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PERFORMANCE EVALUATION OF SIMPLE AERO GAS TURBINE CYCLE WITH TRANSPIRATION COOLING OF GAS TURBINE BLADES

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ABSTRACT

Aero gas turbine engines have been using mainly film cooling and convection cooling techniques. For operating at high gas temperature amongst different cooling options the transpiration cooling technique has emerged as the most promising technique to improve the gas turbine cycle performance by allowing higher turbine inlet temperatures. This paper investigates the advantages of transpiration cooling technique over film cooling used in simple aero gas turbine cycle. The effects of variation of TIT and compressor pressure ratio on cycle performance are presented. TSFC with transpiration cooling is slightly higher than with film cooling in higher TIT range. However comparatively greater specific thrust is obtained with transpiration cooling than with film cooling. At a TIT of 1900 K and compressor pressure ratio of 21 the specific thrust developed with film cooling is 980 N-s/kg while that with transpiration cooling is 1013 N-s/kg.

Keywords: Aero gas turbine cycle, Transpiration cooling, Film cooling, Performance analysis.

1. INTRODUCTION

For last many decades the gas turbines are being used to produce either specific thrust power in an aircraft jet engine or shaft power to turn an electric generator. The thermal efficiency and power output of a gas turbine are critically dependent on the gas temperature at the turbine inlet and on compressor pressure ratio. With the use of higher gas turbine inlet temperature, the gas turbine blades are exposed to a continuous flow of gas that may enter the turbine at a temperature between 850^{o}C to 1700^{o}C. However the turbine blade material limitations do not allow for use of such higher TITs. To withstand higher gas temperatures, the aero gas turbine engines have been using various blade cooling techniques e.g. convection cooling, film cooling, impingement cooling techniques etc.

In film cooling the coolant is ejected from a few rows of holes on the blade surface forming a film of low temperature coolant which prevents the blade surface from coming in contact of hot expanding gases.

In transpiration cooling technique the coolant is ejected out from a perforated like blade surface and completely shrouds the blade surface by formation of a large number of small conjoint films of coolant, all along the blade surface, so the cooling effectiveness is better. At steady state operation a thermal equilibrium between the coolant and the solid wall is reached. The coolant film partially absorbs the convective wall heat flux and thus reduces the heat flux conducted into wall to a certain amount, just like actual film cooling. The heated coolant at the film surface is transported downstream by the momentum of the hot gas flow and is continuously replaced with fresh coolant flowing out of the wall. Due to mixing of the coolant with the expanding gases a balance is to be obtained between the adverse effect of the coolant diluting expanding gases and highest possible cooling effectiveness for attainment of highest TIT. Studies have shown that transpiration cooling technique uses the coolant more effectively than the film and convection cooling techniques [1].

Present study investigates the advantages of transpiration cooling technique over film cooling used in simple aero gas turbine cycle. A three stage gas turbine cycle is considered for the performance analysis. The key performance parameters include specific thrust, thrust specific fuel consumption, coolant requirement etc.

2. SIMPLE COOLED AERO GAS TURBINE CYCLE

A simple aircraft gas turbine consists of a diffuser, compressor, combustion chamber, gas turbine and nozzle. To prevent air separation on the compressor blades, the airflow entering the compressor should have a low Mach number, in the range of 0.4 to 0.7. For this purpose diffuser is used which reduces the high-speed airflow to lower speeds and also recovers pressure. The compressor increases the pressure of the gas entering into it. The high-pressure air delivered by the compressor is fed into the combustor where it is mixed with fuel and ignited to raise its temperature and enthalpy. The enthalpy of high temperature gas is converted to shaft work by the gas turbine, which is used to run the compressor. The high pressure air exiting the turbine is accelerated to high velocity in the nozzle thus producing thrust. The nozzle exit static pressure is the same as the inlet static pressure. A part of the compressed air is bled from the compressor for gas turbine blade cooling. This air bled from compressor is not available for the combustion process and the mixing of the coolant with the expanding gases causes loss of performance in the gas turbine. The compression and expansion process with fluid friction have been considered polytropic [2, 3]. Thermodynamic modeling considers air admitted in compressor to behave as an ideal gas.

