Def Sci J, Vol 38, No. 4, October 1988, pp 363-380

Advanced Combustor Liner Cooling Technology for Gas Turbines

Aspi R. Wadia

General Electric Company, Cincinnati, Ohio–45215

ABSTRACT

This paper briefly reviews some of the work on advanced liner cooling techniques – specifically laminated porous wall cooling, angled-multihole (effusion) cooling and composite metal matrix liner cooling. The concept definition, heat transfer design procedure and design problems including key materials and fabrication considerations associated with each basic concept will be reviewed. Published rig and engine experience of aircraft engine manufacturers and research organizations will be cited. Some logical extensions of the current liner cooling schemes are suggested for future applications.

. INTRODUCTION

The projected designs for future aircraft engines are very aggressive in terms of thrust-to-weight, rotor speed and temperature goals[1]. Conventional wall cooling methods (e.g. film convection cooling) are incapable of providing satisfactory durability without using excessive amounts of cooling air, which in turn severely restricts air available for temperature pattern control. Engine envelope demands, especially for small aircraft engines, further burden the designer by mandating foldback (reverse flow) combustor designs, which reduce engine length and weight but substantially increase the combustor surface area to be cooled. Although attention to cooling problems has traditionally been focused on the HP turbine for obvious reasons, there is a need to view combustor liner cooling as a technical problem of equal importance if projected performance goals and durability improvements are to be achieved.

Received 19 April 1988

2. LINER COOLING SCHEMES

The principal schemes representing both the current and future candidate approaches to combustor liner wall cooling are shown conceptually in Fig. 1.

Simple slot film cooling has received the most attention in terms of application and fundamental studies. While the significant body of research work has greatly enhanced the understanding of film flow behaviour, significant reduction of cooling flows in pure slot film cooled combustors has not been forthcoming. This is attributed in part to the difficulty in maintaining film integrity in a turbulent environment characterized by recirculating flows. Also, the insulating film primarily affects the convective component of heat transfer and has little effect on the radiative heat transfer from the luminous gas.

Colladay[2] was the first to recognize that significant reductions in coolant flow could be achieved if the heat sink capability inherent to the coolant is more fully utilized in the active mode prior to injection. This led to the development of the film-convection cooling system.

Further improvements in cooling air usage requires a departure to more effective mass transfer cooling schemes, such as transpiration or effusion cooling. The heat transfer development work leading to incorporation of such systems into combustors is reviewed in the following sections.

3. LAMINATED POROUS WALL COOLING

3.1 General Description

Improvements in metal joining techniques has led to the development of fabricated multiple-laminate porous structures. One such structure, Lamilloy[3]*, consists of several diffusion-bonded, photoetched metal sheets shown in Fig. 2. This cooling scheme has considerable design flexibility relative to flow resistance control and optimization of heat transfer performance. Design variables include hole size and spacing, laminate thickness, number of laminates, pedestal height, pedestal diameter and spacing. Large internal heat transfer surface area to volume ratio and hence improved thermal effectiveness levels (defined as actual coolant temperature rise divided by ideal rise) can be achieved by reducing all the dimensions so that the internal structure approaches that of a fine capillary mesh. The reduction in the dimensions is however limited by passage clogging considerations and by fabrication limitations.

3.2 Heat Transfer Analysis

One of the key inputs in the heat transfer analysis of laminated porous wall structures is the knowledge of the internal (matrix) heat transfer coefficient. The heat transfer data is generally obtained by testing simple disk specimens, mounted in a fixture and heated by a high intensity radiant energy source[3]. Some typical heat transfer trends for selected Lamilloy configurations relevant to combustor liner

*Lamilloy is a registered trademark of General Motors Corporation.

