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Numerical analysis on effects of inlet pressure and temperature non-uniformities on aero-thermal performance of a HP turbine

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ABSTRACT

Unsteady numerical studies were conducted on the effects of combustor exit total pressure non-uniformity and hot streak (HS) on the aero-thermal characteristics of a high-pressure (HP) turbine stage. Firstly, four total pressure profiles including uniform, turbulent boundary layer, realistic engine combustor exit and radial reverse total pressure (higher total pressure near the endwall) were applied to reveal the vortex structures and heat transfer characteristics of the turbine. Then the four total pressure profiles coupled with inlet HS were specified to study the HS migration mechanisms. The results indicated that the turbulent boundary layer total pressure intensifies the passage vortices (PVs) while the higher total pressure near the endwall induces counter rotating vortices (CRVs), in both the stator and rotor passages. The heat transfer on the vane and blade surfaces was affected by the vortices and the extent of the impact directly depends on the intensity of the vortices, which is determined by the gradient of the total pressure. The heat transfer coefficients (HTCs) on the vane and blade pressure surface are increased by 104% and 20–30%, respectively, due to the CRV effects. The increase of tip leakage flow for the higher total pressure near the tip enhances the tip pressure side HTC by 4% comparing with the uniform case. The PVs and CRVs from the stator passage persistently exist in the rotor passage near the suction side endwall while have negligible influence on the rotor heat transfer. The migration paths of the hot streak are significantly affected by the PVs and CRVs in the stator passage, and the rotor blade surface temperature is dominated by both of the redistribution of HS in the vane passage and the vortices in the rotor passage. The PVs extend the HS radial migration and increase the vane and blade surface temperature. While the CRVs circulate the hot fluid away from the vane surface and increase the HS attenuation, thus decrease the vane and blade surface temperature significantly. The total pressure non-uniformity also decreases the turbine adiabatic efficiency by 0.6% at uniform inlet temperature condition.

1. Introduction

The temperature of the hot gas leaving the combustor exit of a modern gas turbine is well above the melting point of the airfoil metal. Long exposure to such severe heat transfer environments would result in turbine airfoils experiencing thermal fatigues and even failure. Due to the interacted effects of discrete fuel injectors, dilution and cooling flows from the liner surface and the swirling flow, combustor exit profiles of a modern gas turbine generally show total pressure and temperature distortions, which have both radial and circumferential gradients. The total temperature distortion is also known as hot streak (HS). The distortions of total temperature and pressure profiles would affect the airfoil passage secondary flow structures, and lead to complex heat transfer characteristics on the airfoil surfaces. The turbine aerodynamic and heat transfer characteristics under these non-uniform inlet conditions would significantly differ from that in uniform conditions. The turbine aerodynamic performance and heat load distributions considering the effects of these severe distortion profiles are of great significance to turbine designers for improving the turbine cooling schemes and efficiency.

The total pressure gradient is the driving factor in determining the airfoil secondary flow structures. Hermanson and Thole [1] numerically assessed the effects of three total pressure distributions. In the uniform inlet condition, no significant passage vortex (PV) was developed. The secondary flow patterns under turbulent boundary layer total pressure were consistent with the secondary flow models that proposed by Langston [2], Sharma and Butler [3].

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Nomenclature

A_n	nodal area (m^2)
AC, *Ca*	axial chord (m)
Cp	pressure coefficient (–)
CRV	counter rotating vortex
HP	high pressure
HS	hot streak
HTC, *h*	heat transfer coefficient (W/(m^2·K))
Ma	Mach number
P	pressure (Pa)
PV	passage vortex
PS	pressure surface
q	wall heat flux (W/m^2)
$\dot{q}_n$	heat flux per unit area (J/(m^2·s))
$\dot{Q}$	turbine heat transfer rate (J/s)
Re	Reynolds number
SS	suction surface
T	temperature (K)
T_u	turbulence intensity (%)
V	velocity (m/s)
$\dot{W}_{Sis}$	shaft power at isentropic condition (J/s)
X,Y,Z	Cartesian coordinates
y+	normalized distance from the solid wall (–)
$\Delta\dot{H}$	rotor exit enthalpy difference between the adiabatic and non-adiabatic expansion (J/s)
η	efficiency (%)
θ	normalized temperature (K)
ρ	density (kg/m^3)

