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THE EFFECT OF TBC UTILIZATION IN THE DESIGN OF ROBUST AIRCRAFT COMBUSTORS

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ABSTRACT

As performance objectives of new and derivative military engines require combustion systems to operate at higher pressures and temperatures, balancing conflicting demands of improved durability, stability, and operability becomes more difficult without technology improvements in combustor liner designs. Since combustor thermo-mechanical fatigue is currently a significant contributor to engine life cycle costs, improved structural durability must be achieved, without compromise of other combustor requirements. This paper addresses the application and verification of liner cooling schemes and their interaction with thermal barrier coatings (TBCs) for the design of robust aircraft turbine engine combustor liners to meet the above mentioned demands. An analytical investigation was conducted to determine the effect of TBCs on the average metal temperature for a full annular, semi-transpiration cooled combustor liner. The perspective is from a customer's viewpoint, a combustor liner designer who is continuously challenged to increase combustor temperature rise capability and operability for new products while maintaining cooling flow levels.

LIST OF SYMBOLS

A	area
σ	Stefan-Boltzmann constant
α	absorptivity
ρ	density
δ	thickness
μ	viscosity
θ	cooling effectiveness
Cp	specific heat
D	combustor diameter
eff	effective
ε	emissivity
fc	flame chamber
f/a	fuel air ratio
h	heat transfer coefficient
k	thermal conductivity
L	luminosity factor
l_b	mean beam length of radiation path
$\dot{m}$	mass flow rate
η	thermal effectiveness
P	pressure
q''	heat flux
T	temperature

Subscripts

a	air
AN	annulus
c	casing
g	gas
w	wall
WH	flame side
WC	cold side
3	combustor inlet
4	combustor exit

1.0 INTRODUCTION

Air dominance is maintained by fielding affordable and durable high performance air platforms capable of delivering payload when and where needed in the field command. Key to successful air platforms is the propulsion system. The major key to higher specific thrust is higher engine operating temperature. Current combustion systems still operate well below the stoichiometric chemical limits of kerosene based fuels. In the future, major increases to these values will be required (Hill, 1997) [1]. It is estimated that overall pressure ratios are expected to grow from a present 20 to over 40, and the turbine entry temperatures from 1800 to 2400°K. As a result of these higher combustion pressure and gas temperatures, the heat flux levels in these engines will also increase making the task of combustor liner durability design extremely challenging. Therefore, a critical technology need for the development of advanced gas turbine engines will be the control of heat transfer. Since thermal barrier coatings are an effective approach to heat transfer control, advanced materials will need to be integrated with robust thermal barrier coatings and innovative cooling techniques to minimize increases in cooling flow requirements. For a combustor

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liner system, TBCs act as thermal insulators reducing the amount of heat transmitted to the combustor wall from the products of combustion. The heat flux gets dissipated through radiation back into the hot gas path, conduction across the coating and substrate, and convection through back side cooling air.

2.0 THE PROBLEM

Hot section components are significant maintenance items in today's engines. Almost 7 percent of unscheduled removals of F100-200 engines in the F-16 due to core components during 1991 were caused by combustor or turbine problems. In fact, during this period of time, the mean time between failures of the combustor in the -200/F16 system did not meet the Engine Structural Integrity Program requirements (Turner, 1996) [2]. TBCs can alleviate this problem, however, at a cost. Currently the Air Force spends on average $143.3 per kilogram ($65/lb) for TBC bond materials. Each combustor consumes around 0.9 - 1.7 kilograms (2-3 lbs) of bond coat. For a typical top coat the cost is about $28.7 per kilogram ($13/lb) and each combustor consumes around 1.7 kilograms (3 lbs). Labor time for the application of these TBC systems is approximately 1.84 hours per combustor.

3.0 CHALLENGES

In the future, the entire combustor section of a high performance aircraft engine, as shown in figure 1, will be subjected to an increasingly severe thermal environment which will eventually approach stoichiometric conditions.