For the given the flight speed, and the free-stream static temperature and pressure, at a given altitude, the total pressure and temperature at the end of diffusion are given by [4].

$$\frac{p_{a,dif,ex}}{p_{a,dif,in}} = \left(1+\eta_{dif}\frac{\gamma-1}{2}M^2\right)^{\frac{\gamma}{\gamma-1}} \tag{1}$$

$$\frac{T_{a,dif,ex}}{T_{a,dif,in}} = \left(1+\frac{\gamma-1}{2}M^2\right) \tag{2}$$

From the diffuser the air goes to compressor. The temperature of air at compressor exit and coolant bleed air at any stage during compression is found as:

$$\frac{T_{wa,cmp,ex}}{T_{a,cmp,in}} = \left(r_p\right)^{\frac{1}{\eta_{p,cmp}}\left(\frac{\gamma-1}{\gamma}\right)} \tag{3}$$

The specific heat of atmospheric air varies with temperature and thus the ratio of specific heats (γ) also varies with temperature.

Using mass and energy balance the work input to the compressor and fuel burning rate is found as described in [5]. The governing equations are,

$$\dot{m}_{a,cmp,in} = \dot{m}_{a,cmp,ex} + \sum_{i=1}^{n}\dot{m}_{ci} \tag{4}$$

$$W_{cmp} + \dot{m}_{a,cmp,in}\cdot h_{a,cmp,in} = \dot{m}_{wa,cmp,ex}\cdot h_{wa,cmp,ex} + \sum_{i=1}^{n}\dot{m}_{ci}.h_{ci} \tag{5}$$

$$\dot{m}_{a.cc.in} + \dot{m}_f = \dot{m}_g \tag{6}$$

$$af.h_{a,cc,in} + \eta_{cc}.LHV = (af+1).h_g \tag{7}$$

Assuming no extra power is extracted to drive turbine auxiliaries, the gas turbine should generate enough power to drive the compressor.

$$\dot{m}_{g,trb,in}.h_{g,trb,in} + \sum_{i=1}^{n}\dot{m}_{ci}h_{ci} - \dot{m}_{g,trb,ex}.h_{g,trb,ex} = \frac{1}{\eta_{mech}}\left[\dot{m}_{wa,cmp,ex}.h_{wa,cmp,ex} + \sum_{i=1}^{n}\dot{m}_{ci}.h_{ci} - \dot{m}_{a,cmp,in}.h_{a,cmp,in}\right] \tag{8}$$

Considering the expansion process in gas turbine to be polytropic, the pressure at turbine exit is calculated as

$$\frac{TIT}{T_{g,trb,ex}} = \left(\frac{p_{g,trb,in}}{p_{g,trb,ex}}\right)^{\eta_{p,trb}\left(\frac{\gamma-1}{\gamma}\right)} \tag{9}$$

For cooling of turbine blades air is bled from the compressor and is introduced into coolant injection holes on the blade surface where it mixes with expanding gases. Assuming adiabatic mixing of coolant and main flow the mixture properties i.e. temperature, specific heat, enthalpy etc are calculated as function of temperature and composition of mixed gas as described in [5].

The total pressure losses in mixing of coolant and mainstream, is calculated as [6].

$$\frac{\Delta p}{p_{trb,in}} = -\frac{\dot{m}_c}{\dot{m}_g}\gamma\frac{Ma_g^2}{2}.\left\{1+\frac{T_c}{T_g}-2\left(\frac{T_c}{T_g}\right)^{1/2}\cos\alpha\right\} \tag{10}$$

The coolant requirement for film cooling is given by [7]

$$\frac{\dot{m}_c}{\dot{m}_g} = \frac{c_{pg}}{c_{pc}}\lambda.St_g.\left[\frac{(T_g-T_{bo})-\varepsilon_{aw}\left[T_g-\{T_{ci}+\eta_c(T_b-T_{ci})\}\right]}{\eta_c(T_b-T_{ci})}\right] \tag{11}$$

And the coolant requirement for transpiration cooling of gas turbine blades is given as [5].

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$$\frac{\dot{m}_c}{\dot{m}_g} = \lambda.St_g.$$
$$\ln\left[\frac{c_{pg}}{c_{pc}} \cdot \frac{(T_g - T_b) - \varepsilon_{aw}\left[T_g - \{T_{ci} + \eta_c(T_b - T_{ci})\}\right]}{\eta_c(T_b - T_{ci})} + 1\right] \tag{12}$$

Assuming no loss in passing of the gas from turbine exhaust to nozzle, the temperature of exhaust gas from nozzle exit is given by

$$\frac{T_{g,noz,ex}}{T_{g,noz,in}} = \left(\frac{p_{g,noz,ex}}{p_{g,noz,in}}\right)^{\eta_{noz}\left(\frac{\gamma-1}{\gamma}\right)} \tag{13}$$

