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ANALYTICAL INVESTIGATION OF THERMAL BARRIER COATINGS ON ADVANCED POWER GENERATION GAS TURBINES

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WESTINGHOUSE ELECTRIC CORPORATION
GENERATION SYSTEMS DIVISION

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16. Abstract

An analytical investigation of present and advanced gas turbine power generation cycles incorporating thermal barrier turbine component coatings was performed. Approximately 50 parametric points considering simple, recuperated, and combined cycles (including gasification) with gas turbine inlet temperatures from current levels through 1644°K (2500°F) were evaluated. The results indicated that thermal barriers would be an attractive means to improve performance and reduce cost of electricity for these cycles. A recommended thermal barrier development program has been defined.

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SECTION 1
SUMMARY

1.1 OBJECTIVE

The objective of the present study is to:

> Evaluate by means of cycle efficiency and cost of electricity calculations the effect of the thermal barrier turbine blade coatings on current production and proposed advanced design gas turbine systems.

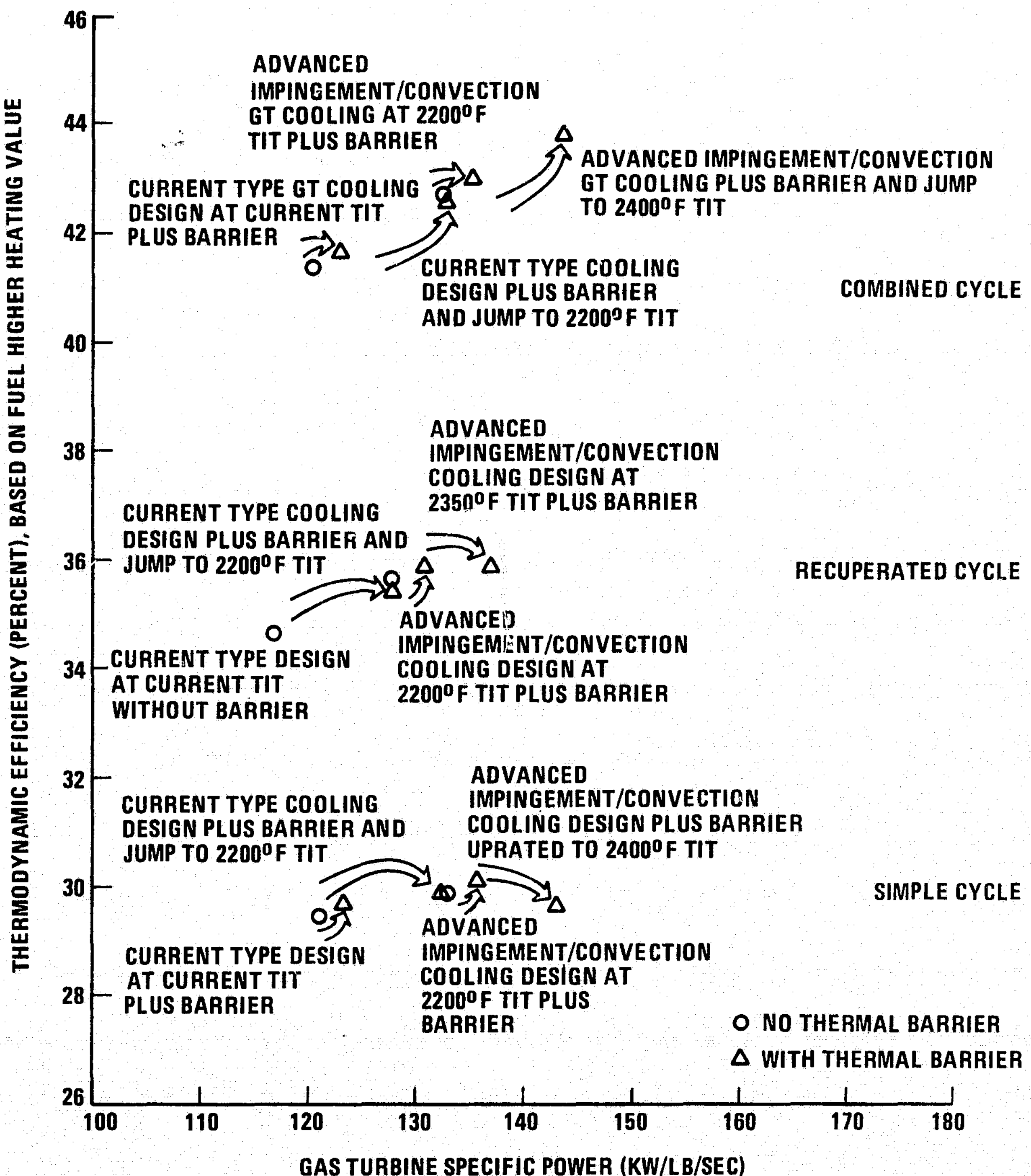
The gas turbines considered in this work are all components of land based power generation systems and the study has focused on the use of heavy duty power generation gas turbines of the single shaft category. Three distinct types of power generation systems have been considered including: simple cycle peaking gas turbines, recuperative cycle gas turbines, and combined gas-steam turbine cycles, including as a special case, the use of an integrated coal gasification subsystem.

The so-called thermal barrier coating under consideration is a duplex system consisting of a thin NiCRAlY bond coat and a yttria stabilized zirconia outer layer applied by plasma spray techniques over the external surfaces of turbine element blades and other components. This layer acts as a heat insulator and serves to provide a substantial reduction of the blade metal temperature below that of the exposed surface of the ceramic materials.

1.2 CYCLE PERFORMANCE AND COST OF ELECTRICITY RESULTS

The effect of thermal barrier addition and simultaneous increase of turbine inlet temperatures, T_{IT}, for current type designs is shown in Figure 1. In this analysis, turbine inlet temperatures have been increased stepwise from the current design levels of 1366 - 1422°K(2000 - 2100°F) to 1588°K (2400°F) with the addition of advanced impingement/convection cooling techniques, thermal barriers, and improved materials. It is seen that important efficiency gains for the simple and recuperated cycles are obtained by advancing turbine inlet temperatures to 1488°K (2200°F). Very significant combined cycle performance improvements are obtained at turbine inlet temperatures all the way to 1588°K (2400°F), the limit selected for uprating current-type gas turbine frames.

The corresponding cost of electricity comparisons resulting from simultaneous increase in turbine inlet temperature and addition of thermal barrier coating is illustrated by



EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON CYCLE EFFICIENCY
Figure 1

Figure 2 and the effect of firing residual fuel oil in place of distillate fuel oil in terms of cost of electricity is shown by Figure 3. (Thermal barriers are felt by NASA to offer corrosion resistance for residual firing.) Additional calculations have been made on the cost of electricity effects of direct retrofit of thermal barriers on existing turbines as well as on the effect of barrier replacement interval.

1.3 CONCLUSIONS AND RECOMMENDATIONS

The following principal conclusions have been drawn as a result of this study:

1. Thermal barrier application is an attractive potential means to improve performance and reduce cost of electricity for simple, recuperated and combined cycles.

2. The largest payoff in terms of efficiency increase and cost of electricity improvement associated with turbine inlet temperature increase and thermal barrier introduction occurs with the combined cycle.

3. The largest single jump in efficiency increase and cost of electricity decrease of those steps considered occurs with all three cycles as current type cooling systems are uprated by the addition of the thermal barrier and turbine inlet temperature increase to 1477 $^{\circ}$K (2200 $^{\circ}$F).

4. Significant performance improvements and cost of electricity reductions can be achieved with near-term advanced versions of today's combined cycles at turbine inlet temperatures to 1588 $^{\circ}$K (2400 $^{\circ}$F).