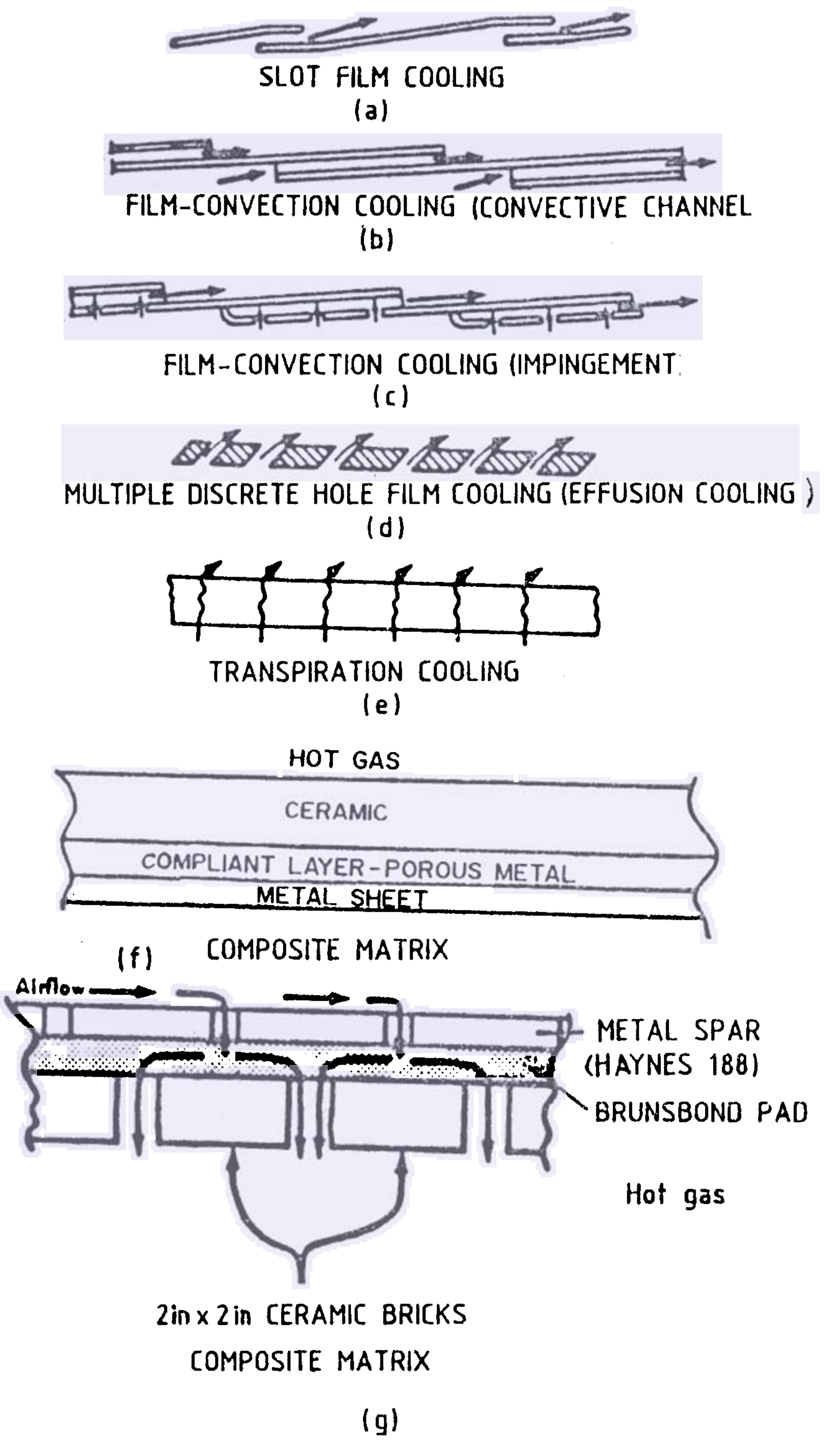


Figure 1 Combustor liner cooling schemes.

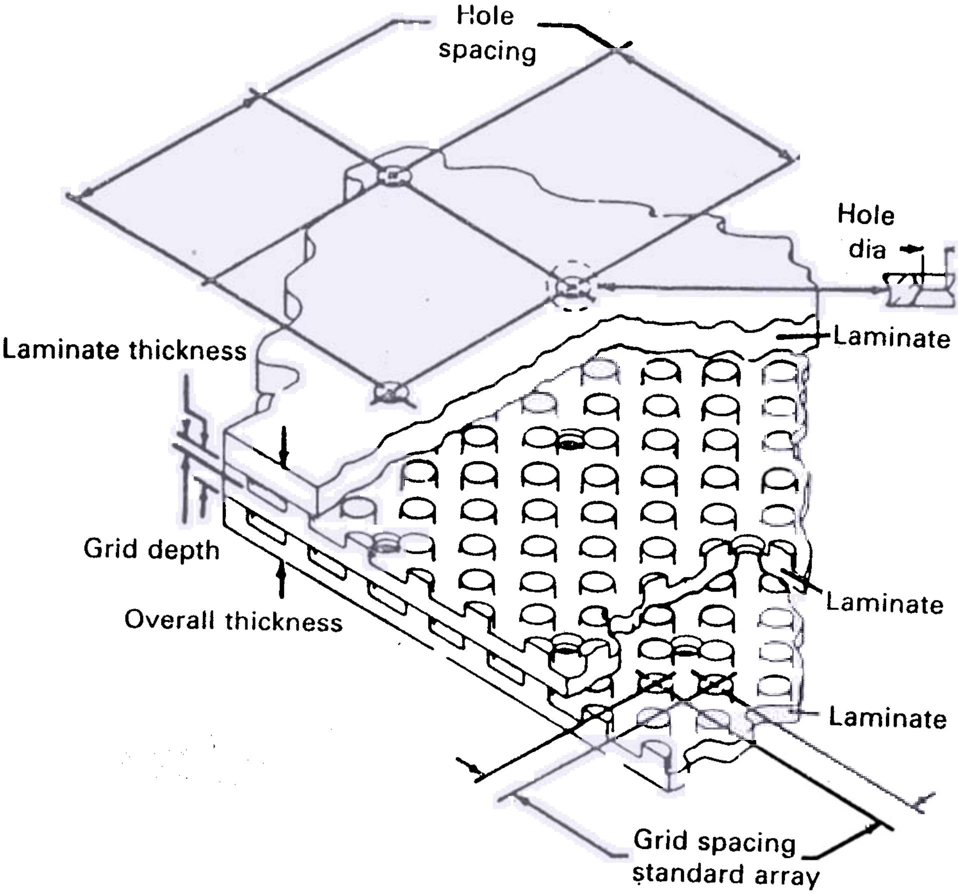


Figure 2. Typical geometric arrangement of laminated porous wall structure

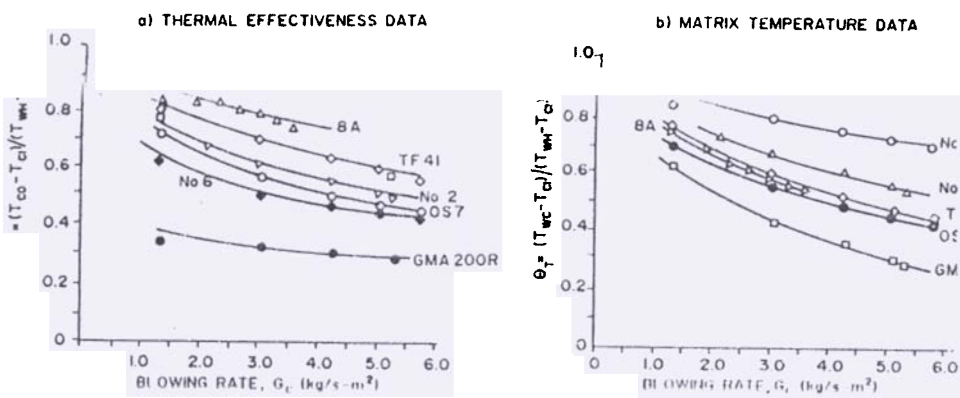


Figure 3. ... us wall

application[3] are shown in Fig. 3. The internal 'heat exchanger' effectiveness parameter (η_T) is especially important since liner cooling requirements will be inversely proportional to it. Fig. 3 indicates the possibility of developing a unique geometric configuration that approaches the ideal transpiration cooling effectiveness of unity. The difference in the performance trends of the configurations is due to differences in the hole/pedestal geometry of the specimen.

3.3 Mechanical Design Considerations

While transpiration cooling potentially represents the most thermodynamically efficient approach to combustor cooling, practical implementation of the method has been hampered by limitations of porous materials. The most glaring drawback of this cooling technique has been its lack of structural strength and resistance to oxidation. Considerable difficulty in predicting or controlling local permeability has also been encountered, and susceptibility to foreign particle clogging continues to be a problem.

Laminated porous wall structures[3] have been fabricated from nickel-and-cobalt-based superalloys in addition to a full spectrum of other materials, including Haselloy X and Haynes 188. The structure requires very thin sheet stock ranging from 0.01 to 0.025 in.

A typical laminated porous wall structure (Lamilloy) manufacturing sequence[3] is illustrated in Fig. 4. The process is complex and expensive.

3.4 Engine Experience

From the data published in the open literature, two aircraft engine companies appear to be the leaders in the application of porous wall structures.