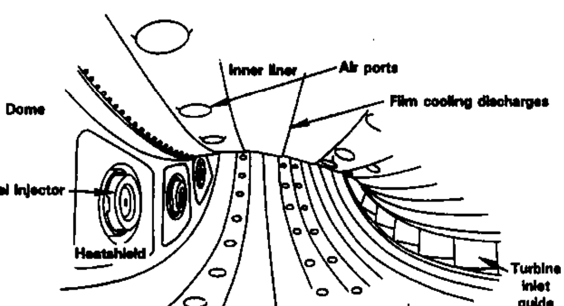

Figure 1. Internal view of a typical annular combustor system

Two major factors will significantly impact the ability to cool the combustor liner:

- *Combustor Inlet Temperature*: As the inlet temperature to the combustor increases, gas temperature also increases and, therefore, the liner temperature increases proportionally requiring more air to maintain a given liner temperature. TBCs can ameliorate this issue by enabling a reduction in the amount of radiant heat flux from the flame to the liner (figure 2).

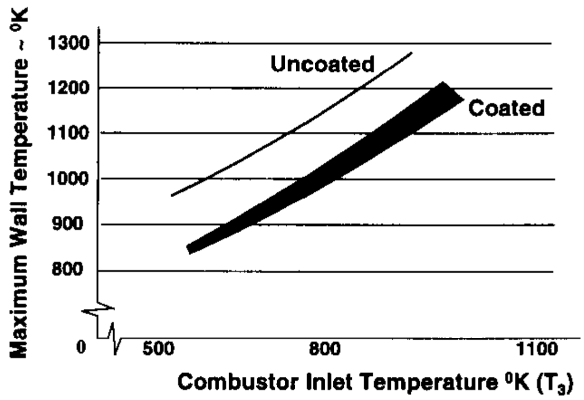

Figure 2. Typical TBC payoff for combustor liners

- *Combustor Temperature Rise*: It is estimated that for the high temperature rise combustors which will be required to operate at an equivalence ratio of 80 percent, over 85% of the entire combustor air flow will be required to be introduced in the primary combustion zone to completely react all of the fuel. As a result, the amount of air remaining for liner cooling and combustor exit temperature control becomes scarce (figure 3). If the cooling air is not reduced, there would be no dilution air available to reduce hot streaks. This further complicates the task of maintaining the liner temperature at that allowed for conventional metal materials.

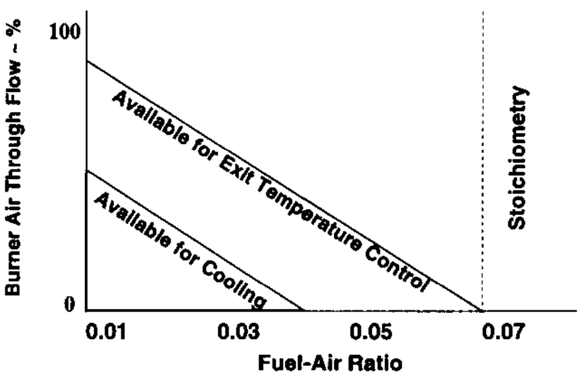

Figure 3. Increased combustor exit temperatures reduce available dilution and cooling air

From the practical combustor design standpoint, the objective is not to use improved cooling methods to lower wall temperatures below the levels achieved with present methods, but rather to achieve the same wall temperatures despite reductions in the amount of coolant flow as shown in figure 4.

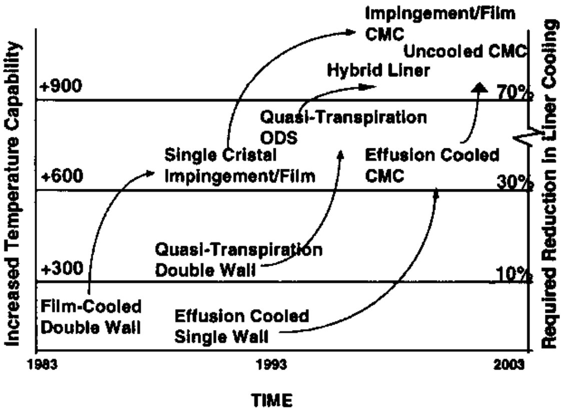

Figure 4. Combustor liner material/cooling progression for the past 15 years.