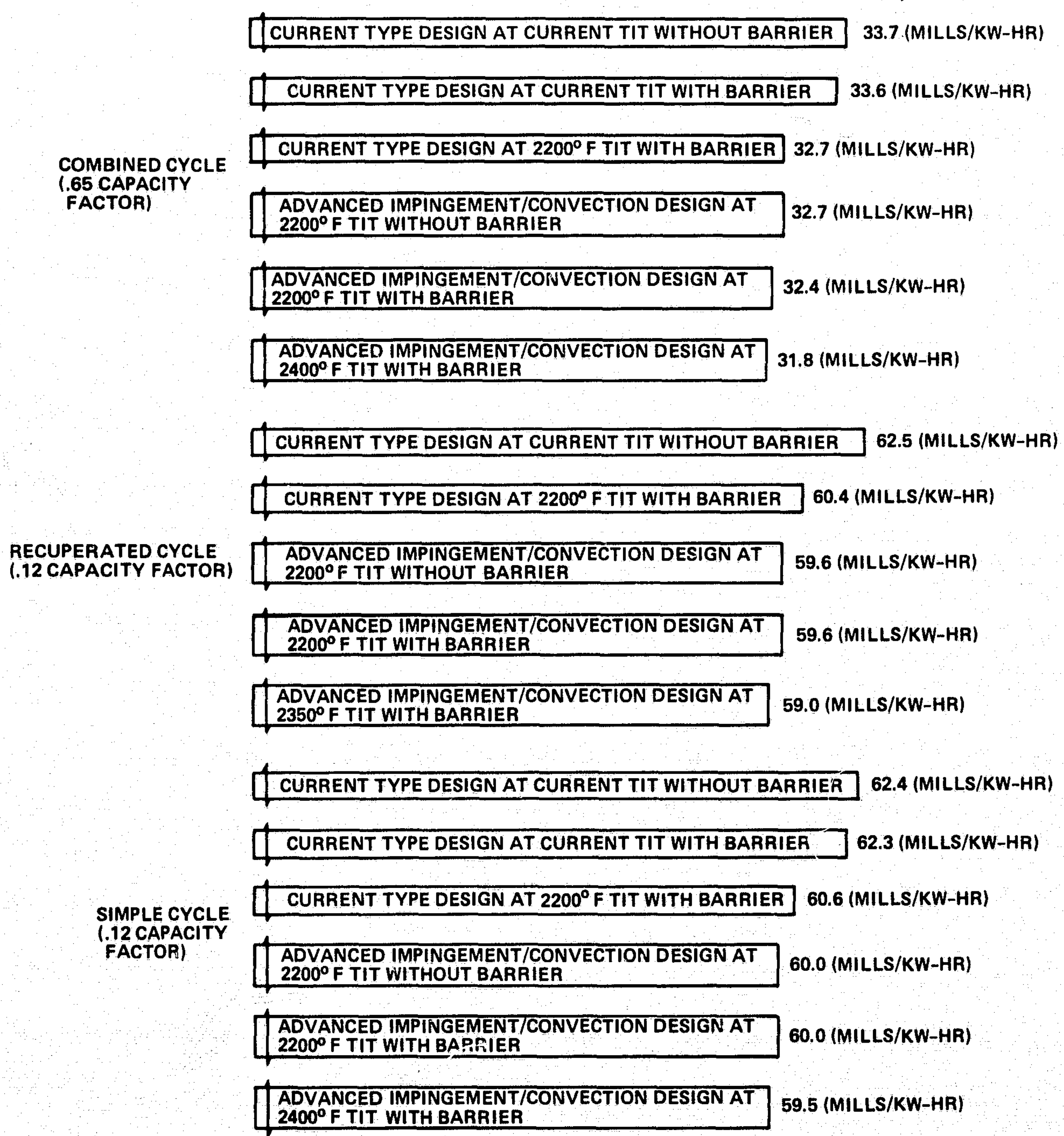
5. With the given fuel prices, substitution of residual fuel results in significant cost of electricity decreases for all three types of cycles.

It is recommended that further development of the thermal barrier concept incorporate the following:

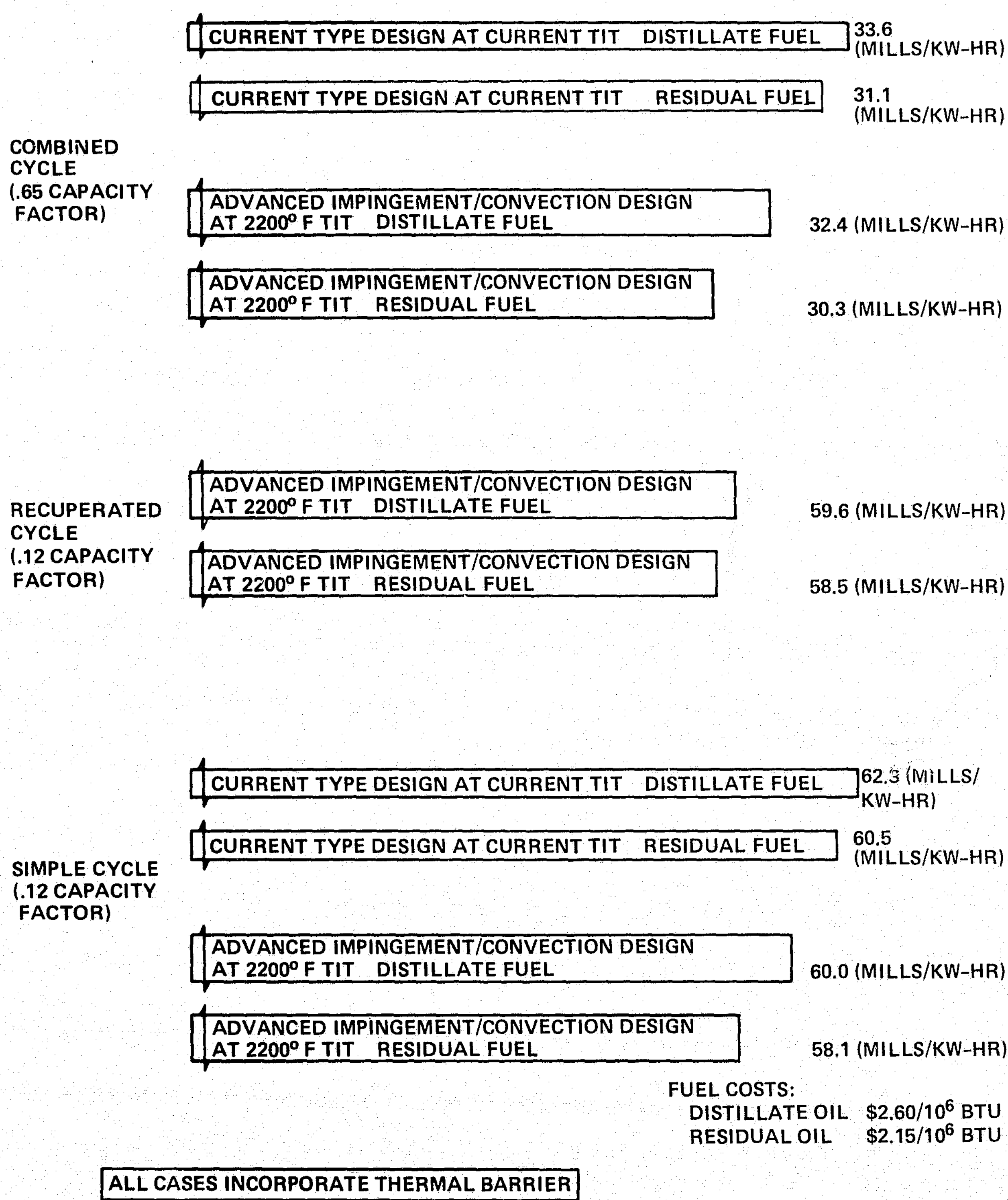
1. The first goal in implementing thermal barrier development should be to apply the barrier to current type cooling designs and uprate to 1477 $^{\circ}$K (2200 $^{\circ}$F).