Allison Gas Turbines[3,4] has used Lamilloy combustors for several engine models; the most recent application being for the T800/ATE109 (LHX) 1200 hp turboprop engine.

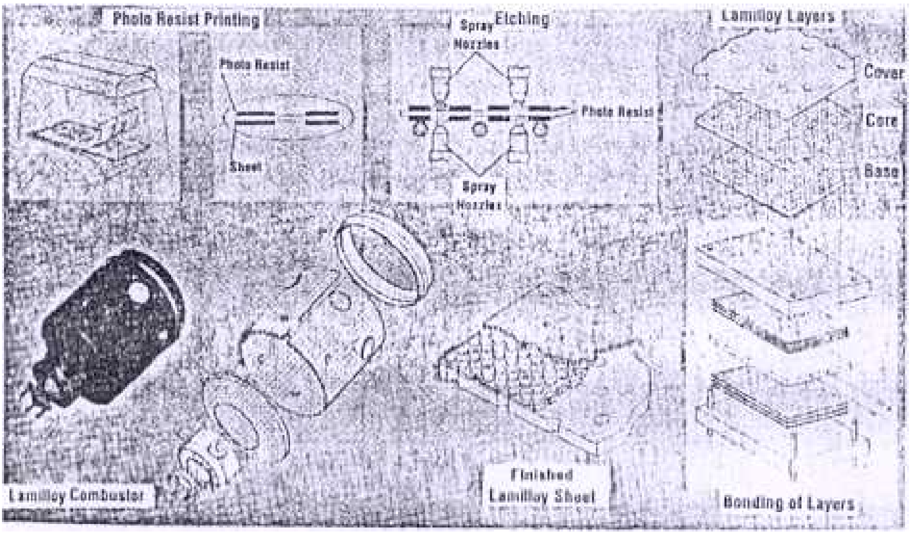


Figure 4. Laminated porous wall manufacturing sequence

Rolls Royce has developed their own technology[5], using 'Transply' and rig and engine tests have dispelled doubts concerning the blockage of transpiration cooled systems by airborne particles, and Spey combustors incorporating sections of Transply have achieved flight certification for commercial airline operations.

4. ANGLED-MULTIHOLE EFFUSION COOLING

4.1 General Description

A more practical arrangement, but one that still approximates a porous wall, is a wall perforated by a large number of small holes as shown in Fig. 1(d). Ideally, the individual holes should be large enough to remain free of blockage by dirt, but small enough to prevent excessive penetration of the airjets.

The cooling performance is controlled by a relatively few geometric parameters, namely : sheet thickness, hole size, spacing and plunge angle. In theory, the effusion cooling concept can approach transpiration cooling performance in the limit as the hole diameters and spacing-to-diameter ratios are reduced to zero. Parametric studies[3] have shown that when 'realistic minimum' hole sizes, spacings and plunge angles are considered effusion cooling performance lies between film-convection cooling and laminated porous wall cooling. The principal advantage of this approach lies in its inherent simplicity, ease of manufacturing and cost-effectiveness, a major factor under present economic conditions.

4.2 Heat Transfer Analysis

The modeling of heat transfer in effusion cooling systems is largely empirical. The basic working relationships between cooling flow rate, hole geometry and cooling performance can be established from the work of Wadia and Nearly[6]. Although the work described in ref. 6 is focused on the airfoil leading edge problem, the modeling of internal wall heat transfer processes is equally applicable to the combustor liner problem. Further experiments with respect to the internal wall heat transfer characteristics are underway at the University of Leeds under the direction of Andrews[7] and their results confirm the importance of the hole approach heat transfer.

The cooling performance trends for an angled, multihole wall can best be illustrated by a parametric study presented in Table 1 using a model very similar to that described in ref. 8 with the appropriate boundary conditions. Table 1 shows the cooling flow requirements for several alternate effusion cooling configurations compared with that for a laminated porous wall design. While this comparative study was made[3] for the TF41 combustor liner, the results can be considered as at least representative and can be used to illustrate the influence of the principal geometric variables on effusion cooling performance. The following key observations are summarized below.

The best practical effusion geometry (a = 20°, d = 0.015 in.) requires about 60 per cent more cooling flow than does the laminated porous wall baseline configuration. Halving the wall thickness increases the cooling flow requirements by about 30 per cent (configurations A versus E and B versus F, etc.). The effect of hole angle is

Table 1. Effusion cooling – performance trends (refs. 3 and 7)

Configuration	Description	Cooling flux requirement lb/s-in^2	Hole spacing to diameter ratio	Number to holes (per liner)	Wall temperature gradient – °F
L	Laminated porous wall (baseline)	0.010	4.0	6,812	302
A	$a = 20$ deg $d = 0.015$ $\tau = 0.060$	0.0160	7.12	11,052	145
B	$a = 45$ deg $d = 0.015$ $\tau = 0.060$	0.0247	5.90	16,117	146
C	$a = 20$ deg $d = 0.030$ $\tau = 0.060$	0.0250	5.78	4,199	150
D	$a = 45$ deg $d = 0.030$ $\tau = 0.060$	0.0389	4.73	6,255	153
E	$a = 20$ deg $d = 0.015$ $\tau = 0.030$	0.0214	6.24	14,390	75
F	$a = 45$ deg $d = 0.015$ $\tau = 0.030$	0.0332	5.13	21.306	75
G	$a = 20$ deg $d = 0.030$ $\tau = 0.030$	0.0332	5.05	5,488	77
H	$a = 45$ deg $d = 0.030$ $\tau = 0.030$	0.0513	4.14	8,172	78

a – effusion hole plunge angle (measured from horizontal)

d – effusion hole diameter (in.)

τ – liner wall thickness (in.)

significant indicating the importance of active cooling in the effusion cooling system. Decreasing the hole angle from 45° to 20° (hole length-to-diameter ratio increases from 5.7 to 11.7) reduces the cooling flow requirements by about 35 per cent (configuration A versus B, C versus D, etc.). The significant cooling performance advantage of small holes is partially offset by the requirement to drill significantly more holes (about 2-1/2 times more holes for the case cited). Wall normal temperature gradients in effusion-cooled liners are considerably lower than in laminated porous walls with the same overall thickness (configurations A, B, C, D versus L).

4.3 Shaped Holes

Shaped holes have been effectively used in turbine airfoil cooling applications to eliminate the problem of low film cooling effectiveness associated with lift-off. Two possible means of preventing[9,10] film 'lift-off' are illustrated in Fig. 5.