4.0 GENERAL REQUIREMENTS AND DESIGN CONSIDERATIONS

The primary technical objective of combustor research and development is to provide the technology that will enable the designers attain combustor performance, operability, weight and cost objectives for a particular mission and cycle definition.

As systems become more efficient and probably more complex, the next generation of combustor designs will need to balance conflicting demands such as increased temperature while improving durability, maintainability, and affordability. In order to achieve these demands an integrated product/process must be implemented. This process should enable the production of a robust design capable of operating efficiently in a wide variety of environments dictated by the intended air vehicle. To satisfy the requirements of future cycle parameters, technology improvements are required. Advanced planning allow us to establish combustor performance objectives that will support overall engine system improvements. Some combustor specific performance objectives are:

- 50% weight reduction
- 100% increase in volumetric heat release rate
- $+900^0$ F maximum combustor exit temperature
- $+400^0$ F combustor inlet temperature
- 300% increase in fuel/air turndown ratio
- 99% efficiency at all operating conditions

Other combustor system design considerations:

- 2000 engine flight hours and 4000 Low Cycle Fatigue TACS - hot parts
- 4000 engine flight hours and 8000 Low Cycle Fatigue TACS - cold parts
- 10 hour of operation at any steady-state flight condition
- 1 hour of operation at simultaneous T_3/T_4 condition
- 350 hour of test cell operation

Other requirements:

- Buckling margin of safety as dictated by combustor geometry (shape and wall thickness) shall exceed 50 percent under maximum pressure and temperature

- 5% creep limit without loss of mechanical alignment, as dictated by structural geometry to mitigate the interaction of thermal and mechanical stresses

- Component high cycle fatigue life greater than 10^7 cycles, as determined by both material and structure

- -3σ material properties (average for crack propagation)

- 500 0-maximum-0 cycles at maximum power

5.0 COMBUSTOR TBC EVOLUTION

Since the birth of thermal barrier coatings in the early 1960's, combustors and augmentors have benefited from a series of incremental growths in this technology. This has resulted in an increased ability to withstand higher temperatures with a decrease in the severity of oxidation damage, and an improved cracking resistance to severe thermal cycling for longer periods of time (Harris, 1991)[3]. Early combustor coatings consisted of an outer ceramic insulative layer of air plasma deposited, 22 weight percent MgO stabilized ZrO_2, and an inner metallic bond coat of air plasma deposited Ni-Cr or Ni-Al (Gupta, et al.,1994)[4].

As these temperatures in the combustor have been increased, the durability of this early combustor coating has become inadequate. The maximum use temperature of the plasma sprayed magnesia-stabilized zirconia coating is on the order of 1255^{0}K since magnesia-stabilized zirconia crystallographically destabilizes above 1227^{0}K.

By replacing the 22 weight percent magnesia with 7 weight percent yttria composition, the spallation life at temperature above 1255^{0}K has been improved substantially (Stecura, 1979)[5]. This improvement is approximately 4 times that of 22 percent magnesia fully stabilized zirconia coating at 1367^{0}K (Gupta et al, 1991)[6].

Since the operating temperature of the combustor was increased to take advantage of the increased temperature capability of the yttria stabilized zirconia, the underlying bond coat oxidation became a problem. By adopting a more oxidation-resistance NiCoCr AlY bond coat composition from the turbine section of the engine, the performance of the combustor has been improved (Gupta et al., 1994)[4]. Figure 5 illustrates the progression of TBC's for hot section applications.

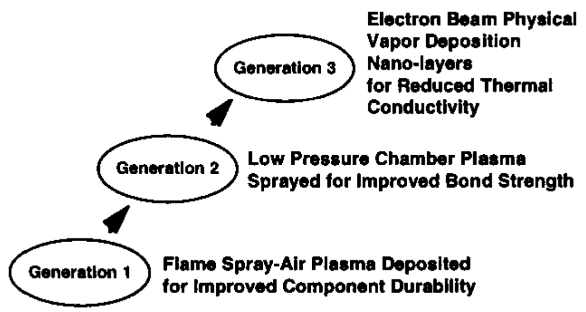

Figure 5. TBC Progression for Hot Section Applications

6.0 TBC APPLICATION ADVANTAGES

Let us look at other potential benefits of applying TBC's in the combustor section:

6.1 Efficiency and Idle Emissions

As summarized by the results of Mularz et al., 1978 [7], from which figure 6 is reproduced, the application of thermal barrier coating along the inside of the combustor liner allows combustion gases near the wall to be at higher temperatures. This minimizes wall quenching effects of the combustion chemical kinetics and therefore, reducing unwanted exhaust emissions such as carbon monoxide and unburned hydrocarbons. Since the design intent of cooling air is to protect the liner surface, air introduced through the liner walls downstream of the dilution jets may not be effective in the combustion process, therefore, contributing to potential combustor inefficiency.