2. Ultimate development of present type combined cycle designs to 1588 $^{\circ}$K (2400 $^{\circ}$F) turbine inlet temperature in conjunction with thermal barrier introduction should be pursued.

3. The potential application of a temperature uprate - 1477 $^{\circ}$K (2200 $^{\circ}$F) to 1588 $^{\circ}$K (2400 $^{\circ}$F) - of a current type design gas turbine using thermal barriers should be evaluated for use in coal and oil gasification combined cycles.



EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON COST OF ELECTRICITY
Figure 2

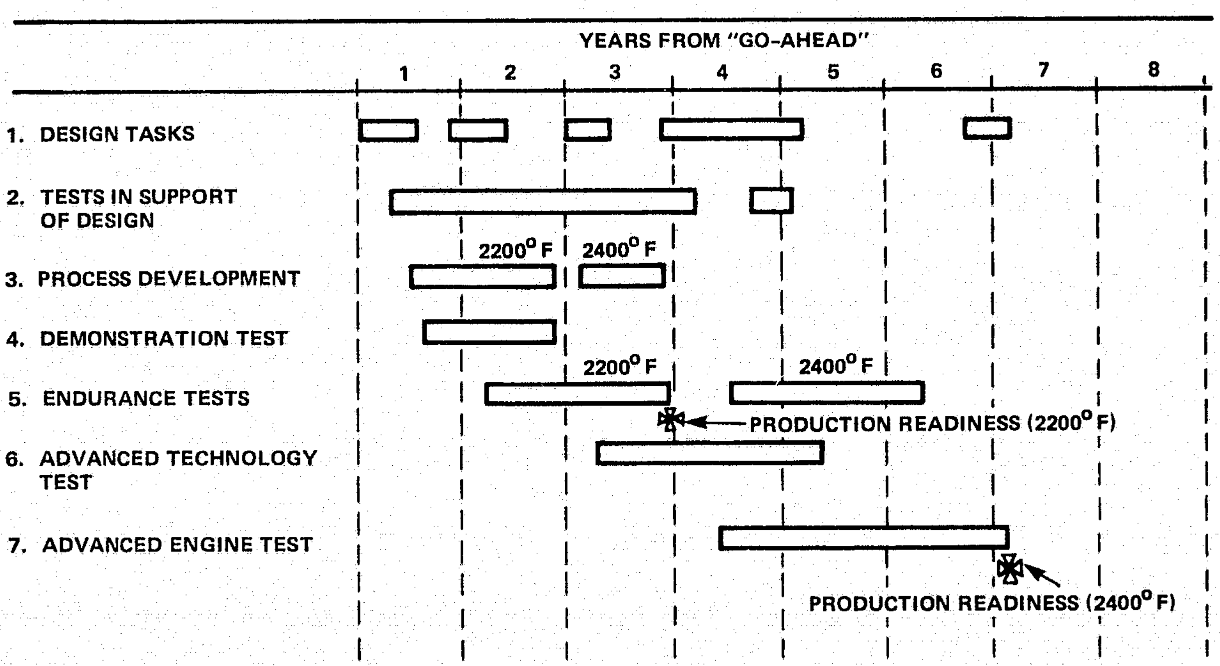


EFFECT OF RESIDUAL OIL FIRING ON COST OF ELECTRICITY
Figure 3

4. Determination of the effects of typical residual fuel firing on thermal barrier coating integrity should be investigated with high priority.

5. The potential for retrofitability of thermal barrier coatings to existing utility machines for clean fuel and ultimately residual and coal-derived liquids should be evaluated in more depth.

A recommended thermal barrier development program has been prepared which incorporates the essential elements identified above. Figure 4 displays the suggested tasks included and the associated time schedule. The program incorporates a two phase approach in which current type power generation gas turbines would be uprated to a near-term turbine inlet temperature target of 1477°K (2200°F) and production readiness displayed in three years; and subsequently an increase to 1588°K (2400°F) production readiness after a total elapsed time of six years is proposed.



RECOMMENDED THERMAL BARRIER DEVELOPMENT PROGRAM SCHEDULE
Figure 4

SECTION 2
INTRODUCTION

2.1 THERMAL BARRIER DEVELOPMENT BACKGROUND AND STATUS OF TECHNOLOGY

Prior studies in conjunction with rocket nozzle design and aircraft turbine engine research had indicated the desirability of protecting certain hot section components with a thin ceramic surface layer. (References 9.1, 9.2)

Early development work associated with the application of ceramic thermal barriers to gas turbine hot components focused on the approach of applying cast and hot pressed zirconia and hot pressed alumina to metallic honeycomb substructures (Reference 9.3). Tests of this approach included thermal shock testing, ballistic impact evaluation and erosion testing. It was concluded that the application of a ceramic thermal barrier has the potential for significant reduction of cooling air expenditure for both combustor and blading components.

More recent experimental work has been conducted for the purpose of investigating the resistance of the ceramic thermal barrier material to corrosive attack (Reference 9.4). In this work the attention was focused upon zirconia* (ZrO_2) stabilized by the oxides calcia, CaO, magnesia, MgO, and yttria, Y_2O_3. Tests included exposure of solid ZrO_2 specimens to Na_2SO_4 at 1172°K (1650°F) for up to 1000 hours as well as exposure of zirconia coated metal specimens in corrosive atmospheres in special burner test passages. For these tests, the coatings were applied via plasma spray to approximately 0.5mm (.020 inch) thick utilizing a system of multiple bond coats compared to application over a special honeycomb matrix used in the work cited above. The test results indicated good resistance to corrosive attack as well as thermal insulation, although continued effort to optimize coating adherence was recommended.

A research program for thermal barrier development has been initiated at the NASA-Lewis Research Center and has included successful laboratory tests of thermal barrier coatings in an operating aircraft type gas turbine engine (Reference 9.5).

In this work, as well as in the previously cited example, a stabilized zirconia has been applied by means of a plasma spray technique. This ceramic coating has been applied to turbine blading directly over a metallic bond coat of NiCRAlY to a total coating

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*The zirconia has been selected partly due to the fact that its coefficient of thermal expansion is much higher than other oxides and thereby more nearly matches that of the substrate metal.

thickness of approximately 0.5mm (.020 in). (Reference 9.6) Related work has evaluated the effect of thermal barrier surface roughness on cascade aerodynamic performance (Reference 9.7) as well as on the durability of the coating in cyclic engine operation. (Reference 9.8)

Summarizing major thermal barrier test results obtained by the NASA-Lewis Research Center to date, the engine test portions of the program have shown that the coated blades of a specially modified aircraft test engine withstood 35 start-stop cycles to acquire a total of 150 hours of testing at turbine inlet temperatures as high as 1644°K (2500°F) and 500 two min. cycles from 1644°K (2500°F) to flameout with no signs of deterioration. In addition, the coating system exhibited good adherence to blade and vane metals when cycled in a static burner test facility 40 times between the temperatures of 1367°K (2000°F) and 300°K (80°F). Temperature measurements made on coated and uncoated blading during the engine tests corroborated the theoretically predicted insulating effect of the coating.

2.2 ENERGY CONVERSION ALTERNATIVES STUDY (ECAS) BACKGROUND

As an adjunct to the planning efforts of the Federal Government on advanced energy concepts, the Energy Conversion Alternatives Study (ECAS) was initiated in 1974. The ECAS study, supported by the United States Energy Research and Development Administration (ERDA), the National Science Foundation (NSF), and NASA, had as its goal the comparative evaluation of advanced electrical generation concepts utilizing coal or coal-derived fuels. Westinghouse Electric Corp. was one of the participating contractors in the ECAS study.