Subscripts

0	stagnation parameter
3	rotor exit
ad	adiabatic condition
aw	adiabatic wall
max	maximal value
min	minimal value
midspan	50% span
Non-adia	non-adiabatic condition
Uni	uniform inlet condition
w	solid wall
*	mass-flow averaged parameters at the turbine inlet

The turbulent boundary layer total pressure enhanced the PV comparing with the uniform case. However, when the total pressure decreases from the endwall to midspan in the radial direction, counter rotating vortices (CRVs) were generated. The CRVs drive the endwall fluid migrating toward the midspan and influence the heat transfer patterns there markedly. Hermanson and Thole [4] studied the combined effects of combustor exit circumferential and radial total pressure non-uniformities. The flow characteristics in airfoil passage in cases with both circumferential and radial total pressure gradients could be compared and predicted from the cases considering only radial total pressure gradient [4]. Colban et al. [5] simulated the flow profiles at a combustor model exit by varying the combustor liner film cooling flow and junction slot flow. The followed experimental studies [6] confirmed the numerical results of Hermanson and Thole [1,4].

A combustor simulator was designed by Barringer et al. [7,8] to study the interacted influences of total pressure and temperature radial gradients on the HP turbine vane heat transfer. The total pressure gradient has marked influence on the vane heat transfer [9]. Due to the effects of different radial total pressure and temperature profiles, the vane heat transfer could be increased by 10–20% or decreased by 30–40% at the two vane spans [9]. It should be noted that the heat transfer driving temperature which is needed to determine the local HTC is taken from the turbine inlet temperature at the relevant vane spans. However, the effects of radial migration of temperature profiles on local heat transfer driving temperature were not considered. The radial migration of fluid would be very significant under the effects of corresponding inlet total pressure profiles. As also pointed out by Barringer et al. [10], the cold fluid at high stagnation pressure near endwall region was directed toward the lower total pressure mid-region.

Hot streak is considered individually in plenty of literatures to reveal its unsteady migration mechanisms and its effects on airfoil heat transfer. Butler et al. [11] observed the accumulation of hot gas toward the rotor blade PS which is resulting from the "segregation effect". Shang and Epstein [12], and Ong and Miller [13] reported the radial migration of hot streak toward the rotor inner endwall due to the buoyancy effect. The OTDF (overall temperature distortion factor) simulator was designed by Chana et al. [14], and was upgraded to the EOTDF (enhanced overall temperature distortion factor) by Povey and Qureshi [15] to simulate the engine combustor exit realistic temperature profiles. The subsequent studies revealed the influences of HS on the heat transfer of the HP turbine vane [16], the HP turbine blade [17], and the intermediate pressure vane [18]. The HS affected the airfoil heat load by redistributing the hot and cold gases in the airfoil passages. However, the HTC was essentially unaffected. Several passive techniques were proposed to alleviate the HS adverse effects. The mixture of HS with the vane wake weakened the segregation of hot gas towards the rotor PS when aligns the HS at the vane leading edge [19]. The decrease of the HS count at turbine inlet weakened the effects of the HS circumferential position while enhanced the transient fluctuation of heat load on the rotor blade surface [20]. Feng et al. [21] reported that higher turbine efficiency and lower heat load on the second stage stator vane could be obtained by matching the HS and second stage stator vane clocking positions. Prasad and Hendricks [22] and Rahim et al. [23] certified that the radial migration of HS in the rotor passage could be diminished by using a reverse twist vane and a compound leaned vane. However, the influence of total pressure profiles on airfoil passage secondary flow structures and thus the variation of hot streak migration mechanisms was not considered in these studies [11–23].

The residual swirling flow from the lean-burn low-NOx combustor exit would affect turbulence level, total temperature and pressure distributions at the turbine inlet [24]. Two combustor simulators with axial swirlers were designed in the University of Florence and University of Oxford [25,26], respectively, to study the effects of swirling flow on the turbine aero-thermal performance. The measured total pressures at the simulator exit showed both radial and circumferential total pressure gradients. Qureshi et al. [27] presented that the inlet swirl increases the stator vane surface *Nu* by 50–200% with the greatest changes near the leading edge, while its effects on the rotor blade are less significant than on the stator vane [28]. It is noted that the experimental works of Qureshi et al. [27,28] only investigate a swirl-only case. In the unsteady numerical simulations of Khanal et al. [29] and Rahim et al. [30], the interacted swirl and hot streak influence on turbine heat transfer were analyzed.

Obviously, the liner surface dilution and film cooling flows would mainly affect the combustor exit radial total pressure