0.272	0.072
0.85	0.85	0.67	663	1.09	51.9	0.249	0.067
0.85	0.85	0.66	714	1.09	52.0	0.233	0.063
0.85	0.85	0.66	827	1.09	51.9	0.245	0.066
0.85	0.85	0.66	874	1.09	51.8	0.243	0.065
0.85	0.85	0.66	956	1.10	51.9	0.235	0.064
0.85	0.85	0.66	1034	1.10	51.8	0.230	0.062
0.95	0.95	0.70	641	1.08	52.5	0.215	0.066
0.95	0.95	0.69	698	1.09	52.7	0.202	0.062
0.95	0.95	0.69	805	1.09	52.9	0.198	0.062
0.95	0.95	0.69	854	1.09	52.8	0.200	0.062
0.95	0.95	0.69	959	1.09	52.8	0.200	0.062
0.95	0.95	0.68	1012	1.09	52.8	0.197	0.061
1.05	1.05	0.70	725	1.08	53.1	0.178	0.060
1.05	1.05	0.70	790	1.09	53.3	0.168	0.057
1.05	1.05	0.70	867	1.08	53.3	0.181	0.061
1.05	1.05	0.69	914	1.08	53.2	0.177	0.059
1.05	1.05	0.69	1047	1.09	53.3	0.173	0.058

Table E.33: HS1C Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = +10.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	0.892	-1.900	0.063	0.198	0.860
0.031	0.848	-1.689	0.115	0.195	0.866
0.061	0.671	-0.852	0.175	0.199	0.859
0.104	0.679	-0.886	0.231	0.219	0.818
0.155	0.682	-0.902	0.296	0.218	0.819
0.209	0.668	-0.836	0.361	0.226	0.802
0.265	0.661	-0.806	0.424	0.242	0.765
0.327	0.666	-0.828	0.483	0.252	0.742
0.386	0.669	-0.840	0.540	0.265	0.708
0.443	0.667	-0.832	0.598	0.292	0.637
0.497	0.672	-0.855	0.648	0.307	0.596
0.546	0.645	-0.728	0.704	0.318	0.562
0.591	0.616	-0.597	0.754	0.347	0.473
0.631	0.602	-0.534	0.807	0.365	0.415
0.667	0.591	-0.484	0.859	0.386	0.344
0.699	0.589	-0.475	0.906	0.423	0.211
0.733	0.580	-0.435	1.000	0.547	-0.289
0.763	0.577	-0.423			
0.798	0.589	-0.477			
0.835	0.567	-0.376			
0.871	0.571	-0.395			
0.899	0.580	-0.434			
0.928	0.557	-0.335			
0.956	0.556	-0.328			
1.000	0.565	-0.367			

Table E.34: HS1C Cascade Blade Loading at $M_2 = 0.47$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	1.187	-1.826	0.063	0.225	0.919
0.031	1.192	-1.840	0.115	0.222	0.924
0.061	1.054	-1.444	0.175	0.228	0.916
0.104	0.880	-0.895	0.231	0.256	0.871
0.155	0.868	-0.859	0.296	0.261	0.862
0.209	0.856	-0.817	0.361	0.267	0.851
0.265	0.846	-0.785	0.424	0.291	0.807
0.327	0.853	-0.808	0.483	0.298	0.793
0.386	0.864	-0.844	0.540	0.313	0.764
0.443	0.874	-0.878	0.598	0.345	0.697
0.497	0.884	-0.911	0.648	0.357	0.670
0.546	0.865	-0.849	0.704	0.382	0.612
0.591	0.828	-0.727	0.754	0.418	0.525
0.631	0.804	-0.649	0.807	0.445	0.455
0.667	0.788	-0.597	0.859	0.486	0.344
0.699	0.787	-0.595	0.906	0.539	0.192
0.733	0.771	-0.542	1.000	0.724	-0.386
0.763	0.772	-0.545			
0.798	0.792	-0.608			
0.835	0.753	-0.481			
0.871	0.768	-0.531			
0.899	0.778	-0.564			
0.928	0.742	-0.446			
0.956	0.741	-0.442			
1.000	0.742	-0.446			

Table E.35: HS1C Cascade Blade Loading at $M_2 = 0.65$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	1.232	-1.495	0.063	0.244	0.942
0.031	1.348	-1.743	0.115	0.242	0.943
0.061	1.332	-1.711	0.175	0.238	0.949
0.104	1.092	-1.161	0.231	0.261	0.917
0.155	0.986	-0.884	0.296	0.267	0.908
0.209	0.951	-0.788	0.361	0.277	0.893
0.265	0.916	-0.693	0.424	0.306	0.844
0.327	0.932	-0.738	0.483	0.320	0.821
0.386	0.943	-0.768	0.540	0.342	0.781
0.443	0.966	-0.829	0.598	0.375	0.718
0.497	0.999	-0.919	0.648	0.395	0.676
0.546	0.994	-0.904	0.704	0.424	0.613
0.591	0.963	-0.821	0.754	0.463	0.527
0.631	0.946	-0.774	0.807	0.492	0.457
0.667	0.927	-0.723	0.859	0.539	0.342
0.699	0.923	-0.713	0.906	0.605	0.169
0.733	0.911	-0.680	1.000	0.857	-0.529
0.763	0.914	-0.688			
0.798	0.934	-0.744			
0.835	0.905	-0.663			
0.871	0.921	-0.707			
0.899	0.933	-0.741			
0.928	0.884	-0.606			
0.956	0.881	-0.597			
1.000	0.868	-0.561			

Table E.36: HS1C Cascade Blade Loading at $M_2 = 0.78$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	1.236	-1.294	0.063	0.274	0.927
0.031	1.399	-1.609	0.115	0.261	0.946
0.061	1.398	-1.607	0.175	0.265	0.940
0.104	1.205	-1.229	0.231	0.285	0.912
0.155	1.089	-0.968	0.296	0.288	0.907
0.209	1.024	-0.814	0.361	0.300	0.889
0.265	0.977	-0.699	0.424	0.327	0.846
0.327	0.982	-0.710	0.483	0.340	0.824
0.386	0.981	-0.708	0.540	0.367	0.776
0.443	1.012	-0.784	0.598	0.406	0.702
0.497	1.052	-0.881	0.648	0.427	0.659
0.546	1.080	-0.947	0.704	0.457	0.597
0.591	1.057	-0.894	0.754	0.497	0.510
0.631	1.046	-0.868	0.807	0.532	0.429
0.667	1.031	-0.830	0.859	0.577	0.324
0.699	1.033	-0.835	0.906	0.648	0.148
0.733	1.034	-0.837	1.000	0.970	-0.682
0.763	1.043	-0.860			
0.798	1.074	-0.935			
0.835	1.052	-0.882			
0.871	1.066	-0.915			
0.899	1.076	-0.939			
0.928	1.020	-0.803			
0.956	1.012	-0.785			
1.000	0.976	-0.695			

Table E.37: HS1C Cascade Blade Loading at $M_2 = 0.92$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	1.236	-1.247	0.063	0.288	0.916
0.031	1.433	-1.612	0.115	0.279	0.928
0.061	1.418	-1.588	0.175	0.279	0.928
0.104	1.228	-1.230	0.231	0.304	0.892
0.155	1.123	-1.003	0.296	0.303	0.893
0.209	1.051	-0.839	0.361	0.310	0.882
0.265	0.998	-0.711	0.424	0.332	0.846
0.327	0.998	-0.711	0.483	0.344	0.826
0.386	0.998	-0.711	0.540	0.366	0.788
0.443	1.038	-0.808	0.598	0.408	0.709
0.497	1.077	-0.898	0.648	0.434	0.658
0.546	1.105	-0.964	0.704	0.466	0.593
0.591	1.094	-0.937	0.754	0.508	0.501
0.631	1.092	-0.933	0.807	0.543	0.421
0.667	1.084	-0.916	0.859	0.595	0.300
0.699	1.097	-0.945	0.906	0.666	0.126
0.733	1.117	-0.991	1.000	1.061	-0.862
0.763	1.126	-1.011			
0.798	1.149	-1.062			
0.835	1.152	-1.067			
0.871	1.199	-1.168			
0.899	1.245	-1.264			
0.928	1.193	-1.156			
0.956	1.147	-1.057			
1.000	1.060	-0.859			