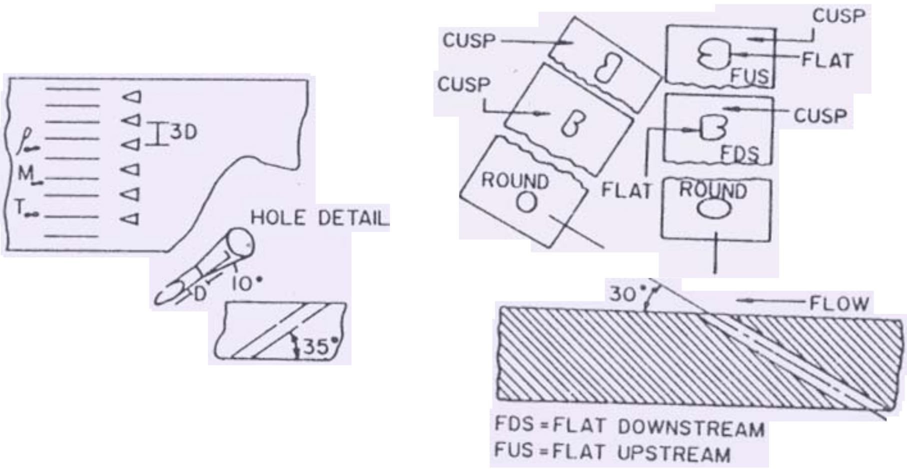


Figure 5. Shaped hole design configurations (a) Goldstein (Ref. 9), (b) Pappel (Ref. 10).

The improved performance of shaped holes relative to cylindrical holes is illustrated in Fig. 6. The shaped hole contributes to improved lateral spreading of the coolant flow so that the film cooling effectiveness between holes is significantly greater. The larger discharge area of shaped holes reduces the injection velocity of the coolant flow resulting in improved effectiveness. The lower effective blowing ratio causes the flow to stay nearer to the wall with less penetration into the mainstream.

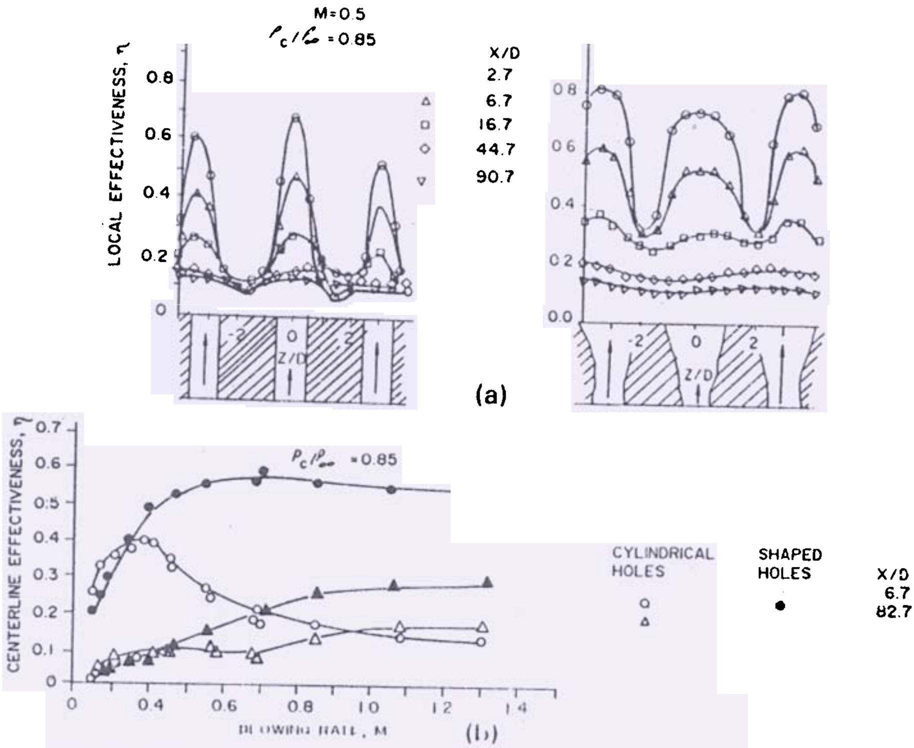


Figure 6. Comparison of cylindrical and shaped hole cooling effectivenc ... (a) loc ... effectiveness, (b) centre line film cooling effectiveness.

While most published data on effusion liners has been for straight cylindrical holes, it may be possible to improve the thermal performance by using shaped holes. Trade-off studies between the improvement in performance and manufacturing cost are needed to fully evaluate the shaped hole concept for liner cooling.

4.4 Mechanical Design Considerations

Of the three liner cooling concepts described in this paper, effusion cooling is the simplest and most cost-effective to implement using either laser or electron beam technology in conjunction with computer controlled manufacturing[3]. Messer Griesheim's (Germany) Electron Beam Drilling Machine (EBDPULS) is capable of drilling 20 holes (0.02 in. dia, 0.175 in. long) per second through both nickel and cobalt steels at angles down to 20° from the surface. Similar hole drilling speeds and quality are achieved by pulsating low divergence Nd : YAG lasers at very high peak powers with little or no thermal distortion.

The manufacturing process is simple, since only a single thickness material need be rolled or formed to the final combustor shape. All holes, including both cooling and combustion air holes, can then be drilled at the same time.

4.5 Rig/Engine Experience

One of the users of this concept has been Allison Gas Turbines. An excellent back-to-back comparison of the laminated porous wall cooling concept with the effusion cooling concept is presented in the paper by Mongia and Reider[4]. A 135° segment of effusion liner was used for the primary zone wall region of an ATDE/GMA500 reverse-flow combustor manufactured using laminated porous walls. The effusion patch tested had three-hole densities in equal areas of the segment giving cooling air fluxes of 0.0055, 0.007 and 0.01 lb/in^2 sec., respectively, at sea level design point. The baseline laminated porous wall configuration has a coolant flux of 0.0056 lb/in^2–sec. Wall thermocouple readings, for the burner tested to a 2598°F combustor exit temperature are shown in Fig. 7.

This test indicates that for the same coolant flux, the effusion liner runs about 200°F hotter than the laminated porous wall configuration, and it takes approximately 45 per cent more coolant flow for the effusion liner to attain the same temperature as the porous wall liner. However, the estimated through-the-wall temperature gradient and the axial temperature gradient in the effusion cooling system is about 75 and 70 per cent less, respectively, relative to the porous wall concept, indicating significant life improvement benefit of the effusion liner.

5. COMPOSITE MATRIX LINER COOLING

5.1 General Description

Two typical composite matrix cooling schemes[4] are shown in Figs. 1(f) and 1(g). The compliant layer is included in the matrix to act as a strain relieving medium between the ceramic and the solid metal support structure.