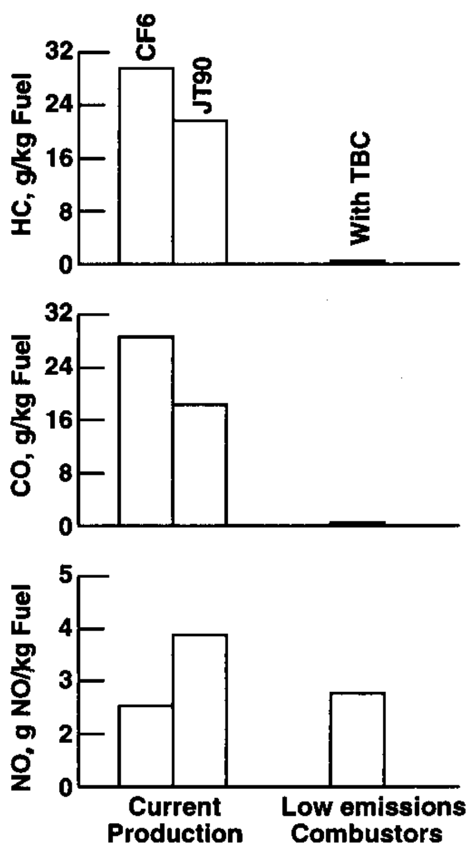

Figure 6. Efficiency Benefits for Idle Engine Operating Condition

6.2 Exhaust Smoke Emissions

Exhaust smoke numbers decrease slightly with thermal barrier coatings as reported by Butze, et al., 1976 [8] from which figure 7 is reproduced, for a cruise condition. Soot is the primary source of flame emissivity at high pressures. The intense radiation from the ceramic coating back to the flame results in a decrease in soot concentrations due to reduction of the amount of soot formed initially and through burnup of the soot formed.

6.3 Combustor Liner Life

The average cooling temperature that is required in a typical production combustor today is graphically illustrated in figure 8. The gas temperature referred to in this figure represents the maximum combustor exit temperature. The allowable operating temperature for the metal liner is about half way between the maximum gas temperature and the temperature of the cooling flow (T3). Also shown in this figure is the relationship of the allowable design metal temperature levels to the incipient metal temperatures. As shown, the allowable metal temperature levels are near the incipient melt point of nickel based superalloys. Therefore, any breakdown in the cooling layer would be detrimental.

