NUMERICAL ANALYSIS OF FILM COOLING EFFECTIVENESS USING COMPOUND COOLING HOLES AT THE END OF GAS TURBINE ENGINE COMBUSTOR

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To my beloved my father and my mother

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This thesis could not have been written without the assistance of my supervisors; Dr. *Nor Azwadi Che Sidik*Who persuaded and challenged me through my academic program.

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ABSTRCT

By using the well-known Bryton cycle, great turbine industries strive to extend the turbine inlet temperature and increase engine performance. However the turbine inlet temperature increment creates harsh environment for the downstream components of the combustor and so it is needed to design a cooling technique. The blowing ratio increase, caused to cooling effectiveness enhancement, however, for the traditional cooling system, the coolant not attached better on the surface at higher blowing ratios and therefore, it is required to restructure the cooling holes. Compound cooling holes is the useful way to this achievement. But, Most of the previous studies paid attention on the using compound cooling holes on the turbine blades and there is a lack of research on the application of these holes at the end wall of combustor. This study was accomplished in order to investigate the effects of cylindrical and row compound cooling holes with alignment angle of 30 degree, 60 degree and 90degree. The combustor used in the study is Pratt and Whitney gas turbine engine. This model was simulated and analyzed with a commercial finite volume package ANSYS FLUENT 14.0 to gain fundamental data.The entire findings of the study showed that with using the row compound cooling holes near the end wall surface, film cooling effectiveness is doubled compared to the cooling performance of baseline case. To conclude, as different arrangements of cooling holes affect the film cooling performance, it is strongly recommended to use a combination of compound cooling holes with different alignment angles for cooling panels.

ABSTRAK

Dengan menggunakan kitar Bryton yang terkenal, industri turbin berusaha untuk menambahkan lagi suhu di alur masuk turbin dan meningkatkan prestasi enjin. Walau bagaimanapun pertambahan suhu di alur masuk turbin mewujudkan persekitaran yang kasar untuk komponen-komponen di hiliran pembakar dan menyebabkan perlunya kepada mereka-bentuk teknik penyejukan. Peningkatan nisbah penyemburan, menyebabkan peningkatan pada keberkesanan penyejukan, walaubagaimanapun, untuk sistem penyejukan tradisional, bahan penyejuk tidak melekat dengan baik di permukaan pada nisbah penyemburan yang tinggi, dan disebabkan itu, ia memerlukan penyusun semula lubang-lubang penyejukan. Penyejukan lubang secara kompoun adalah cara yang berguna untuk pencapaian ini. Tetapi, kebanyakan kajian terdahulu memberi perhatian terhadap penyejukan secara kompoun pada bilah turbin dan kurang penyelidikan berkenaan dengan penggunaan lubang ini di dinding akhir ruang pembakar. Kajian ini dijalankan untuk menyiasat kesan silinder dan deretan lubang penyejukan secara kompoun dengan sudut penjajaran 30 darjah, 60 darjah dan 90 darjah. Pembakar yang digunakan dalam kajian ini adalah Pratt dan Whitney enjin turbin gas. Model ini telah simulasi dan dianalisis dengan menggunakan ANSYS FLUENT 14.0 komersial untuk mendapatkan data-data asas. Seluruh hasil kajian menunjukkan bahawa dengan menggunakan deretan lubang-lubang penyejukan secara kompaun berhampiran dengan permukaan dinding akhir, keberkesanan selaput penyejukan adalah dua kali ganda berbanding dengan prestasi penyejukan kes asas. Kesimpulannya, penyusunan lubang penyejukan yang berlainan memberi kesan kepada prestasi selaput penyejukan, sangat disyorkan untuk menggunakan kombinasi penyejukan lubang secara kompoun dengan sudut penjajaran berbeza untuk panel-panel penyejukan.

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AR	-	Aspect Ratio
CFD	-	Computational Fluid Dynamics
CIP	-	Cubic Interpolated Pseudo-particle
CV	-	Control Volume
FDM	-	Finite Difference Method
FEM	-	Finite Element Method
FVM	-	Finite Volume Method
GUI	-	**G**raphical User Interface
LES	-	Large Eddy Simulation
NSE	-	Navier-Stokes Equation
PIV	-	Particle Image Velocimetry
PDE	-	Partial Differential Equation
PSP	-	Pressure Sensitive Paint
TLC	-	Transient Liquid Crystal