One of the most significant results to come from the ECAS study (References 9.9, 9.10, 9.11) was that for advanced electrical generation systems the use of gas turbines figure prominently. One particular example of an attractive cycle utilizing gas turbines is the gas turbine-steam turbine combined cycle system employing an integrated coal gasification subsystem.

A further key result of the ECAS study was that very significant gains in overall energy conversion efficiency are realized as the gas turbine inlet temperature is increased, particularly so in the case of combined cycle systems, to a lesser degree for recuperated cycles, and to a very limited degree for simple cycles.

During the course of the ECAS contract work at Westinghouse, with subcontract architect/engineering duties performed by C. T. Main, Inc. of Boston, several advanced concept plant designs were prepared and methods of estimating capital cost and cost of electricity were prepared and evaluated by NASA, ERDA and other special advisory groups.

It is against this backdrop of the ECAS study that the present Thermal Barrier evaluation has been executed. An important thrust of the thermal barrier is of course to facilitate higher turbine inlet temperatures for more efficient power generation via gas turbines in addition to providing potential for longer component life, increased tolerance to dirty fuels, and lowering cooling flow requirements. This is particularly the case with respect to near-term potentials for early implementation as a consequence of the promising results obtained to date on the thermal barrier concept. In addition, the present study has been structured for capital cost estimation and cost of electricity calculations in such a manner that the basic groundrules developed during the ECAS study have been utilized (Discussed in Section 6.1).

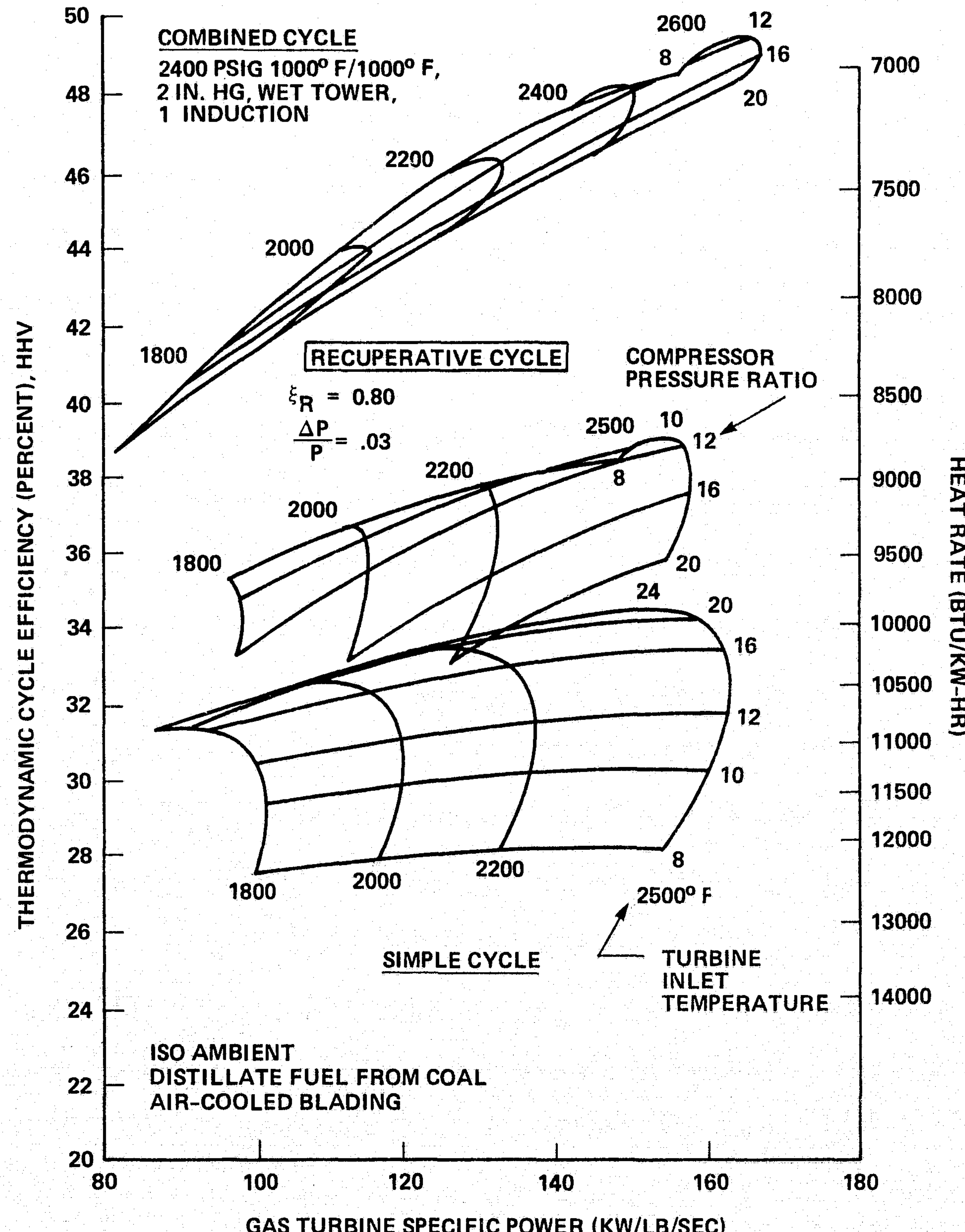
2.3 FUEL CONSERVATION POTENTIAL OF PRESENT AND ADVANCED GAS TURBINE CYCLES

Fossil fired steam power plants have essentially plateaued at thermal efficiency values of 36 - 40% depending on the presence and/or method of emission control of flue gases. Present gas turbine combined cycles, by comparison, can have higher thermal efficiency and considerable additional energy conservation potential exists with advancements of these cycles. Specifically, by increasing turbine inlet temperatures from current levels of 1366 - 1422^{o}K (2000 - 2100^{o}F) to say 1700^{o}K (2600^{o}F) as much as four additional cycle efficiency percentage points could be added in the case of the combined cycle as shown in Figure 5.*

It is important to note the differences among the three types of cycles as turbine inlet temperatures are increased. For a given compressor pressure ratio it is seen that the simple cycle responds with only a very modest efficiency increase at higher turbine inlet temperatures (and for some values of pressure ratio the efficiency actually decreases at the higher temperatures). Recuperated and combined cycles respond more decisively to higher turbine inlet temperatures. It is seen from Figure 5 that the combined cycle derives the greatest efficiency improvement from higher turbine inlet temperatures while the recuperated cycle falls somewhat between the simple and combined cycles.

In order to bring the conservation potential of higher turbine inlet temperatures into sharper focus it is useful to consider the following examples. By increasing gas turbine inlet temperature in a combined cycle powerplant from current levels to 1588^{o}K (2400^{o}F) overall power plant efficiency could be improved by about 3 efficiency points (or about 500 Btu/KW-HR). For a nominal combined cycle output of 300MW operated at 65% capacity factor and using fuel costs of \$2.46/MJ (\$2.60/10^{6} Btu) this improvement is equivalent to a savings of approximately 2.2 million dollars in fuel costs alone each year. The following example relates to the potential for advanced combined cycle fuel savings on the national level. By the end of 1975 there was approximately 5700MW of combined cycle capacity in the United States (Reference 9.12). With the same improvement in

*The results of Figure 5 were taken from the ECAS Task I Parametric Analysis Study (Reference 9.9). They correspond to mathematical design point models and although efficiency levels may not precisely concur with some existing powerplants the differentials and trends are quite valid.



PARAMETRIC CYCLE PERFORMANCE COMPARISON FROM ECAS TASK 1
Figure 5

efficiency described in the example above and with those powerplants operating at base load (65% capacity factor) the savings in fuel alone would be over 2.5 million barrels of fuel oil per year.

At 5700MW, today's combined cycles represent only 1.6% of the installed 1975 year-end fossil fuel base load capacity of 351,245MW (Reference 9.13). It is expected, however, that the combined cycle will enjoy greatly increased capacity additions in the future, thus multiplying the potential fuel oil savings even more.