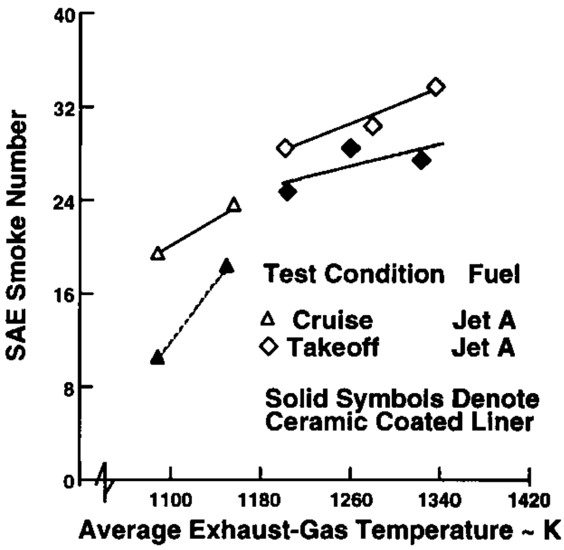

Figure 7. Effect of ceramic coating on SAE smoke number

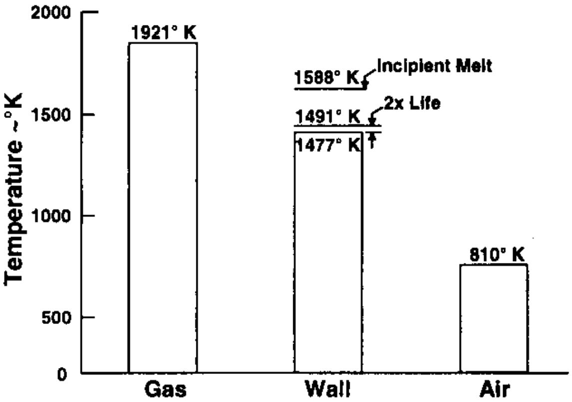

Figure 8. Exhaust, Wall and Air Temperature Comparison

7.0 HEAT MODEL

According to experimental results from Butze et al., 1976 [9], as shown in figure 9, the liner metal temperature reductions available through the use of ceramic coatings can be accurately predicted using a one-dimensional heat transfer model. Having that in mind, a one-dimensional heat transfer analysis was performed in order to estimate the thermal performance of TBCs and compare the capability of TBC improvements against cooling effectiveness improvements and T3 reductions. The heat transfer model outlined in this section closely follows the development approach as detailed by Lefevbre, et al., 1960 [11].

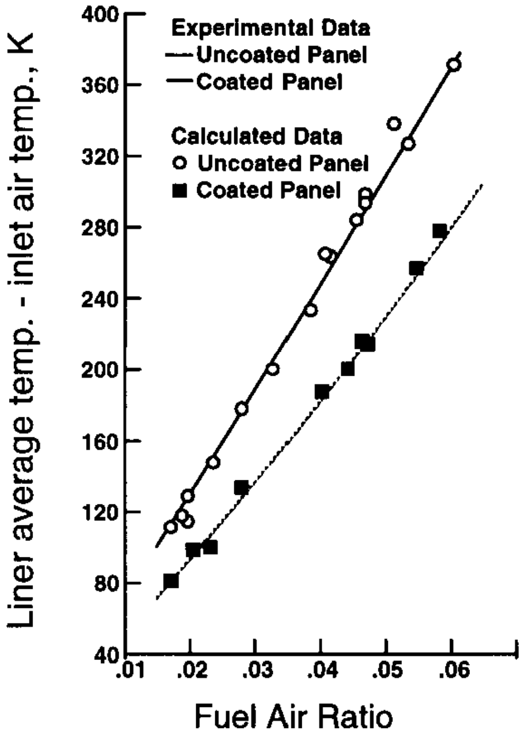

Figure 9. Liner Temperature Prediction vs. Experimental Results

Figure 10 illustrates the schematic of radiant heat flow through a TBC at isothermal conditions. For this analysis the temperature drop through the ceramic was assumed to be linear since yttria-stabilized zirconia thermal conductivity is nearly constant over the temperature range of interest (1.5±0.1 W/m•K over 1200 to 2260^{0}K). Emittance and transmittance were assumed invariant with temperature. The cycle conditions chosen for the analysis are as follow:

P_3 = 3040KPa, T_3 = 1144^{0}K, coolant mass flux = 3.416 Kg/s-m^2

It was assumed that the gas temperature next to the wall is equal to the average flame temperature, having a value close to stoichiometric. This assumes the possibility of a hot streak occurring next to the wall due to combined effects of a localized low annulus velocity on one side of the wall coinciding with a breakdown in the cooling layer on the other.

