**MEASUREMENTS AND COMPUTATIONS OF ROUGHNESS
EFFECTS ON PERFORMANCE OF A HP TURBINE CASCADE IN
COMPRESSIBLE FLOW**

by

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ABSTRACT

Experimental and numerical studies have been conducted to investigate surface roughness effects on aerodynamic performance of a high-pressure (HP) turbine cascade with compressible flow. This work continues earlier work at Carleton University on roughness effects on performance of a large-scale turbine cascade with incompressible flow.

The experiments were conducted in the Carleton University High Speed Wind Tunnel. The surface roughness consisted of randomly distributed sand grains adhered to the suction surface of the blades. The roughness height, the width of the roughness band and the band location relative to the leading edge were varied. The profile loss coefficient and exit flow deviation angle were determined from wake traverse measurements. The tests were run for exit Mach numbers ranging from 0.40 to 1.13, corresponding to exit Reynolds number, based on true chord and exit flow velocity, ranging from 4.0×10^5 to 1.0×10^6 .

Roughness on the suction surface produces large increases in profile losses, but has little effect on the exit flow deviation angle and blade pressure distribution or loading. The losses are strongly influenced by the roughness height and by location of the roughness band relative to the leading edge. Mach number is found to have an important effect on the magnitude of the losses.

The FLUENT code was employed for the numerical computations. The Spalart-Allmaras turbulence model with wall functions was used for smooth and roughened blades. The predicted trends for profile loss coefficient and exit flow deviation angle for roughened blades were essentially the same as the experimental trends. The agreement between predicted and experimental results is fairly good.

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NOMENCLATURE

A, B	constants in logarithmic law of the wall, $A=1/0.41$, $B=5.0$, (see Equation 2.3)
A_T, A_F, A_w, A_f	reference area, total roughness frontal area, roughness element average windward wetted surface area, roughness element average frontal area
$AVDR$	axial velocity density ratio, (see Equation 3.12)
b_m, s_m	length of roughness element spanwise, length of roughness element streamwise, (see Figure 2.5)
ΔB	roughness function, (see Equation 5.7)
c	blade chord length
C	function of roughness spacing, (see Equation 2.3)
C_x	axial blade chord length
C_{ks}	roughness constant
C_f	skin friction coefficient, $(=\tau_w/0.5\rho V_e^2)$
C_{P0}	total pressure coefficient, $(=(P_{02}-P_{01})/q_2)$
$C_{b1}, C_{b2}, C_{v1}, C_{b2}, \sigma_v$	S-A model constants
C_{w1}, C_{w2}, C_{w3}	S-A model constants
d	sand grain nominal diameter or roughness height, (see Equation 3.2)
d	distance from the wall, (see Equation 5.3)
G_v, Y_v, S_v	production of turbulent viscosity, destruction of turbulent viscosity, user-defined source term in S-A model transport Equation 5.1
h_0, h, h_s	stagnation enthalpy, enthalpy, isentropic enthalpy
k, k_s	actual roughness height or nominal sand grain size, equivalent sand roughness height
k_s^+	$= k_s u_\tau / \nu$
K_1, K_{12}, K_{13}	three-hole probe coefficients

$K_{23}, K_{\Phi 2}, K_{\Phi 3}$	three-hole probe coefficients
M	Mach number
M_{2y}	downstream local Mach number
n	total number of sand grains within the areas
N	cascade blade number
P_0, P_s	total pressure, static pressure,
P_s	blade surface static pressure
P_{01r}, P_{1r}, P_{2r}	reference cascade inlet total pressure, reference inlet static pressure, reference outlet static pressure
$P_{02y}, P_{02yc}, P_{2y}$	true downstream local total pressure upstream of the shock, measured downstream local total pressure, measured downstream local static pressure
q	dynamic pressure, $(=0.5\rho V^2)$
Re	Reynolds number
s	blade pitch
Δs	arc width of roughness band
S	deformation tensor
$\hat{S}$	transformed deformation tensor
T	temperature
u, U_e	streamwise velocity in boundary layer, streamwise freestream velocity
u^+	$= u / u_\tau$
u_τ	friction velocity, $(=\sqrt{\tau_w / \rho})$
ΔU	norm of the difference between the velocity at the transition trip that at the filed
V	cascade flow velocity
V_e	freestream velocity
W	test section inlet duct width
X	axial distance from blade leading edge

x, y	chordwise distance from blade leading edge, blade-to-blade pitchwise distance
y	distance above the blade surface normal, (see Equation 2.3)
y^+	$= yu_\tau / \nu$
Y_t	midspan total pressure loss coefficient, (see Equation 2.1)

Abbreviations

CFD	computational fluid dynamics
HP	high-pressure
S-A	Spalart-Allmaras
UDFs	user-define functions

Greek letters

α	flow angle
ξ	enthalpy loss coefficient, (see Equation 2.2)
μ	molecular viscosity
μ_t	turbulent viscosity
μ_t/μ	turbulent viscosity ratio
ν	kinematic viscosity
$\hat{\nu}$	modified turbulent viscosity
γ	specific heat
ρ	density
κ	Von Karman's constant
δ	boundary layer thickness, (see Equation 2.9)
δ	exit flow deviation angle, (=exit metal angle- α_2)
$\lambda, \Lambda, \Lambda_s$	Simpson and Dvorak roughness spacing parameter, Waigh and Kind roughness shape/spacing parameter, Sigal and Danberg roughness shape/density parameter (see section 2.2.1)
τ_w	wall shear stress
Φ	calibration yaw angle of three-hole probe

Subscripts

1	cascade inlet
2	cascade outlet
s	isentropic
le	leading edge
te	trailing edge

1. INTRODUCTION

Blade surfaces in operating gas turbine engines will often become rough due to corrosion, erosion, deposition and/or thermal barrier coating spallation. In new turbine engines, surface coating and manufacturing surface finish also have surface roughness. Measurements on in-service turbine blades of aero-engines by Taylor (1990) showed that roughness varied greatly around the blade; the large roughness was distributed on the pressure side near the midchord and trailing edge, and on the suction side near the leading edge. Tarada and Suzuki (1993) also measured the roughness on 50 used aero-engine, industrial, and marine turbine blades, indicating that typical roughness heights were found to be from 20 to 150 μ m centerline average after a few thousand operation hours.

The roughness can significantly increase the turbine blade surface skin friction, and thus cause deteriorated turbine engine performance. Severe blade surface roughening is recognized by engine health monitoring systems as one of three most common forms of blade damage, these being blade surface roughness, blade tip damage and leading or trailing edge damage.

Roughness effects interest many researchers. Unfortunately, few investigations into roughness effects on turbine blade profile loss and flow deviation in compressible flow have been reported. However, high-pressure (HP) turbines working at conditions of high turning angle and transonic Mach number are of great interest to engine manufacturers for the advantages of compactness, lower cost and weight and high efficiency. Therefore, a study on roughness effects under such conditions is important but

challenging, especially with transonic conditions near the blade-row exit due to complex flow with shock waves, shock-boundary layer interaction and possible flow separation.

The present study examines the effect of surface roughness on the aerodynamic performance of a HP turbine cascade with compressible flow. The objectives are to understand the effects on profile losses and flow turning of surface roughness with different partial blade coverage by various roughness heights for compressible flow through HP turbine cascades, and to assess the potential for Computation Fluid Dynamics (CFD) codes such as FLUENT to predict these effects.

The experiments have been performed in the Carleton University High Speed Wind Tunnel, on the HS1C turbine cascade designed by Corriveau and Sjolander (2003). The exit Mach number in the experiments was in the range of 0.40-1.13, with corresponding exit Reynolds number based on chord length and exit flow velocity between 4.0×10^5 and 1.0×10^6 . Uniform size sand grains were randomly stuck to portions of the blade suction surface to act as roughness. The roughness heights and distributions were chosen to approximately match the values used by Kind *et al.* (1998) in their studies of roughness effects on performance of a large-scale turbine cascade blade with incompressible flow. Three roughness heights with three band widths starting from two locations near the blade leading edge were used. The non-dimensional roughness spacing parameter had a similar value in all cases. The profile losses and flow deviations were determined by downstream wake traverse measurements across a full blade-to-blade pitch.

The numerical investigations were carried out using the commercial CFD code FLUENT. The Spalart-Allmaras (1992) turbulence model with wall functions was used

together with the Navier-Stokes equations for compressible flow. The wall functions represent the flow behavior very near the blade surface where the law of the wall is presumed to apply. Solutions were obtained for flow over smooth and rough blades. The objective of this part of the work was to assess if roughness effects in compressible flow could be predicted simply by modifying the wall functions in the same way as for incompressible flow.

This thesis is divided into an introduction and five subsequent chapters. Chapter 2 reviews the pertinent literature on the loss sources and on roughness effects on aerodynamic performance. Chapter 3 describes the wind tunnel setup and measurement instrumentation as well as the data acquisition and reduction procedures. Chapter 4 presents the experimental results and discussions for smooth blades and for blades with various roughness configurations. Chapter 5 presents the CFD methods and results for smooth and rough blades. Finally, conclusions and recommendations for future work are summarized in chapter 6.

2. LITERATURE REVIEW

2.1 Profile Loss in Compressible Flow Turbine Cascades

Profile loss of a turbine blade row is defined as the loss due to boundary layer growth on the blade surface and to dissipation in the blade wake. The flow outside the endwall region is considered to be two-dimensional, and the profile loss can be taken to be the loss in the midspan region in cascade wind tunnel tests.

The most common loss coefficient is the stagnation pressure loss coefficient Y_t , defined by:

$$Y_t = (P_{01} - P_{02}) / q_2 \quad (2.1)$$

Where P_{01} , P_{02} denote the blade row upstream and far downstream total pressure respectively, while q_2 denotes exit mixed-out mass average dynamic pressure ($= 0.5\rho V^2$).

The other commonly used loss coefficient is the enthalpy loss coefficient ξ , defined by:

$$\xi = (h_2 - h_{2s}) / (h_{02} - h_2) \quad (2.2)$$

where h_{02} , h_2 are the blade row downstream stagnation enthalpy and enthalpy respectively, and h_{2s} is the enthalpy that the flow would reach as a result of isentropic expansion through the blade row. In this thesis, total pressure loss coefficient Y_t will be used.

Mee *et al.* (1992) separated the profile loss of a transonic cascade into three components according to the loss generation mechanism: boundary layer loss, shock loss and trailing edge mixing loss. Figure 2.1 shows some smooth cascade test results for each component over a range of exit Mach number at design exit Reynolds number 1.0×10^6 .

Figure 2.1 illustrates that boundary layer loss dominates at subsonic Mach numbers, whereas mixing loss and shock loss become the major contributions at supersonic Mach numbers. The boundary layer loss decreases monotonically with Mach number, and continues to decrease for Mach numbers at which a recompression shock impinges on the suction surface boundary layer. The impingement shock does not lead to a sharp increase in the boundary layer loss in this case since there are no regions of shock-induced boundary layer separation as shown from Figure 2.2 of the surface Mach number distributions. According to Mee *et al.* (Figure 2.1), the shock loss is negligible for Mach number lower than 0.95, but becomes substantial for Mach number between 0.95 and 1.00, and shows a levelling off and possible drop at higher Mach number. The interpretation of these losses as shock losses is open to question because the loss across normal and oblique shocks at Mach number near 1.0 is known to be very small, being proportional to $(M^2 - 1)^3$ (Denton, 1993; Anderson, 2001). As discussed later in this section, shocks do strongly affect the viscous flow and thus cause losses by other mechanisms. The mixing loss makes up about 1/3 of total loss for subsonic Mach number, and rises to about 50% at supersonic Mach number to become the largest component in the total loss.

Figure 2.1 Individual components of profile loss, $Re_{ex}=1,000,000$ (Mee *et al.*, 1992)

Figure 2.2 Blade surface Mach number distributions (Mee *et al.*, 1992)

Many factors, such as velocity distribution, Reynolds number, turbulence level, surface roughness and Mach number, will influence the profile loss. In this section only the effect of Reynolds number and Mach number will be discussed.

Most cascade tests are performed in the Reynolds number range of $10^5 < Re < 10^6$ with Reynolds number based on exit velocity and blade chord. The Reynolds number affects the boundary layer loss significantly. A typical loss variation measured by Ladwig and Fottner (1993) is illustrated in Figure 2.3. For Re less than 2×10^5 , the loss decreases rapidly with increasing Re ; after that, the loss decreases less rapidly and eventually becomes independent of Re .

Figure 2.3 Influence of exit Reynolds number on the total pressure loss coefficient at $M_2=0.59$ and different degrees of turbulence intensities (Ladwig and Fottner, 1993)

The discussions by Denton (1993) and Mayle (1991) depict clearly the blade boundary layer loss associated with Reynolds number. At very low Re ($Re < 10^5$), the blade surfaces are predominantly laminar and the loss increases rapidly due to the high dissipation in laminar boundary layers and possibly due to laminar separation without reattachment. Within the transition range ($2 \times 10^5 < Re < 6 \times 10^5$), the loss is a combination of general decrease with increasing Re and an increase in skin friction as the transition point moves upstream. Transition occurs via either the bypass or flow separation mode depending on Reynolds number. At low Re , a short laminar separation bubble may appear on the suction surface and then be terminated by a turbulent reattachment. With increasing Re , the location of transition moves forward without bubble growth. The loss for flow with separation bubble then reattachment is only slightly higher than that of fully attached flow. At still higher Re ($Re > 6 \times 10^5$), the turbulent flow covers most of blade surface, the loss varies less as Re increases and eventually does not depend on Reynolds number.

Denton (1993) described the effect of Mach number on profile loss. The loss of turbine blades increases rapidly as exit Mach number approaches unity, and usually continues to rise with increasing Mach number in low supersonic outflow. The increased loss comes from the complex shock pattern formed around the trailing edge. Figure 2.4 illustrates the general flow system. In transonic turbine flow, shock waves do occur. At low Mach number, the shocks form within the blade passages. They are usually oblique so they generate little direct loss. As Mach number increases, the base pressure behind the trailing edge becomes lower. The flow will expand around the trailing edge to this low pressure and then recompress by a strong shock wave at the point where the suction

and pressure side flows meet. Large amounts of entropy are generated from the intense viscous dissipation at the separated region behind the trailing edge and from the strong shock formed at the close of this region.

Figure 2.4 Structure of supersonic trailing edge flow (Denton *et al.*, 1990)

The other source of loss associated with shock waves is the shock boundary layer interaction. The shock from the trailing edge impinges on the suction surface of the adjacent blade, resulting in interaction between the shock and the boundary layer. A boundary layer separation bubble will usually be formed at the foot of the shock. If the impingement shock is strong enough or the boundary layer is laminar, the boundary layer will separate, resulting in higher loss. Otherwise, the change of loss is slight.

2.2 Experimental Studies of Flow over Rough Surfaces

Experimental studies show that effects of roughness lie in two aspects. It increases the turbulent boundary layer skin friction factors and the local heat transfer coefficients, and also affects the transition from laminar to turbulent flows, causing transition to occur at lower flow Reynolds numbers as compared to smooth surfaces.

2.2.1 Roughness Effects on Boundary Layers

As outlined in White (1991), surface roughness has a direct effect only on the flow in the inner region of the boundary layer. Assuming that its height, k , is substantially less than the thickness of the boundary layer, there are three roughness regimes based on the value of k_s^+ ($k_s^+ = k_s u_\tau / \nu$), where k_s is the equivalent sand grain roughness height and u_τ is the friction velocity. As proposed by Schlichting (1979), k_s is the size of close-packed sand grains, as used in Nikuradse's (1950) pipe flow experiment, that produce the same flow resistance as the actual roughness. The actual roughness is usually characterized in terms of the equivalent sand grain roughness height, k_s . For $k_s^+ < 5$, the roughness is within the viscous sublayer and thus has no influence on the flow, and the surface is in effect hydraulically smooth. For $k_s^+ > 5$, the roughness will protrude above the thin viscous sublayer causing an increase in skin friction and thus in u_τ . This will result in a downward shift of u/u_τ relative to the smooth-surface law of the wall. For $5 < k_s^+ < 70$, the shift is influenced by the Reynolds number of the flow over the roughness elements and the surface is considered as transitionally rough. For $k_s^+ > 70$, the shift is

independent of Reynolds number because the wall resistance is dominantly due to form drag on the roughness elements and is only affected by roughness geometric properties; the surface is then called fully rough.