Using energy balance the gas jet velocity at nozzle exit is

$$h_{g,noz,in} - h_{g,noz,ex} = \frac{c_j^2}{2} \tag{14}$$

Thrust , F = $m_{g,noz,ex}.c_j - m_{a,dif,in}.c_i$ (15)

Propulsive power, $P = \frac{1}{2}\left[m_{g,noz,ex}.c_j^2 - m_{a,dif,in}.c_i^2\right]$ (16)

Cycle thermal efficiency is

$$\eta = \frac{P}{\dot{m}_f.LHV.\eta_{cc}} \tag{17}$$

Thrust specific fuel consumption is

$$TSFC = \frac{3600.\dot{m}_f}{F} \tag{18}$$

Table 1: Aero gas turbine cycle input parameters

Altitude	6000 (m)
Aircraft speed	800 (km/hr)
Diffuser efficiency	95%
Polytropic efficiency of compressor ($\eta_{p,cmp}$)	90.5 %
Combustor efficiency (η_{cc})	98.5 %
Combustor Pressure loss	3 % of entry pressure
Polytropic efficiency of gas turbine ($\eta_{p,trb}$)	90 %
Mechanical efficiency of transmission (η_{mech})	99 %
Nozzle efficiency	95%
Internal cooling efficiency (η_c)	0.75
Adiabatic wall film effectiveness (ε_{aw})	0.4
Allowable gas turbine blade temperature	830^{o}C
Air composition (by % volume)	O_2 : 21; N_2 : 79
LHV of fuel	43,000 (kJ/kg)
Stoichiometric air-fuel ratio	15:1
Coolant injection angle (α)	30^o

3. RESULTS AND ANALYSIS

Performance evaluation of simple turbojet gas turbine cycle with film and transpiration cooled gas turbine blades has been carried out for input data given in Table1.

For cooling of gas turbine blades compressed air bled from compressor is used as coolant. With increasing TIT the coolant requirement increases for both film as well as transpiration cooling as depicted in Fig 1.

Fig. 2 shows the variation of specific thrust for film cooled and transpiration cooled blades with TIT for different compressor pressure ratio. As expected specific thrust increases for both the film as well as transpiration cooling with increasing TIT and compressor pressure ratio. Comparatively greater specific thrust is obtained with transpiration cooling than with film cooling. At a TIT of 1900 K and compressor pressure ratio of 21 the specific thrust developed with film cooling is 980 N-s/kg while that with transpiration cooling is 1013 N-s/kg. This is attributed to availability of more working gas for expansion and, a reduction in dilution losses due to lesser coolant requirement in transpiration cooling.

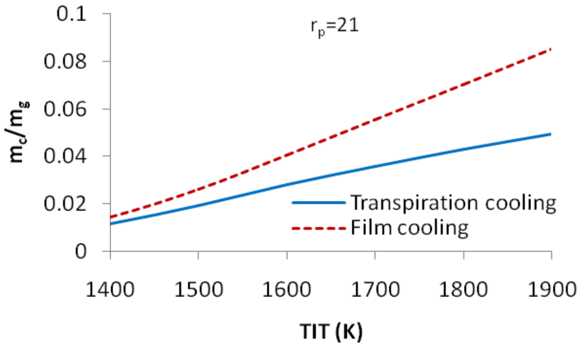

Fig.1. Variation of coolant to gas mass flow ratio with TIT for different cooling techniques

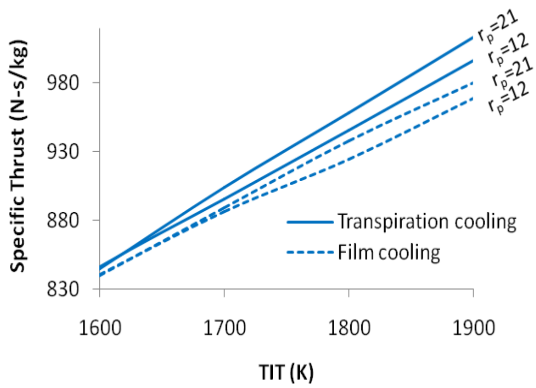

Fig. 2 Variation of Specific thrust with TIT and compressor pressure ratio

Fig. 3 shows the variation of thrust specific fuel consumption for film cooled and transpiration cooled blades with TIT at different compressor pressure ratio. TSFC increases for both the film as well as transpiration cooling with increasing TIT and decreases with increasing compressor pressure ratio. TSFC with transpiration cooling is slightly higher than with film cooling because in transpiration cooling greater amount of compressed air passes through combustor due to reduced coolant requirement, hence extra fuel is burned to achieve the required TIT.