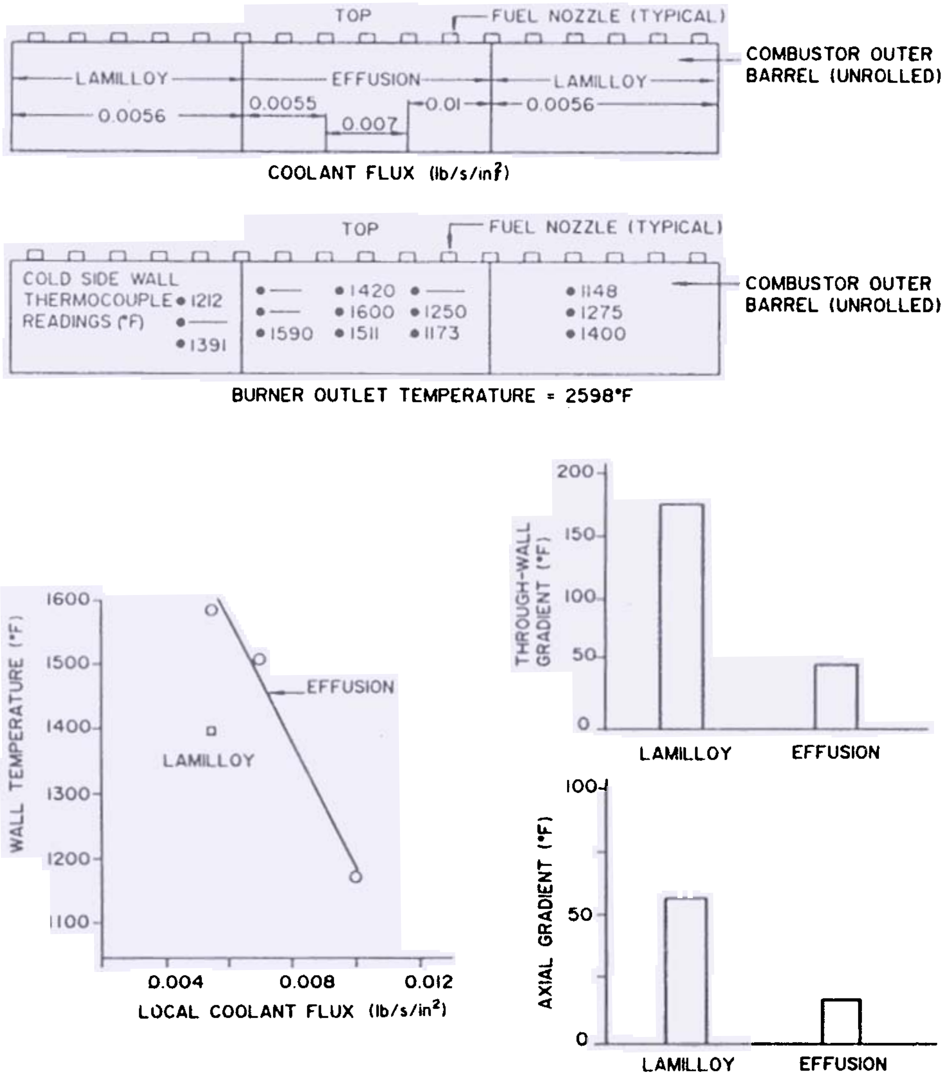


Figure 7. ATDE/GMA 500 reserve flow combustor test results with laminated porous wall and effusion cooling[4].

The solid surface or the conduction and backside convection cooling only design is the simplest configuration from a mechanical design point as illustrated in ref. 11. The entire cooling process is conduction from the hot side wall through the ceramic and the compliant layer to the metal shell support.

The interrupted surface or transpiration cooled concept as illustrated in ref. 4 may be required if the ceramic-compliant layer interface is to be maintained at or below 1750°F, above which, oxidation of the compliant layer, especially the portion adjacent to the ceramic, will become a significant problem. With a proper design and a very small amount of through flow air, the ceramic-compliant layer interface temperature can be more easily maintained at the desired level.

5.2 Heat Transfer Analysis

The convection-cooling concept can be optimized using one-dimensional heat balance equations at selected discrete locations (primary zone, dilution zone, etc.) around the combustor liner. The objective here is to design the cooling configuration (e.g., ceramic coating thickness, compliant layer thickness, etc.) that satisfy the thermal durability requirements (e.g. interface temperatures between pad and ceramic coating, thermal wall gradients, etc.) with minimum cooling airflow. The boundary conditions are empirical in nature.

The selection of the ceramic and compliant pad thickness is critical to the success of the convection cooled concept. Using a typical geometry and airflow distribution for a NASA reverse flow combustor (Fig. 3 in ref. 12), the effect of wall thickness on the temperature distribution through the lower walls of the combustor liner for the primary zone was calculated using the method proposed in ref. 13. The limiting hot streak temperature was assumed to be stoichiometric. For constant metal structure and ceramic coating thicknesses, Fig. 8 shows that the ceramic/pad interface temperature increases by about 200°F, when the pad thickness is increased from 0.025 to 0.1 in. However, for constant metal structure and pad thicknesses, Fig. 9 shows that the ceramic/pad interface temperature decreases by about 500°F by increasing the ceramic thickness from 0.025 to 0.1 in. From an idealistic heat transfer viewpoint,