For fully rough flow, the logarithmic law of the wall for actual roughness of height k becomes,

$$\frac{u}{u_\tau} = A \ln \frac{y}{k} + B - C \quad (2.3)$$

where u is the flow velocity at distance y above the wall, A and B are the usual log-law constants with values of $1/0.41$ and 5.0 respectively, and C is a function of roughness geometry, the density and the arrangement of the roughness elements. C has a constant value for any particular roughness geometry. For Nikuradse standard sand roughness, C is -4.0 (Grigson, 1984), thus the equivalent sand roughness height, k_s , is obtained by equating the intercepts yielding,

$$\frac{k_s}{k} = \exp\left(\frac{C+4}{A}\right) \quad (2.4)$$

Therefore, the logarithmic law of the wall for equivalent sand roughness height, k_s , can be written as,

$$\frac{u}{u_\tau} = A \ln \frac{y}{k_s} + B + 4.0 \quad (2.5)$$

Thus the effect of roughness on boundary layer flow can be represented by its effect on the law of the wall, that is by the value of C associated with the roughness.

Many attempts have been made to correlate C with geometric characteristics of the surface roughness. Dvorak (1969) quantified C with relation to the roughness spacing parameter λ , which he defined for two-dimensional (square bars) roughness as the ratio of space between bars to bar width or height. The correlation was given as

$$\begin{aligned} C &= 17.35(1.625 \log_{10} \lambda - 1) & \text{for } 1 \leq \lambda \leq 5 \\ C &= -5.95(1.103 \log_{10} \lambda - 1) & \text{for } \lambda > 5 \end{aligned} \quad (2.6)$$

The C increased with increasing λ in the dense spacing regime and vice versa in the sparse regime. Simpson (1973) suggested a more universal roughness density parameter λ , which was defined as the ratio of reference surface area to total roughness frontal area, $\lambda = A_T / A_F$. Sigal and Danberg (1988) proposed another roughness shape/density parameter Λ_s , $\Lambda_s = (A_T / A_F)(A_f / A_w)^{-1.6}$. A_f and A_w are roughness element frontal area and roughness windward wetted surface area, respectively, and the ratio A_f / A_w could be interpreted as an element-shape parameter. They proposed the following correlations:

$$\begin{aligned} C &= 17.35(1.634 \log \Lambda_s - 1) & 1.4 \leq \Lambda_s \leq 4.89 \\ C &= 2.2 & 4.89 \leq \Lambda_s \leq 13.25 \\ C &= 9.55(1.0 - 0.686 \log \Lambda_s) & 13.25 \leq \Lambda_s \leq 100.0 \end{aligned} \quad (2.7)$$

Waigh and Kind (1998) constructed another correlation through a more detailed consideration of the shape of the roughness elements. They found that previous spacing parameters did not serve as a good discriminator between the dense and sparse regimes. There exists data scatter for λ near 5. A new parameter Λ , which measured the size of the

intervening spaces between roughness elements, was introduced, and it led to the correlation,

$$\Lambda = \lambda k / s_m$$

$$C = 10.56 \log_{10} \left[\lambda (k / b_m)^{0.87} (A_w / A_f)^{0.44} \right] - 7.59 \quad \Lambda < 6$$

$$C = -5.75 \log_{10} \left[\lambda (k / b_m)^{0.55} (A_w / A_f)^{1.38} \right] + 5.78 \quad \Lambda > 6 \quad (2.8)$$

where b_m and s_m are length of roughness element spanwise and streamwise respectively (See Figure 2.5 for symbols). This correlation reduced the scatter by a factor of about 2 for those roughness configurations included in their study.

Figure 2.5 Geometry of a rough wall (Waigh & Kind, 1998)

Only regular roughness configurations were considered by Waigh and Kind (1998) and previous authors, with the exception of the standard sand configuration. However, the actual roughnesses on gas turbine engine blades have different features for different roughness formation mechanisms, presenting a random, non-uniform and irregular appearance. It is difficult for a single parameter such as equivalent sand roughness height to describe precisely the surface roughness geometry of such roughness and its resistance effect on turbulent flows. Research on this subject is still being performed.

Surface roughness has only an indirect effect on the outer region of boundary layers through the skin friction stress. For example, for zero-pressure-gradient boundary layers, the same defect law holds for both smooth and rough surfaces. The defect law (Clauser, 1956) can be written,

$$\frac{U_e - u}{u_\tau} = f\left[\frac{y}{\delta}\right] \quad (2.9)$$

where U_e is the velocity outside the boundary layer (i.e. at $y=\delta$) and δ is the boundary layer thickness. Surface roughness results in a larger value of u_τ and thus indirectly affects the distribution of u in the outer part of the boundary layer.

2.2.2 Roughness Effects on Blade-Row Performance and Aerodynamics

Measurements by Bammert and Sandstede (1972) showed that sand grain surface roughness on turbine blades with k_s/c in the ranging from 10^{-3} to 10^{-2} decreased stage efficiency by 7 to 14 percent compared to smooth blades. Boynton *et al.* (1993) found that a decrease in the roughness height of a surface finish coating on turbine rotor blades from 10.16 to 0.76 μm (root mean square) caused a 2.5 percentage point increase in the

efficiency of a high pressure fuel turbopump for a rocket engine. Suder *et al.* (1995) measured the roughness of paint coated on high-speed compressor rotor blades, and found that a 0.025 mm thick rough coating with a surface finish of 2.54-3.18 μm resulted in approximately twice the performance deterioration found with a smooth coating with a surface finish of 0.254-0.508 μm .

Roughness height and roughness distribution on blades are clearly important. Suder *et al.* (1995) applied a rough coating to different portions of the rotor blade surface of a transonic turbine. They found that the blade leading edge and front half of the suction surface were most sensitive to the thickness/roughness variations; roughness on these regions accounted for more than 70% of the performance degradation found for the fully coated blade, and there was little performance degradation when the leading edge was left uncoated. Kind *et al.* (1998) assessed roughness effects on a low-speed turbine cascade with partial surface coverage of various roughness heights, roughness band widths and locations. Surface roughness was produced by uniformly distributed sand grains glued to the blade surfaces. The results showed that roughness on the suction surface could cause large increases in profile loss while roughness on the pressure surface had relatively little effect. Roughness relatively close to the leading edge on the suction surface caused substantially greater loss increment than the same roughness located further aft. Loss increment increased less slowly than the width of the roughness strip.

Surface roughness effects are substantially influenced by Reynolds number. This is shown in Figure 2.6 in which the blade profile loss variation with Reynolds number and roughness height are plotted (Denton, 1993). A small roughness height, which has no influence on loss at lower Reynolds number, causes higher loss at higher Reynolds

number. This can be explained that roughness elements at lower Reynolds number are immersed in the viscous sublayer, but can protrude through this layer at higher Reynolds number. An interesting case was inspected by Leipold *et al.* (2000) on a highly loaded compressor cascade with 0.3mm surface roughness engraved in a precision forged blade. The separation bubble was only slightly affected by surface roughness at low Reynolds number 3.0×10^5 ; however at high Reynolds number of 6.0×10^6 , roughness completely suppressed the laminar separation, and at the same time caused the turbulent boundary layer to separate from the blade. Another experiment on effects of surface roughness ($k_s/c=4.8 \times 10^{-4}$) on turbine vanes was conducted by Boyle *et al.* (2003). The measurements were made for Reynolds number from 1.5×10^5 to 1.8×10^6 and exit Mach number from 0.3 to 0.9 at three inlet turbulence levels of 3%, 8% and 17%. The results showed that roughness effects strongly depended on Reynolds number and were not sensitive to higher inlet turbulence intensity for low Reynolds number.

Figure 2.6 Variation of profile loss with Reynolds number and surface roughness (Denton, 1993)

The influence of roughness on the shock waves was examined by Bölcs *et al.* (1988) on a transonic turbine cascade. A deposit was put onto the pressure surface and on the forward 40% of the suction surface with roughness height around 7-12 μm . Flow visualization indicated that the shock wave fluctuations were not significantly influenced by the roughness and thickness of the deposit in his case. One example cascade flow at exit isentropic Mach number of 1.09 is shown in Figure 2.7. Compared with the smooth cascade, the cascade with deposit showed that the sonic line departed from the suction side to the pressure side, and a pre-shock originated from the neighboring blade came into view on the suction side. It seemed that the impinged shock didn't show noticeable increase in the boundary layer thickness. The post-shock that occurred at the trailing edge didn't appear remarkably different between the smooth blade and the deposit blade.

Figure 2.7 Sonic line distribution for cascade flow at $\beta_1=10^\circ$, $M_{2s}=1.09$ (Bölcs *et al.*, 1998)

Experiments from Kind *et al.* (1998), Speidel (1954) and Bammert and Sandstede (1980) show that roughness has only a small effect on blade surface static pressure distribution. One example from Kind *et al.* is shown in Figure 2.8. Moreover the

explanation for this phenomenon is clearly documented. Roughness on the blade surface thickens the boundary layer and increases the blade-to-blade passage blockage, thus causing higher velocity in the inviscid core flow. The increase of velocity at the suction surface is more noticeable than that at the pressure surface due to the larger velocity near the suction surface. However the thickening of the suction surface boundary layer also produces a de-cambering effect, which would tend to cause a compensatory slight decrease in loading over the rear portion of the blades.

The small effect of roughness on blade loading undoubtedly implies that roughness has little effect on exit flow deviation angle. Kind *et al.* (1998) found not more than one degree increase of deviation angle for all their investigated cases, provided that roughness didn't cause substantial flow separation.

Figure 2.8 Measured static pressure distributions with and without roughness at design incidence (Kind *et al.*, 1988)

An investigation of boundary layer development along a rough turbine blade at a Reynolds number of 5.6×10^5 and a Mach number of 0.14 was presented by Bammert and Sandstede (1980). Four sand grain roughness heights (k_s/c) ranging from 0.001 to 0.01 were tested. Results showed that the boundary layers were significantly affected by roughness. Boundary layer momentum thickness is shown in Figure 2.9. The momentum thickness at the pressure side first increased to a maximum then decreased. At the suction surface the momentum thickness increased slowly near the leading edge then increased rapidly downstream, and showed considerably greater thickness than on the pressure side. This is one reason why the roughness on the pressure surface has little effect on the loss. More importantly, the velocity at the pressure surfaces is relatively low. As compared to the smooth surface, the boundary layer momentum thickness for the largest size roughness was up to about three times as high as that on smooth surface. Figure 2.10 shows the boundary layer shape factor. On the pressure surface the location of transition was almost un-affected by the surface roughness. But the laminar to turbulent transition was more rapid for the rough blade than for the smooth blade. On the suction side the shape factor differed for different roughness size. The transition for the smooth surface occurred in the deceleration area, whereas it occurred in the acceleration area for the rough surface and moved upstream with increasing roughness size. For the largest roughness the transition point was found to be at the leading edge.

(a) Pressure side

(b) Suction side

Figure 2.9 Momentum thickness at (a) pressure side (b) suction side: (○) smooth, (Δ) $k_s/c=0.57 \times 10^{-3}$, (▽) $k_s/c=1.02 \times 10^{-3}$, (◇) $k_s/c=3.31 \times 10^{-3}$ (Bammert and Sandstede, 1980)

(a) Pressure side

(b) Suction side

Figure 2.10 Shape factor at (a) pressure side (b) suction side. Legend see Figure 2.9 (Bammert and Sandstede, 1980)

2.3 Flow Prediction over Rough Surfaces

At present the effects of roughness on aerodynamic performance are predicted by two different approaches: the correlation approach and the discrete-element approach.

2.3.1 Correlation Approach

This approach uses the law of the wall (Equations 2.3 to 2.5) or turbulence models in which the wall boundary condition is correlated with equivalent sand roughness height (k_s), to model the effect of roughness. The rough surface is characterized by an equivalent sand roughness height which depends on the actual roughness height, shape and spacing. The boundary layer flow over rough surfaces can be predicted using either integral methods or differential methods.

Dvorak (1969) employed a rough surface skin friction law based on Equations of 2.3 and 2.9, in conjunction with the boundary layer momentum integral equation and Head's (1958) entrainment function to calculate the turbulent boundary layers over rough surfaces in pressure gradients. This rough surface skin friction law was also applied by Kind *et al.* (1998) to compute the profile losses and flow deviation on their low speed turbine cascade, using a coupled viscous/inviscid interaction analysis.

Cebeci and Chang (1978) were first to derive a correlation of the near-wall mixing length with the equivalent sand roughness height. The boundary layer development over a rough wall was predicted by incorporating this correlation into a mixing length/eddy viscosity turbulence model in a two-dimensional finite difference boundary layer code. Boyle (1994) employed Cebeci and Chang's roughness mixing

length model combined with the quasi-three-dimensional Navier-Stokes equations to predict the efficiency loss due to roughness measured by Boynton (1993). In 2003, Boyle *et al.* (2003) developed a roughness transition model. Using this model along with Cebeci and Chang's roughness model, the loss prediction achieved the best agreement with experiments. Flow with rough boundaries can also be computed using the correlation approach, together with a turbulence model, in finite-difference or finite-volume solutions of the Navier-Stokes equations.

Wilcox (1998) reported that his $k-\omega$ turbulence model had an advantage over $k-\epsilon$ models in yielding the smooth wall law of wall with no need for any special wall-region terms if the wall boundary condition was set rationally. Roughness effects can easily be incorporated into the $k-\omega$ turbulence model through the surface boundary condition on ω . It was suggested by Wilcox that this approach should only be applied for k_s^+ up to 400.

Tarada (1990) proposed a low-Reynolds number $k-\epsilon$ model applicable to flow over rough surfaces to determine skin friction coefficients and flow transition. The model was first calibrated by comparison with data obtained with deterministic roughness elements and then applied to random surface roughness conditions.

Taylor *et al.* (1985) argued that roughness effects would be overestimated by using Schlichting's equivalent sand roughness height since there were some erroneous assumptions in Schlichting's (1937) experimental data reduction. However correct results should be obtained by using the re-evaluated equivalent sand roughness heights reported by Coleman *et al.* (1984) for Schlichting's (1937) data and Grigson (1984) for Nikuradse's data.

2.3.2 Discrete Element Approach

The discrete element approach was first suggested by Finson (1982) and developed by Taylor *et al.* (1985). The effects of roughness are considered by including the form drag of the roughness elements and blockage effects in the governing partial differential equations. The boundary conditions and turbulence model do not rely on the equivalent sand grain roughness height. The basic boundary layer conditions for rough wall flow are assumed to be the same as those for smooth wall. The turbulence closure is achieved by using the Prandtl mixing length model with van Driest's damping function. The drag of roughness elements is incorporated as a sink term in the momentum equations. The relationships for drag of the individual roughness elements are based on well-accepted empirical data for ordered and deterministic roughness elements.

McClain *et al.* (2003) improved the discrete element approach so that it could be used with random rough surfaces. The random roughness elements have varying heights and frontal area shapes, and also have changing plan-form sizes and shapes of roughness elements. The "no slip" condition is set at the reference mean elevation. The total drag on a randomly rough surface is then the sum of the drag on the roughness elements and the shear force acting on the plan-form area at the mean elevation where not blocked by roughness elements. An element-eccentricity factor is incorporated into the discrete element model to account for random roughness-element plan form shapes and roughness anisotropy by adding a term in the local element drag coefficient. McClain *et al.* applied this method to two rough surfaces found on high-hour gas-turbine blades, and obtained excellent agreement of measured and predicted skin friction coefficient.

3. DESCRIPTION OF EXPERIMENT

3.1 High Speed Wind Tunnel

The present measurements were performed in the Carleton University High Speed Wind Tunnel. It is a transonic blow-down wind tunnel. The Mach number and Reynolds number can be varied independently.

Figure 3.1 shows the diagram of the wind tunnel. The air is compressed, dried and then filled into the four tanks. Air flows from the tanks through a control valve (throttling valve) into the test section. When the tanks are full with a tank pressure reaching 100 psig, the control valve is opened and air is discharged through the test section at constant blowing pressure. The blowing pressure is typically set within 1.1–1.7 bar absolute, enabling exit Mach numbers approximately between 0.4 and 1.2. The desired test section blowing stagnation pressure is established through the control system in about 10 to 30 seconds, and is maintained constant for a duration of 20 to 30 seconds, depending on the blowing pressure and cascade outlet Mach number. Therefore, a high speed data acquisition system and fast response probes are employed for measurement during such a short runtime. The inlet turbulence intensity of the test section is about 4%. An ejector-diffuser system is mounted on the outlet of the test section to allow the downstream static pressure to be controlled independently of the inlet blowing pressure, and thus to allow Reynolds number to be varied for fixed outlet Mach number. The ejector-diffusion system was not used in the present measurements. The air from the test section exhausted

directly to the laboratory at atmospheric conditions. A more detailed description of the wind tunnel can be found in Jeffries (2001).