2.4 CANDIDATE APPLICATIONS OF THERMAL BARRIERS IN NEAR TERM AND ADVANCED POWER GENERATION GAS TURBINES

A typical state-of-the-art electric power generation gas turbine engine is shown in Figure 6. This unit, the Westinghouse model W501D, is representative of the large, single shaft type of industrial design power generation gas turbines. Typical performance specifications for this unit are listed below in Table 1.

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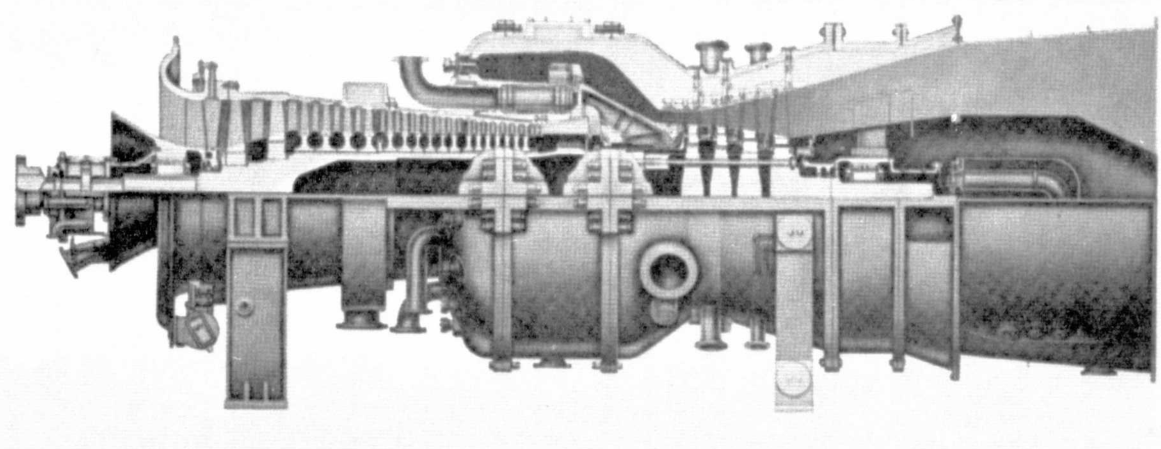


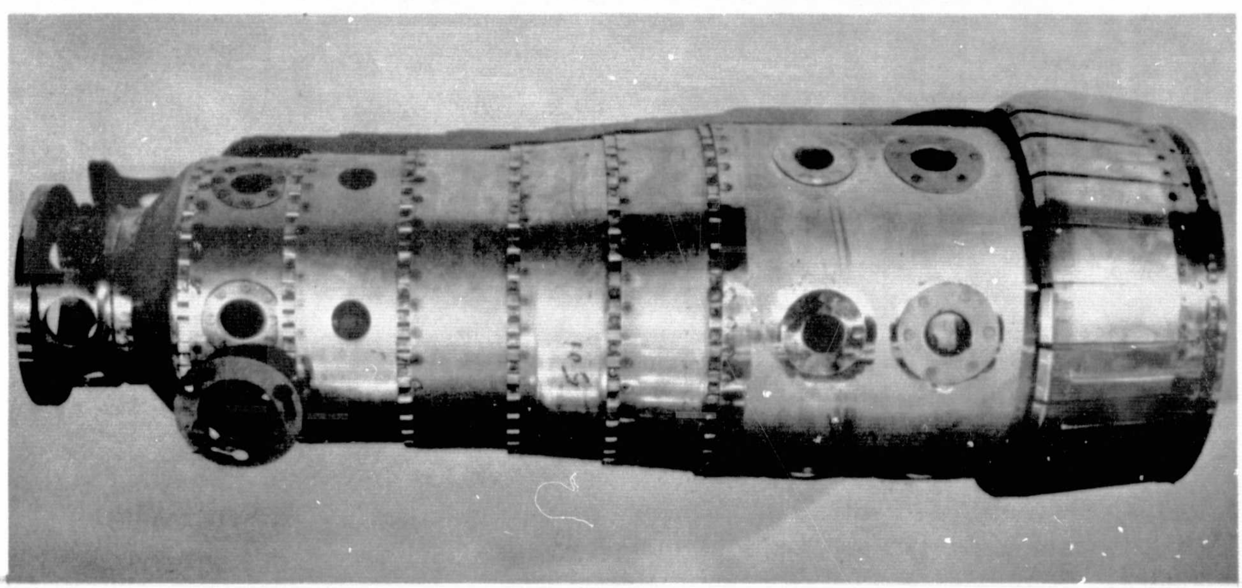
STATE-OF-THE-ART ELECTRIC POWER GENERATION GAS TURBINE ENGINE
Figure 6

The candidate applications for thermal barrier coatings of course, are found in the gas turbine "hot section" consisting of the gas turbine combustor liners, transition pieces (linking combustor with first row stationary vane), stationary vanes and rotating blades.

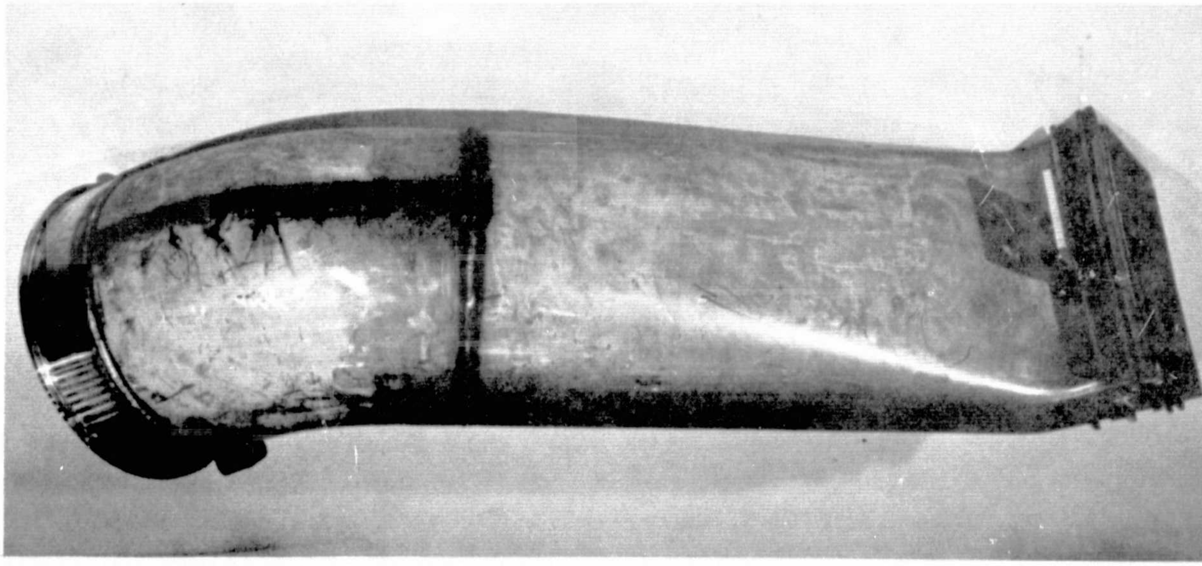
TABLE 1
TYPICAL PERFORMANCE SPECIFICATIONS W501D GAS TURBINE
ISO PEAK RATING-SIMPLE CYCLE

Shaft roational speed	3600 RPM
Inlet air flow	350 KG/SEC (780 LB/SEC)
Compressor pressure ratio	12.7:1
Combustion System	Can-Annular (16 Cans)
Number of Turbine Stages	4
Turbine Inlet Temperature (Base Load)	1366 - 1422 ^{o}K (2000 - 2100 ^{o}F)
Peak Load Output	98.5 MW
Dimensions	7.6m (25 ft) x 3.5m (11.5 ft) x 3.5m (11.5 ft)
Mass (Weight)	127000 KG (280000 LB)

Illustrated by Figures 7 and 8 are current design combustor liners and transition pieces. These elements receive very high convective and radiative heat loads and therefore require substantial convective and film cooling. These areas represent ideal potential applications for thermal barrier coatings. The coating would be applied to the inner surfaces of both the transition pieces and the combustor liners.