Table E.38: HS1C Cascade Blade Loading at $M_2 = 1.01$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	1.233	-1.218	0.063	0.295	0.908
0.031	1.449	-1.615	0.115	0.280	0.931
0.061	1.435	-1.591	0.175	0.278	0.932
0.104	1.234	-1.220	0.231	0.296	0.907
0.155	1.119	-0.976	0.296	0.302	0.898
0.209	1.052	-0.824	0.361	0.305	0.892
0.265	1.001	-0.701	0.424	0.338	0.840
0.327	0.999	-0.697	0.483	0.352	0.817
0.386	1.001	-0.702	0.540	0.371	0.783
0.443	1.036	-0.786	0.598	0.413	0.706
0.497	1.083	-0.894	0.648	0.442	0.649
0.546	1.117	-0.971	0.704	0.468	0.594
0.591	1.114	-0.964	0.754	0.508	0.507
0.631	1.113	-0.962	0.807	0.545	0.425
0.667	1.113	-0.963	0.859	0.596	0.305
0.699	1.136	-1.013	0.906	0.672	0.120
0.733	1.178	-1.104	1.000	1.154	-1.053
0.763	1.229	-1.210			
0.798	1.311	-1.372			
0.835	1.313	-1.375			
0.871	1.261	-1.275			
0.899	1.287	-1.325			
0.928	1.295	-1.341			
0.956	1.295	-1.341			
1.000	1.154	-1.053			

Table E.39: HS1C Cascade Blade Loading at $M_2 = 1.09$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.004	1.234	-1.241	0.063	0.282	0.924
0.031	1.451	-1.641	0.115	0.278	0.930
0.061	1.440	-1.622	0.175	0.275	0.934
0.104	1.229	-1.232	0.231	0.292	0.909
0.155	1.117	-0.989	0.296	0.292	0.910
0.209	1.050	-0.834	0.361	0.302	0.894
0.265	0.997	-0.708	0.424	0.335	0.841
0.327	0.998	-0.710	0.483	0.347	0.821
0.386	1.004	-0.726	0.540	0.373	0.775
0.443	1.030	-0.787	0.598	0.408	0.711
0.497	1.084	-0.915	0.648	0.430	0.666
0.546	1.117	-0.989	0.704	0.458	0.608
0.591	1.113	-0.980	0.754	0.508	0.501
0.631	1.111	-0.975	0.807	0.543	0.422
0.667	1.108	-0.968	0.859	0.594	0.301
0.699	1.135	-1.030	0.906	0.667	0.124
0.733	1.177	-1.122	1.000	1.217	-1.207
0.763	1.235	-1.243			
0.798	1.341	-1.450			
0.835	1.433	-1.612			
0.871	1.545	-1.787			
0.899	1.523	-1.755			
0.928	1.362	-1.488			
0.956	1.349	-1.464			
1.000	1.233	-1.239			

Table E.40: HS1C Cascade Blade Loading at $M_2 = 1.16$, and $i - i_{des} = +10.0^\circ$.

Appendix F

HS1D Cascade Results

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.531	0.333	507	0.98	0.044	56.6	0.056	0.052
0.725	0.405	690	0.98	0.035	57.1	0.051	0.045
0.867	0.428	820	0.98	0.028	57.6	0.048	0.040
0.996	0.442	920	0.98	0.039	57.2	0.076	0.060
1.100	0.443	1003	0.97	0.042	57.0	0.092	0.069
1.191	0.439	1064	0.97	0.050	56.3	0.124	0.090
1.298	0.439	1106	0.97	0.058	54.5	0.166	0.114

Table F.1: HS1D Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = -10.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.491	-0.056	0.834	0.848
0.696	-0.054	0.701	0.723
0.841	-0.049	0.605	0.630
0.964	-0.048	0.519	0.550
1.047	-0.217	0.402	0.499
1.147	-0.163	0.357	0.441
1.234	-0.174	0.300	0.394

Table F.2: HS1D Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = -10.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^{-3})$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.60	0.36	401	0.98	56.3	0.075	0.014
0.60	0.60	0.37	511	0.97	56.6	0.066	0.012
0.60	0.60	0.36	583	0.98	56.7	0.050	0.009
0.60	0.60	0.35	867	1.00	56.7	0.040	0.008
0.85	0.85	0.43	701	0.97	57.5	0.053	0.016
0.85	0.85	0.43	944	0.98	57.5	0.054	0.016
0.85	0.85	0.43	1100	0.98	57.5	0.057	0.017
0.95	0.95	0.44	723	0.96	57.7	0.066	0.022
0.95	0.95	0.44	881	0.97	57.8	0.057	0.019
0.95	0.95	0.44	1081	0.97	57.8	0.057	0.019
1.05	1.05	0.44	879	0.98	56.7	0.088	0.032
1.05	1.05	0.44	985	0.98	56.7	0.092	0.034
1.05	1.05	0.43	1265	1.00	56.6	0.084	0.031

Table F.3: HS1D Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = -10.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.361	-0.219	0.061	0.223	0.533
0.026	0.434	-0.727	0.110	0.175	0.722
0.060	0.511	-1.334	0.168	0.175	0.720
0.105	0.570	-1.837	0.223	0.181	0.699
0.161	0.615	-2.236	0.289	0.192	0.661
0.216	0.645	-2.505	0.355	0.207	0.602
0.275	0.662	-2.658	0.418	0.226	0.520
0.339	0.673	-2.764	0.478	0.249	0.418
0.398	0.664	-2.675	0.536	0.275	0.287
0.453	0.644	-2.492	0.595	0.302	0.140
0.505	0.614	-2.222	0.645	0.323	0.019
0.551	0.596	-2.060	0.701	0.341	-0.091
0.593	0.575	-1.881	0.752	0.361	-0.218
0.632	0.555	-1.708	0.805	0.383	-0.368
0.668	0.546	-1.623	0.857	0.406	-0.528
0.698	0.545	-1.621	0.905	0.434	-0.729
0.730	0.543	-1.605	1.000	0.518	-1.392
0.763	0.544	-1.609			
0.797	0.542	-1.593			
0.832	0.544	-1.612			
0.864	0.543	-1.601			
0.895	0.543	-1.602			
0.924	0.546	-1.631			
0.954	0.539	-1.566			
1.000	0.516	-1.377			

Table F.4: HS1D Cascade Blade Loading at $M_2 = 0.49$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.435	-0.145	0.061	0.303	0.439
0.026	0.539	-0.706	0.110	0.248	0.634
0.060	0.659	-1.423	0.168	0.239	0.660
0.105	0.744	-1.959	0.223	0.244	0.647
0.161	0.819	-2.435	0.289	0.257	0.602
0.216	0.862	-2.707	0.355	0.281	0.521
0.275	0.889	-2.879	0.418	0.301	0.449
0.339	0.919	-3.061	0.478	0.328	0.340
0.398	0.909	-3.003	0.536	0.364	0.187
0.453	0.885	-2.855	0.595	0.395	0.046
0.505	0.855	-2.664	0.645	0.425	-0.097
0.551	0.834	-2.531	0.701	0.456	-0.255
0.593	0.804	-2.336	0.752	0.490	-0.435
0.632	0.769	-2.115	0.805	0.520	-0.599
0.668	0.762	-2.072	0.857	0.558	-0.818
0.698	0.764	-2.083	0.905	0.599	-1.058
0.730	0.766	-2.099	1.000	0.736	-1.908
0.763	0.771	-2.127			
0.797	0.772	-2.133			
0.832	0.780	-2.187			
0.864	0.783	-2.207			
0.895	0.782	-2.202			
0.924	0.792	-2.263			
0.954	0.779	-2.183			
1.000	0.731	-1.874			

Table F.5: HS1D Cascade Blade Loading at $M_2 = 0.70$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.455	-0.078	0.061	0.335	0.414
0.026	0.583	-0.697	0.110	0.255	0.672
0.060	0.707	-1.362	0.168	0.246	0.698
0.105	0.815	-1.961	0.223	0.254	0.675
0.161	0.899	-2.422	0.289	0.267	0.637
0.216	0.952	-2.711	0.355	0.290	0.565
0.275	0.987	-2.894	0.418	0.322	0.460
0.339	1.037	-3.156	0.478	0.355	0.340
0.398	1.036	-3.151	0.536	0.392	0.196
0.453	1.012	-3.025	0.595	0.434	0.017
0.505	0.977	-2.843	0.645	0.467	-0.131
0.551	0.967	-2.790	0.701	0.505	-0.310
0.593	0.946	-2.676	0.752	0.542	-0.494
0.632	0.899	-2.422	0.805	0.587	-0.721
0.668	0.890	-2.372	0.857	0.632	-0.956
0.698	0.896	-2.405	0.905	0.681	-1.219
0.730	0.905	-2.452	1.000	0.884	-2.337
0.763	0.919	-2.531			
0.797	0.926	-2.568			
0.832	0.946	-2.677			
0.864	0.958	-2.740			
0.895	0.965	-2.779			
0.924	0.993	-2.926			
0.954	0.970	-2.803			
1.000	0.878	-2.308			