Figure 3.1 Carleton University High Speed Wind Tunnel (Corriveau and Sjolander, 2003)

3.1.1 Cascade Test Section

A planview of the cascade test section is shown schematically in Figure 3.2. The cascade of blades is mounted on a turntable. By rotating the turntable, the incidence angle can be changed from -10° to 15° relative to the design value. When the incidence is changed, the two side walls of the test section can be moved to adjust the inlet flow. Downstream of the cascade, the transition walls can be rotated to align with the downstream flow direction. A probe traversing mechanism is mounted to the downstream transition walls, and the measuring probe is attached to the probe holder which attaches to the end of the probe stem. The vertical and horizontal traversing gears are driven separately by individual stepper motors, allowing the probe to be moved separately along the cascade's spanwise and pitchwise directions. The probe orientation relative to the exit metal angle can be checked by a probe angle alignment mechanism, which is shown in Figure 3.3. A more detailed description of the probe angle alignment mechanism is given by Jouini (2000).

Normally in cascade tests, the inlet flow is adjusted by moving various wind tunnel control surfaces to produce an equal mass flow rate in each blade-to blade passage. In the current wind tunnel, inlet flow adjustment is achieved by moving the test section side walls. Other than the transition walls there are no control surfaces to adjust the outlet flow. The outlet flow discharges into a wider plenum as compared to the cascade inlet. In order to obtain good flow conditions, the suggestions from Corriveau and Sjolander (2002) were adopted. The two test section side walls were set parallel to each other and were aligned with the leading edge of the cascade end blocks. The distance between the

side walls, that is the inlet duct width, W , was set to the value calculated from Equation 3.1,

$$W = (N + 1)s \cos \alpha_1 \quad (3.1)$$

where N is the cascade blade number, s is the blade pitch and α_1 is the desired inlet flow angle. Downstream of test section, the transition walls were rotated to align with the blade outlet metal angle. With this procedure, a good inlet flow uniformity and outlet flow periodicity were achieved (See section 3.6). The vertical traversing gear position was fixed at the midspan of the cascade, while the horizontal traversing gear system was calibrated to give 3000 motor steps in one full blade-to-blade pitch of probe travel. By using a 30X pocket microscope, the probe deviation relative to the exit blade metal angle for all measuring cases was found to be 0.3° .

3.1.2 Test Cascade

The investigated HS1C cascade consists of seven blades and eight blade passages. HS1C is an aft-loaded modification by Corriveau and Sjolander (2003) of the HS1A cascade which has mid-loaded blades. HS1A is the midspan section of a high pressure turbine from a Pratt & Whitney Canada engine. The HS1C blade geometry and nomenclature are summarized in Figure 3.4, and its coordinates are given in Appendix A. For the roughness tests, the number of roughened blades was determined based on the cascade downstream flow conditions. Flow periodicity downstream of the measured blade must be achieved. More discussion will be given in Section 3.6.2.

Figure 3.2 Plan view of the cascade test section (Corriveau and Sjolander, 2003)

Figure 3.3 Probe alignment mechanism (Jouini, 2000)

Cascade Parameter			
Chord length, c	40.4mm	Outlet metal angle, β_2	59.0°
Blade span, H	61.0mm	Leading edge wedge angle, We_{le}	15.0°
Blade pitch, s	29.14mm	Trailing edge wedge angle, We_{te}	6.0°
Leading edge ellipse ratio, a/b	2.1	Design incidence, i_{des}	-4.5°
Leading edge ellipse minor axis, b	1.02mm	Stagger angle, ζ	22.5°
Trailing edge thickness, t	1.26mm	Aspect ratio, H/c	1.51
Throat opening, o	15.9mm	Unguided turning, θ_u	14.5°
Inlet metal angle, β_1	50.5°		

Figure 3.4 Summary of blade geometry and nomenclature (Corriveau and Sjolander, 2003)

3.2 Test Matrix and Roughness Configurations

The tested conditions are given in Table 3.1. Approximately round natural silica sand grains (7030 Industrial Quartz) were used to act as the surface roughness elements. Nine sandgrain roughness configurations were chosen. The sandgrains were sparsely distributed in spanwise-oriented bands of various widths and various locations on the suction surface of the blades. Three sizes of sandgrains were used, corresponding to k/c values of 0.0022, 0.0041 and 0.0064, where k is the nominal grain diameter and c is the true chord length. These roughness sizes are in the range of typically in-service roughness level (Kind *et al.* 1998). The roughness locations near the leading edge and roughness band widths are shown in Figure 3.5. For each size of sandgrain, the smaller band widths were obtained by removing sand from the wider bands. The experiments on smooth and rough blades were conducted at design incidence for exit Mach number between 0.40 and 1.13. The corresponding Reynolds number based on exit flow velocity and true chord varies from 4.0×10^5 to 10^6 . The measurements consisted primarily of static and total pressure and flow direction downstream of the test blades. These were taken by traversing pneumatic probes that are described in the following pages.

Table 3.1 Test conditions

Case No.	Roughness configuration				Incidence	Exit Mach number	Reynolds number	Inlet flow turbulence intensity
	Size (k/c) 10^{-3}	Band start location (x/c)	Band width ($\Delta s/c$)	Spacing (λ)				
1	6.4	0.1	0.65	~ 20	Design incidence	0.40-1.13	$4.0 \times 10^5 - 10^6$	4%
2	6.4	0.25	0.475	~ 20				
3	6.4	0.25	0.25	~ 20				
4	4.1	0.1	0.65	~ 20				
5	4.1	0.25	0.475	~ 20				
6	4.1	0.25	0.25	~ 20				
7	2.2	0.1	0.65	~ 20				
8	2.2	0.25	0.475	~ 20				
9	2.2	0.25	0.25	~ 20				
10	Smooth

Figure 3.5 Locations and roughness band widths on the blade profile

3.3 Instrumentation

3.3.1 Pressure Measuring Probe

The probes used in the present downstream wake measurements are miniature fast response static and three-hole probes designed by Islam (1999). Figure 3.6 shows their dimensions. They are miniature fast response probes. The tip of the three-hole probe has a width of 1.37 mm which is roughly 4.7% of the blade pitch, and a thickness of 0.46 mm. The static probe has a cone tip of angle 15° and a diameter 1.02 mm, which is 3.5% of the blade pitch. The two static holes of 0.3 mm in diameter are located 12 times the probe tip diameter from the tip. All probes have a maximum probe blockage area less than 0.0009 of the blade-to-blade passage area, which implies a percentage change in Mach number of less than 1.0% (Jouini, 2000). Therefore, the probe blockage effects can be neglected.

A United Sensor Model PAC-T-8-C/C-KL Pitot-static probe was used for cascade inlet total and static pressure measurements. The diameter of the probe head and stem is 3.175 mm. The total pressure hole of 1.0 mm in diameter is located in the nose of the probe, 14 stem diameters upstream of the stem, and the static holes of 0.3 mm in diameter are located 4 probe diameters downstream of the total pressure hole.

Pressures were transduced into voltages by a single miniature fast response Kulite pressure transducer (Model XCQ-062-25A), which was placed inside a Scanivalve system located outside the test section as close as possible to the probes. Approximately 2 meters of 0.86 mm inner diameter plastic tubing connected the probes to the Scanivalve.

The full-scale range of the transducer was 0 to 25 psi absolute. Calibrations of the transducer and probes are discussed in Section 3.5.5.

Figure 3.6a Carleton University Static Probe (Islam, 1999)

Figure 3.6b Carleton University Three-hole Probe (Islam, 1999)

3.3.2 Pressure Control System

The cascade exit Mach number is changed through the adjustment of the test section blowing pressure. A schematic diagram of the test section blowing pressure control system is shown in Figure 3.7. It is a closed-loop system. The control algorithm is of the proportional-derivative-integral type. The demanded blowing pressure is sent from a computer to the HP3497A data acquisition and control unit (DACU). The real blowing pressure is measured by an Omega transducer located downstream of the control valve on the upper wall of the cascade test section. The voltage output of the transducer is recorded by a microprocessor HP3437A system voltmeter and then communicated to the DACU. The difference signal between the demanded pressure and the real pressure is converted into an analog signal through unit HP44429A. The control valve is then actuated according to the control algorithm. With the use of the control system, a period of constant test section blowing pressure is maintained during each run. More information can be found in Jeffries (2000) and Jouini (2000). In the present tests the blowing pressure ranged from 1.1 to 1.6 bar absolute and was maintained constant with fluctuations within $\pm 0.4\%$ during any particular test run.

Figure 3.7 Test section pressure control system (Jouini, 2000)

3.3.3 Data Acquisition System

Figure 3.8 shows the wind tunnel data acquisition system. The data are recorded using a Hewlett-Packard HP3852A high speed data-acquisition system controlled by a microcomputer. This system is equipped with a HP44702B, 13-bit high speed voltmeter capable of sampling rates up to 100 kHz, and a HP44701A, 5½ digit integrating voltmeter and a HP44708A, 20-channel relay multiplexer with thermocouple compensation. As mentioned earlier, all pressure measurements were made using a single Kulite pressure transducer mounted in a 48-port scannivalve. The scanivalve had a scan rate of 15 ports per second. The output from the transducer is recorded by the HP44702B. The pressure signals were sampled for 0.1 second at a rate of 2000 samples per second. The scan and sampling rate were selected as the same as the experiments made by

Jeffries (2000), who did a wide investigation on this issue. Using the HP44708A, the test section inlet total temperature was measured using a type K-thermocouple. The signal output of the HP44708A was recorded by the HP44701A. A Hewlett-Packard Interface Bus (HP-IB) card interfaced the data acquisition system with the computer. Thus, the operation of data acquisition system could be controlled by the computer. Additionally, the computer sent commands from a parallel port to a motor controller to move the traversing mechanism to the next data point after measuring the current point.

Figure 3.8 Data acquisition system

3.4 Rough Surface Modeling

3.4.1 Roughness Selection Procedure

As mentioned earlier, three sizes of approximately round natural silica sand grains were used in the present study. With the test cascade chord being 40 mm, the k/c values of 0.0064, 0.0041 and 0.0022 correspond to nominal sand grain diameters of 255 μm , 165 μm and 90 μm . The sand was sorted using sieves and a sieve shaker. The sieves were stacked such that the top sieve (pass sieve) had mesh holes slightly larger than the desired nominal size and the lower sieve (retain sieve) had mesh holes slightly less than the desired nominal size. The sand which passed through the pass sieve and remained in the retain sieve was used. The nominal grain size of the sand was taken as the average of the size of the mesh holes of the pass sieve and the retain sieve. In this way reasonably uniform sand grains were obtained. Table 3.2 lists the nominal size of the sand grains and the mesh opening sizes of the pass and retain sieves.

Table 3.2 Sand grain sizes and related sieve sizes

Classification	Non-dimensional sand grain size, k/c	Nominal sand grain size, k (μm)	# of pass sieve / mesh opening size (μm)	# of retain sieve / mesh opening size (μm)
Large	0.0064	255	#50 / 300	#70 / 210
Medium	0.0041	165	#80 / 180	#100 / 150
Small	0.0022	90	#140 / 105	#200 / 74

3.4.2 Roughness Application Procedure

A roughness application apparatus was developed to enable the sand grains to be uniformly applied to the blade surface with a desired average spacing. The system, shown in Figure 3.9, consisted of a Mini-Shaker 4810, a Sieve, three Foam-Blocks, a Sweep/Function Generator 180, a Power Amplifier 2706, a Frequency Counter and an Oscilloscope. The mini-shaker has a permanent magnet vibration table at one of its ends, through which the sieve was tightly attached to the mini-shaker and could vibrate horizontally and synchronously with the table. Three foam-blocks were used to support the sieve and also to act as cushions to reduce its vertical movements. The shaker was driven by a power amplifier controlled by the sweep/function generator which provided a sweep sinusoidal signal. This enabled the frequency and amplitude of vibration to be adjusted. The frequency counter and the oscilloscope displayed and monitored these parameters. The blade to be roughened with sand was put underneath the sieve, and collected the sand grains which dropped from the sieve.

Figure 3.9 Roughness application apparatus

An automotive anti-rust clear paint (Tremclad[®], Tremco Company) was used as the glue to adhere the sand grains to the blade surface. Figure 3.10 is a photograph of typical adhered sand grains. It indicates clearly that the paint has no significant distortion effects on the shape of the attached sand grains and blade profile contour. Moreover, the paint provided a strong adhesive strength. A test was performed in which the roughened blades was subjected to an air blast from a nozzle operating at about 90-100 psig. No sand was observed to be removed from the blade surface. In addition, the paint is easily removed by acetone.

Figure 3.10 Adhered medium sand grains on blade surface

Application of sand grains to the blade surface was a trial-and-error process. Due to the desire for only partial coverage, masking tape was employed to cover the non-rough area on the blade surface. The area to be roughened was sprayed with a very thin coat of the adhesive paint. Then the blade was placed under the sieve, from which sand grains were sprinkled falling steadily and were stuck to the wet adhesive area on the

blade. The masking tape was removed after the paint coating completely dried in about one minute. If the roughness on the blade was not uniformly distributed and the spacing parameter, λ , varied by more than ± 1.5 from the desired value of 20, the roughness was washed out and application was repeated. The expected roughness spacing could be achieved by adjusting both the vibration frequency and the amplitude of the mini-shaker. For the large size sand grains, No. 50 sieve was used; the frequency was controlled around 10.8 KHZ and the gain control of the power amplifier was set at the half range. The blade was exposed for 6 seconds to the sand. For the medium size sand grains, the sieve size of No. 80, the frequency of around 12.8 KHZ, the gain at half range and an exposure of around 6 seconds were required. For the small size sand grains, the sieve size of No. 140, the frequency of around 14.2 KHZ, the gain at half range and an exposure of around 6 seconds were needed. As a precaution, each roughened blade was tested by an air blast as mentioned above, before it was installed into the cascade wind tunnel.

3.4.3 Determination of Roughness Spacing Parameter

The roughness uniformity on the blade surfaces was assessed visually by using square grids of 10 x 10 mm drawn on a transparent film, which was closely pressed onto the blade surface. Photographs of sample roughened blades covered with the large, medium and small sand grains are given in Figure 3.11, Figure 3.12 and Figure 3.13, respectively.

Figure 3.11 Rough blade covered with large size sand grains (10x10 mm grid superimposed in upper image)

Figure 3.12 Rough blade covered with medium size sand grains (10x10 mm grid superimposed in upper image)

Figure 3.13 Rough blade covered with small size sand grains (10x10 mm grid superimposed in upper image)

The value of the roughness spacing parameter, λ , was determined from a count of sand grains over a number of sample areas in the blade surface. A transparent film drawn with square grids of 5 x 5 mm was used to cover the blade surface. Fourteen grid squares near the midspan were chosen as the sample areas. The number of sand grains in these areas was counted with the help of an Edmund 12x transparent base magnifier. Knowing the average particle size and the number of individual sand grains, the spacing parameter was calculated using Simpson's (1973) definition of λ , that is,

$$\lambda = \frac{A_T}{A_F} = \frac{A_T}{\left(\frac{\pi}{4}d^2\right)n} \quad (3.2)$$

where A_T is the total sample area and A_F is the sand grain total average frontal area, n is the total number of sand grains within the areas and d is the sand nominal diameter or roughness height. Appendix B contains a figure showing the location of the fourteen areas and a table with the calculated spacing parameters for the various roughness applications.

Some efforts were made to use the image processing software ImageJ to determine the spacing parameter. The image of the sample surface was captured by a digital camera, and then converted into an 8-bit greyscale image. A threshold was set to separate the sand grains and the background sample surface. The sand grains were then counted, numbered and outlined. This method seemed convenient, but it required the image to have an intense color contrast between the sand particle and the sample surface. In practice, suitable images were hard to acquire. The threshold had to be set by judging

and this resulted in an unacceptable variation in the calculated spacing parameter. Consequently this approach was abandoned.

As mentioned before, the equivalent sand roughness height, k_s , is the significant parameter for the prediction of surface roughness effects. Dvorak-Simpson's (1969/1973) correlation for the roughness constant C is used to determine the k_s . For spacing parameter $\lambda=20$, the C is calculated by Equation 2.6, and the non-dimensional k_s/k is subsequently obtained by Equation 2.4. These equations are repeated here as Equations 3.3 and 3.4.

$$C = -5.95(1.103 \log_{10} \lambda - 1) \quad (3.3)$$

$$\frac{k_s}{k} = \exp\left(\frac{C+4}{2.439}\right) \quad (3.4)$$

Therefore, the non-dimensional k_s/k is 1.78 for spacing parameter $\lambda=20$.