TYPICAL CURRENT DESIGN GAS TURBINE COMBUSTOR LINER
Figure 7



TYPICAL CURRENT DESIGN GAS TURBINE TRANSITION PIECE
Figure 8

Gas turbine stationary vane design practice today is generally characterized by the use of cast airfoils which are often joined at the hub and tip shroud regions to form a multi-vane segment as shown by Figure 9. In this application, the coating would be applied over the external airfoil and shroud surfaces. It should be pointed out that the plasma spray process used to apply the thermal barrier coating, in its present status of technology, requires direct line-of-sight access for complete coverage. This may not be possible for the multivane segments as shown however. Thus a routine design change would be required to change the vane configuration to a single vane arrangement.

The gas turbine rotor blades, except for size, are similar to those already coated and operated in the NASA-Lewis thermal barrier test program. Figure 10 illustrates the size of a typical bladed turbine rotor employed in a power generation gas turbine engine.

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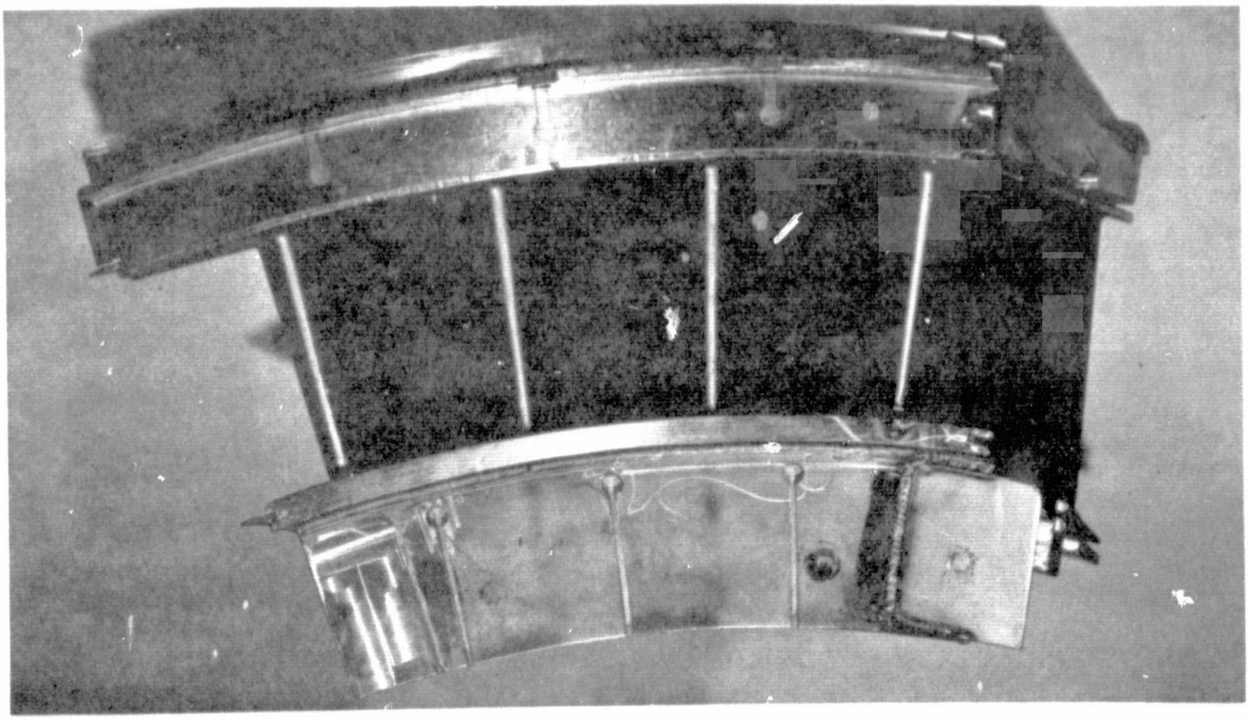
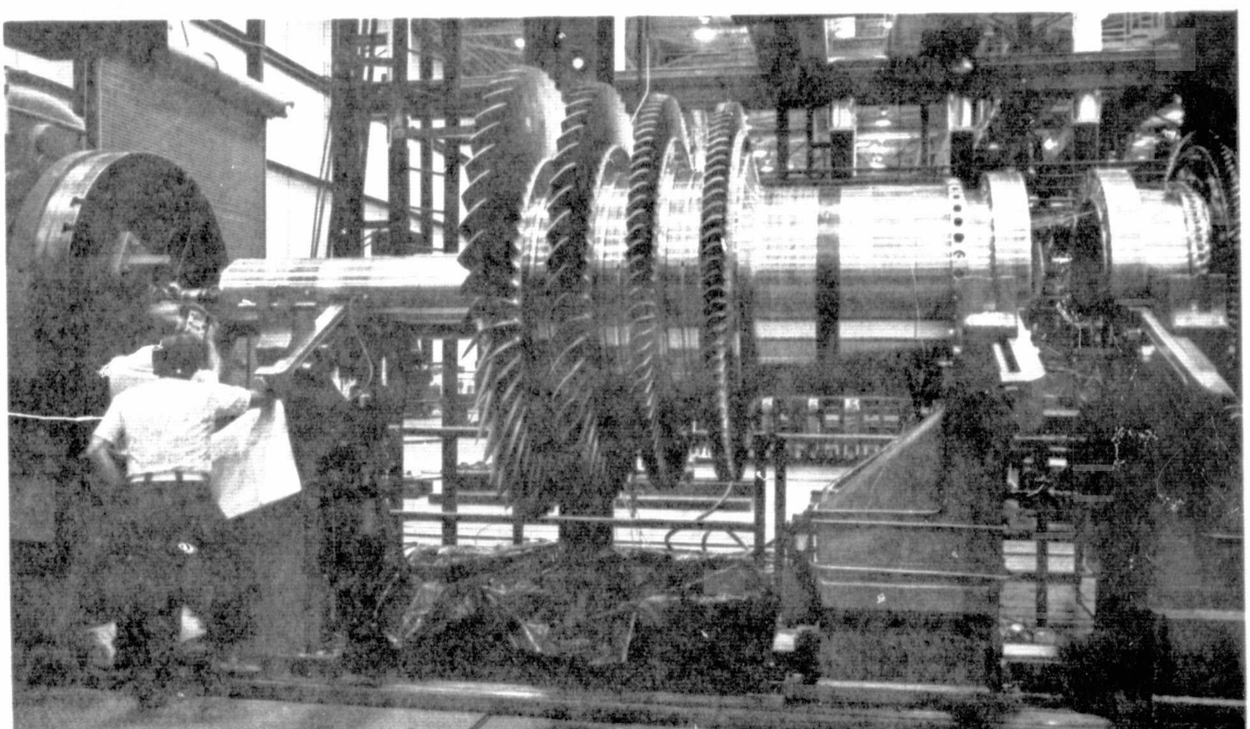


TYPICAL CURRENT DESIGN MULTIPLE STATIONARY VANE SEGMENT
Figure 9



BLADED POWER GENERATION GAS TURBINE ELEMENT ROTOR ASSEMBLY
Figure 10

SECTION 3
HEAT TRANSFER ANALYSIS

3.1 ONE-DIMENSIONAL CONDUCTION ANALYSIS

3.1.1 Objectives

The objectives of the one-dimensional heat transfer conduction analysis are to:

(a) Determine the temperature drops attainable through the thermal barrier coating as applicable to blades characteristic of current design industrial power generation gas turbines.