Table F.6: HS1D Cascade Blade Loading at $M_2 = 0.84$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.461	-0.053	0.061	0.353	0.380
0.026	0.591	-0.664	0.110	0.280	0.619
0.060	0.727	-1.365	0.168	0.266	0.659
0.105	0.844	-1.985	0.223	0.268	0.653
0.161	0.932	-2.448	0.289	0.292	0.583
0.216	0.993	-2.762	0.355	0.320	0.492
0.275	1.028	-2.934	0.418	0.349	0.392
0.339	1.085	-3.215	0.478	0.378	0.284
0.398	1.092	-3.244	0.536	0.413	0.148
0.453	1.075	-3.167	0.595	0.458	-0.043
0.505	1.038	-2.986	0.645	0.489	-0.180
0.551	1.037	-2.979	0.701	0.530	-0.370
0.593	1.029	-2.943	0.752	0.566	-0.544
0.632	0.988	-2.736	0.805	0.616	-0.792
0.668	0.980	-2.695	0.857	0.671	-1.072
0.698	0.992	-2.758	0.905	0.727	-1.368
0.730	0.992	-2.755	1.000	1.017	-2.880
0.763	1.009	-2.840			
0.797	1.016	-2.874			
0.832	1.052	-3.055			
0.864	1.081	-3.193			
0.895	1.098	-3.275			
0.924	1.136	-3.455			
0.954	1.216	-3.811			
1.000	1.014	-2.868			

Table F.7: HS1D Cascade Blade Loading at $M_2 = 0.96$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.462	-0.055	0.061	0.348	0.400
0.026	0.590	-0.654	0.110	0.279	0.625
0.060	0.727	-1.355	0.168	0.265	0.663
0.105	0.845	-1.982	0.223	0.269	0.654
0.161	0.932	-2.433	0.289	0.285	0.608
0.216	0.995	-2.756	0.355	0.310	0.527
0.275	1.035	-2.956	0.418	0.336	0.439
0.339	1.092	-3.232	0.478	0.371	0.314
0.398	1.100	-3.269	0.536	0.411	0.159
0.453	1.083	-3.186	0.595	0.456	-0.030
0.505	1.051	-3.031	0.645	0.493	-0.193
0.551	1.055	-3.050	0.701	0.530	-0.365
0.593	1.056	-3.055	0.752	0.570	-0.557
0.632	1.015	-2.854	0.805	0.618	-0.795
0.668	1.026	-2.912	0.857	0.668	-1.049
0.698	1.066	-3.105	0.905	0.727	-1.358
0.730	1.129	-3.402	1.000	1.238	-3.885
0.763	1.213	-3.778			
0.797	1.254	-3.952			
0.832	1.121	-3.364			
0.864	1.115	-3.337			
0.895	1.134	-3.428			
0.924	1.169	-3.586			
0.954	1.254	-3.951			
1.000	1.219	-3.805			

Table F.8: HS1D Cascade Blade Loading at $M_2 = 1.05$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.468	-0.088	0.061	0.355	0.367
0.026	0.595	-0.693	0.110	0.281	0.615
0.060	0.732	-1.403	0.168	0.272	0.642
0.105	0.849	-2.024	0.223	0.273	0.639
0.161	0.937	-2.490	0.289	0.286	0.599
0.216	0.996	-2.791	0.355	0.316	0.504
0.275	1.040	-3.008	0.418	0.345	0.404
0.339	1.100	-3.302	0.478	0.378	0.282
0.398	1.106	-3.327	0.536	0.418	0.122
0.453	1.085	-3.228	0.595	0.457	-0.040
0.505	1.046	-3.037	0.645	0.492	-0.198
0.551	1.052	-3.070	0.701	0.536	-0.404
0.593	1.052	-3.070	0.752	0.575	-0.592
0.632	1.008	-2.852	0.805	0.621	-0.825
0.668	1.026	-2.939	0.857	0.674	-1.098
0.698	1.068	-3.145	0.905	0.733	-1.406
0.730	1.126	-3.425	1.000	1.298	-4.171
0.763	1.213	-3.817			
0.797	1.307	-4.209			
0.832	1.420	-4.634			
0.864	1.530	-4.998			
0.895	1.462	-4.778			
0.924	1.322	-4.266			
0.954	1.345	-4.357			
1.000	1.309	-4.215			

Table F.9: HS1D Cascade Blade Loading at $M_2 = 1.15$, and $i - i_{des} = -10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.469	-0.084	0.061	0.354	0.377
0.026	0.597	-0.686	0.110	0.279	0.624
0.060	0.731	-1.373	0.168	0.269	0.655
0.105	0.847	-1.982	0.223	0.271	0.650
0.161	0.933	-2.430	0.289	0.292	0.585
0.216	0.996	-2.754	0.355	0.318	0.504
0.275	1.033	-2.937	0.418	0.350	0.392
0.339	1.096	-3.237	0.478	0.379	0.286
0.398	1.102	-3.265	0.536	0.415	0.145
0.453	1.082	-3.173	0.595	0.456	-0.026
0.505	1.040	-2.968	0.645	0.488	-0.168
0.551	1.053	-3.033	0.701	0.528	-0.352
0.593	1.053	-3.033	0.752	0.568	-0.544
0.632	1.008	-2.813	0.805	0.617	-0.785
0.668	1.022	-2.881	0.857	0.672	-1.065
0.698	1.062	-3.075	0.905	0.732	-1.378
0.730	1.125	-3.374	1.000	1.476	-4.768
0.763	1.208	-3.746			
0.797	1.305	-4.149			
0.832	1.414	-4.557			
0.864	1.526	-4.929			
0.895	1.577	-5.080			
0.924	1.389	-4.467			
0.954	1.428	-4.606			
1.000	1.432	-4.621			

Table F.10: HS1D Cascade Blade Loading at $M_2 = 1.23$, and $i - i_{des} = -10.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.520	0.381	502	1.00	0.060	55.4	0.076	0.071
0.711	0.483	686	0.99	0.041	56.0	0.060	0.053
0.857	0.519	819	1.00	0.030	56.6	0.050	0.042
0.984	0.542	923	0.97	0.039	56.7	0.074	0.059
1.075	0.543	996	0.98	0.044	55.9	0.094	0.072
1.180	0.546	1067	0.96	0.045	56.0	0.110	0.080
1.248	0.543	1124	0.96	0.055	54.7	0.147	0.104

Table F.11: HS1D Cascade Performance Data at Design Incidence, $i - i_{des} = 0.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.563	-0.045	0.788	0.806
0.742	-0.038	0.673	0.694
0.882	-0.031	0.583	0.603
0.982	-0.037	0.512	0.540
1.068	-0.146	0.416	0.471
1.152	-0.128	0.370	0.420
1.214	-0.182	0.310	0.382

Table F.12: HS1D Cascade Base Pressure Data at Design Incidence, $i - i_{des} = 0.0^\circ$.