3.4.4 Uncertainty Analysis of Spacing Parameter/Equivalent Sand Roughness Height

The uncertainty of a parameter is a function of the uncertainty in each of the individual measurements used in calculating this parameter. For spacing parameter, the definition formula is,

$$\lambda = \frac{A_T}{\left(\frac{\pi d^2}{4}\right)n} = \frac{A_T}{nA_f} \quad (3.5)$$

where A_T is the sample area and A_f is average frontal area of one grain and n is the number of particles within the area. The uncertainty of spacing parameter λ will be,

$$\Delta\lambda = \sqrt{\left(\frac{\partial\lambda}{\partial A_T} \Delta A_T\right)^2 + \left(\frac{\partial\lambda}{\partial n} \Delta n\right)^2 + \left(\frac{\partial\lambda}{\partial A_f} \Delta A_f\right)^2} \quad (3.6)$$

where ΔA_T , Δn and ΔA_f are the corresponding uncertainties of A_T , n and A_f . The individual derivatives are,

$$\frac{\partial\lambda}{\partial A_T} = \frac{\lambda}{A_T}, \quad \frac{\partial\lambda}{\partial n} = -\frac{\lambda}{n}, \quad \frac{\partial\lambda}{\partial A_f} = -\frac{\lambda}{A_f} \quad (3.7)$$

It can be roughly estimated that the uncertainty of $\Delta A_T/A_T$ is zero, the uncertainty of $\Delta n/n$ is ± 0.3 and the uncertainty of $\Delta A_f/A_f$ is ± 0.2 . Thus the uncertainty of λ is calculated as ± 7.2 , and the uncertainty of the corresponding non-dimensional equivalent roughness height, k_s/k , is ± 1.0 by using Equations 3.3 and 3.4.

3.5 Experimental Procedure and Data Reduction

The experimental and data-reduction procedures in the present study are essentially the same as those used by Corriveau and Sjolander (2003) and the computer software used for calculating the exit mixed-out flow quantities was developed by Jeffries (2000) and Corriveau and Sjolander (2003).

3.5.1 Cascade Inlet and Outlet Flow Measurements

As discussed by Corriveau and Sjolander (2002), lack of good inlet flow uniformity yields erroneous measured cascade performance. Cascade inlet flow measurements were made to examine the uniformity of the inlet flow. Cascade outlet flow measurements were made to assess flow periodicity and to determine profile losses and deviation angles. Figure 3.14 shows the locations of the upstream and downstream traverse measurements made for these purposes. A row of static taps was located at a distance of one chord length upstream of the leading edge. The taps are 0.25 mm in diameter and have a spacing of 7.3 mm with four taps per blade pitch. The inlet static pressure distribution in the pitchwise direction was obtained from these taps in one run. The measurement results are presented later.

The profile loss and flow deviation were evaluated from downstream traverse measurements at the middle blade of the cascade. The three-hole probe and the static probe were traversed separately in a plane located 40% true chord downstream of the blade row (Figure 3.14). Both probes were described previously. The local static pressure was obtained by the static probe, while the local total pressure and the flow angle were obtained by the three-hole probe. Besides the traverse measurements, three reference pressures were taken during each run. The first was the reference inlet total pressure (P_{01r})

Figure 3.14 Cascade blade row measurement locations (Corriveau and Sjolander, 2003)

, measured by a United Sensor Model CT-156 three-hole Cobra-Type probe located at the inlet test section (approximately 1.4 m upstream of the cascade). This probe was also equipped with a type K thermocouple to allow the test section total temperature to be measured during each run. The second was the reference inlet static pressure (P_{1r}), measured by a static tap located at one blade true chord upstream of the cascade. The third was the reference downstream static pressure (P_{2r}), measured by a static tap at the downstream measuring plane in the cascade base plate. The inlet reference pressures (P_{01r} and P_{1r}) were used to monitor the cascade inlet isentropic Mach number for each run, while the downstream reference pressure (P_{2r}) was used to monitor the pressure ratio across the cascade during each run. All reference pressures were also used to match the operating conditions for the different runs required for one complete blade pitch traverse.

Because the P_{01r} reading could not adequately represent the cascade inlet total pressure, additional measurements by a Pitot-static probe were conducted upstream of the cascade. The Pitot-static probe, described earlier, was placed at one blade true chord upstream of the cascade, and fixed at center position (the biggest black point at the cascade inlet shown in Figure 3.14) of the inlet wall with the probe aligned with the inlet flow angle. When downstream traverses were conducted, the Pitot-static probe was removed so as not to disturb the inlet flow. The measurements from the Pitot-static probe were related to the reference pressures P_{01r} and P_{1r} . The cascade inlet total pressures could then be deduced from the reference pressure measurements using the ratio of the Pitot-static probe to reference probe readings.

Due to the short runtimes for the wind tunnel, a full downstream traverse was completed in four runs (Figure 3.14). During each run, 10 measurements were made. In order to have a better definition of the downstream wake, non-uniform spacing was used. The step displacements within the blade wake region were half of those in the non-wake region. The start point for the downstream traverse was chosen so that the center of the wake was around the middle of one blade pitch at low exit Mach number running conditions. This point was then fixed for all subsequent wake traverses. The measuring points in the static-probe traverses were thereby insured to be at the same locations as the measuring points in the three-hole probe traverses.

3.5.2. Outlet Flow and Profile Loss Determination

The downstream total pressure measured by the three-hole probe at exit Mach number greater than unity must be corrected to account for the effect of the bow shock in front of the probe mouth, since the calibration of this probe is limited to exit Mach number less than unity. The calibration of three-hole probe will be discussed in Section 3.5.5. The true downstream local total pressure (P_{02y}) is the pressure upstream of the shock. By knowing the measured local total pressure (P_{02yc}) and the measured local static pressure (P_{2y}), the local Mach number upstream of the shock (M_{2y}) is obtained from the Rayleigh Pitot tube formula, i.e. Equation 3.8. Thus, the true downstream local total pressure (P_{02y}) is calculated from the normal shock relationship, i.e. Equation 3.9. The nomenclature used in the equations is shown in Figure 3.15.

$$\frac{P_{02yc}}{P_{2y}} = \left(\frac{(\gamma + 1)^2 M_{2y}^2}{4\gamma M_{2y}^2 - 2(\gamma - 1)} \right)^{\frac{\gamma}{\gamma - 1}} \frac{1 - \gamma + 2\gamma M_{2y}^2}{\gamma + 1} \quad (3.8)$$

$$\frac{P_{02y}}{P_{02yc}} = \left(\frac{1 + \frac{\gamma - 1}{2} M_{2y}^2}{\frac{\gamma + 1}{2} M_{2y}^2} \right)^{\frac{\gamma}{\gamma - 1}} \left(\frac{1 - \gamma + 2\gamma M_{2y}^2}{1 + \gamma} \right)^{\frac{1}{\gamma - 1}} \quad (3.9)$$

Figure 3.15 Total pressure probe in supersonic flow

The downstream flow angle in the pitch plane was directly extracted from the measurements of the three-hole probe. The probe was calibrated for this purpose as outlined in Section 3.5.5. The downstream Mach number and dynamic pressure were calculated by combining the static pressure measurements obtained from the static probe with the total pressure measurements obtained from the three-hole probe.

The loss coefficient and flow deviation angle for cascades are based on conditions far downstream of the blade row, where mixing has caused the flow to become uniform. Since the downstream measurements are taken relatively close to the blade row, it is necessary to calculate the mixed-out flow conditions. The mixed-out flow quantities were calculated by using the method of Amecke and Safarik (1995). This method assumes a mixing process from the measured plane to uniform conditions in a constant area channel without friction, thus the uniform flow quantities can be calculated by solving the equations of conservation of mass, momentum and energy for a control volume between the measured plane and the uniform plane. A sample of downstream traverse

measurements and calculated mixed-out quantities for smooth blades at an exit Mach number of 1.04 is shown in Figure 3.16. All of the calculations were based on the assumption that the total temperature was constant across the cascade. In order to reduce heat transfer during a test run, several pre-runs were performed to cool the cascade blades and the walls of the test section before data were collected.

The loss coefficient and Reynolds number were obtained using the mixed-out downstream quantities with the following equations,

$$Y_t = \frac{(P_{01} - P_{02})}{q_2} \quad (3.10)$$

where P_{01} is the cascade inlet total pressure, P_{02} is the mixed-out cascade outlet total pressure and q_2 is the mixed-out cascade downstream dynamic pressure ($=0.5\rho V^2$).

$$Re = \frac{c\rho_2\sqrt{\gamma RT_2}M_2}{\mu} \quad (3.11)$$

where γ and R are the air property constants, μ is the molecular viscosity, c is the true chord length of the blade and M_2 , T_2 and ρ_2 are, respectively, the mixed-out exit Mach number, the mixed-out exit static temperature and the mixed-out exit density.

Figure 3.16 Mixed-out and pitchwise distributions of the measured flow quantities, HS1C smooth cascade, $M_2=1.04$

3.5.3 Axial Velocity Density Ratio

The value of the axial velocity density ratio (AVDR) was calculated from the upstream and downstream mixed-out flow quantities. The equation is

$$\text{AVDR} = \frac{\rho_2 V_2 \cos \alpha_2}{\rho_1 V_1 \cos \alpha_1} \quad (3.12)$$

where V_1 and V_2 are the upstream and mixed-out downstream flow velocity, respectively, ρ_1 and ρ_2 are the upstream and mixed-out downstream air densities, respectively, and α_1 and α_2 are the upstream and mixed-out downstream flow angle, respectively.

Using the isentropic assumptions, the AVDR can be expressed as follows,

$$\text{AVDR} = \frac{P_2 \cos \alpha_2 \sqrt{\left(\frac{P_{02}}{P_2}\right)^{\frac{\gamma-1}{\gamma}} \left[\left(\frac{P_{02}}{P_2}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}}{P_1 \cos \alpha_1 \sqrt{\left(\frac{P_{01}}{P_1}\right)^{\frac{\gamma-1}{\gamma}} \left[\left(\frac{P_{01}}{P_1}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}} \quad (3.13)$$

where P_1 and P_2 were upstream and mixed-out downstream static pressure respectively, P_{01} and P_{02} were the upstream and mixed-out downstream total pressure respectively.

3.5.4 Blade Surface Pressure and Base Pressure Measurements

The blade surface pressures were measured using instrumented blades. According to the blade numbering scheme shown in Figure 3.14, blade 3 is instrumented with 16 pressure taps in its pressure surface, and blade 5 is instrumented with 25 pressure taps in its suction surface. The last pressure tap in blade 5 is located at the trailing edge to

measure the blade base pressure. All the static taps have a diameter of 0.254 mm. This diameter is 20% of the trailing edge thickness. The locations of the static taps on the blades are given in Appendix C.

Once the traverse measurements were completed, blade loading and trailing edge base pressure data were then collected. All reference pressures were also collected in order to obtain the operating conditions. In the present study, surface static pressure measurements were performed only for the smooth blades.

The surface static pressure coefficients were calculated by the following relation:

$$C_{ps} = \frac{(P_s)_x - P_{lr}}{q_{lr}} \quad (3.14)$$

where $(P_s)_x$ is the pressure at the blade surface static tap X, P_{lr} and q_{lr} are the reference upstream static pressure and upstream dynamic pressure respectively. The surface pressure distribution was also expressed as the surface isentropic Mach number distribution, using the following equation:

$$(M_s)_x = \sqrt{\left(\frac{2}{\gamma-1}\right) \left(\left(\frac{P_{01}}{(P_s)_x} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right)} \quad (3.15)$$

where P_{01} is the cascade upstream total pressure.

3.5.5 Probe and Transducer Calibration

The calibration for the three-hole probe was done by Jeffries (2000) and Corriveau and Sjolander (2003) in one degree steps over a range of $\pm 10^\circ$ of flow

misalignment in yaw at Mach number $M \leq 1$. In the present study the calibration data was taken from Corriveau and Sjolander (2003). The probe was calibrated in the cascade test section by removing the cascade and inserting a probe calibration rig designed by Corriveau (1998). This rig is a convergent nozzle to allow the probe to be calibrated in yaw for flow velocities up to sonic. The calibration procedure used the method of Lewis (1996). The calibration coefficients of the three-hole probe are determined as a function of yaw angle, Φ . The coefficients are:

$$\begin{aligned} K_1 &= \frac{P_1 - P_s}{P_0 - P_s} & K_{12} &= \frac{P_1 - P_2}{P_0 - P_s} & K_{13} &= \frac{P_1 - P_3}{P_0 - P_s} \\ K_{23} &= \frac{P_2 - P_3}{P_0 - P_s} & K_{\phi 2} &= \frac{P_1 - P_2}{P_1 - P_3} & K_{\phi 3} &= \frac{P_1 - P_3}{P_1 - P_2} \end{aligned} \quad (3.16)$$

where P_1 , P_2 and P_3 are the three port pressures as defined in Figure 3.17, and P_0 and P_s are the total pressure and static pressure at probe respectively. The variation of the calibration coefficients with yaw angle are shown in Figure 3.18. The coefficient K_1 was used to calculate the total pressure from three-hole probe measurements, and $K_{\phi 2}$ and $K_{\phi 3}$ were used to calculate the flow angle. Since the present calibration was made only for subsonic conditions, for supersonic conditions, the total pressure measured by the three-hole probe center port was corrected to consider the effect of normal shock at mouth of the probe port (see Equations 3.8 and 3.9). For flow angle, the calibration made by Jeffries (2000) suggested that there is no requirement for Mach number correction. More details about calibration of the Carleton University three-hole probe can be found in Jeffries (2000).

Figure 3.17 Three-hole probe numbering system and orientation (Jeffries, 2000)

Figure 3.18 Three-hole probe calibration coefficients versus yaw angle

The static probe was assumed to give correct readings without use of calibration factors. Neither the present nor previous authors did calibration measurements on the static probe.

The Kulite miniature transducer was calibrated between 0.5-2.0 bar. Figure 3.19 shows the transducer calibration apparatus. The pressure was applied gradually, first increasing to 2.0 bar absolute, then decreasing to 0.5 bar, and then increasing to atmospheric pressure at the end. No hysteresis effects were detected. As shown in Figure 3.20, the transducer calibration is highly linear. The calibration was performed every two weeks. It was found that the calibration slope and zero changed slightly. The uncertainty due to the transducer calibration is less than 0.003 bar.

Figure 3.19 Transducer calibration apparatus (Jouini, 2000)

Figure 3.20 Kulite transducer calibration curve (Model XCQ-62)

3.5.6 Leak and Continuity Checks

Leak and continuity checks on the pressure measuring system of probes, taps, pneumatic tubes and scanivalve were regularly conducted. At first the transducer calibration apparatus (See Figure 3.19) was employed to do these checks. It was found that this apparatus was not a sensitive indicator of leaks due to the large volume air chamber. An improvement was made by adding a T-connector to the air tube so that the air chamber could be isolated from the measurement system when checking for leaks. The leak and continuity check apparatus is shown in Figure 3.21. Leaks were indicated by falling of the mercury manometer height when the air tube connecting the air chamber was pinched and the probe or tap mouth was blocked; while lack of continuity (i.e. tube blockage) was shown by constant mercury height when the probe or tap mouth was left open. Continuity checks were necessary because the present tests involved sand particles

adhered to the cascade blades, opening the possibility of detached particles blocking probe or tap mouths.

Figure 3.21 Leak and continuity check apparatus

3.5.7 Measurement Uncertainty

Corriveau and Sjolander (2003) summarized the uncertainty for the turbine cascade measurements in the Carleton University High Speed Wind Tunnel. The uncertainty values were estimated using the method of Moffat (1988) for single-sample uncertainty analysis. The measurement uncertainty for the inlet and outlet flow angles is $\pm 1.0^\circ$. The uncertainty on the static pressure measurements is $\pm 2\%$ of the local dynamic pressure. The uncertainty in the cascade inlet Mach number is ± 0.0015 . The cascade exit Mach number uncertainty is about ± 0.005 for Mach numbers greater than 0.8. For lower Mach numbers the uncertainty in the Mach number increases somewhat, but the amount

of increase has not been established. The uncertainty for the mixed-out total pressure loss coefficient is estimated as ± 0.006 for Mach numbers between 0.85 and 1.1. For higher transonic Mach numbers the uncertainty is higher due to the complex shock structures. The amount of increase has not been established. The uncertainty in the AVDR values is ± 0.004 .

3.6 Cascade Flow Conditions

The cascade flow conditions of uniformity, periodicity and axial velocity density ratio (AVDR) were assessed at the midspan of the smooth cascade and roughened cascade, at the design incidence.