LIST OF SYMBOLS

A_{jet}	-	Area of the cooling jet
BR	-	Blowing ratio
C_d	-	Discharge coefficient
C_p	-	Specific heat
d	-	Trench depth
D	-	Film-cooling hole diameter
D_1	-	First row dilution hole diameter
D_2	-	Second row dilution hole diameter
DR	-	Density ratio
E	-	Specific energy
g	-	Gravitationalacceleration
h	-	Specific enthalpy
h_0	-	Specifictotal enthalpy
H_{in}	-	Combustor inlet height
I	-	Momentum flux ratio
k	-	Thermal conductivity
L	-	Combustor length
$\dot{m}$	-	Mass flow rate
N	-	Number of film-cooling holes
p	-	Pressure
p_r	-	Prandtl number
P_0	-	Total pressure
P_s	-	Static pressure
q	-	Heat flux
Ra	-	Rayleigh number
S_M	-	Source of energy
S_p	-	Pitch wise film-cooling hole spacing

S_s	-	Stream wise film-cooling hole spacing
t	-	Time
T	-	Local temperature
T_c	-	Temperature of coolant
T_∞	-	Temperature of mainstream
u	-	Velocity in x direction
u_{jet}	-	Velocity of cooling jet
u_∞	-	Velocity of mainstream
U	-	Dimensionless velocity in x direction
v	-	Velocity in y direction
V	-	Dimensionless velocity in y direction
VR	-	Velocity ratio
w	-	Velocity in z direction
W	-	Combustor width
W_T	-	Trench width
x	-	Stream wise distance
X	-	Dimensionless stream wise distance
y	-	Pitch wise distance
Y	-	Dimensionless pitch wise distance
z	-	Span wise (vertical) distance
Z	-	Dimensionless span wise (vertical) distance

Greek Symbols

α	-	Thermal diffusivity
β	-	Volumetric thermal expansion coefficient
Γ	-	Diffusion coefficient
δ	-	Height of the constriction
∇	-	Nabla operator
θ	-	Dimensionless temperature
μ	-	Dynamic viscosity
ν	-	Kinematic viscosity
ρ	-	Density
ρ_{jet}	-	Density of cooling jet

ρ_∞	-	Density of mainstream
τ	-	Viscous stress
τ_t	-	Dimensionless time
σ	-	Tangential velocity
$\emptyset$	-	General variable
ψ	-	Stream function
Ψ	-	Dimensionless stream function
ω	-	Vorticity
Ω	-	Dimensionless vorticity

Superscript

n	-	Current value
$n+1$	-	Next step value

Subscript

i	-	x direction node
j	-	y direction node
$max\ i$	-	x direction maximum node
$max\ j$	-	y direction maximum node

CHAPTER 1

INTRODUCTION

1.1 Introduction

The turbine propulsion science history came back to 150 B.C; In that time,the first steam engine the aeolipile is shown in Figure (1.1) was created by the Egyptian mathematicians and philosophers called Hero. In this engine, a simple closed spherical vessel was set up on bearings and this mechanism allowed it to have rotational movement due to the exerted tangential forces which are created by the steam discharge of the nozzles.Thereafter, in 1930, Frank Whittle received the patent for the first gas turbine engine invention.

Modern gas turbine industries strive for higher engine efficiencies and power to weight ratio. Brayton cycle is a key to this study. According to this cycle, the turbine inlet temperature should increase [1] to gain more efficiency. However, the operating temperature is such above that all materials cannot resist against this value of temperature [2]. Furthermore, increasing the turbine inlet temperature creates an extremely harsh environment for critical downstream components such as turbine vanes.

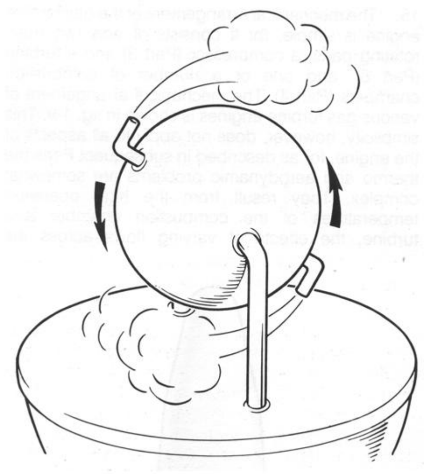

Figure 1.1 Schematic of aeolipile engine

Figure (1.2) shows that how the incidence of such condition can destroy the critical components downstream the combustor. On the other hand, based on our observation, the interaction between film cooling and mainstream makes a complex temperature and velocity profiles.