M ₂ nominal	M ₂ actual	M ₁	Re ₂ (x10 ⁻³)	AVDR	α ₂	Y _t	Y _{pwc}
0.60	0.59	0.42	340	1.03	54.1	0.111	0.019
0.60	0.60	0.42	399	1.03	54.9	0.078	0.015
0.60	0.60	0.42	504	1.01	55.4	0.070	0.013
0.60	0.60	0.42	577	1.00	55.6	0.065	0.013
0.60	0.60	0.43	700	1.00	55.7	0.059	0.011
0.60	0.60	0.43	911	0.99	55.6	0.071	0.013
0.60	0.60	0.43	925	0.99	55.6	0.066	0.012
0.85	0.85	0.52	625	1.00	56.2	0.069	0.021
0.85	0.85	0.52	689	0.99	56.4	0.065	0.020
0.85	0.85	0.52	756	1.00	56.3	0.059	0.018
0.85	0.85	0.51	807	1.01	56.5	0.053	0.014
0.85	0.85	0.51	938	1.00	56.5	0.056	0.017
0.85	0.85	0.52	997	1.00	56.4	0.053	0.016
0.85	0.85	0.52	1113	1.00	56.5	0.059	0.018
0.95	0.95	0.54	718	0.98	56.7	0.071	0.024
0.95	0.95	0.53	793	0.99	56.8	0.063	0.021
0.95	0.95	0.52	886	1.00	56.8	0.064	0.018
0.95	0.95	0.53	938	0.99	56.9	0.061	0.021
0.95	0.95	0.53	1059	0.98	56.9	0.061	0.020
0.95	0.95	0.53	1149	0.99	56.9	0.057	0.019
1.05	1.05	0.54	866	1.00	55.9	0.092	0.033
1.05	1.05	0.54	983	0.99	55.8	0.094	0.037
1.05	1.05	0.53	1266	1.00	55.7	0.094	0.035

Table F.13: HS1D Cascade Performance Data at Design Incidence at Design Incidence, $i - i_{des} = 0.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.617	-1.064	0.061	0.186	0.822
0.026	0.650	-1.258	0.110	0.196	0.798
0.060	0.723	-1.689	0.168	0.200	0.787
0.105	0.761	-1.923	0.223	0.213	0.754
0.161	0.782	-2.051	0.289	0.226	0.719
0.216	0.786	-2.070	0.355	0.242	0.672
0.275	0.776	-2.015	0.418	0.268	0.592
0.339	0.778	-2.022	0.478	0.291	0.514
0.398	0.762	-1.927	0.536	0.314	0.430
0.453	0.733	-1.753	0.595	0.348	0.297
0.505	0.693	-1.512	0.645	0.365	0.225
0.551	0.671	-1.383	0.701	0.387	0.131
0.593	0.646	-1.234	0.752	0.412	0.018
0.632	0.625	-1.110	0.805	0.437	-0.099
0.668	0.617	-1.064	0.857	0.465	-0.239
0.698	0.612	-1.038	0.905	0.495	-0.392
0.730	0.614	-1.044	1.000	0.595	-0.941
0.763	0.614	-1.047			
0.797	0.613	-1.039			
0.832	0.617	-1.063			
0.864	0.616	-1.057			
0.895	0.615	-1.053			
0.924	0.619	-1.078			
0.954	0.611	-1.032			
1.000	0.594	-0.934			

Table F.14: HS1D Cascade Blade Loading at $M_2 = 0.56$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.735	-0.991	0.061	0.244	0.788
0.026	0.794	-1.253	0.110	0.238	0.801
0.060	0.899	-1.713	0.168	0.236	0.805
0.105	0.963	-1.993	0.223	0.249	0.777
0.161	0.984	-2.083	0.289	0.262	0.746
0.216	0.986	-2.090	0.355	0.284	0.693
0.275	0.976	-2.048	0.418	0.310	0.624
0.339	0.981	-2.067	0.478	0.337	0.549
0.398	0.964	-1.994	0.536	0.371	0.447
0.453	0.932	-1.857	0.595	0.412	0.314
0.505	0.892	-1.683	0.645	0.439	0.218
0.551	0.869	-1.585	0.701	0.476	0.086
0.593	0.839	-1.451	0.752	0.510	-0.045
0.632	0.809	-1.318	0.805	0.544	-0.178
0.668	0.802	-1.285	0.857	0.583	-0.338
0.698	0.800	-1.280	0.905	0.626	-0.517
0.730	0.807	-1.309	1.000	0.773	-1.157
0.763	0.812	-1.331			
0.797	0.813	-1.338			
0.832	0.823	-1.380			
0.864	0.822	-1.374			
0.895	0.820	-1.366			
0.924	0.832	-1.418			
0.954	0.812	-1.332			
1.000	0.775	-1.166			

Table F.15: HS1D Cascade Blade Loading at $M_2 = 0.74$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.780	-0.958	0.061	0.271	0.766
0.026	0.845	-1.218	0.110	0.260	0.791
0.060	0.976	-1.732	0.168	0.252	0.807
0.105	1.065	-2.067	0.223	0.265	0.779
0.161	1.088	-2.149	0.289	0.279	0.750
0.216	1.082	-2.127	0.355	0.309	0.678
0.275	1.070	-2.085	0.418	0.342	0.595
0.339	1.079	-2.116	0.478	0.375	0.504
0.398	1.063	-2.059	0.536	0.411	0.398
0.453	1.037	-1.964	0.595	0.456	0.257
0.505	0.997	-1.812	0.645	0.488	0.150
0.551	0.986	-1.773	0.701	0.526	0.019
0.593	0.956	-1.656	0.752	0.566	-0.128
0.632	0.922	-1.526	0.805	0.611	-0.296
0.668	0.919	-1.513	0.857	0.652	-0.455
0.698	0.930	-1.554	0.905	0.702	-0.650
0.730	0.940	-1.593	1.000	0.909	-1.473
0.763	0.951	-1.638			
0.797	0.959	-1.666			
0.832	0.986	-1.771			
0.864	0.992	-1.794			
0.895	1.002	-1.831			
0.924	1.029	-1.933			
0.954	0.994	-1.802			
1.000	0.914	-1.492			

Table F.16: HS1D Cascade Blade Loading at $M_2 = 0.88$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.782	-0.903	0.061	0.283	0.752
0.026	0.854	-1.181	0.110	0.262	0.797
0.060	1.001	-1.738	0.168	0.255	0.812
0.105	1.104	-2.102	0.223	0.266	0.790
0.161	1.119	-2.156	0.289	0.285	0.749
0.216	1.113	-2.135	0.355	0.319	0.671
0.275	1.098	-2.083	0.418	0.356	0.577
0.339	1.117	-2.149	0.478	0.387	0.492
0.398	1.108	-2.118	0.536	0.423	0.386
0.453	1.080	-2.020	0.595	0.468	0.247
0.505	1.040	-1.880	0.645	0.499	0.143
0.551	1.042	-1.884	0.701	0.534	0.025
0.593	1.027	-1.831	0.752	0.573	-0.114
0.632	0.992	-1.702	0.805	0.621	-0.288
0.668	1.001	-1.736	0.857	0.677	-0.497
0.698	1.016	-1.790	0.905	0.734	-0.717
0.730	1.025	-1.823	1.000	1.036	-1.863
0.763	1.028	-1.836			
0.797	1.035	-1.859			
0.832	1.072	-1.991			
0.864	1.097	-2.079			
0.895	1.118	-2.153			
0.924	1.163	-2.302			
0.954	1.237	-2.539			
1.000	1.026	-1.829			

Table F.17: HS1D Cascade Blade Loading at $M_2 = 0.98$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.788	-0.934	0.061	0.257	0.806
0.026	0.856	-1.199	0.110	0.235	0.849
0.060	1.003	-1.753	0.168	0.217	0.882
0.105	1.109	-2.132	0.223	0.236	0.848
0.161	1.129	-2.200	0.289	0.288	0.740
0.216	1.114	-2.149	0.355	0.314	0.680
0.275	1.102	-2.108	0.418	0.347	0.598
0.339	1.125	-2.187	0.478	0.382	0.502
0.398	1.112	-2.144	0.536	0.428	0.367
0.453	1.086	-2.053	0.595	0.471	0.231
0.505	1.048	-1.917	0.645	0.502	0.131
0.551	1.060	-1.962	0.701	0.539	0.005
0.593	1.046	-1.910	0.752	0.581	-0.147
0.632	1.005	-1.761	0.805	0.629	-0.324
0.668	1.027	-1.842	0.857	0.678	-0.509
0.698	1.070	-1.995	0.905	0.739	-0.741
0.730	1.130	-2.203	1.000	1.223	-2.509
0.763	1.213	-2.478			
0.797	1.260	-2.622			
0.832	1.173	-2.347			
0.864	1.132	-2.212			
0.895	1.160	-2.304			
0.924	1.201	-2.441			
0.954	1.273	-2.663			
1.000	1.194	-2.416			