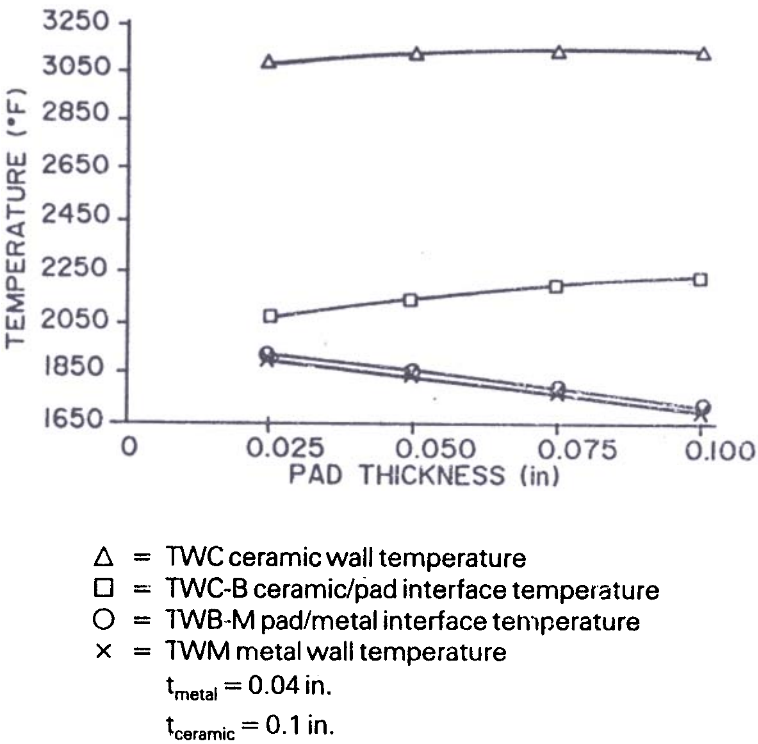
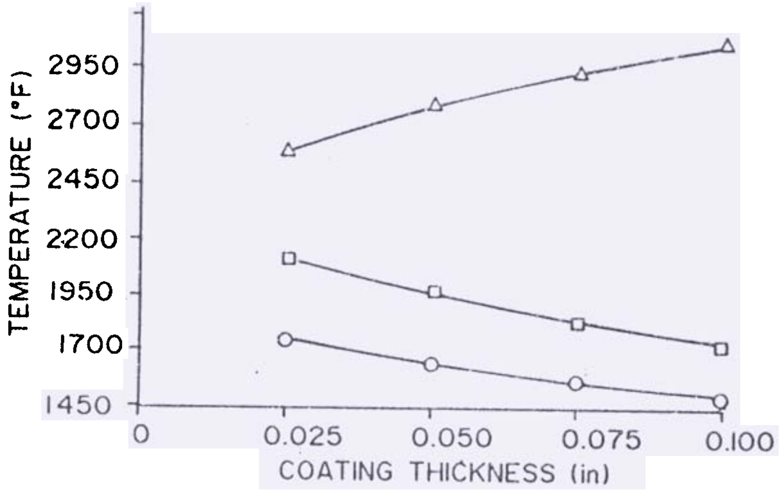
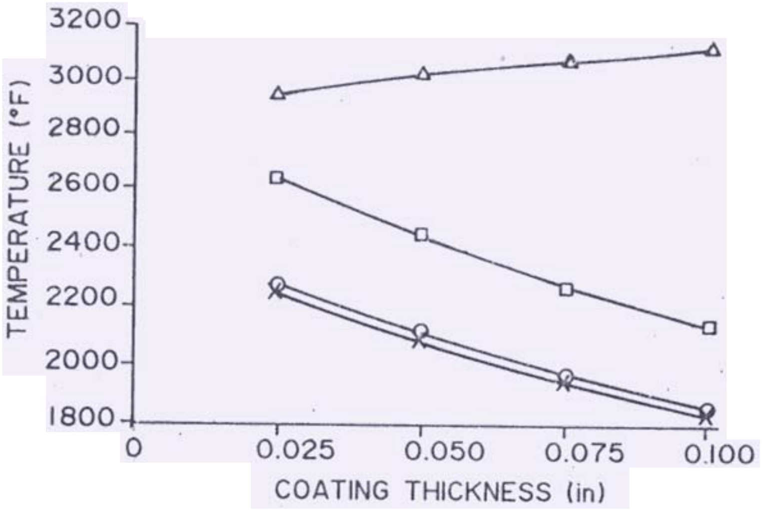


Figure 8. Effect of compliant layer thickness on the wall temperatures fc convection cooled design.



△ = TWC ceramic wall temperature
□ = TWC-B ceramic/pad interface temperature
○ = TWB-M pad/metal interface temperature
× = TWM metal wall temperature
$t_{metal} = 0.04$ in.
$t_{pad} = 0.05$ in.

Figure 9. Effect of ceramic coating thickness on the wall temperature for the convection-cooled design.



Zone 1, lower wall, $W_c = 0.78\%$
△ = TWC coating hot side wall temperature
□ = TWC-B ceramic/pad interface temperature
○ = TWM coolant side wall temperature

Figure 10. Effect of coating thickness on the layer temperatures for composite matri› porous wall cooling design[8].

the convection-cooled concept would require the minimum possible pad thickness and the maximum possible coating thickness. However, thicker coatings will be subjected to gteater wall thermal gradients, including cracks much sooner due to cyclic variations, reducing the overall life of the component.

The composite matrix transpiration-cooled concept can also be evaluated by using the procedure developed in ref. 8 and assuming a thermal effectiveness level of 0.5. For the GMA500 combustor liner, Mongia and Reider[4] using this analytical technique have shown a 38.7 per cent reduction in coolant flow relative to a similar laminated porous wall design.

Figure 10 shows the effect of coating thickness on the pad/ceramic interface temperature in the primary zone for the transpiration-cooled concept. Increasing the coating thickness from 0.025 to 0.1 in. reduces the pad/ceramic interface temperature from 2110 to 1749°F while the gradient across the coating increases from 483 to 1803°F.

Figure 11 illustrates the asymptotic effect of variations in the flow rate on the wall temperatures in the primary zone of the transpiration-cooled concept.

5.3 Mechanical Design Considerations

Mechanical design studies of the composite matrix concept are described elsewhere[14,15]. These studies outline the application of the ceramics to the combustor liners by plasma spraying Yttria Stabilized Zirconia (YSZ) on a compliant nickel alloy substrate. The compliant metal substrate is designed to yield at relatively low levels of stress, thereby absorbing the differential expansion which develops between the metal and the ceramic as the material is heated. The compliant metal substrate is made from randomly oriented fibers which are sintered for strength. A large number of physical and mechanical properties are obtainable by controlling product alloy, fiber diameter, porosity and sintering conditions.

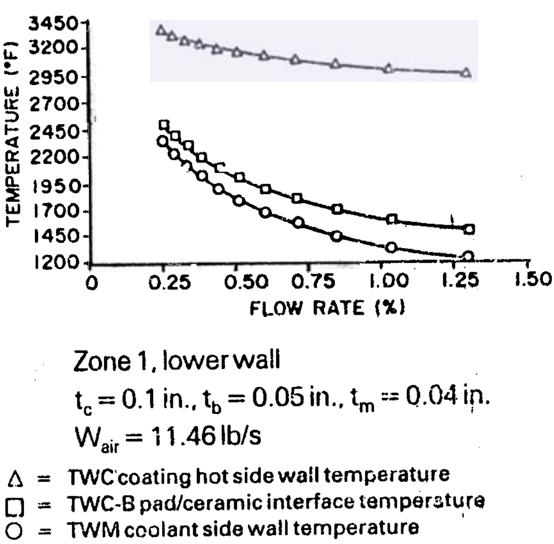


Figure 11. Effect of variation in the flow rate on the layer temperatures of a NASA type reserve flow annular combustor linear wall[8].