3.6.1 Inlet Flow Uniformity

The inlet flow uniformity was assessed from pitchwise static pressure measurements. The variation of the inlet static pressure coefficient in the pitchwise direction at exit isentropic Mach number $M_{2s}=1.1$ is shown in Figure 3.22. This shows that the central passages possess a very good uniform inlet flow. Non-uniformity in the inlet flow is only present for the outer passages near the test section side walls. Figure 3.22 indicates that good inlet flow quality is achieved.

Figure 3.22 Pitchwise variation of static pressure coefficient across the cascade inlet plane

3.6.2 Outlet Flow Periodicity

The outlet flow periodicities were examined from wake traverse measurements at mid-span of three blades, namely blades 3, 4 and 5. For the smooth cascade, the pitchwise variation of the outlet flow total pressure coefficient, Mach number and flow angle at exit Mach numbers of 0.68 and 1.04 are shown in Figures 3.23 and 3.24, respectively. For the roughened cascade, the same downstream flow quantities at exit Mach numbers of 0.71 and 1.05 are shown in Figures 3.25 and 3.26, respectively. Periodicity with rough blades was only assessed with the three middle blades roughened with medium size sand grains with the wide band width of roughness coverage. As shown by these results, good periodicity is present in both the smooth and the rough cases. This implies that three roughened blades, instead of all seven, are enough to obtain reliable results. Accordingly, only blades 3, 4 and 5 were roughened in the rough cascade tests.

(a) Outlet pitchwise Mach number variation

(b) Outlet pitchwise flow angle variation

(c) Outlet pitchwise total pressure coefficient variation

Figure 3.23 Outlet flow periodicity for smooth cascade at $M_2=0.68$, design incidence

(a) Outlet pitchwise Mach number variation

(b) Outlet pitchwise flow angle variation

(c) Outlet pitchwise total pressure coefficient variation

Figure 3.24 Outlet flow periodicity for smooth cascade at $M_2=1.04$, design incidence

(a) Outlet pitchwise Mach number variation

(b) Outlet pitchwise flow angle variation

(c) Outlet pitchwise total pressure coefficient variation

Figure 3.25 Outlet flow periodicity for roughened cascade at $M_2=0.71$, design incidence

(a) Outlet pitchwise Mach number variation

(b) Outlet pitchwise flow angle variation

(c) Outlet pitchwise total pressure variation

Figure 3.26 Outlet flow periodicity for roughened cascade at $M_2=1.05$, design incidence

The outlet periodicity appears poorer when the exit Mach number is 1.18. This can be seen in Figure 3.27, which shows the total pressure traverse measurements downstream of the middle blade of the smooth cascade at exit Mach number ranging from 0.5 to 1.2. Therefore the present tests were restricted to exit Mach numbers below about 1.13.

Figure 3.27 Variation of total pressure downstream of middle blade for various exit Mach numbers

3.6.3 Flow Two-Dimensionality and Axial Velocity Density Ratio

The axial velocity density ratio (AVDR) is an approximate measure of the two-dimensionality of the flow through the blade passages at the midspan. It is necessary for the AVDR to be equal to 1.0 for the flow to be two-dimensional, but it is not a sufficient condition. The measurements of AVDR for smooth blades and roughened blades covered with medium sand grains are presented in Figure 3.28. It can be seen that the AVDR is

nearly equal to 1.0 over the whole range of test Mach numbers. This suggests that two dimensionality in the flow measurement region is good for both the smooth and the roughened cascade experiments.

Figure 3.28 Effect of Mach number on the axial velocity ratio (AVDR), for the cascade with and without roughness

3.6.4. Repeatability of Data

Repeatability measurements were made on the middle blade (blade 4) for two roughness configurations: medium size roughness with band width $\Delta s/x=0.65$ starting at $x/c=0.10$ and medium size roughness with band width $\Delta s/x=0.475$ starting at $x/c=0.25$. As mentioned earlier, the second roughness configuration was obtained by removing a strip from the first roughness configuration. For the two repeatability trials, new roughness was applied to the smooth blades using the same nominal roughness parameters, and the spacing parameter was found to differ by 12%. Figure 3.29 shows the test results for the wide band roughness configuration and Figure 3.30 shows the

results for the narrow band roughness configuration. Good consistency was clearly achieved in these two trials.

Figure 3.29 Repeated test for total pressure loss coefficient and exit flow deviation angle versus exit Mach number for roughness configuration $k/c=0.0042$, $x/c=0.10$ and $\Delta s/x=0.65$

Figure 3.30 Repeated test for total pressure loss coefficient and exit flow deviation angle versus exit Mach number for roughness configuration $k/c=0.0042$, $x/c=0.25$ and $\Delta s/x=0.475$

4. EXPERIMENTAL RESULTS AND DISCUSSIONS

The results for measured mixed-out total pressure loss coefficient and exit flow deviation angle for smooth blades and roughened blades will be presented and discussed in this chapter. All the results were obtained at exit Mach number ranging from 0.40 to 1.13, corresponding Reynolds number based on chord length and exit velocity ranged from 4.0×10^5 to 1.0×10^6 . The results for the smooth blades will be presented first, and are used as a basis for the analysis of the roughened blade results. The comparison of results for different roughness configurations will then be presented. The complete flow data for the smooth blades and roughened blades are tabulated in Appendix D and Appendix E respectively. The smooth blade cascade was previously investigated by Corriveau and Sjolander (2003). Appendix F presents the comparison of results obtained by the present author and Corriveau and Sjolander (2003). The agreement is very good, except for one small local discrepancy.

The blade surface static pressure distributions will be referenced in the discussion of the total pressure loss coefficient results. Only the smooth blade surface pressure distributions were measured. Loading measurements for the roughened blades were not performed because of instrumentation constraints. On the present instrumented blades, the pressure tapping holes are dispersed around the midspan, so that a wide area near the midspan couldn't be covered with roughness. The scattered arrangement for the holes was to minimize aerodynamic interference between holes. For this reason, the roughened blade surface static pressures were not evaluated in direct measurements. As mentioned

before, it is generally thought that roughness has little effect on pressure distribution around the blades and on exit flow deviation angle. It can be assumed that surface static pressure distribution for the roughened blades is similar to that for the smooth blades. This hypothesis is supported by the fact that the present measurements of exit flow angle show little change between the smooth blades and roughened blades. The detailed discussion about exit flow angle will be presented later in section 4.3.

4.1 Loading Distribution for Smooth Blades

Blade loading measurements were performed for different exit Mach numbers. The isentropic Mach number distributions on the blade surface are plotted in Figure 4.1. Note that increasing isentropic Mach number corresponds to decreasing static pressure and vice versa. The farthest downstream points in all cases are obtained from the base pressure measurements. At lower exit Mach numbers, the blade displays a small region of constant pressure on the rear part of the suction surface and then a gradual diffusion before the trailing edge. This might be evidence of a separation bubble, with laminar separation and turbulent reattachment occurring on the suction surface. This result suggests that the transition on the suction surface in these cases might be of the separated-flow type, occurring around 75% axial chord length. As the exit Mach number becomes greater than unity, the passage chokes and the suction surface loading distributions remain constant over about the forward 75% of axial chord length for further increase in Mach number. For exit isentropic Mach numbers above unity, a small region of adverse pressure gradient is evident in Figure 4.1. This may be due to an embedded shock, interacting with the boundary layer, or to an oblique shock originating

at the trailing edge of the adjacent blade pressure surface (see Figure 2.4) impinging on the suction surface and interacting with the boundary layer. A recompression is observed on the suction surface at the trailing edge, with the base pressure being higher than the pressure at the rear point of the suction surface. This is somewhat at variance with Figure 2.4, suggesting that there is no expansion wave at the trailing edge on the suction surface of the present cascade. No regions of shock-induced boundary layer separation are evident.

Figure 4.1 Effects of exit isentropic Mach number on blade loading

4.2. Loss Coefficient

The total pressure loss coefficients for smooth blades and roughened blades with the nine different roughness configurations are plotted versus the exit Mach number. The roughness height, roughness location near the leading edge and roughness band width for

each roughness configuration can be found from the Test Matrix in Section 3.2. The total pressure loss coefficients for the large ($k/c=0.0064$), medium ($k/c=0.0041$) and small ($k/c=0.0022$) roughness height, each with three band widths (the wide band $x/c=0.1$, $\Delta s/c=0.65$, the medium band $x/c=0.25$, $\Delta s/c=0.475$ and the small band $x/c=0.25$, $\Delta s/c=0.25$) are shown in Figure 4.2, Figure 4.3 and Figure 4.4 respectively. The total pressure loss coefficient for constant band width and various roughness heights (large $k/c=0.0064$, medium $k/c=0.0041$ and small $k/c=0.0022$) are shown for each band width in Figures 4.5, 4.6 and 4.7. The smooth results are plotted in every figure for a comparison base.

4.2.1 Loss for Smooth Blades

As seen in Figure 4.2, the loss coefficient for the smooth blades shows a slight decrease with increasing exit Mach number and reaches a minimum at roughly $M_2=0.8$, after which it shows a steady increase as exit Mach number rises. There is a particularly steep rise at exit Mach number around unity. These trends are typical (Denton, 1993) and can be explained with reference to the measured blade surface isentropic Mach number distributions plotted in Figure 4.1. As mentioned previously, the boundary layer loss dominates the overall loss for low exit Mach number. For exit isentropic Mach number lower than 0.85, the loading measurements show a similar trend of adverse pressure gradient downstream of the throat, and the effects of Mach number on the skin friction are generally considered to be small, so the variation of loss with exit Mach number is relatively small. As exit Mach number increases to around 0.95, the loading distributions in Figure 4.1 suggest that a shock occurs within the blade passages near the trailing edge.

Figure 4.2 Total pressure loss coefficient with the large roughness height ($k/c=0.0064$) for various bands, versus exit Mach number

Figure 4.3 Total pressure loss coefficient with the medium roughness height ($k/c=0.0041$) for various bands, versus exit Mach number

Figure 4.4 Total pressure loss coefficient with the small roughness height ($k/c=0.0022$) for various bands, versus exit Mach number

Figure 4.5 Total pressure loss coefficient with the wide band ($x/c=0.1$, $\Delta s/c=0.65$) for various roughness heights, versus exit Mach number

Figure 4.6 Total pressure loss coefficient with the medium band ($x/c=0.25$, $\Delta s/c=0.475$) for various roughness heights, versus exit Mach number

Figure 4.7 Total pressure loss coefficient with the narrow band ($x/c=0.25$, $\Delta s/c=0.25$) for various roughness heights, versus exit Mach number

The total pressure loss across this shock is proportional to $(M^2-1)^3$, so this loss is very small for Mach number near 1.0. The loss due to this shock mainly comes from the effect of shock-induced thickening of the boundary layer. At exit Mach number above unity, a shock system forms at the trailing edge (see Figure 2.4), and the base pressure behind the trailing edge becomes lower as seen in Figure 4.1. The presence of the low base pressure tends to generate a very large trailing edge loss; as was stated by Denton (1993), this part of the loss comes from the intense viscous dissipation associated with the mixing that occurs downstream of the trailing edge and from the strong shock forming near this region. The rapid rise of the overall loss as exit Mach number rises above unity is attributed mainly to this large trailing edge loss. At transonic speeds, the trailing edge loss makes up a major amount of the overall loss.

4.2.2 Effect of Roughness

4.2.2.1 Exit Mach Number

The influence of the exit Mach number on total pressure loss coefficient for the nine measured roughness configurations are shown in one format in Figure 4.2 to 4.4 and in another format in Figure 4.5 to 4.7. It is observed that all the curves show a similar trend. The loss coefficient decreases to a minimum, then displays a slight increase at exit Mach number around 0.85-0.95, then as exit Mach number reaches unity it displays a rapid rise. Comparing with the smooth blades results, it is clear that roughness on the suction surface does substantially increase the total pressure loss at exit Mach number below unity. However, the loss increments due to roughness are relatively small for exit Mach numbers between 1.0 and 1.13, the highest value reached in the present tests.

At exit Mach number below about 1.0, the curves for the different roughness configurations are separated; that is, the losses are sensitive to the roughness height and width of the roughness bands. As seen from Figure 4.4, the smallest loss increment occurs in the case of the small roughness height with the narrow band, as would be expected. This roughness configuration causes the total pressure loss coefficient to increase by 43% at exit Mach number $M_2=0.5$ relative to the smooth case; at the same exit Mach number, the largest increase of total pressure loss coefficient is found from Figure 4.2 to be 134% in the case of the large roughness height with the wide band width. From Figure 4.8, which shows the wake measurements of total pressure for smooth blades and roughened blades, it is seen that the wakes for roughened cases are deeper and wider than for the smooth case. This indicates that the roughness thickens the boundary layer as would be expected. The roughness causes increased momentum loss in the flow, corresponding to increased entropy production and drag on the blades, thus increases losses. Some of the entropy production has already occurred upstream of the measurement station and more will occur in the mixing-out process. Recall that the latter losses are included in the loss coefficients since the data reduction includes the mixing-out analysis outlined in Chapter 3.

Figure 4.8 Pitchwise total-pressure coefficient distributions in the blade wake for various roughness configurations with $M_2 \approx 0.5$

As exit Mach number rises above unity, all the curves, regardless of the roughness configuration, tend to converge and lie close to the curve for the smooth blades. Some curves for roughened cases intersect; this is due to the measurement uncertainty. Thus, for M_2 above unity the total pressure loss coefficients are approximately the same for all the different roughness cases. Compared with the smooth blades, the largest increase of total pressure loss coefficient at about $M_2=1.03$ is only 18% for the case of large roughness height with the wide band, as seen in Figure 4.2. These results indicate that roughness has an only relatively weak effect on the total pressure loss coefficient near sonic exit conditions. This could be interpreted as being because of the presence of the trailing edge shock system (see Figure 2.4). Denton (1990) suggested that for choked turbine blades with realistic trailing edge thickness, the loss due to blade boundary layers is likely to be much less than that due to the trailing edge shock system. The trailing edge shock system is known to be influenced by the back pressure. The presence of low base

pressure at the trailing edge leads to the creation of strong trailing edge shocks thus giving a higher trailing edge loss. Investigations by Sieverding *et al.* (1980), and Xu and Denton (1988) proposed that at transonic speeds, higher values of boundary layer thickness at the trailing edge result in higher values of base pressure. This might suggest that a higher base pressure is present in the case of roughened blades and this would contribute a favorable influence on loss. As a result, the increased loss in the boundary layer due to roughness may be offset by a reduction in trailing edge losses. The net result is only a slight increase in the total pressure loss coefficient for roughened blades as compared with smooth blades. Figure 4.9 shows total pressure wake measurements for the smooth blades and various roughened blades for M_2 around 1.03. It is seen that the wake depth and width for the roughened blades are, in comparison to Figure 4.8, relatively little greater than for the smooth blades. The loss results reflect this.

Figure 4.9 Pitchwise total-pressure coefficient distributions in the blade wake for various roughness configurations with $M_2 \approx 1.03$

When exit Mach number rises above about 1.1, the curves for the roughened blades sometimes tend to diverge from the smooth case. This is particularly evident in Figure 4.2 and 4.5 for the wide band of large roughness, where the loss at $M_2=1.12$ is observed to be approximately 41% higher than for the smooth blades. The roughness effect on the total pressure loss seems to be aggravated. This might be attributed to the flow separation from the suction surface near the trailing edge. With increased thickness of the suction surface boundary layer due to the roughness effect, the blockage would increase, and this would tend to result in increased velocity near the trailing edge on the pressure surface. Therefore, a stronger pressure-side trailing edge shock would impinge on the adjacent blade suction surface resulting in a stronger diffusion on the suction surface. The interaction between the impinging shock and the thickened suction surface boundary layer then results in a further increase in trailing edge boundary layer thickness. The impinging shock might cause boundary layer separation on the suction surface near the trailing edge and the overall loss would then inevitably be increased even if the higher base pressure that would then tend to occur is favorable for the overall loss. Such phenomena might occur in the cases of the large roughness height with the wide band and with the medium band. Some cases do not show a larger loss increase, and this might be due to an absence of shock-induced boundary layer separation.

4.2.2.2 Roughness Height

As mentioned earlier, three sizes of roughness, applied in different band widths on the suction surface, were investigated. Figure 4.5 compares the effects of the different roughness heights for the wide band width. It is clear that roughness height has a substantial influence on total pressure loss. As roughness size increases, the increase in

loss becomes larger as might be expected. For example, compared with the total pressure loss coefficient for the smooth blades at exit Mach number $M_2=0.5$, the total pressure loss coefficient sustains an increase of 75% for the small roughness height, an increase of 94% for the medium roughness height and an increase of 134% for the large roughness height.