Table F.18: HS1D Cascade Blade Loading at $M_2 = 1.07$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.786	-0.913	0.061	0.261	0.800
0.026	0.853	-1.175	0.110	0.248	0.827
0.060	0.999	-1.725	0.168	0.245	0.833
0.105	1.107	-2.110	0.223	0.258	0.807
0.161	1.120	-2.152	0.289	0.274	0.774
0.216	1.106	-2.105	0.355	0.307	0.700
0.275	1.095	-2.069	0.418	0.338	0.623
0.339	1.117	-2.144	0.478	0.371	0.537
0.398	1.106	-2.106	0.536	0.408	0.432
0.453	1.086	-2.036	0.595	0.454	0.291
0.505	1.041	-1.878	0.645	0.486	0.187
0.551	1.053	-1.920	0.701	0.530	0.042
0.593	1.038	-1.866	0.752	0.573	-0.109
0.632	0.994	-1.707	0.805	0.618	-0.273
0.668	1.017	-1.790	0.857	0.670	-0.471
0.698	1.066	-1.966	0.905	0.730	-0.698
0.730	1.123	-2.165	1.000	1.269	-2.631
0.763	1.210	-2.450			
0.797	1.300	-2.723			
0.832	1.415	-3.041			
0.864	1.524	-3.306			
0.895	1.447	-3.122			
0.924	1.343	-2.846			
0.954	1.328	-2.805			
1.000	1.281	-2.669			

Table F.19: HS1D Cascade Blade Loading at $M_2 = 1.15$, and $i - i_{des} = 0.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.792	-0.946	0.061	0.265	0.791
0.026	0.862	-1.219	0.110	0.257	0.807
0.060	1.001	-1.746	0.168	0.258	0.804
0.105	1.107	-2.125	0.223	0.269	0.782
0.161	1.126	-2.188	0.289	0.285	0.749
0.216	1.109	-2.132	0.355	0.309	0.692
0.275	1.093	-2.076	0.418	0.345	0.602
0.339	1.120	-2.169	0.478	0.375	0.522
0.398	1.108	-2.127	0.536	0.417	0.402
0.453	1.080	-2.029	0.595	0.466	0.249
0.505	1.038	-1.880	0.645	0.500	0.137
0.551	1.053	-1.935	0.701	0.536	0.013
0.593	1.036	-1.873	0.752	0.576	-0.129
0.632	1.000	-1.743	0.805	0.623	-0.300
0.668	1.023	-1.825	0.857	0.671	-0.482
0.698	1.067	-1.984	0.905	0.731	-0.713
0.730	1.127	-2.194	1.000	1.443	-3.129
0.763	1.212	-2.472			
0.797	1.306	-2.758			
0.832	1.417	-3.063			
0.864	1.523	-3.323			
0.895	1.604	-3.500			
0.924	1.440	-3.123			
0.954	1.433	-3.104			
1.000	1.409	-3.042			

Table F.20: HS1D Cascade Blade Loading at $M_2 = 1.21$, and $i - i_{des} = 0.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.527	0.417	494	1.05	0.075	54.5	0.097	0.091
0.714	0.524	671	1.05	0.054	55.2	0.080	0.071
0.858	0.578	801	1.03	0.049	55.8	0.082	0.070
0.981	0.595	908	1.02	0.045	56.2	0.086	0.069
1.072	0.596	978	1.02	0.050	55.8	0.107	0.082
1.172	0.600	1049	0.99	0.051	56.2	0.124	0.091
1.262	0.597	1093	0.99	0.062	54.8	0.168	0.118

Table F.21: HS1D Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = +5.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.539	-0.066	0.796	0.820
0.721	-0.061	0.678	0.707
0.851	-0.054	0.590	0.623
0.970	-0.018	0.524	0.547
1.050	-0.166	0.417	0.498
1.136	-0.135	0.374	0.447
1.208	-0.171	0.315	0.408

Table F.22: HS1D Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = +5.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (x10^3)$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.60	0.45	339	1.09	53.8	0.084	0.015
0.60	0.60	0.45	392	1.08	54.3	0.086	0.016
0.60	0.60	0.45	498	1.07	54.6	0.078	0.014
0.60	0.60	0.46	566	1.06	54.8	0.087	0.016
0.60	0.60	0.47	691	1.04	54.9	0.086	0.016
			0				
0.85	0.85	0.57	610	1.05	55.5	0.077	0.023
0.85	0.85	0.57	738	1.04	55.7	0.084	0.025
0.85	0.85	0.58	788	1.03	55.7	0.083	0.025
0.85	0.85	0.58	924	1.03	55.7	0.087	0.026
0.85	0.85	0.58	991	1.03	55.7	0.088	0.026
0.85	0.85	0.57	1131	1.04	55.6	0.076	0.023
			0				
0.95	0.95	0.59	700	1.03	55.9	0.081	0.027
0.95	0.95	0.59	771	1.02	56.0	0.089	0.029
0.95	0.95	0.59	865	1.03	56.0	0.084	0.028
0.95	0.95	0.59	924	1.02	56.1	0.088	0.029
0.95	0.95	0.59	1049	1.03	56.1	0.083	0.027
0.95	0.95	0.59	1131	1.04	56.0	0.074	0.024
			0				
1.05	1.05	0.60	786	1.01	56.0	0.112	0.040
1.05	1.04	0.60	846	1.01	56.0	0.108	0.039
1.05	1.05	0.60	943	1.02	56.0	0.102	0.037
1.05	1.05	0.60	1019	1.01	56.0	0.109	0.039
1.05	1.05	0.60	1206	1.02	55.9	0.109	0.039

Table F.23: HS1D Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = +5.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.752	-1.583	0.061	0.149	0.920
0.026	0.741	-1.524	0.110	0.165	0.891
0.060	0.794	-1.815	0.168	0.174	0.874
0.105	0.809	-1.897	0.223	0.195	0.828
0.161	0.812	-1.913	0.289	0.201	0.816
0.216	0.796	-1.826	0.355	0.224	0.760
0.275	0.777	-1.719	0.418	0.250	0.691
0.339	0.764	-1.651	0.478	0.271	0.629
0.398	0.730	-1.466	0.536	0.298	0.547
0.453	0.686	-1.223	0.595	0.328	0.444
0.505	0.657	-1.066	0.645	0.349	0.371
0.551	0.636	-0.956	0.701	0.368	0.297
0.593	0.612	-0.832	0.752	0.394	0.196
0.632	0.600	-0.771	0.805	0.414	0.112
0.668	0.591	-0.721	0.857	0.442	-0.009
0.698	0.590	-0.716	0.905	0.477	-0.164
0.730	0.590	-0.719	1.000	0.574	-0.637
0.763	0.592	-0.728			
0.797	0.595	-0.741			
0.832	0.595	-0.741			
0.864	0.595	-0.743			
0.895	0.601	-0.776			
0.924	0.593	-0.735			
0.954	0.591	-0.720			
1.000	0.580	-0.665			

Table F.24: HS1D Cascade Blade Loading at $M_2 = 0.54$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.929	-1.475	0.061	0.194	0.920
0.026	0.930	-1.481	0.110	0.206	0.900
0.060	1.040	-1.886	0.168	0.211	0.892
0.105	1.070	-1.996	0.223	0.232	0.856
0.161	1.056	-1.944	0.289	0.243	0.834
0.216	1.018	-1.809	0.355	0.266	0.788
0.275	0.985	-1.689	0.418	0.303	0.706
0.339	0.970	-1.631	0.478	0.327	0.648
0.398	0.936	-1.503	0.536	0.362	0.558
0.453	0.885	-1.309	0.595	0.404	0.441
0.505	0.857	-1.199	0.645	0.433	0.354
0.551	0.839	-1.132	0.701	0.461	0.266
0.593	0.808	-1.011	0.752	0.498	0.143
0.632	0.788	-0.933	0.805	0.529	0.038
0.668	0.780	-0.900	0.857	0.565	-0.087
0.698	0.781	-0.904	0.905	0.608	-0.245
0.730	0.785	-0.922	1.000	0.757	-0.813
0.763	0.787	-0.930			
0.797	0.792	-0.950			
0.832	0.796	-0.962			
0.864	0.797	-0.968			
0.895	0.806	-1.003			
0.924	0.796	-0.964			
0.954	0.790	-0.939			
1.000	0.766	-0.849			