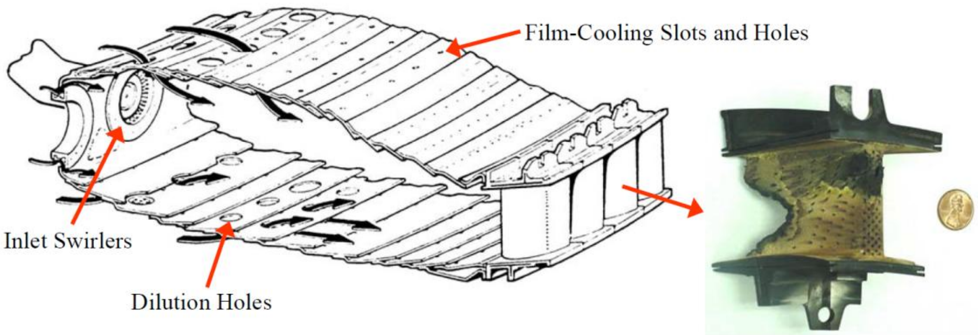

Figure 1.2 Schematic of annular combustor and the turbine first vane damage

In addition, while, the existence of complete uniform temperature and velocity profile at the end of combustor simulator is desirable, this condition is rarely achieved as a result of non-uniformities at the end of combustion chamber. The hot layers lead to early premature wear and turbine components failure. Also, turbine inlet non combusted fuel mixes with cooling layers leading to catastrophic failures of the engine. So the rate of exit velocity and temperature profiles of the combustor is critical in the turbine inlet secondary flow simulation. Therefore, a cooling technique must be applied to prevent the thermal degradation of critical components.

While, the early gas turbine engines functioned at temperature range of 1200°Cto 1500°C, the advanced engines operated at the turbine inlet temperature of 1950°Cto 2010°C. However, Figure (1.3) shows that the turbine inlet temperature increased above 2000°C with new patterns of cooling since the first of 21^{th}century [3].

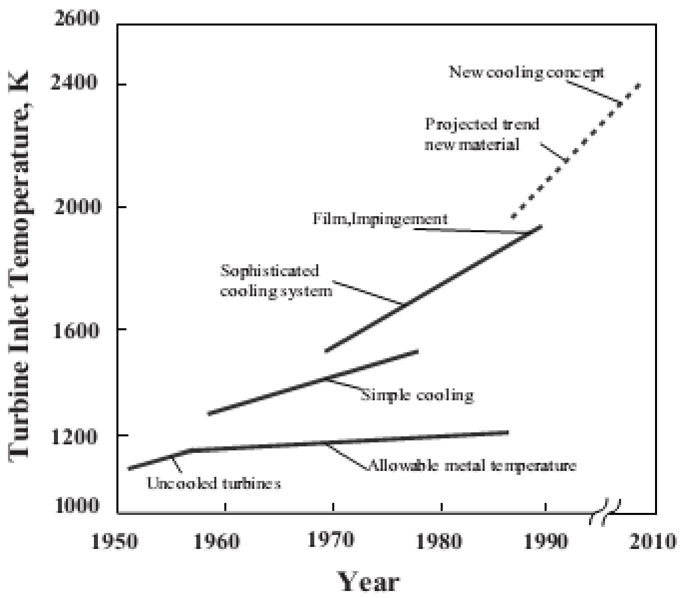

Figure 1.3 Gas turbine inlet temperature developments per year

Gas turbine cooling classified into two different schemes: internal cooling and external cooling. In the internal cooling method, coolant provided by the compressor, is forced into the cooling flow circuits inside turbine components. In the external way, the injected coolant is directly perfused from coolant manifold to save downstream components against hot gases. In the external cooling, coolant is used to quell the heat transfer from hot gas stream to a component. External cooling contains several ways. Film cooling is the most well-known method of preservation. Figure (1.4) shows that in this system, a low temperature thin boundary layer such as buffer zone is formed by cooling holes and attached on the protected surface. To improve the film cooling effectiveness, it is needed to increase the mass flux ratio. However, it is required to achieve a better attachment of coolant on the surface especially at higher blowing ratios. By trenching cooling holes, the flow behavior and thermal Characteristics are modified. In the trenched cooling holes, the injected coolant is suddenly spread before exiting the cooling holes and entering the main flow and as a result enhances the film cooling effectiveness [4].