Corresponding results for the medium width band are shown in Figure 4.6. The effect on losses is less, as would be expected since less of the surface is roughened. Also the medium width bands start at a larger value of x/c , so transition induced by the roughness, which probably occurs, occurs farther downstream. As an example, at exit Mach number $M_2=0.5$, total pressure loss coefficients are estimated to be 50%, 67% and 75% larger than the smooth case, for the small, medium and large roughness height, respectively. This trend continues as the width of the band is further reduced, as shown in Figure 4.7. It is interesting that in Figure 4.7, the total pressure loss coefficients for the large, medium and small roughness height are nearly the same. This may be an indication that the most important effect of roughness is to induce early transition and that even the small roughness is sufficient to induce transition near the start of the band.

The above results indicate that the extent and location of the roughness, as well as its height, are important. The discussion about the effect of roughness location and roughness band width will be presented below.

4.2.2.3 Roughness Band Width

The current study made measurements using three widths of roughness bands: $\Delta s/c=0.65$ (wide), $\Delta s/c=0.475$ (medium) and $\Delta s/c=0.25$ (narrow). The medium and narrow bands both begin at the same location from the leading edge, $x/c=0.25$. The total pressure loss coefficient versus exit Mach number for the cases of the large, medium and small roughness heights with these three bands are shown respectively in Figure 4.2, Figure 4.3 and Figure 4.4. The width of the medium-width roughness band is 1.9 times the width of the narrow roughness band, but the loss for the medium-width band is only a little larger than that for the narrow band. For example, at $M_2=0.5$, the total pressure loss coefficient is 11% higher for the large roughness height, 5% higher for the medium roughness height and almost identical for the small roughness height. These results suggest that roughness location plays a more important role than roughness strip width in loss creation. The present results are consistent with the result of Suder *et al.* (1995) who found that there was little difference in transonic compressor degradation between the case of fully coated blades and the case of 50% of the suction surface coated and 10% of the pressure surface coated from the leading edge. This is also consistent with findings of Kind *et al.* (1998) in their studies of roughness effects on incompressible turbine cascades. They found that the loss increments increase less than the width of the roughness strip. This is probably because the ratio of roughness height to boundary layer thickness becomes smaller as distance from the start of the roughness increases, and also because the fluid near the surface loses most of its momentum near the start of the roughened band. Consequently, the skin friction increment becomes lower with increasing distance from the start of the roughness band.

4.2.2.4 Roughness Location Relative to the Leading Edge

As already mentioned, the location of the start of the roughness band is clearly important. This discussion is based mainly on the measurements for the roughness bands starting at $x/c=0.10$ with band width $\Delta s/c=0.65$ (wide) and at $x/c=0.25$ with band width $\Delta s/c=0.475$ (medium). Although the roughness band width for these two cases is different, the comparison is still thought to be instructive because the previous discussion has shown that roughness band width has only a relatively small effect. Examining Figures 4.2, 4.3 and 4.4 for these two roughness start-locations, it is apparent that there is a substantially larger effect when the roughness band starts at $x/c=0.1$, as compared to $x/c=0.25$. The loss increment increases as the roughness height increases. For example at $M_2=0.5$, for the large roughness height, the roughness with start-location $x/c=0.10$ gives a total pressure loss coefficient 47% higher than that roughness with start-location $x/c=0.25$, whereas 28% higher loss coefficient occurs for the medium roughness height and 21% for the small roughness height. The results indicate that roughness location is very important and roughness closer to the leading edge on the suction surface will result in more increase of loss. This is also consistent with the findings of Kind *et al.* (1998). The explanation can be quoted from Kind *et al.* (1998) that the roughness is relatively high compared to the boundary-layer thickness near the leading edge and it accordingly extracts more momentum from the flow. Also, it may well be that the roughness induces transition at the start of the bands and the entropy or loss production in turbulent boundary layers is substantially greater than in laminar boundary layers (Denton, 1993).

4.3. Exit Flow Deviation Angle

The exit flow deviation angle for smooth blades and roughened blades with the nine roughness configurations are plotted versus exit Mach number with the figure organizations as for the profile loss coefficient. Deviation angle, δ , is defined in Figure 3.4. Figure 4.10, 4.11 and 4.12 show the exit flow deviation angle for the large ($k/c=0.0064$), medium ($k/c=0.0041$) and small ($k/c=0.0022$) roughness height respectively, for the three bands (wide band $x/c=0.1$, $\Delta s/c=0.65$, medium band $x/c=0.25$, $\Delta s/c=0.475$ and narrow band $x/c=0.25$, $\Delta s/c=0.25$). Figure 4.13, 4.14 and 4.15 show the exit flow deviation angle for various roughness heights (large $k/c=0.0064$, medium $k/c=0.0041$ and small $k/c=0.0022$) for the wide band ($x/c=0.1$, $\Delta s/c=0.65$), medium band ($x/c=0.25$, $\Delta s/c=0.475$) and narrow band ($x/c=0.25$, $\Delta s/c=0.25$) respectively. The smooth results are plotted in every figure as a comparison base.

4.3.1 Smooth Blades

As observed from Figure 4.10, the exit flow deviation angle decreases with exit Mach number until it reaches a minimum at an exit Mach number close to 1.0, then increases with further increase of exit Mach number. The flow turning towards the axial direction when the downstream flow becomes supersonic is due to the requirement for increased flow area.

4.3.2 Effect of Roughness

It is seen from Figure 4.10 to Figure 4.15 that for all the investigated roughened cases, the variation of exit flow deviation angle with exit Mach number is nearly the same

as for the smooth case. As compared to the smooth blades, the roughened cases show only a slight change in the exit flow deviation angle and this change is not more than 1 degree. It is apparent that roughness has very little effect on the flow deviation angle. The same result was also obtained by Kind *et al.* (1998) in their studies of roughness effects on incompressible turbine cascade flow. The flow turning angle is directly related to the blade loading and thus to the pressure distribution. Therefore the fact that roughness has little effect on the flow turning indicates that roughness only slightly influences the static pressure distribution around the blades. For this reason, the hypothesis that the pressure distribution around the roughened blades being approximately the same as for the smooth blades, made in Section 4.1, should be reliable and valid.

Figure 4.10 Exit flow deviation angle with large roughness height ($k/c=0.0064$) for various bands, versus exit Mach number

Figure 4.11 Exit flow deviation angle with medium roughness height ($k/c=0.0041$) for various bands, versus exit Mach number

Figure 4.12 Exit flow deviation angle with small roughness height ($k/c=0.0022$) for various bands, versus exit Mach number

Figure 4.13 Exit flow deviation angle with the wide band ($x/c=0.10$, $\Delta s/c=0.65$) for various roughness heights, versus exit Mach number

Figure 4.14 Exit flow deviation angle with the medium band ($x/c=0.25$, $\Delta s/c=0.475$) for various roughness heights, versus exit Mach number

Figure 4.15 Exit flow deviation angle with the narrow band ($x/c=0.25$, $\Delta s/c=0.25$) for various roughness heights, versus exit Mach number

5. COMPUTATIONAL MODELLING AND ANALYSIS

Computational fluid dynamics (CFD) results for total pressure loss coefficient and exit flow deviation angle for the HS1C cascade with smooth blades and roughened blades will be presented in this chapter. The computational analyses were performed using the commercial Navier-Stokes code FLUENT (version 6.0). The Spalart-Allmaras turbulence model with wall functions was used for both smooth blades and roughened blades. Roughness effects were modelled by modifying the wall functions in the same way as used by Kind *et al.* (1998) for incompressible flow. The FLUENT flow field results were processed using the same methods as in the software used for the experimental data to obtain predicted loss coefficients and exit flow deviation angles. The objective of the CFD work was to assess if this approach is also satisfactory for compressible flow. This was assessed by comparison of the CFD results with the experimental results.

5.1 Turbulence Model

The compressible, Reynolds-averaged Navier-Stokes equations for conservation of mass, momentum and energy were used as the governing equations to model the cascade flow. By using the Boussinesq hypothesis, the turbulent stress is related to the mean velocity gradients using turbulent viscosity μ_t , and the turbulent heat flux is treated similarly by using the concept of Reynolds' analogy. In FLUENT, the available turbulence models to calculate the turbulent viscosity for the Reynolds-averaged Navier-Stokes equations include Spalart-Allmaras (1992), k - ϵ and its variants (White, 1991), and k - ω and its variants (White, 1991). The two-equation models in FLUENT were found to

have a stagnation point problem because their production terms for turbulence kinetic energy are expressed as $\mu_t S^2$ where S is the modulus of the mean rate-of-strain tensor. This results in an excessive production of the turbulent kinetic energy in stagnation point flows. These unphysical levels of turbulent kinetic energy convected downstream would influence the entire boundary layer significantly. For this reason, and also because it is well established as a good model for flows where viscous effects are concentrated in relatively thin boundary layers, the one-equation Spalart-Allmaras turbulence model was employed for the present computations.

5.1.1 Spalart-Allmaras Model

The Spalart-Allmaras (S-A) model is designed specifically for aerospace applications involving wall-bounded flows and has been shown to give good results for boundary layers subjected to adverse pressure gradients. This model is designed to be applied throughout the boundary layer including the viscous sublayer. It solves a transport equation for a modified turbulent viscosity quantity, $\hat{\nu}$, which is identical to the turbulent kinematic viscosity, ν_t , except in the near-wall (viscous-affected) region. In FLUENT, the transport equation for $\hat{\nu}$ is

$$\frac{D\rho\hat{\nu}}{Dt} = G_v + \frac{1}{\sigma_v} \left[\frac{\partial}{\partial x_j} \left\{ (\mu + \rho\hat{\nu}) \frac{\partial \hat{\nu}}{\partial x_j} \right\} + C_{b2}\rho \left(\frac{\partial \hat{\nu}}{\partial x_j} \right)^2 \right] - Y_v + S_v \quad (5.1)$$

where G_v is the production of turbulent viscosity and Y_v is the destruction of turbulent viscosity that occurs in the near-wall region due to wall blocking and viscous damping. The terms in the square brackets represent molecular and turbulent diffusion of $\hat{\nu}$. C_{b2} ,

and $\sigma_{\hat{v}}$ are model constants and μ is the molecular viscosity. $S_{\hat{v}}$ is a user-defined source term. The turbulent viscosity, μ_t , is computed from

$$\mu_t = \rho \hat{v} f_{v1}, \quad f_{v1} = \frac{\chi^3}{\chi^3 + C_{v1}^3}, \quad \chi \equiv \frac{\hat{v}}{\nu} \quad (5.2)$$

where C_{v1} is a model constant, f_{v1} is the viscous damping function and $\nu = \mu / \rho$. The production term, G_v , is modelled as

$$G_v = C_{b1} \rho \hat{S} \hat{v}, \quad \hat{S} \equiv S + \frac{\hat{v}}{\kappa^2 d^2} f_{v2}, \quad f_{v2} = 1 - \frac{\chi}{1 + \chi f_{v1}} \quad (5.3)$$

where C_{b1} is a model constant, κ is Von Karman's constant (0.419) in FLUENT, d is the distance from the wall, and S is a scalar measure of the deformation tensor, given by,

$$S = \sqrt{2\Omega_{ij}\Omega_{ij}}, \quad \Omega_{ij} = \frac{1}{2} \left(\frac{\partial u_i}{\partial x_j} - \frac{\partial u_j}{\partial x_i} \right) \quad (5.4)$$

The destruction term, Y_v , is modelled as,

$$Y_v = C_{w1} \rho f_w \left(\frac{\hat{v}}{d} \right)^2, \quad f_w = g \left[\frac{1 + C_{w3}^6}{g^6 + C_{w3}^6} \right]^{\frac{1}{6}},$$

$$g = r + C_{w2} (r^6 - r), \quad r \equiv \frac{\hat{v}}{\hat{S} \kappa^2 d^2} \quad (5.5)$$

where C_{w1} , C_{w2} , and C_{w3} are model constants, and $\hat{S}$ is given by Equation 5.3 and 5.4.

5.1.2 Roughness Modeling

As mentioned above, the S-A model is a low-Reynolds-number model, enabling the viscous-affected region of the boundary layer to be properly resolved (i.e. the model

will yield the correct law-of-the-wall behavior), provided that the near-wall mesh is sufficiently fine. In FLUENT, the S-A model is also implemented such that wall functions can be used to bridge the viscous-affected region between the wall and the fully-turbulent region without solving the viscous-affected inner region. The wall function approach was employed in the present studies. The roughness effect could then be modelled by using the rough-surface form of the law-of-the wall without modifying the S-A model. The roughness parameters for the roughened surface could be easily incorporated through the wall boundary condition in FLUENT.

Law-of-the-wall function for smooth surface

Assuming that the centroid of the wall-adjacent cell falls within the logarithmic region of the boundary layer, the law-of-the-wall is expressed as follows in FLUENT:

$$\frac{u}{u_\tau} = \frac{1}{\kappa} \ln E \left(\frac{\rho u_\tau y}{\mu} \right) \quad (5.6)$$

where u is the velocity parallel to the wall, u_τ is the shear velocity, y is the distance from the wall, κ is the von Kármán constant 0.4187, and $E = 9.793$.

Law-of-the-wall function for rough surface

As described in section 2.2.1, the mean velocity distribution near rough walls, when plotted in the usual semi-logarithmic scale, has the same slope $1/\kappa$ as for smooth walls, but a different intercept. Thus, the law-of-the-wall for mean velocity, when modified for roughness has the form

$$\frac{u}{u_\tau} = \frac{1}{\kappa} \ln E \left(\frac{\rho u_\tau y}{\mu} \right) - \Delta B \quad (5.7)$$

where ΔB is a roughness function that quantifies the shift of the intercept due to roughness effects. ΔB depends on the type and size of the roughness. For a sand-grain roughness, ΔB takes different forms depending on the value of nondimensional equivalent roughness height, $k_s^+ = k_s u_\tau / \nu$, where k_s is the equivalent standard-sand roughness height. In FLUENT, the whole roughness regime is subdivided into the three regimes, and the formulas proposed by Cebeci and Bradshaw (1977) based on Nikuradse's (1950) data are adopted to compute the roughness function, ΔB , for each regime.

For the hydrodynamically smooth regime ($k_s^+ < 2.25$):

$$\Delta B = 0 \quad (5.8)$$

For the transitional regime ($2.25 < k_s^+ < 90$):

$$\Delta B = \frac{1}{\kappa} \ln \left[\frac{k_s^+ - 2.25}{87.75} + C_{ks} k_s^+ \right] \times \sin \{ 0.4258 (\ln k_s^+ - 0.811) \} \quad (5.9)$$

where C_{ks} is a roughness constant, and depends on the type of the roughness.

In the fully rough regime ($k_s^+ > 90$):

$$\Delta B = \frac{1}{\kappa} \ln (1 + C_{ks} k_s^+) \quad (5.10)$$

Accordingly the wall roughness effects can be modelled by specifying two roughness parameters on selected wall boundaries: the equivalent standard-sand roughness height k_s and the roughness constant C_{ks} . For the present type of sand roughness, $C_{ks} = 0.5$ is recommended by FLUENT. The value of k_s for each of the roughnesses used in the present experiments is determined as discussed in Section 3.4.3, using Equations 3.3 and 3.4.

5.1.3 Transition

The S-A model implemented in FLUENT, as outlined in section 5.1.1 has been modified to be a fully turbulent model. The original S-A model (Spalart and Allmaras, 1992) is designed to yield a relatively smooth laminar-to-turbulent transition, with the transition point specified by the user. In the original model a tripping function is imposed at the transition point, and is turned off after that. This is done by adding a source term into the transport equation in only a small domain which should not extend outside the boundary layer or far beyond the transition point. The modified transport equation for $\hat{v}$ is given below,

$$\frac{D\rho\hat{v}}{Dt} = G_v - c_{b1}f_{t2}\rho\hat{S}\hat{v} + \frac{1}{\sigma_{\hat{v}}}\left[\frac{\partial}{\partial x_j}\left\{(\mu + \rho\hat{v})\frac{\partial\hat{v}}{\partial x_j}\right\} + C_{b2}\rho\left(\frac{\partial\hat{v}}{\partial x_j}\right)^2\right] - Y_v + \frac{c_{b1}}{\kappa^2}f_{t2}\rho\left[\frac{\hat{v}}{d}\right]^2 + S_{\hat{v}}$$

$$S_{\hat{v}} = \rho f_{t1}\Delta U^2 \quad f_{t1} = c_{t1}g_t \exp\left(-c_{t2}\frac{\omega_t^2}{\Delta U^2}[d + g_t^2 d_t^2]\right) \quad g_t = \min(0.1, \Delta U / \omega_t \Delta x)$$

$$f_{t2} = c_{t3} \exp(-c_{t4}\chi^2) \quad (5.11)$$

where the nomenclature in the above equation has the same definitions as for Equation 5.1, ΔU is the norm of the difference between the velocity at the trip and that at the field point being considered, ω_t is the vorticity at the wall at the trip point, d_t is the distance from the field point to the trip point, Δx is the grid spacing along the wall at the trip and c_{t1} and c_{t2} are constants. Compared with Equation 5.1, Equation 5.11 has added two f_t terms. The f_{t1} term is the transition trip term and f_{t2} is a damping term used to ensure that the turbulent viscosity is zero in the laminar flow region upstream of the transition. In

general, the original S-A model (Equation 5.11) can initiate the appropriate turbulent viscosity at the desired trip location and then spread downstream.