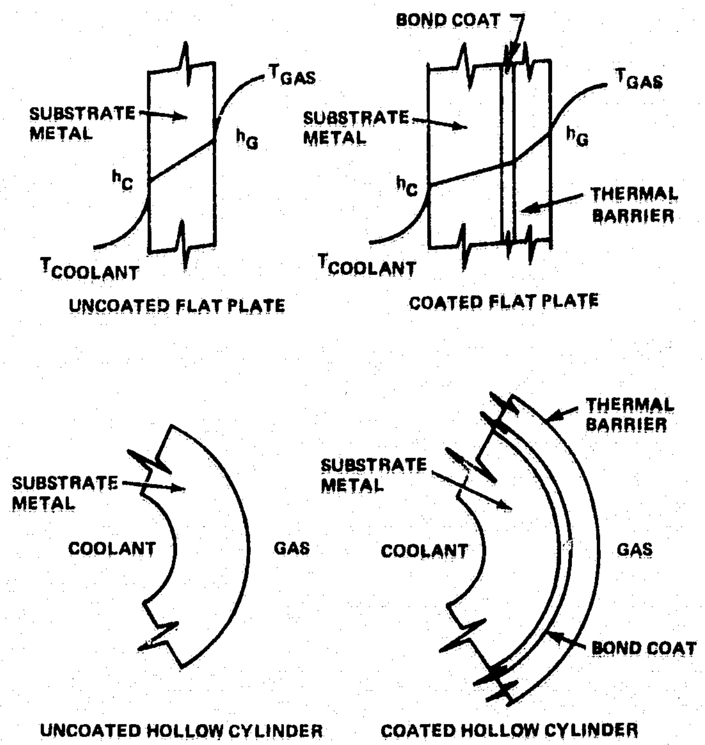
(b) Estimate the potential reduction of cooling flow applicable to power generation gas turbines as a result of the introduction of the thermal barrier coating.

3.1.2 Approach

In order to accomplish these objectives, one-dimensional analysis models were used as shown in Figure 11. The thermal barrier models utilized the two layer configuration with material conductivities of both the bond coat and ceramic layer coat specified by NASA.

Table 2 lists the thermal conductivities of the bond coat and thermal barrier material as a function of mean metal temperature in addition to the base metal, Mar M509, used in the one-dimensional heat transfer analyses.

The individual layer thicknesses are, speaking strictly from an analytical viewpoint, independent variables. However, NASA has specified for the purposes of this study those thicknesses - bond coat 0.10mm (.004 in.), and ceramic overcoat 0.38mm (.015 in.) - which lie within the range successfully applied and tested in actual aircraft engine tests run by NASA. An additional case with arbitrarily selected coating thickness .76mm (.030 in.) has also been evaluated. The analysis techniques developed for this portion of the one-dimensional analysis, however, are completely general and can be extended to other coating thicknesses.



ONE DIMENSIONAL HEAT TRANSFER MODELS
Figure 11

TABLE 2
THERMAL CONDUCTIVITY OF BASE METAL, BOND COAT, AND THERMAL BARRIER MATERIALS

Material	Thermal Conductivity	Temperature
MAR M509 (Base Metal)	$k(\frac{W}{m\text{-}K}) = 0.169T(^oK) + 6.8$	$366\text{-}1255^oK$
	$k(\frac{Btu}{hr\ ft^2\ \frac{^oF}{ft}}) = 5.36 \times 10^{-3}T(^oR) + 4.04$	$(200\text{-}1800^oF)$
NiCRAlY (Bond Coat)	$k(\frac{W}{m\text{-}K}) = 0.0083T(^oK) + 6.7$	$400\text{-}1400^oK$
	$k(\frac{Btu}{hr\ ft^2\ \frac{^oF}{ft}}) = 2.67 \times 10^{-3}T(^oR) + 3.87$	$(250\text{-}2050^oF)$
Y_2O_3 Stabilized ZrO_2 (Thermal Barrier)	$k(\frac{W}{m\text{-}K}) = 0.00022T(^oK) + 1.09$	$400\text{-}2200^oK$
(CaO Stabilized ZrO_2 properties used)	$k(\frac{Btu}{hr\ ft^2\ \frac{^oF}{ft}}) = 7.06 \times 10^{-5}T(^oR) + 0.63$	$(250\text{-}3000^oF)$

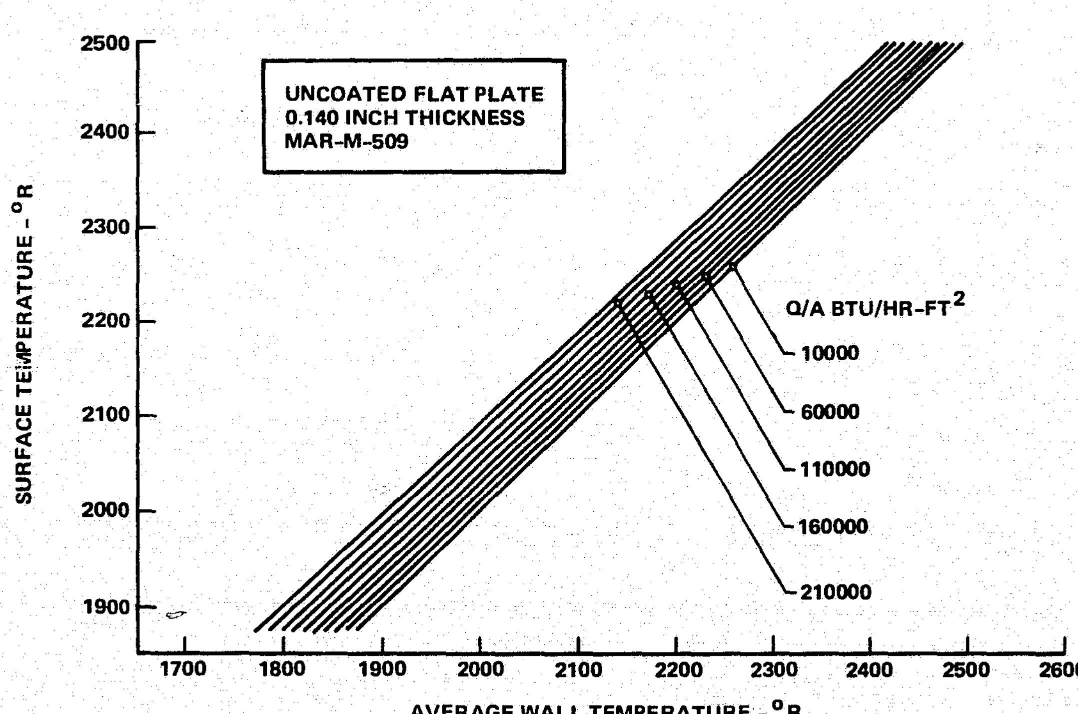
For the uncoated flat plate model the one-dimensional relationship