Table F.25: HS1D Cascade Blade Loading at $M_2 = 0.72$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.003	-1.405	0.061	0.209	0.931
0.026	1.008	-1.419	0.110	0.221	0.912
0.060	1.158	-1.883	0.168	0.222	0.911
0.105	1.244	-2.127	0.223	0.243	0.876
0.161	1.215	-2.045	0.289	0.262	0.844
0.216	1.123	-1.778	0.355	0.287	0.796
0.275	1.055	-1.568	0.418	0.323	0.722
0.339	1.063	-1.594	0.478	0.357	0.644
0.398	1.044	-1.536	0.536	0.394	0.554
0.453	0.995	-1.378	0.595	0.439	0.436
0.505	0.962	-1.270	0.645	0.466	0.360
0.551	0.955	-1.248	0.701	0.501	0.259
0.593	0.925	-1.145	0.752	0.543	0.132
0.632	0.904	-1.077	0.805	0.583	0.008
0.668	0.898	-1.055	0.857	0.629	-0.142
0.698	0.902	-1.069	0.905	0.682	-0.320
0.730	0.912	-1.102	1.000	0.890	-1.029
0.763	0.918	-1.123			
0.797	0.930	-1.163			
0.832	0.943	-1.207			
0.864	0.950	-1.231			
0.895	0.963	-1.274			
0.924	0.961	-1.266			
0.954	0.945	-1.214			
1.000	0.901	-1.067			

Table F.26: HS1D Cascade Blade Loading at $M_2 = 0.85$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.015	-1.340	0.061	0.233	0.906
0.026	1.030	-1.387	0.110	0.237	0.899
0.060	1.204	-1.891	0.168	0.243	0.890
0.105	1.335	-2.226	0.223	0.264	0.853
0.161	1.304	-2.151	0.289	0.282	0.822
0.216	1.164	-1.781	0.355	0.311	0.766
0.275	1.070	-1.509	0.418	0.346	0.692
0.339	1.102	-1.602	0.478	0.375	0.624
0.398	1.087	-1.560	0.536	0.410	0.541
0.453	1.047	-1.438	0.595	0.459	0.412
0.505	1.014	-1.336	0.645	0.490	0.327
0.551	1.020	-1.357	0.701	0.525	0.225
0.593	0.998	-1.289	0.752	0.570	0.093
0.632	0.977	-1.221	0.805	0.615	-0.048
0.668	0.981	-1.233	0.857	0.667	-0.211
0.698	0.990	-1.264	0.905	0.725	-0.402
0.730	1.009	-1.321	1.000	1.004	-1.307
0.763	1.019	-1.353			
0.797	1.030	-1.388			
0.832	1.064	-1.490			
0.864	1.086	-1.557			
0.895	1.115	-1.640			
0.924	1.154	-1.752			
0.954	1.148	-1.735			
1.000	1.007	-1.315			

Table F.27: HS1D Cascade Blade Loading at $M_2 = 0.97$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.026	-1.312	0.061	0.245	0.894
0.026	1.042	-1.360	0.110	0.250	0.886
0.060	1.215	-1.848	0.168	0.251	0.884
0.105	1.368	-2.221	0.223	0.270	0.852
0.161	1.333	-2.140	0.289	0.291	0.815
0.216	1.156	-1.689	0.355	0.316	0.765
0.275	1.078	-1.469	0.418	0.355	0.684
0.339	1.113	-1.570	0.478	0.384	0.618
0.398	1.106	-1.549	0.536	0.421	0.530
0.453	1.070	-1.443	0.595	0.470	0.402
0.505	1.038	-1.350	0.645	0.502	0.314
0.551	1.052	-1.390	0.701	0.537	0.216
0.593	1.035	-1.340	0.752	0.581	0.085
0.632	1.013	-1.274	0.805	0.626	-0.051
0.668	1.033	-1.334	0.857	0.674	-0.201
0.698	1.068	-1.440	0.905	0.736	-0.398
0.730	1.127	-1.607	1.000	1.175	-1.742
0.763	1.187	-1.774			
0.797	1.147	-1.663			
0.832	1.096	-1.518			
0.864	1.116	-1.577			
0.895	1.150	-1.672			
0.924	1.196	-1.797			
0.954	1.260	-1.962			
1.000	1.191	-1.783			

Table F.28: HS1D Cascade Blade Loading at $M_2 = 1.05$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.017	-1.314	0.061	0.235	0.906
0.026	1.032	-1.360	0.110	0.237	0.903
0.060	1.210	-1.868	0.168	0.242	0.896
0.105	1.367	-2.258	0.223	0.253	0.877
0.161	1.334	-2.181	0.289	0.272	0.844
0.216	1.134	-1.661	0.355	0.295	0.802
0.275	1.080	-1.505	0.418	0.341	0.708
0.339	1.118	-1.614	0.478	0.375	0.633
0.398	1.106	-1.578	0.536	0.413	0.541
0.453	1.062	-1.451	0.595	0.461	0.418
0.505	1.030	-1.353	0.645	0.489	0.342
0.551	1.042	-1.391	0.701	0.527	0.234
0.593	1.027	-1.345	0.752	0.572	0.100
0.632	1.007	-1.284	0.805	0.614	-0.028
0.668	1.029	-1.352	0.857	0.667	-0.193
0.698	1.072	-1.479	0.905	0.731	-0.400
0.730	1.130	-1.647	1.000	1.252	-1.978
0.763	1.207	-1.860			
0.797	1.290	-2.074			
0.832	1.407	-2.348			
0.864	1.468	-2.479			
0.895	1.272	-2.029			
0.924	1.255	-1.987			
0.954	1.295	-2.086			
1.000	1.273	-2.033			

Table F.29: HS1D Cascade Blade Loading at $M_2 = 1.14$, and $i - i_{des} = +5.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.030	-1.341	0.061	0.247	0.888
0.026	1.040	-1.369	0.110	0.257	0.873
0.060	1.214	-1.863	0.168	0.255	0.876
0.105	1.372	-2.249	0.223	0.272	0.847
0.161	1.346	-2.190	0.289	0.287	0.819
0.216	1.124	-1.616	0.355	0.316	0.763
0.275	1.080	-1.489	0.418	0.348	0.696
0.339	1.115	-1.589	0.478	0.378	0.628
0.398	1.102	-1.553	0.536	0.413	0.545
0.453	1.066	-1.448	0.595	0.462	0.419
0.505	1.031	-1.344	0.645	0.496	0.325
0.551	1.045	-1.386	0.701	0.533	0.222
0.593	1.029	-1.336	0.752	0.577	0.092
0.632	1.010	-1.279	0.805	0.625	-0.055
0.668	1.033	-1.347	0.857	0.676	-0.214
0.698	1.074	-1.470	0.905	0.739	-0.418
0.730	1.136	-1.648	1.000	1.373	-2.253
0.763	1.211	-1.854			
0.797	1.299	-2.078			
0.832	1.410	-2.335			
0.864	1.507	-2.534			
0.895	1.584	-2.677			
0.924	1.449	-2.418			
0.954	1.392	-2.294			
1.000	1.399	-2.311			

Table F.30: HS1D Cascade Blade Loading at $M_2 = 1.21$, and $i - i_{des} = +5.0^\circ$.

M_{2MO}	M_1	$Re_2 (x10^{-3})$	AVDR	ξ	α_2	Y_t	Y_{pwc}
0.516	0.441	453	1.10	0.185	51.3	0.269	0.255
0.702	0.575	617	1.10	0.156	51.8	0.251	0.227
0.834	0.634	748	1.08	0.116	53.5	0.202	0.174
0.942	0.664	864	1.06	0.091	54.9	0.173	0.143
1.045	0.674	956	1.04	0.080	55.5	0.170	0.133
1.133	0.682	1028	1.03	0.081	55.3	0.191	0.144
1.226	0.679	1089	1.00	0.074	55.6	0.196	0.141

Table F.31: HS1D Cascade Performance Data at Off-Design Incidence,
 $i - i_{des} = +10.0^\circ$.

M_{2MO}	C_b	P_b/P_{o1}	P_2/P_{o1}
0.476	-0.129	0.810	0.856
0.651	-0.125	0.686	0.752
0.785	-0.088	0.606	0.666
0.888	-0.067	0.545	0.599
0.994	-0.032	0.490	0.532
1.071	-0.153	0.396	0.485
1.171	-0.096	0.360	0.428

Table F.32: HS1D Cascade Base Pressure Data at Off-Design Incidence,
 $i - i_{des} = +10.0^\circ$.