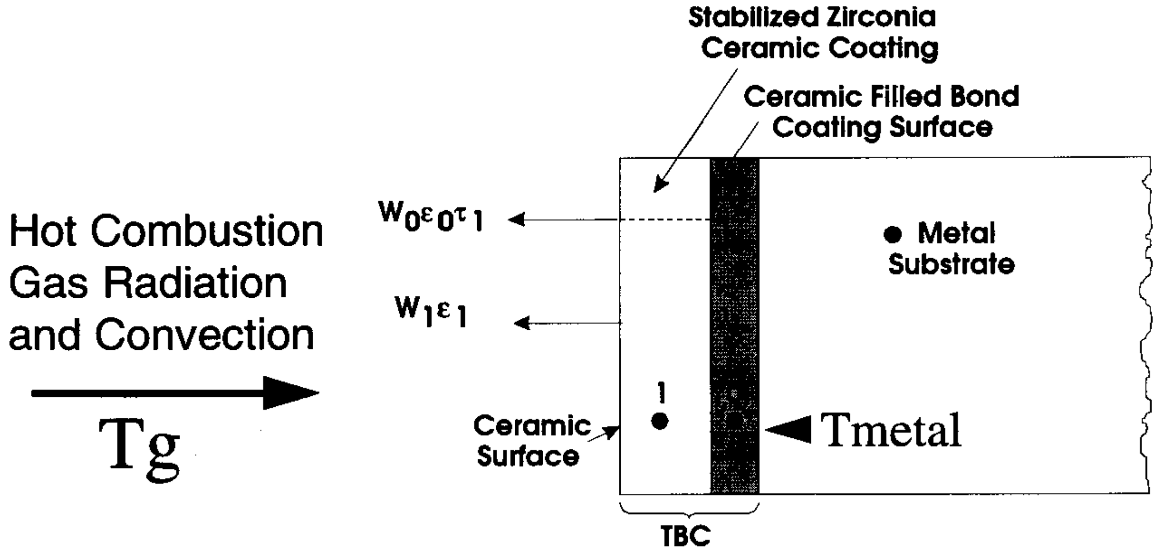

Figure 10. Schematic of radiant heat flow through the TBC at isothermal conditions.

7.1 Internal Radiant Heat Flux

The radiation heat flux was calculated for the case of a TBC-coated liner using the results obtained by Liebert, 1978 [12] where emittance of both the ceramic translucent layer and the material substrate were taken into consideration.

$$q''_{RAD} = \sigma(0.5)\varepsilon_g T_g^{1.5}(T_g^{2.5} - T_{WH}^{2.5}) \quad \{1\}$$

The radiation heat flux was calculated for the case of an uncoated liner using reference [11]:

$$q''_{RAD} = \sigma\frac{(1+\alpha_w)}{2}\varepsilon_g T_g^{1.5}(T_g^{2.5} - T_{WH}^{2.5}) \quad \{2\}$$

The flame emissivity, ε_g, was calculated using reference [11]:

$$\varepsilon_g = 1 - \exp(-290 P_g L\sqrt{(f/a)l_b}\, T_g^{-1.5}) \quad \{3\}$$

where the luminosity factor was calculated using reference [13]:

$$L = 0.0691(C/H - 1.82)^{2.71} \quad \{4\}$$

where C/H is the fuel carbon/hydrogen ratio by weight. For future high pressure ratio engines, the practicality of using this correlation could be somewhat dubious. More experimental data will need to be generated.

7.2 External Radiant Heat Flux

Radiant heat flow from the combustor liner to the casing is given by:

$$q''_{RAD} = \sigma\frac{\varepsilon_W\varepsilon_C\left(T_{WC}^4 - T_3^4\right)}{\varepsilon_C + \varepsilon_W(1-\varepsilon_C)\left(A_W/A_C\right)} \quad \{5\}$$

7.3 Internal Convective Heat Flux

Convective heat flow to the combustor liner from the gases is given by:

$$q''_{CONV} = 0.02\frac{k_g}{D_{fc}^{0.2}}\left(\frac{\dot{m}}{A_w\mu}\right)_g^{0.8}\left(T_g - T_{WH}\right) \quad \{6\}$$

7.4 External Convective Heat Flow

Convective heat flow from the combustor liner to the annulus air is given by:

$$q''_{CONV} = 0.02\frac{k_a}{D_{AN}^{0.2}}\left(\frac{\dot{m}}{A\mu_a}\right)_{AN}^{0.8}(T_{WA} - T_3) \quad \{7\}$$

7.5 Conduction Heat Flux

The radial heat flux conducted through the liner wall is given by:

$$q''_{COND} = \left(\frac{k}{\delta}\right)_{eff} (T_{WH} - T_{WC}) \qquad \{8\}$$

where $$\left(\frac{k}{\delta}\right)_{eff} = \frac{1}{\left(\frac{\delta}{k}\right)_{TBC} + \left(\frac{\delta}{k}\right)_{Metal}} \qquad \{9\}$$