5.4 Rig/Engine Experience

NASA/Lewis has provided the leadership for the research[11,12] and devel of the composite matrix liner. Using Brunswich Technetic's (Florida) p BRUNSBOND Hoskins-875 compliant pad, an experimental combustor was fat and tested[11] at NASA. The composite consisted of a 0.025 in. thick Hastelloy substrate, 0.06 in. thick compliant pad and 0.06 in. thick YSZ ceramic coatin convective cooling was used during the test.

The test results show that the composite matrix liner required 35 per ce coolant flow than a similar laminated porous wall liner. The outer liner appe indicate good short term durability with no indication of spalling while the inn did encounter spalling of the ceramic as a result of stresses within the coating

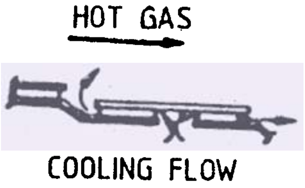
The combustor was capable of providing higher exit temperatures than cu possible with conventional cooling schemes and further increases in temperatu be achieved by incorporating transpiration cooling to the metal ceramic interf

6. OTHER CONCEPTS

Liner cooling concepts other than those described above are briefly cited section.

To meet the challenging durability goals of the NASA/E^3 combustor pro General Electric Company and Pratt and Whitney have devised new configurations which incorporate advanced film plus impingement cooling.

GE's patented design concept[16] utilizes a double-walled 'shingle liner' that co of a load carrying 360° turning which supports individual heat shield or shingles shingles are segmented axially and circumferentially to reduce stress and provide life. The support liner, in addition to supporting shingles, provides impinge cooling to the shingle. Details of the 'support foot' configuration and the method to control leakage between adjacent shingles is illustrated in great detail in ref.



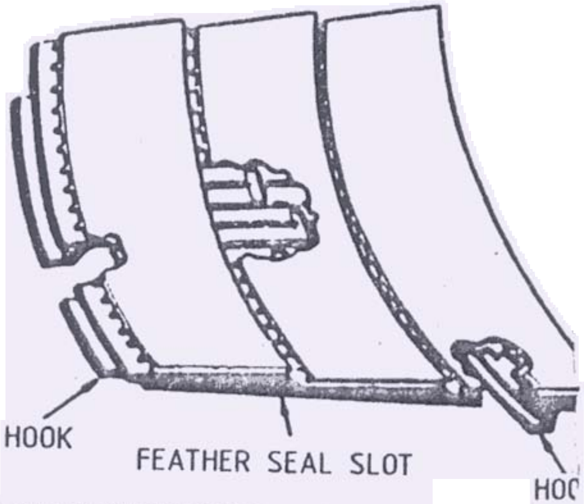


Figure 12. Segmented, counter-parallel FINWALL cooling scheme

Pratt and Whitney's patented design concept[17] called 'FINWALL' also has a segmented construction and counter-parallel wall cooling. Fig. 12 shows the convective/film cooling technique used by the liner. Airflow enters through slots in the cold wall and flows upstream and downstream in discrete cooling passages. The coolant is discharged on the hot side to form an additional protective film. Complete details of the application of this technique including test results are presented in ref. 18.

Garrett's advanced liner cooling concepts[19] are summarized in Fig. 13. Their experiments indicate that cold side convection, augmented by rectangular fins and combined with film cooling (Fig. 13(a)) offers the largest potential reduction in coolant-flux requirements relative to the other configurations in Fig. 13.

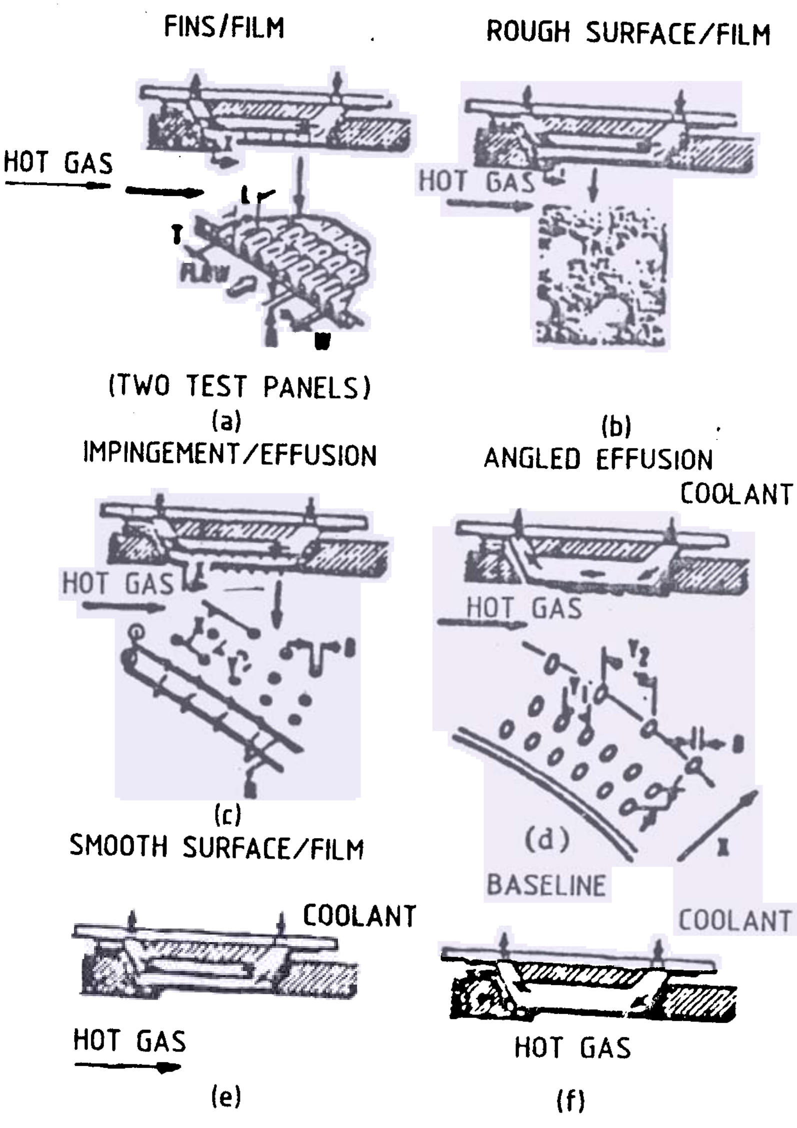


Figure 13. Liner cooling configurations tested at Garrett[19].