M_2 nominal	M_2 actual	M_1	$Re_2 (\times 10^3)$	AVDR	α_2	Y_t	Y_{pwc}
0.60	0.60	0.50	318	1.13	50.2	0.296	0.048
0.60	0.60	0.51	367	1.12	50.7	0.283	0.046
0.60	0.60	0.51	467	1.13	50.8	0.250	0.042
0.60	0.60	0.51	535	1.11	51.2	0.256	0.043
0.60	0.60	0.51	638	1.10	50.9	0.280	0.046
0.60	0.60	0.52	788	1.10	50.7	0.313	0.051
0.85	0.85	0.64	582	1.09	53.3	0.200	0.055
0.85	0.85	0.64	643	1.09	53.4	0.197	0.055
0.85	0.85	0.64	709	1.09	53.6	0.195	0.054
0.85	0.85	0.64	770	1.09	53.6	0.194	0.054
0.85	0.85	0.64	898	1.09	53.4	0.199	0.055
0.85	0.85	0.64	951	1.09	53.3	0.202	0.056
0.85	0.85	0.64	1051	1.10	53.2	0.202	0.056
0.95	0.95	0.66	684	1.08	54.4	0.165	0.052
0.95	0.94	0.66	748	1.07	54.5	0.176	0.055
0.95	0.95	0.66	864	1.07	54.6	0.171	0.054
0.95	0.95	0.66	925	1.07	54.5	0.178	0.056
0.95	0.95	0.66	1033	1.08	54.4	0.174	0.055
0.95	0.94	0.66	1094	1.08	54.5	0.175	0.055
1.05	1.04	0.67	770	1.06	55.0	0.164	0.057
1.05	1.04	0.67	836	1.06	55.1	0.164	0.057
1.05	1.05	0.67	951	1.05	55.1	0.169	0.058
1.05	1.05	0.68	1025	1.05	55.1	0.171	0.059
1.05	1.05	0.67	1191	1.05	55.1	0.173	0.060

Table F.33: HS1D Cascade Performance Data at Design Incidence at Off-Design Incidence, $i - i_{des} = +10.0^\circ$ (Reynolds Number Study).

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	0.876	-2.106	0.061	0.133	0.953
0.026	0.804	-1.734	0.110	0.144	0.936
0.060	0.816	-1.797	0.168	0.148	0.931
0.105	0.804	-1.732	0.223	0.182	0.868
0.161	0.781	-1.611	0.289	0.183	0.866
0.216	0.745	-1.424	0.355	0.203	0.824
0.275	0.701	-1.197	0.418	0.238	0.741
0.339	0.679	-1.081	0.478	0.253	0.702
0.398	0.641	-0.891	0.536	0.274	0.644
0.453	0.600	-0.684	0.595	0.308	0.542
0.505	0.584	-0.607	0.645	0.322	0.495
0.551	0.570	-0.538	0.701	0.343	0.423
0.593	0.566	-0.517	0.752	0.368	0.334
0.632	0.552	-0.449	0.805	0.387	0.263
0.668	0.550	-0.442	0.857	0.409	0.180
0.698	0.561	-0.495	0.905	0.442	0.042
0.730	0.545	-0.419	1.000	0.540	-0.394
0.763	0.553	-0.456			
0.797	0.568	-0.529			
0.832	0.549	-0.435			
0.864	0.562	-0.496			
0.895	0.583	-0.598			
0.924	0.552	-0.450			
0.954	0.559	-0.483			
1.000	0.558	-0.477			

Table F.34: HS1D Cascade Blade Loading at $M_2 = 0.48$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.256	-2.147	0.061	0.180	0.972
0.026	1.119	-1.756	0.110	0.201	0.943
0.060	1.178	-1.930	0.168	0.206	0.937
0.105	1.148	-1.843	0.223	0.237	0.888
0.161	1.071	-1.609	0.289	0.243	0.877
0.216	0.949	-1.219	0.355	0.262	0.845
0.275	0.908	-1.082	0.418	0.294	0.784
0.339	0.884	-1.001	0.478	0.315	0.740
0.398	0.838	-0.846	0.536	0.340	0.685
0.453	0.790	-0.682	0.595	0.387	0.574
0.505	0.769	-0.609	0.645	0.411	0.514
0.551	0.755	-0.562	0.701	0.436	0.447
0.593	0.754	-0.559	0.752	0.474	0.341
0.632	0.734	-0.491	0.805	0.504	0.254
0.668	0.734	-0.489	0.857	0.536	0.158
0.698	0.755	-0.560	0.905	0.584	0.008
0.730	0.729	-0.473	1.000	0.730	-0.478
0.763	0.744	-0.523			
0.797	0.770	-0.612			
0.832	0.743	-0.522			
0.864	0.757	-0.568			
0.895	0.788	-0.673			
0.924	0.747	-0.535			
0.954	0.754	-0.559			
1.000	0.753	-0.555			

Table F.35: HS1D Cascade Blade Loading at $M_2 = 0.65$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.321	-1.842	0.061	0.186	0.999
0.026	1.272	-1.734	0.110	0.209	0.972
0.060	1.355	-1.916	0.168	0.218	0.960
0.105	1.343	-1.889	0.223	0.252	0.912
0.161	1.312	-1.823	0.289	0.264	0.893
0.216	1.114	-1.344	0.355	0.287	0.856
0.275	1.018	-1.085	0.418	0.324	0.789
0.339	0.974	-0.962	0.478	0.352	0.735
0.398	0.944	-0.875	0.536	0.384	0.668
0.453	0.909	-0.776	0.595	0.432	0.561
0.505	0.889	-0.718	0.645	0.455	0.506
0.551	0.873	-0.671	0.701	0.488	0.426
0.593	0.866	-0.650	0.752	0.532	0.312
0.632	0.853	-0.610	0.805	0.569	0.214
0.668	0.850	-0.602	0.857	0.610	0.098
0.698	0.871	-0.665	0.905	0.666	-0.061
0.730	0.857	-0.623	1.000	0.854	-0.615
0.763	0.875	-0.676			
0.797	0.905	-0.763			
0.832	0.877	-0.683			
0.864	0.895	-0.735			
0.895	0.921	-0.810			
0.924	0.883	-0.701			
0.954	0.880	-0.690			
1.000	0.878	-0.684			

Table F.36: HS1D Cascade Blade Loading at $M_2 = 0.79$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.323	-1.662	0.061	0.210	0.989
0.026	1.328	-1.671	0.110	0.231	0.962
0.060	1.387	-1.789	0.168	0.237	0.955
0.105	1.381	-1.777	0.223	0.259	0.923
0.161	1.288	-1.588	0.289	0.277	0.897
0.216	1.201	-1.396	0.355	0.298	0.863
0.275	1.124	-1.213	0.418	0.336	0.796
0.339	1.083	-1.111	0.478	0.372	0.729
0.398	1.028	-0.973	0.536	0.406	0.659
0.453	0.991	-0.874	0.595	0.462	0.536
0.505	0.959	-0.788	0.645	0.500	0.446
0.551	0.949	-0.762	0.701	0.538	0.354
0.593	0.940	-0.738	0.752	0.576	0.257
0.632	0.930	-0.710	0.805	0.617	0.149
0.668	0.935	-0.723	0.857	0.664	0.024
0.698	0.956	-0.781	0.905	0.721	-0.132
0.730	0.950	-0.765	1.000	0.956	-0.781
0.763	0.972	-0.823			
0.797	0.998	-0.893			
0.832	0.998	-0.893			
0.864	1.022	-0.955			
0.895	1.041	-1.005			
0.924	1.027	-0.970			
0.954	1.011	-0.928			
1.000	0.972	-0.825			