However, the S-A model as implemented in FLUENT omits the transition term. Therefore modification was required for present purposes. The User-defined functions (UDFs) available in FLUENT provide possibilities to customize FLUENT. A defined custom source term for the $\hat{\nu}$ transport equation (see Equation 5.1) was programmed by the author to be linked to the FLUENT solver. However, the variables which can be accessed from the FLUENT solver are limited, and difficulties were encountered in trying to compute the source term as defined in Equation 5.11. Instead a constant value over a small circle domain was used as the source term. The chosen value was determined from its recommended order of magnitude, $\rho U_e \delta \Delta U^2$, where δ is the boundary layer thickness and U_e is the freestream velocity. It was found that there was no difference in the solution if the source term was given the value of 100 or 10000. In the present study, a source term with value of 1000 was “manually” added at the defined transition points in a small circle domain having a diameter of 0.738 mm. The S-A model in FLUENT also drops the f_{t2} term. It is impossible to add this term into the FLUENT solver. However, it was found to be not necessary to add it. The destruction term in the transport Equation 5.1 appears to provide sufficient control to keep the flow laminar until it is tripped. The computed eddy viscosity in the laminar flow region was found to remain very low, especially when the freestream modified turbulent viscosity $\hat{\nu}$ was set roughly to be $\nu/10$ in the boundary conditions.

It is known that transition on the suction surface will tend to occur, in reality, near the suction peak on the smooth blades, and is expected to be triggered by the surface

roughness near the leading edge of the roughness bands for the present roughness types and Reynolds number (Kind *et al.*, 1998). The flow is expected to remain laminar on the pressure surface. If the solutions from the S-A fully turbulent model show un-reasonable transition, a trip term should be defined at desired transition points. For the smooth blades, it was found that the computed flow was laminar throughout on the pressure surface and that transition occurred near the suction peak on the suction surface, so no trip term was added into the FLUENT code for these computations. For the roughened blades, the trip term was added as outlined above so as to make the transition on the suction surface occur at the beginning of the roughness strips. The computed distributions of turbulent viscosity for smooth blades and roughened blades will be presented later.

5.2 Solver Implementation

FLUENT uses a control-volume-based technique to convert the governing equations to algebraic equations that can be solved numerically. There are two numerical methods: segregated solver and coupled solver. The segregated solver was employed in the present computations to solve the continuity, momentum and energy equations sequentially. The pressure-velocity coupling was achieved through SIMPLE (Patankar and Spalding, 1972) algorithms. Second-order upwinding was used to compute all convection terms of each governing equation. The segregated solver used under-relaxation to control the update of computed variables at each iteration. The default under-relaxation factors of 0.3 for continuity, 0.7 for momentum, 1.0 for density and 0.8 for modified turbulent viscosity were found to give better convergence rates. The convergence was judged not only by examining the residual levels, but also by monitoring the relevant integrated quantities of mass flow rate and lift coefficient. The

convergence criteria were set to 10^{-6} decrease of the scaled residuals for all equations except the energy equation, for which the criterion was 10^{-8} . The iteration would stop when the values of monitoring net mass flow rate through inlet and outlet and monitoring lift coefficient for blade upper and lower surface stopped changing. The initial conditions were specified by FLUENT solutions of the incompressible laminar flow. Convergence was easily achieved after a reasonable number of iterations. All the computations in this thesis were run on a computer with dual 2.8GHz Zeon processors, and a typical running time for one case was about 15 minutes.

5.3 Grid Generation and Boundary Conditions

The mesh for smooth blades is shown in Figure 5.1. It consists of 16502 cells. The mesh was generated by GAMBIT (version of 2.0) and imported to FLUENT. The cell qualities of node point distribution, smoothness, and skewness were checked and improved in FLUENT. A hybrid grid was adopted. Quadrilateral cells were used near the blade wall. These cells were aligned along the main flow direction so as to reduce the numerical diffusion error. Triangular cells were used over the rest of domain to adapt the geometry. The distance from the centroid of the wall-adjacent cell to the wall was determined to maintain y^+ falling in the wall log-law range. As recommended by FLUENT, a value of $30 \leq y^+ \leq 80$ was maintained for all the first cells. A refined mesh consisting of 59197 cells was checked and found to result in a negligible change in the solutions. The mesh for roughened blades was generated the same way as for the smooth blades except for the distance from the first cells centroid to the wall. This distance was

set to be greater than the equivalent roughness height, k_s , while at the same time ensuring that the value of y^+ for the first cells fell in the log-law range, 30-1000.

The domain for the present computations is shown in Figure 5.1. Pressure inlet and pressure outlet boundary conditions were used. The inlet flow properties of total pressure, total temperature and flow direction, along with outlet static pressure were specified directly from the experimental measurements. The outlet static pressure was used only for a subsonic outlet flow condition. For supersonic flow, the outlet pressure and the other outlet flow quantities would be extrapolated from the flow in the interior. The turbulence parameter at the inlet was specified in the form of turbulent viscosity ratio μ_t/μ . FLUENT would internally compute the free-stream value of the transported turbulence quantity, that is, the modified turbulent viscosity ($\hat{\nu}$), by combining turbulent viscosity ratio with the appropriate values of density and molecular viscosity. Spalart and Allmaras (1992) suggested that good results could be obtained by setting the freestream value of $\hat{\nu}$ to be $\nu/10$, therefore, the turbulent viscosity ratio at the inlet was set to 0.1. Periodic boundary conditions were used between the blades to achieve the repeated flow fields. On the blade walls, no-slip was prescribed, and the effects of wall roughness on selected wall strips were specified by the equivalent standard-sand roughness height and roughness constant, as outlined in section 5.1.2.

Figure 5.1 Mesh for HS1C blade

5.4 Results and Discussions

The cases for prediction are based on the experimental work. Not all the experimental cases were computed. Only a small portion of the cases covering the key aspects were computed and presented. A discussion will be given regarding the predictions for smooth blades and roughened blades for the surface loading, skin friction, total pressure loss coefficient and exit flow deviation angle.

5.4.1 Smooth Blades

The fully turbulent S-A model with low freestream turbulent viscosity ratio was primarily employed for smooth blades to evaluate its ability for prediction of transition. Figure 5.2 shows the contours of turbulent viscosity for exit Mach number $M_2=0.45$, $M_2=0.8$ and $M_2=1.1$. It is found that the turbulent viscosity has a very low value on the pressure surface and undergoes a rapid increase on the suction surface near $X/C_x=0.645$ ($X=0.024\text{m}$). From Figure 5.3 for the predicted skin friction coefficient (C_f) for $M_2=0.45$, $M_2=0.8$ and $M_2=1.1$, it is also found that C_f starts to increase at $X/C_x=0.68$, and the position of C_f increase moves forward with increasing M_2 . These results imply that the model predicts laminar flow on the pressure surface and transition on the suction surface occurring near the suction peak, as expected. Thus, there is no need to use a trip to obtain transition for the current application to smooth blades.

Figure 5.4 shows contours of predicted static pressure (gauge) for $M_2=0.45$ and $M_2=1.1$. A trailing edge shock is well captured for the case of $M_2=1.1$. A strong shock wave is evident on the suction surface just before the trailing edge, and also a shock from the trailing edge pressure side is observed to impinge on the adjacent suction surface.

Figure 5.5 shows the predicted surface isentropic Mach number distributions for various exit Mach numbers and the comparisons with the measured distributions. Discrepancies are small and mainly due to a small difference in running conditions between the predicted cases and the experimental cases. The CFD running conditions differ slightly from the experimental ones because the inlet and exit pressure boundary conditions were set at those of the smooth-blade experiments for all the CFD work. The small day to day variations in the experimental inlet and exit pressure were thus not accounted for. In general, the computational results show good agreement with the measured results. The shock location and base pressure are also well predicted, showing virtually identical values with the experiments. Significant disagreement is only evident at the trailing edge for the case of maximum exit Mach number $M_{2s}=1.14$, shown in Figure 5.5. Compared with the experiment, it is found that the trailing edge shock on the suction surface is predicted a little bit stronger and a little bit farther downstream.

Figure 5.6 and 5.7 respectively show the predicted and measured total pressure loss coefficient and exit flow deviation angle with the variation of exit Mach number. Both parameters are seen to be well predicted, agreement being within 0.01 and 1.5° , respectively. Trends are particularly well predicted. The predicted total pressure loss coefficient for $M_2=1.1$ is somewhat more above experiment than for the lower Mach numbers. This is probably due to the predicted stronger trailing edge shock, which might result in a higher total pressure loss. Overall, the disagreement is comparable with the experimental error of ± 0.006 and $\pm 1.0^\circ$ respectively for total pressure loss coefficient and flow deviation angle.

(a) $M_2=0.45$ (b) $M_2=0.80$ (c) $M_2=1.1$

Figure 5.2 Contour plots of turbulent viscosity for smooth blades at various exit Mach numbers

Figure 5.3 Predicted skin friction coefficient for smooth blades at various exit Mach numbers

(a) $M_2=0.45$ (b) $M_2=1.1$

Figure 5.4 Contour plots of gauge static pressure (Pa) for smooth blades at various exit Mach numbers

Figure 5.5 Predicted and measured surface isentropic Mach number distributions for smooth blades at various exit Mach numbers

Figure 5.6 Predicted and measured total pressure loss coefficient versus exit Mach number for smooth blades

Figure 5.7 Predicted and measured exit flow deviation angle versus exit Mach number for smooth blades

5.4.2 Roughened Blades

For the roughened blades computations, the S-A model was modified as outlined earlier to force transition to occur at the start of the roughness band. Figure 5.8 shows contour plots of turbulent viscosity for roughened cases with the small size of roughness ($k/c=0.0022$) covering wide, medium and small bands at $M_2=0.45$ and $M_2=1.1$. It can be seen that the turbulent viscosity shows a rapid rise at the beginning of the roughness bands. This indicates that transition tripping was successfully achieved in the FLUENT S-A computations at the desired locations.

As outlined previously, roughness effects were modelled by using the rough surface law-of-the wall function through the boundary condition. This approach was assessed by looking at the computed velocity profiles in wall variables. The same grids were used for the computations of the case of smooth blades and the case of roughened blades with a wide band of small roughness. Figure 5.9 shows the predicted near wall streamwise velocity profiles on the suction surface at $X/C_x=0.37$ for smooth blades and roughened blades. The profiles are shown for $X/C_x=0.37$ mainly because the streamwise velocity there is closer to x-velocity, since only x-velocity could be exported from FLUENT. As seen from Figure 5.9, the log law velocity profiles are well reproduced, and also show a downward shift ($\Delta u^+=10.7$) for the roughened case. The downward shift should also be given by Equations 5.8, 5.9 and 5.10 for any value of k_s^+ . For the present small roughness case, k_s^+ is about 169, and Δu^+ is calculated from Equation 5.10 to be 10.6, showing excellent consistency. It can be concluded that wall roughness can be

modelled effectively in FLUENT by using the rough surface law-of-the wall function as outlined earlier.

Figure 5.10 shows comparisons of predicted static pressure (gauge) contours for smooth blades and roughened blades for the small roughness in a wide band, at $M_2=0.45$ and $M_2=1.1$. Figure 5.11 shows comparisons of predicted surface isentropic Mach number distributions for smooth and roughened cases with the small or large roughness in the wide band configuration at $M_2=0.45$, $M_2=0.8$ and $M_2=1.1$. The prediction results show that, as expected, the blade loading is not changed very much by the roughness. The presence of roughness on the suction surface does not significantly affect the isentropic Mach number distribution on the pressure side, but it tends to slightly reduce the isentropic Mach number on the suction side, particularly near suction peaks. This may be due to a de-cambering effect associated with a thicker boundary layer on the aft portion of the suction surface. For $M_2=1.1$, the base pressure is predicted to be higher as roughness size increases. This could support the speculation proposed in section 4.2.2.1 that a higher base pressure might be present in the case of roughened blades. In addition, the predicted trailing edge shock on the suction surface seems to be stronger as roughness size is increased. The increased base pressure would tend to reduce losses, but this tendency is overpowered by the increase in skin friction and thicker boundary layer and wake caused by the roughness.

Figure 5.12 shows the predicted skin friction coefficient (C_f) for the large, medium and small size roughness applied in the wide band configuration at $M_2=0.45$ and $M_2=1.1$, and for the small roughness size applied in the wide, medium and narrow band configuration at $M_2=0.45$. As expected, the skin friction coefficient shows a considerable

increase where the surface is covered with roughness, and increases with the size of roughness. For the case of the large size roughness over a wide band at $M_2=0.45$, the C_f displays a nearly zero value near the trailing edge, which suggests that a flow separation might occur in this case. For the case of the large size roughness over a wide band at $M_2=1.1$, the C_f shows a tendency to reduce rapidly to zero very near the trailing edge; this is consistent with the strong local adverse pressure gradient that is evident here in Figure 5.11.

Figure 5.13 shows the predicted stagnation pressure loss coefficient versus exit Mach number for all test cases for roughened blades. The predicted trends with roughness height and roughness band starting point and band width are all as expected and essentially the same as for the experimental results, discussed in Chapter 4. As seen in Figure 5.13, all the curves converge and lie close to the curve for the smooth blades as M_2 increases above unity. This is consistent with the experimental results which also show that roughness on the suction surface has a substantial influence on the total pressure loss at exit Mach number below unity, and a relatively small effect for exit Mach number between 1.0 and 1.13. Predicted and experimental loss coefficients are compared in Figure 5.14 and Figure 5.15. The agreement between predicted and experimental results for roughened blades is not as good as for smooth blades (Figure 5.6), but the agreement is nevertheless fairly good. The loss increments due to roughness are predicted fairly well and, as already mentioned, the predicted trends agree well with those seen in the experimental results. The predicted losses tend to be higher than the experimental losses but a notable exception is for the wide band of large roughness at exit Mach

number near 0.5. This may be due to flow separation occurring near the trailing edge as suggested but not actually predicted by the computational results (see Figure 5.12a).

Figure 5.16 shows the predicted exit flow deviation angle versus exit Mach number for all test cases for roughened blades. As expected, the predicted exit flow deviation angle is not greatly influenced by the roughness effect. Roughness usually causes a slight increase in the predicted deviation angle, consistent with the slight decrease in blade loading evident in Figure 5.11. From Figure 5.17 and Figure 5.14, it is seen that predicted and measured exit flow deviation angle are in good agreement. As seen from the smooth blades (see Figure 5.7), the predicted deviation angles for the most part are lower than the measured values, the maximum difference being about 1.5° , comparable with the experimental uncertainty. The changes in deviation angle due to roughness are very well predicted.