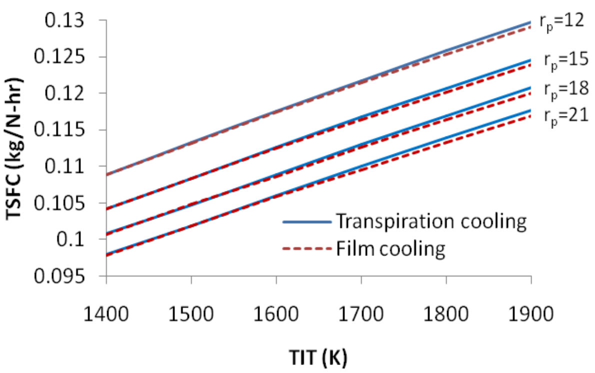

Fig. 3 Variation of thrust specific fuel consumption with TIT and compressor pressure ratio

5. CONCLUSIONS

A general model of simple turbojet gas turbine cycle with film and transpiration cooled turbine blades has been developed and the cycle performance parameters i.e. coolant requirements, specific thrust and thrust specific fuel consumption are compared for varying TIT and compressor pressure ratio. The study shows that the coolant required for transpiration cooling is less than that for film cooling. Comparatively greater specific thrust is obtained with transpiration cooling than with film cooling. TSFC with transpiration cooling is slightly higher than with film cooling in higher TIT range, due to reduced coolant requirement resulting into greater amount of compressed air passing through combustor, hence extra fuel is burned to achieve the required TIT.

REFERENCES

1. Polezhaev, J., 1997, “The transpiration cooling for blades of high temperatures gas turbine”, Pergamon, Energy Convers. Mgmt., Volume 38, pp. 1123-1133.
2. Horlock, J.H., Watson, D.T., Jones T.V., 2001, “Limitations on gas turbine performance imposed by large turbine cooling flows”, ASME Journal of Engineering For Gas Turbines and Power, Volume 123, pp. 487-494.
3. Sanjay, Singh, O., Prasad, B.N., 2008, “Thermodynamic modeling and simulation of advanced combined cycle for performance enhancement”, J. Power and Energy Proc. IMechE, Vol. 222 Part A:, pp. 541-555.
4. Cohen, H., Rogers, G.F.C. and Saranamuttoo, H.I.H. (1996) Gas Turbine Theory. Longman, Harlow, UK.
5. Kumar, S. and Singh, O., 2010, “Thermodynamic performance evaluation of gas turbine cycle with transpiration cooling of blades using air vis-à-vis steam”, Proc. IMechE, Part A: J. Power and Energy, 224 (A8), 1039-1047. DOI 10.1243/09576509JPE964.
6. Kumar, S. and Singh, O., 2011, “Performance evaluation of transpiration-cooled gas turbine for different coolants and permissible blade temperatures considering the effect of radiation”, Proc. IMechE, Part A: J. Power and Energy, 225, 1156-1165. DOI 10.1177/0957650911404305.
7. Sanjay Kumar and Singh, O., 2008, “Thermodynamic evaluation of different gas turbine blade cooling techniques”, IEEE Xplore Conference Proceedings-Second International Conference on Thermal Issues in Emerging Technologies, ThETA ‘08’; 237-244. DOI 10.1109/THETA.2008.5167172.

NOMENCLATURE

Symbol

a	Air	
af	Air-duel ratio	
c_p	Specific heat at constant pressure	(kJ/kg-K)
h	Specific enthalpy	(kJ/kg)
LHV	Lower heating value of fuel	(kJ/kg)
$\dot{m}$	Mass flow rate	(kg/s)
M	Mach number	
p	Pressure	(Pa)
r_p	Compressor pressure ratio	
T	Temperature	(K)
TIT	Turbine inlet temperature	(K)
TSFC	Thrust specific fuel consumption	(kg/N-hr)
W	Work	(kW)

Greek Symbols

α	coolant injection angle
ε	effectiveness
γ	adiabatic index
η	efficiency (%)

Subscripts

a	air
b	blade
ci	coolant in
cc	combustion chamber
cmp	compressor
dif	diffuser
ex	exit
f	fuel
g	gas
in	inlet
mech	mechanical
noz	Nozzle
p	polytropic
trb	turbine
wa	working air

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