7.6 Convective Heat Flux Through Liner

The heat flux picked up by the coolant as it passes through the semi-transpiration cooled wall is expressed by

$$q''_{air} = \dot{m}_{air} Cp_{air} \eta \cdot (T_{WH} - T_3) \qquad \{10\}$$

where η is the thermal effectiveness, which is a function of the internal heat transfer coefficient within the liner. This heat flux is absorbed by the coolant and therefore reduces the heat load that must be removed by q''_{CONV} and q''_{RAD} on the outside liner.

The following figures summarize the results of the investigation:

- Figure 11 shows the potential reduction in metal wall temperature attainable with a refractory coating for the semi-transpiration cooled liner system. It is obvious that changes in TBC thickness have a pronounced effect on combustor liner temperature as reflected in the figure. It is also evident that with the application of the TBC to the liner, the liner wall metal temperature would be capable of meeting life requirements as mentioned in section 4.

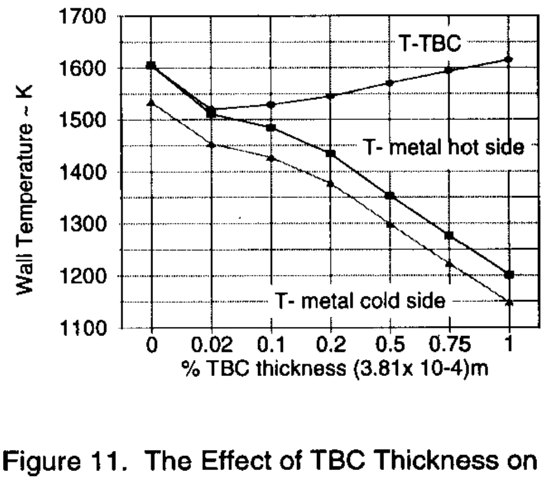

Figure 11. The Effect of TBC Thickness on Liner Wall Temperature

- Figure 12 shows how the heat fluxes vary as the effective conductivity changes. The heat flux due to conduction diminishes as the thickness of the TBC system is increased. For the same case, the outside convective and radiation heat fluxes also decrease as expected. The internal radiation heat flux is maximum for the case of the uncoated liner since a small fraction of the heat gets reflected back to the gases.

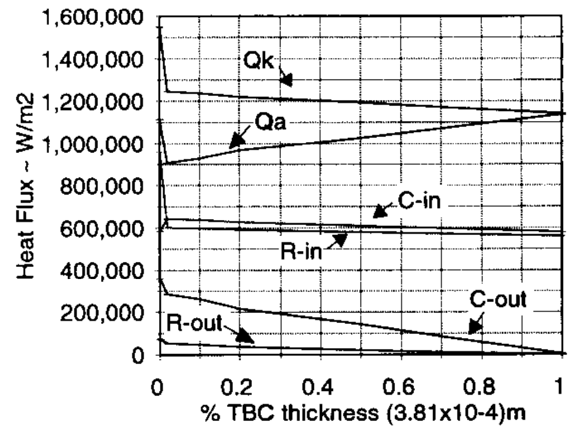

Figure 12. The Effect of TBC Thickness on heat fluxes

- Figure 13 shows the reduction in metal temperatures without TBC as coolant temperature decreases for the same amount of coolant flow rate. Comparing these results with the results from figure 14, it is evident that although the TBC surface temperature is higher, the actual metal temperature (substrate) on the gas side is much lower.