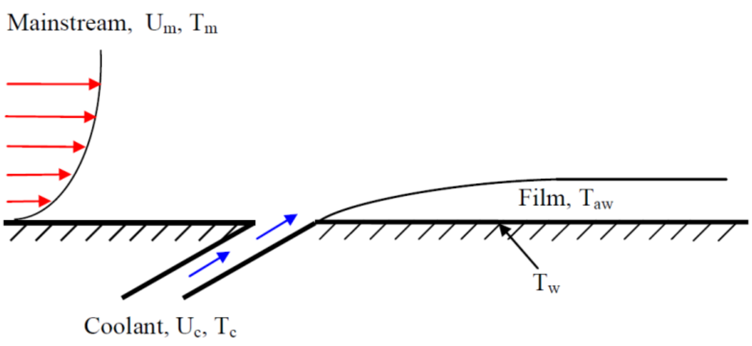

Figure 1.4 Schematic of film cooling

1.2 Problem Statement

End wall of the combustion chamber can be damaged by the hot gases which flow inside a combustor and increasing the film cooling effectiveness above these surfaces is an important issue. Most of the previous studies paid attention on the using compound cooling holes on the turbine blades and the application of these holes at the end wall combustor is very few. The effect of alignment angles of the compound cooling holes has not been tried by past researcher.

1.3 Objective of Research

The aim of this study is to find out the effects of compound cooling holes on the thermal and flow field characteristics near the end wall surface of a combustor. Investigate and analysis compound cooling hole under different alignment angles.

1.4 Research Scopes

The combustor used in the study is representation of a Pratt and Whitney gas turbine engine that is shown in Figure (1.5). The combustor simulator included four different cooling panel holes. Each panel included many cooling holes. However the second and third one contained two rows of dilution jets. In the combustor, the dilution jets and cooling flow staggered in the stream wise direction and aligned in the spanwise direction as well. The present control volume not included high momentum dilution. In this case temperature is 332 degree kelvin. The current study has been performed with turbulence model and flow is compressible. In this study α is 30 and β are, 30, 60 and 90 degree.

In this research, the gas turbine engine was simulated and analyzed with a commercial finite volume package ASYS FLUENT 14.0 to gain fundamental data. The current study has been performed with Reynolds-averaged Navier-Stokes turbulence model (RANS) on internal cooling passages. Furthermore, the two-dimensional representation of a part of combustor endwall was simulated and a program will be written in the finite difference method to solve the problem.

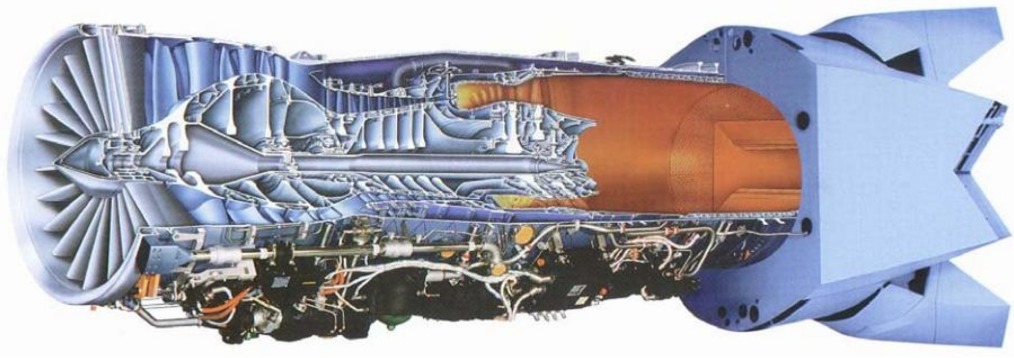

Figure 1.5 Schematic of Pratt & Whitney turbine engine

1.5 Research Contributions

According to the investigations which were done on the previous literatures and the principle objective of the current research, it is needed to have taken new steps to develop a database documenting in this field of study.

The literatures highlighted that, the first area that faced with outlet hot gases is the combustor end wall surface and therefore, cooling this area is very important to protect that and increase the expected life of this critical component. So, it is needed to increase the effectiveness of film cooling. Enhancement of mass flux ratio is the key of this. But, as stated before, with increasing the blowing ratio, the coolant not attached well on the protected surface and changing the structure of the cylindrical cooling holes is an important issue.

As stated in the previous literatures, it is found that the row compound cooling holes have more influence on the film cooling performance compared to the individual compound cases. On the other hand, according to the structure of cylindrical cooling holes placement at the end of combustor simulator, and the effects of different alignment angles of row compound cooling holes, this is another subject that motivated the researcher.

According to the effects of the variety of coolant mass flux ratios on the film cooling performance downstream the cylindrical and row compound cooling holes near the combustor exit end wall surface, it is necessary to study the effects of blowing ratios.

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