Table F.37: HS1D Cascade Blade Loading at $M_2 = 0.89$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.313	-1.596	0.061	0.216	0.986
0.026	1.329	-1.629	0.110	0.231	0.968
0.060	1.402	-1.769	0.168	0.242	0.953
0.105	1.402	-1.770	0.223	0.271	0.911
0.161	1.298	-1.565	0.289	0.288	0.886
0.216	1.254	-1.472	0.355	0.317	0.837
0.275	1.191	-1.331	0.418	0.366	0.749
0.339	1.154	-1.246	0.478	0.394	0.693
0.398	1.088	-1.088	0.536	0.434	0.609
0.453	1.044	-0.977	0.595	0.486	0.494
0.505	1.002	-0.871	0.645	0.519	0.414
0.551	0.991	-0.841	0.701	0.556	0.324
0.593	0.992	-0.844	0.752	0.601	0.209
0.632	0.976	-0.801	0.805	0.647	0.087
0.668	0.994	-0.850	0.857	0.695	-0.041
0.698	1.019	-0.913	0.905	0.756	-0.207
0.730	1.017	-0.908	1.000	1.064	-1.028
0.763	1.041	-0.970			
0.797	1.071	-1.046			
0.832	1.084	-1.077			
0.864	1.121	-1.167			
0.895	1.159	-1.258			
0.924	1.183	-1.313			
0.954	1.178	-1.302			
1.000	1.063	-1.026			

Table F.38: HS1D Cascade Blade Loading at $M_2 = 0.99$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.317	-1.546	0.061	0.215	0.994
0.026	1.331	-1.573	0.110	0.231	0.975
0.060	1.416	-1.733	0.168	0.237	0.966
0.105	1.413	-1.727	0.223	0.266	0.926
0.161	1.310	-1.532	0.289	0.278	0.909
0.216	1.277	-1.464	0.355	0.316	0.850
0.275	1.219	-1.341	0.418	0.361	0.770
0.339	1.180	-1.254	0.478	0.396	0.703
0.398	1.114	-1.102	0.536	0.440	0.612
0.453	1.066	-0.985	0.595	0.491	0.499
0.505	1.023	-0.880	0.645	0.528	0.411
0.551	1.015	-0.860	0.701	0.572	0.304
0.593	1.013	-0.854	0.752	0.611	0.205
0.632	1.006	-0.837	0.805	0.656	0.090
0.668	1.038	-0.917	0.857	0.705	-0.039
0.698	1.081	-1.023	0.905	0.760	-0.187
0.730	1.108	-1.088	1.000	1.241	-1.387
0.763	1.149	-1.184			
0.797	1.160	-1.208			
0.832	1.134	-1.149			
0.864	1.152	-1.190			
0.895	1.199	-1.296			
0.924	1.242	-1.390			
0.954	1.305	-1.522			
1.000	1.231	-1.368			

Table F.39: HS1D Cascade Blade Loading at $M_2 = 1.07$, and $i - i_{des} = +10.0^\circ$.

Suction Surface			Pressure Surface		
x/C_x	M_s	C_{ps}	x/C_x	M_s	C_{ps}
0.003	1.316	-1.574	0.061	0.217	0.988
0.026	1.340	-1.621	0.110	0.233	0.968
0.060	1.423	-1.777	0.168	0.247	0.949
0.105	1.425	-1.781	0.223	0.277	0.907
0.161	1.311	-1.563	0.289	0.286	0.893
0.216	1.280	-1.498	0.355	0.315	0.847
0.275	1.211	-1.350	0.418	0.362	0.761
0.339	1.177	-1.274	0.478	0.394	0.701
0.398	1.117	-1.133	0.536	0.440	0.604
0.453	1.066	-1.008	0.595	0.491	0.490
0.505	1.019	-0.891	0.645	0.524	0.411
0.551	1.011	-0.871	0.701	0.564	0.313
0.593	1.016	-0.885	0.752	0.609	0.199
0.632	1.010	-0.869	0.805	0.653	0.084
0.668	1.042	-0.949	0.857	0.698	-0.036
0.698	1.088	-1.063	0.905	0.759	-0.200
0.730	1.125	-1.154	1.000	1.308	-1.556
0.763	1.208	-1.344			
0.797	1.303	-1.547			
0.832	1.394	-1.724			
0.864	1.499	-1.908			
0.895	1.460	-1.843			
0.924	1.324	-1.589			
0.954	1.343	-1.626			
1.000	1.302	-1.545			

Table F.40: HS1D Cascade Blade Loading at $M_2 = 1.17$, and $i - i_{des} = +10.0^\circ$.

Appendix G

Additional Performance Results at Off-Design Mach Number and Incidence

This appendix presents additional cross-plots of some of the data presented in Chapter 8. It includes plots of the effect of incidence on the losses, outlet flow angle and base pressure at various Mach numbers for the three blade profiles. These figures are convenient for comparing the data in various ways. However, they could not be all included in the main body of the thesis given the amount of data.

In addition to these data, plots showing a comparison of the Mach number distribution for the three profiles at various incidences and outlet Mach numbers are also included.

Finally, the Mach number distributions obtained from the Reynolds number study at off-design incidence are included in this appendix.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.1: Effect of Incidence on the Variation of the Total Pressure Loss Coefficient for Various Mach Numbers.

Figure G.2: Effect of Incidence on Losses for the Three Cascades at $M_{2s} = 0.65$.

Figure G.3: Effect of Incidence on Losses for the Three Cascades at $M_{2s} = 0.85$.

Figure G.4: Effect of Incidence on Losses for the Three Cascades at $M_{2s} = 1.05$.

Figure G.5: Effect of Incidence on Losses for the Three Cascades at $M_{2s} = 1.25$.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.6: Effect of Incidence on the Variation of the Outlet Flow Angle for Various Mach Numbers.

Figure G.7: Effect of Incidence on Outlet Flow Angle for the Three Cascades at $M_{2s} = 0.65$.

Figure G.8: Effect of Incidence on Outlet Flow Angle for the Three Cascades at $M_{2s} = 0.85$.

Figure G.9: Effect of Incidence on Outlet Flow Angle for the Three Cascades at $M_{2s} = 1.05$.

Figure G.10: Effect of Incidence on Outlet Flow Angle for the Three Cascades at $M_{2s} = 1.25$.

Figure G.11: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of -10.0° and an Outlet Mach Number of 0.65.

Figure G.12: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of -10.0° and an Outlet Mach Number of 0.85.

Figure G.13: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of -10.0° and an Outlet Mach Number of 1.05.

Figure G.14: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of -10.0° and an Outlet Mach Number of 1.15.

Figure G.15: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at Design Incidence and an Outlet Mach Number of 0.65.

Figure G.16: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at Design Incidence and an Outlet Mach Number of 0.85.

Figure G.17: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at Design Incidence and an Outlet Mach Number of 1.05.

Figure G.18: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at Design Incidence and an Outlet Mach Number of 1.15.

Figure G.19: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 0.65.

Figure G.20: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 0.85.

Figure G.21: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 1.05.

Figure G.22: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 1.15.

Figure G.23: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 0.65.

Figure G.24: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 0.85.

Figure G.25: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 1.05.

Figure G.26: Comparison of the Mach Number Distribution for Blades HS1A, HS1C and HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 1.15.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.27: Effect of Incidence on the Variation of the Base Pressure Coefficient for Various Mach Numbers.

Figure G.28: Effect of Incidence on Base Pressure Coefficient for the Three Cascades at $M_{2s} = 0.65$.

Figure G.29: Effect of Incidence on Base Pressure Coefficient for the Three Cascades at $M_{2s} = 0.85$.

Figure G.30: Effect of Incidence on Base Pressure Coefficient for the Three Cascades at $M_{2s} = 1.05$.

Figure G.31: Effect of Incidence on Base Pressure Coefficient for the Three Cascades at $M_{2s} = 1.25$.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.32: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of -10.0° and an Outlet Mach Number of 0.60.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.33: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of -10.0° and an Outlet Mach Number of 0.85.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.34: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of -10.0° and an Outlet Mach Number of 0.95.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.35: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of -10.0° and an Outlet Mach Number of 1.05.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.36: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at Design Incidence and an Outlet Mach Number of 0.60.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.37: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at Design Incidence and an Outlet Mach Number of 0.85.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.38: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at Design Incidence and an Outlet Mach Number of 0.95.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.39: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at Design Incidence and an Outlet Mach Number of 1.05.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.40: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 0.60.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.41: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 0.85.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.42: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 0.95.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.43: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+5.0^\circ$ and an Outlet Mach Number of 1.05.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.44: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 0.60.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.45: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 0.85.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.46: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 0.95.

(a) Mid-Loaded Cascade HS1A

(b) Aft-Loaded Cascade HS1C

(c) Front-Loaded Cascade HS1D

Figure G.47: Reynolds Number Effect on the Loading Distribution of Blade (a) HS1A, (b) HS1C and (c) HS1D at an Incidence of $+10.0^\circ$ and an Outlet Mach Number of 1.05.