(a) $M_2=0.45$, smooth (b) $M_2=0.45$, rough (small, $x/c=0.10$, $\Delta s/c=0.65$) (c) $M_2=0.45$, rough (small, $x/c=0.25$, $\Delta s/c=0.475$) (d) $M_2=0.45$, rough (small, $x/c=0.25$, $\Delta s/c=0.25$) (e) $M_2=1.1$, smooth (f) $M_2=1.1$, rough (small, $x/c=0.10$, $\Delta s/c=0.65$)

Figure 5.8 Contour plots of turbulent viscosity for roughen blades at various exit Mach numbers

Figure 5.9 Predicted near wall streamwise velocity profiles at $X/C_x=0.37$ on the suction surface for smooth blades and roughened blades, at $M_2=0.45$

(a) $M_2=0.45$, smooth(b) $M_2=0.45$, rough (small, $x/c=0.10$, $\Delta s/c=0.65$) (c) $M_2=1.1$, smooth(d) $M_2=1.1$, rough (small, $x/c=0.10$, $\Delta s/c=0.65$)

Figure 5.10 Comparisons of predicted contours of gauge static pressure (Pa) for smooth blades and roughened blades

Figure 5.11 Comparisons of predicted surface isentropic Mach number distributions for smooth and roughened blades at various exit Mach numbers

(a) regular scale (left) / expended scale (right)

(b) regular scale (left) / expended scale (right)

(c)

Figure 5.12 Predicted skin friction coefficient for roughened blades at various exit Mach numbers

Figure 5.13 Predicted total pressure loss coefficient versus exit Mach number for roughened blades

(a)

(b)

Figure 5.14 Comparison of prediction and experiment with different roughness heights for the wide band width: (a) total pressure loss coefficient versus exit Mach number (b) exit flow deviation angle versus exit Mach number

Figure 5.15 Comparison of predicted and measured total pressure loss coefficient versus exit Mach number for roughened blades

Figure 5.16 Predicted exit flow deviation angle versus exit Mach number for roughened blades

Figure 5.17 Comparison of predicted and measured exit flow deviation angle versus exit Mach number for roughened blades

6. CONCLUSIONS AND RECOMMENDATIONS

6.1 Conclusions

Experimental and numerical investigations were conducted in the present study to understand the effect of surface roughness on the aerodynamics of turbine cascades in compressible flow with exit Mach numbers ranging from 0.40 to 1.13. The major conclusions to be drawn from this work are:

1. Roughness on the suction surface produces large increases in profile losses. The losses are greatly influenced by the roughness height and location relative to the leading edge of the roughness band. The closer the roughness band is to the leading edge, the more loss will be produced. Compared with the roughness location, roughness band width plays a less important role in loss creation.
2. Mach number is also found to have an important effect on the magnitude of the losses. The loss increments due to roughness on the suction surface are relatively small for exit Mach numbers between 1.0 and 1.13, as compared with exit Mach numbers below unity, provided that no separation is caused by the roughness.
3. Roughness has little effect on the exit flow deviation angle, provided that it does not cause substantial flow separation. It can be inferred that roughness has little effect on static pressure distribution around the blades.
4. The use of the Spalart-Allmaras turbulence model with wall functions, with the law-of-the-wall functions modified to incorporate the effects of roughness, was proved to be effective. The predicted loading distributions over the blades, the skin friction

coefficients and the flow field solutions appear reasonable and support the experimental results.

5. Excellent agreements of predicted profile loss coefficient and flow deviation angle with experimental results were found for smooth blades.

6. The predictions of profile loss coefficient and exit flow deviation angle for the roughened blades give essentially the same trends as for the experiments. The agreement between predicted and experimental results is not as good as for the smooth blades results, but is still fairly good. A failure to predict a suspected flow separation may be responsible for a relatively poor prediction of the losses in one case.

6.2 Recommendations

Further investigations and improvements based on the current work are recommended as follows:

1. Detailed measurements should be performed to examine the blade pressure distribution and base pressure for roughened blades. The possibility of flow separation due to roughness could then be investigated, and roughness impact on the shock structure could also be studied. This would be very helpful in the interpretation of loss measurements.

2. The experimental database for roughness effects on compressible cascade flow should be expanded to include the effect of varying the roughness spacing parameter and off-design incidence.

3. Flow visualization, boundary layer measurements, skin friction measurements and transition measurements should be carried out to obtain a comprehensive awareness of roughness effects on the development of blade boundary layers. This would enable improvement and better validation of prediction models.
4. Other turbulence models should be investigated, particularly ones which have the potential to accurately predict flow separation with or without surface roughness.
5. The commercial code FLUENT has limits on capability to modify and incorporate new models and correlations to solve special problems. A self-developed Navier-Stokes CFD code would be desirable.

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APPENDIX A: HS1C BLADE COORDINATES

Table A.1 HS1C coordinates

Suction Surface				Pressure Surface			
X(in)	Y(in)	X(in)	Y(in)	X(in)	Y(in)	X(in)	Y(in)
-0.2907	0.02913	0.02562	-0.10358	-0.2907	0.02913	0.02562	0.0325
-0.28979	0.02204	0.04069	-0.095	-0.28979	0.03483	0.04069	0.0397
-0.2877	0.01528	0.05575	-0.08471	-0.2877	0.03868	0.05575	0.04752
-0.28347	0.00654	0.07081	-0.07268	-0.28347	0.04158	0.07081	0.05595
-0.27805	-0.00158	0.08588	-0.05898	-0.27805	0.04234	0.08588	0.06499
-0.27172	-0.00949	0.10094	-0.04387	-0.27172	0.0412	0.10094	0.07464
-0.26509	-0.0175	0.116	-0.02771	-0.26509	0.03838	0.116	0.08491
-0.25756	-0.02617	0.13106	-0.01062	-0.25756	0.03365	0.13106	0.09581
-0.24551	-0.03907	0.14613	0.00731	-0.24551	0.02639	0.14613	0.10736
-0.23045	-0.05364	0.16119	0.02599	-0.23045	0.01872	0.16119	0.11961
-0.21538	-0.06656	0.17626	0.04538	-0.21538	0.01243	0.17626	0.13259
-0.20032	-0.07794	0.19132	0.06546	-0.20032	0.00737	0.19132	0.14636
-0.18526	-0.08792	0.20638	0.08634	-0.18526	0.00341	0.20638	0.16098
-0.17019	-0.09661	0.22144	0.10807	-0.17019	0.00045	0.22144	0.17652
-0.15513	-0.10406	0.23651	0.13072	-0.15513	-0.00159	0.23651	0.19307
-0.14007	-0.11034	0.25157	0.15431	-0.14007	-0.00277	0.25157	0.21071
-0.12501	-0.11546	0.26663	0.1789	-0.12501	-0.00314	0.26663	0.22956
-0.10995	-0.11946	0.27748	0.19723	-0.10995	-0.00274	0.27748	0.24395
-0.09488	-0.12236	0.28682	0.21344	-0.09488	-0.0016	0.28682	0.25694
-0.07982	-0.12415	0.29465	0.22736	-0.07982	0.00027	0.29465	0.26808
-0.06475	-0.12483	0.30068	0.23826	-0.06475	0.00284	0.30068	0.2712
-0.04969	-0.1244	0.30429	0.24489	-0.04969	0.00611	0.30429	0.27099
-0.03463	-0.12281	0.3073	0.25046	-0.03463	0.01006	0.3073	0.26969
-0.01957	-0.12002	0.30972	0.25496	-0.01957	0.01469	0.30972	0.26749
-0.0045	-0.11596	0.31122	0.25797	-0.0045	0.01998	0.31122	0.2648
0.01056	-0.11052	0.31182	0.26137	0.01056	0.02592	0.31182	0.26137

PPENDIX B: SPACING PARAMETER CALCULATION

Table B.1 Spacing parameter calculation

Case #	Blade #	No. of sand grains in sample areas														Blade average	Cascade average	Sample area (mm ²)	Grain Size (μm)	λ
		1	2	3	4	5	6	7	8	9	10	11	12	13	14					
1	3	28	20	24	18	25	22	16	28	19	23	20	23	16	27	22.07	22.73	25	255	21.5
	4	18	28	24	17	23	26	19	28	15	19	23	22	24	17	21.64				
	5	29	15	23	25	14	38	27	20	20	32	18	27	26	29	24.50				
4	3	62	53	70	52	71	55	57	51	56	42	53	47	55	78	57.28	60.0		165	19.5
	4	52	78	83	50	54	72	64	52	53	70	46	65	52	85	62.57				
	5	50	63	70	52	87	55	52	51	56	48	66	83	55	54	60.14				
7	3		243	190		200	210		193	212			222	208		209.75	206.25		90	19.1
	4		170	222		206	198		211	179			196	194		197.0				
	5		220	194		180	232		205	240			225	199		212

Suction surface

Figure B.1 Locations of sample areas

APPENDIX C: LOCATION OF STATIC TAPS ON BLADE

Table C.1 Location of static taps on blade

Suction Surface		Pressure Surface	
No.	X/C_x	No.	X/C_x
1	0.004	1	0.063
2	0.031	2	0.115
3	0.061	3	0.175
4	0.104	4	0.231
5	0.155	5	0.296
6	0.209	6	0.361
7	0.265	7	0.424
8	0.327	8	0.483
9	0.386	9	0.54
10	0.443	10	0.598
11	0.497	11	0.648
12	0.546	12	0.704
13	0.591	13	0.754
14	0.631	14	0.807
15	0.667	15	0.859
16	0.699	16	0.906
17	0.733	Base pressure tap	
18	0.763		
19	0.798		
20	0.835		
21	0.871		
22	0.899		
23	0.928		
24	0.956		
25	1		

APPENDIX D: HS1C SMOOTH BLADES RESULTS

Table D.1 HS1C cascade performance data

M_2	M_{2s}	Re	AVDR	Y_t	δ
0.454	0.466	426822	1.018	0.0599	4.21
0.658	0.673	619279	1.005	0.0573	3.23
0.804	0.819	752527	1.004	0.053	2.49
0.931	0.948	860831	0.988	0.06	2.00
1.024	1.051	934600	0.999	0.0865	2.06
1.104	1.140	995986	0.983	0.11	2.84

Table D.2 HS1C blade surface isentropic Mach number distribution

		$M_{2s}=0.50$	$M_{2s}=0.70$	$M_{2s}=0.85$	$M_{2s}=0.96$	$M_{2s}=1.06$	$M_{2s}=1.16$
	X/C_x	M_s	M_s	M_s	M_s	M_s	M_s
Suction Surface	0.004	0.536	0.691	0.746	0.762	0.771	0.758
	0.031	0.551	0.723	0.795	0.824	0.833	0.828
	0.061	0.533	0.699	0.764	0.790	0.801	0.793
	0.104	0.549	0.717	0.788	0.818	0.829	0.824
	0.155	0.575	0.752	0.831	0.860	0.873	0.869
	0.209	0.593	0.778	0.857	0.885	0.899	0.893
	0.265	0.606	0.800	0.882	0.912	0.921	0.920
	0.327	0.632	0.838	0.924	0.956	0.965	0.962
	0.386	0.652	0.875	0.971	1.009	1.018	1.013
	0.443	0.670	0.915	1.032	1.078	1.090	1.087
	0.497	0.682	0.945	1.086	1.137	1.157	1.151
	0.546	0.677	0.956	1.121	1.206	1.236	1.233
	0.591	0.652	0.922	1.090	1.186	1.250	1.255
	0.631	0.623	0.883	1.051	1.143	1.191	1.185
	0.667	0.588	0.835	0.994	1.091	1.142	1.142
	0.699	0.569	0.800	0.956	1.059	1.140	1.134
	0.733	0.560	0.789	0.940	1.040	1.174	1.179
	0.763	0.555	0.782	0.941	1.037	1.217	1.235
	0.798	0.547	0.774	0.944	1.058	1.219	1.330
	0.835	0.547	0.776	0.953	1.092	1.131	1.439
	0.871	0.538	0.767	0.945	1.132	1.155	1.394
	0.899	0.530	0.755	0.929	1.148	1.193	1.244
	0.928	0.524	0.746	0.917	1.171	1.229	1.289
	0.956	0.514	0.732	0.889	1.059	1.261	1.299
	1	0.502	0.702	0.842	0.942	1.092	1.169
Pressure Surface	0.063	0.210	0.276	0.290	0.305	0.322	0.316
	0.115	0.186	0.248	0.256	0.263	0.284	0.276
	0.175	0.176	0.232	0.246	0.258	0.270	0.272
	0.231	0.185	0.242	0.255	0.271	0.286	0.270
	0.296	0.180	0.247	0.259	0.282	0.299	0.271
	0.361	0.192	0.251	0.265	0.298	0.311	0.281
	0.424	0.215	0.276	0.286	0.322	0.332	0.309
	0.483	0.225	0.289	0.307	0.344	0.340	0.329
	0.54	0.247	0.309	0.335	0.363	0.357	0.356
	0.598	0.262	0.334	0.369	0.392	0.392	0.391
	0.648	0.277	0.359	0.399	0.412	0.425	0.422
	0.704	0.298	0.387	0.429	0.450	0.470	0.455
	0.754	0.322	0.423	0.463	0.492	0.510	0.495
	0.807	0.341	0.454	0.501	0.536	0.547	0.535
	0.859	0.368	0.493	0.552	0.585	0.600	0.589
	0.906	0.398	0.536	0.610	0.654	0.666	0.660

APPENDIX E: HS1C ROUGHENED BLADES RESULTS

Table E.1 HS1C roughened blades performance data

Large roughness height, $x/c=0.10$, $\Delta s/c=0.65$							Large roughness height, $x/c=0.10$, $\Delta s/c=0.475$						
M_2	0.55	0.686	0.83	0.984	1.046	1.115	0.467	0.664	0.808	0.933	1.024	1.101	
Re	495023	625799	741238	859989	927634	979023	437504	622340	765541	858502	936389	995077	
AVDR	0.994	0.993	1	0.99	0.984	0.972	0.999	0.992	0.996	0.988	0.984	0.975	
Y_t	0.1328	0.1117	0.0947	0.0946	0.1122	0.1615	0.1082	0.0966	0.0861	0.0876	0.1035	0.1355	
δ	4.19	3.47	2.84	2.36	2.46	3.17	4.14	3.43	2.72	2.26	2.34	2.80	
Large roughness height, $x/c=0.10$, $\Delta s/c=0.25$							Medium roughness height, $x/c=0.10$, $\Delta s/c=0.65$						
M_2	0.415	0.63	0.775	0.898	1.008	1.101	0.5	0.685	0.825	0.943	1.037	1.122	
Re	396352	602852	737109	850917	939531	1008131	467752	638140	764549	869396	946475	1006944	
AVDR	1.003	0.997	0.996	0.986	0.991	0.974	0.997	0.994	1.001	0.994	0.99	0.978	
Y_t	0.0993	0.0896	0.0794	0.0812	0.0964	0.1182	0.1163	0.0984	0.0849	0.0816	0.1002	0.1317	
δ	4.42	3.62	2.82	2.27	2.40	2.63	4.38	3.56	2.87	2.39	2.56	3.08	
Medium roughness height, $x/c=0.10$, $\Delta s/c=0.475$							Medium roughness height, $x/c=0.10$, $\Delta s/c=0.25$						
M_2	0.533	0.71	0.852	0.981	1.06	1.123	0.49	0.682	0.822	0.94	1.036	1.118	
Re	491187	653297	774498	878010	952133	1001438	460048	640352	765541	871588	951638	1015877	
AVDR	1.004	0.994	1.001	0.986	0.984	0.974	0.997	0.994	0.998	0.987	0.991	0.979	
Y_t	0.0978	0.0871	0.0784	0.0815	0.1014	0.1327	0.0918	0.084	0.0769	0.079	0.0951	0.125	
δ	4.07	3.28	2.67	2.17	2.41	3.11	4.16	3.39	2.71	2.21	2.44	2.98	
Small roughness height, $x/c=0.10$, $\Delta s/c=0.65$							Small roughness height, $x/c=0.10$, $\Delta s/c=0.475$						
M_2	0.455	0.657	0.801	0.933	1.026	1.108	0.411	0.629	0.773	0.896	1.013	1.105	
Re	430317	621647	751945	865694	940942	1003112	398892	603497	740867	856836	945753	1012459	
AVDR	0.999	0.995	0.998	0.989	0.984	0.975	1.002	0.997	1	0.989	0.993	0.975	
Y_t	0.1049	0.0925	0.0782	0.0762	0.098	0.1269	0.09	0.0838	0.0736	0.0736	0.0897	0.117	
δ	4.38	3.54	2.72	2.16	2.34	2.83	4.41	3.66	2.89	2.31	2.36	2.67	
Small roughness height, $x/c=0.10$, $\Delta s/c=0.25$													
M_2	0.47	0.667	0.809	0.936	1.025	1.106							
Re	443727	629245	757378	866884	948538	1005044							
AVDR	1.005	0.999	1.004	0.995	0.994	0.985							
Y_t	0.0855	0.0809	0.0739	0.0746	0.0936	0.1218							
δ	4.03	3.27	2.58	2.06	2.20	2.81							

APPENDIX F: HS1C SMOOTH BLADES REPEATABILITY EXPERIMENT

To check experimental techniques and repeatability, tests on the smooth cascade were performed and compared with the data obtained by Corriveau and Sjolander (2003). Figure E.1 shows comparison of the total pressure loss coefficient and exit flow deviation angle versus exit isentropic Mach number. The total pressure loss coefficient compares very well except at exit Mach number near 1.1. A level off at $M_{2s}=1.1$ is displayed in the data of Corriveau and Sjolander (2003). The cause of this local discrepancy is not understood. The present deviation angle data compares well with the data of Corriveau and Sjolander. The discrepancy in deviation angle is within the measurement uncertainty of $\pm 1.0^\circ$.

Figure E.1 Total pressure loss coefficient and exit flow deviation angle versus exit isentropic Mach number, comparison with Corriveau and Sjolander (2003)