7. SUMMARY

Figure 14 summarizes[4] the relative cooling air requirement and some limi: quantity manufacturing 'costs of the combustor liners reviewed in this pap Technically, the composite matrix liner demands the least amount of cooling air a is most suitable for high performance engines where cost may not be an issue. Effusi cooling appears to be the most attractive from a cost-effective viewpoint and application in many of the current production engines suggests its adequacy relat to present design goals.

For the near term growth, laminated porous wall and composite matrix lin will suffice. In the long term, carbon-carbon appears to be the only lightweight mater capable of withstanding close to stoichiometric conditions. However, the oxidati and inspection problems associated with carbon-carbon materials must be address before it is viable for use in combustor liners.

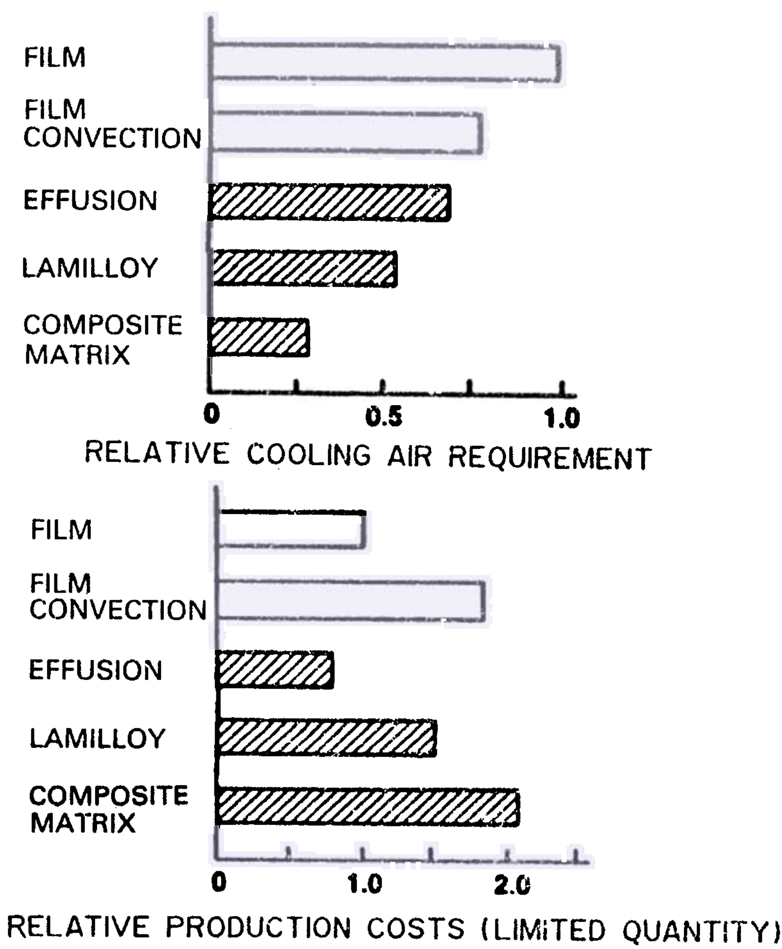


Figure 14. Cooling air requirements and production cost comparison of combustor liner cooling schemes[4].

The technology status of the three liner concepts can be summarized as follows : Transpiration designs are well developed and in use; effusion designs are being developed and are close to being applied; and composite matrix liners are largely in the experimental stage.

ACKNOWLEDGEMENTS

would like to thank General Electric Company for permission to publish this paper.

REFERENCES

Cifone, A.J. & Krueger, E.L., Combustion technology : A navy perspective, AIAA Paper No. 85–1400, July 1988.

2. Colladay, R.S., Importance of combining convection with film cooling, AIAA Paper No. 72–8, January 1972.

3. Nealy, D.A., Reider, S.B. & Mongia, H.C., Alternate cooling configuration for gas turbine combustion systems, AGARD, Propulsion and Energetics Panel, Paper No. 25, May 1985.

4. Mongia, H.C. & Reider, S.B., Allison combustion research and development activities, AIAA Paper No. 85–1402, July 1985.

5. Lefebvre, A.H., Gas Turbine Combustion, (McGraw-Hill, New York), 1983.

6. Wadia, A.R. & Nealy, D.A., to be published in *ASME Trans*, *Journal of Turbomachinery*, (1988).

7. Andrews, G.E., Alikhanizadeh, M., Asere, A.A., Hussian, C.I., Azari, M.S.K. & Mkpadi, M.C., *ASME Trans. Journal of Turbomachinery*, **108** (1987), 283–289.

8. Wadia, A.R. & Nealy, D.A., Development of an engineering design model for airfoil leading edge film cooling, ASME Paper No. 85–GT–120, March 1985.

9. Goldstein, R.J., Eckert, E.R.G. & Burgraff, F., *Intl. J. of Heat and Mass Transfer*, **17** (1974), 595–607.

10. Papell, S.S., Vortex Generating Flow Passage Design for Increased Film Cooling Effectiveness, US Patent No. 4, 529, 358, 1985.

Acosta, W.A. & Norgren, C.T., Small gas turbine combustor experimental study – compliant metal/ceramic liner and performance evaluation, AIAA Paper No. 86–1452, June 1986.

12. Norgren, C.T. & Riddlebaugh, S.M., Advanced liner cooling techniques for gas turbine combustors, AIAA Paper No. 85–1290, July 1985.

13. McAdams, W.H., Heat Transmission, (McGraw-Hill, New York), 1954.

14. Venkat, R.S. & Roffe, G., Testing of Felt-Ceramic Materials for Combustor Applications, NASA–CR 168103, April 1983.

15. Ercegoric, D.B., Walker, C.L. & Norgen, C.T., Ceramic composite liner material for gas turbine combustors, AIAA Paper No. 84–0363, January 1984.

16. Burrus, D.L., Chahrour, C.A., Foltz, H.L., Sabla, P.E., Seto, S.P. & Taylor J.R., E^3 Combustion System Component Technology Development Report NASA–R82AEB401, (NAS3–20643), November 1982.

17. Greene, W., Tanrikut, S. & Sokolowski, D., Development and operating characteristics of an advanced two stage combustor, AIAA Paper No. 82–0191 January 1982.

18. Tanrikut, S., Marshall, R.L. & Sokolowski, D.E., Improved combustor durability – segmented approach with advanced cooling techniques, AIAA Paper No. 81–1356, July 1981.

19. Meyers, G., Van Der Geest, J., Sanborn, J. & Davis, F., Comparison of avanced cooling concepts using color thermography, AIAA Paper No. 85–1289, July 1985.