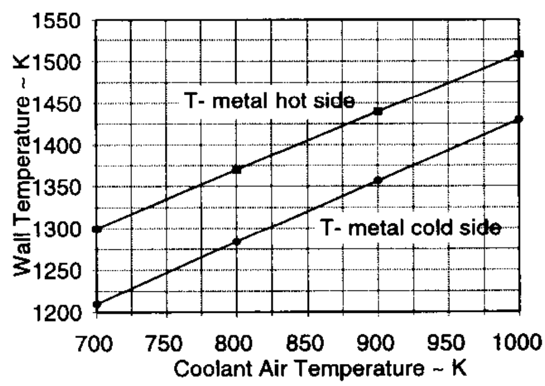

Figure 13. The Effect of Coolant Air Temperature Variation on Liner Wall Temperature (Uncoated)

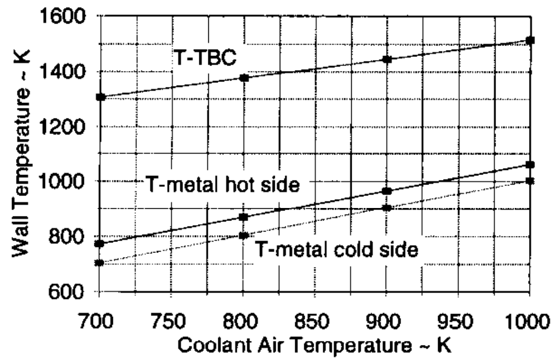

Figure 14. The Effect of Coolant Air Temperature Variation on TBC Liner Wall Temperature

• Figure 15 shows the impact of liner thermal effectiveness variation on wall temperatures. As would be expected, the hot side metal temperature is reduced as the ability of the liner to extract heat becomes greater for the same amount of coolant flow rate

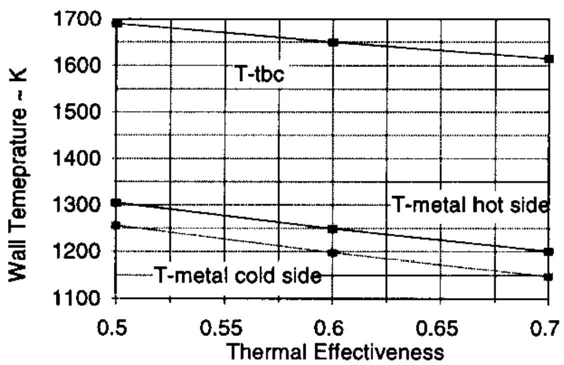

Figure 15. The Effect of Variation of Thermal Effectiveness on TBC Liner Wall Temperature

8.0 SUMMARY

The quest for higher thrust-to-weight ratio will entail increasing combustor inlet and exit temperatures. This will impose very stringent requirements for the development of robust aircraft combustors. A large portion of the heat transferred to the liner wall from the hot combusting gases and particles within the combustor will be by radiation. But significant advances can be obtained by incorporating third generation TBC into the next production of combustor liners. To maximize the potential, there needs to be a focused effort. The thermal effect of the TBC ceramic layer must become an integral element of the combustor liner design system. TBC reliability must be equal or greater than the substrate since the integrity of the coating becomes prime reliable as loss of the coating can result in rapid deterioration of the liner. Also, the ability to apply TBCs to components with complex geometries should be improved. As combustor inlet air temperature approaches the maximum allowable wall temperature, there simply will not be enough cooling potential left to cool the wall adequately. This leaves the possibility of no metal wall structure, no matter how effective, will maintain reasonable metal temperatures at those conditions.

An analytical investigation was conducted to determine, through qualitative trends rather than quantitative values, the effect of a ceramic coating on the average metal temperatures of a full annular, semi-transpiration cooled combustor liner. This investigation was conducted at a pressure of 3040 KPa, inlet air temperatures of 1144^0K and an overall fuel air ratio of 0.67. The insulating effects as well as increased reflectivity of the ceramic coating were responsible for the reduction in heat transfer through the liner walls.

To solve the heat flux equations a non linear algorithm was used. For that purpose a Newton-Raphson non-linear solver was developed to solve the equations in an iterative manner. It should be mentioned that the energy balance equations represent a gross approximation to the heat transfer process taking place between the liner structure and its surroundings. A more rigorous formulation for the heat transfer process may be developed if specific details are required.

It appeared that the reduction in metal temperature at any given coating thickness is considerably greater than any improvements achieved solely by cooling configuration changes.

It was noted that a relationship between liner cooling effectiveness and effective conductivity exists. That relationship was considered in the analysis, however, with some limitations. In other words as cooling effectiveness changes there corresponds a change of effective conductivity.

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