$$\frac{Q}{A} = \bar{k}\,\frac{(T_{SURF} - T_{AVG\ WALL})}{L/2}$$

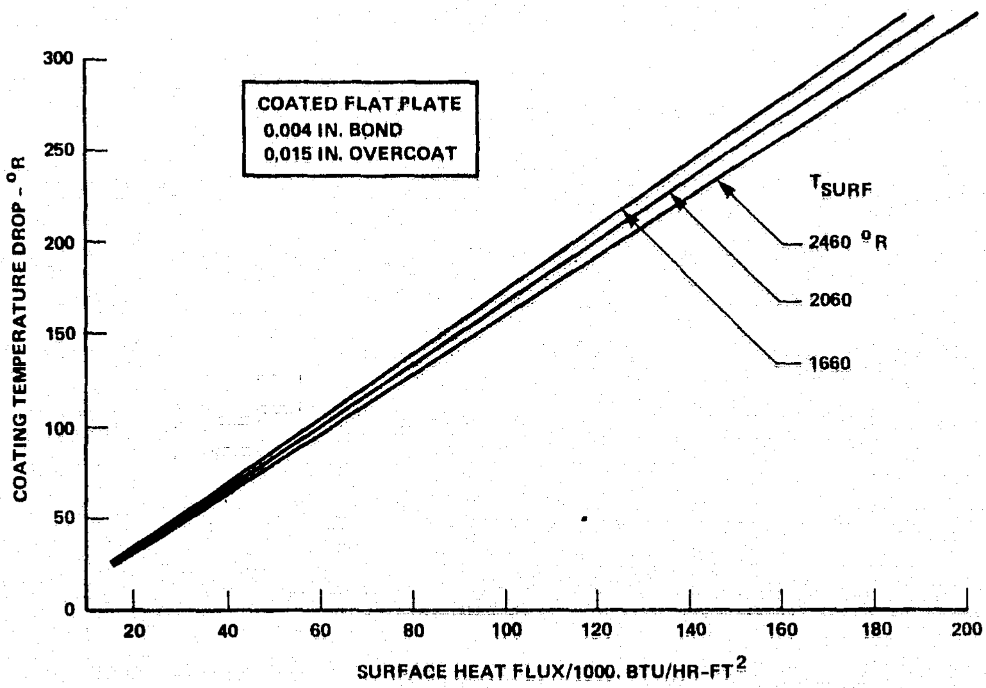
where $\bar{k}$ = average wall thermal conductivity

was used to relate heat flux to surface temperature and average wall temperature for a given wall thickness and the appropriate material thermal conductivity. Parametric variations of these quantities (surface temperature, average wall temperature, and wall thickness) spanning a range of interest in current design were computed. Figure 12 displays the results of one such case. Reference values of average wall temperature, surface temperature, and heat flux were then selected for use in later comparison with the corresponding wall thickness coated flat plate case.

Subsequently, temperature drops through the coating system were computed parametrically. The same basic relationship described above was used to compute the coating system temperature drop as a function of heat flux and coating surface temperature. Figure 13 illustrates typical results.



PARAMETRIC SURFACE TEMPERATURE, AVERAGE WALL TEMPERATURE, AND HEAT FLUX VARIATIONS FOR UNCOATED FLAT PLATE
Figure 12



THERMAL BARRIER COATING SURFACE TEMPERATURE DROP AS A FUNCTION OF SURFACE HEAT FLUX
Figure 13.

Next the iterative portion of the calculation was executed. In this procedure, a new lower heat flux Q/A is assumed for the coated flat plate. From the convective heat transfer relationship

$$\frac{Q}{A} = h\,(T_{GAS} - T_{SURF})$$

(in which h and T_{GAS} are held at the same values as for the uncoated case) a new value for the coating surface temperature is determined. With the current value of Q/A, the coating temperature drop curve, Figure 13, is consulted to give a value for the coating temperature drop. This value can then be used to give the temperature at the inner surface of the coating which is equivalent to the metal outer surface temperature. Using the current working value of Q/A and the desired average wall temperature the same as the reference value identified for the uncoated flat plate of the same thickness the original curve of uncoated flat plate temperature drop, Figure 12, is consulted to give a value for metal surface temperature. This value is then compared with the metal surface temperature predicted from the coating temperature drop analysis. If the values of surface temperature do not agree, a new value of Q/A is selected and the process is repeated until the surface temperatures converge.

In addition to the above flat plate one-dimensional model, the same procedure has been used to predict heat flux reduction associated with thermal barrier coated hollow cylinders of varying radius and wall thickness values. In actual design practice a combination of cylindrical and flat plate simplified models such as these are used for preliminary design heat transfer analysis of cooled airfoil sections.

3.1.3 Predicted Turbine Blade Cooling Requirement Reduction

After compilation, the heat flux values for coated and uncoated one-dimensional models have been related to blading cooling air requirements as a function of cooling effectiveness where:

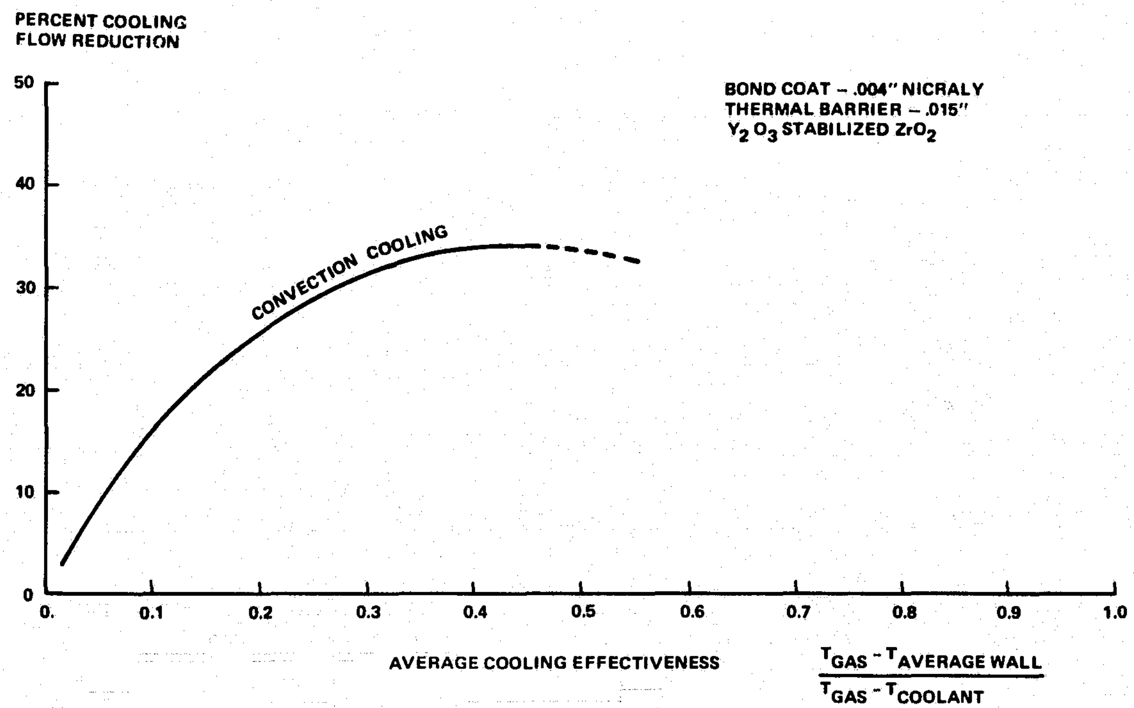
$$\text{Cooling effectiveness} = \frac{T_{GAS} - T_{AVG\ WALL}}{T_{GAS} - T_{COOLANT\ INLET}}$$

Current type industrial gas turbine cooled blading designs typically employ convective or impingement/convective cooling schemes for which the cooling effectiveness values range to approximately 0.4. (Due to the intrinsic stagewise temperature distributions, the earlier cooled stages have highest and latter cooled stages have the lowest cooling effectiveness values.) The results of the one-dimensional heat transfer analysis cooling air reduction potential associated with the thermal barrier are shown in Figure 14. It is seen that cooling air reductions of up to 35%, dependent upon cooling effectiveness, are possible as a result of application of the specified thickness thermal barrier coating. The reason for the levelling of the curve in Figure 14 relates to the efficiency at which cooling air absorbs heat in the convection cooling mode. Cooling effectiveness values higher than say 0.4 are achievable with convective cooling schemes, however, an inordinately large amount of cooling air is consumed in the process reflecting less efficient heat pick up. Although the thermal barrier coating is even more effective with higher heat flux the reduced efficiency by which convective schemes use cooling air provides a net drop-off in percentage cooling air reduction.

3.2 TWO-DIMENSIONAL CONDUCTION ANALYSIS

3.2.1 Objective And Model Formulation

The two-dimensional heat transfer analysis work has been designed to complement the above described one-dimensional heat transfer analysis and to extend the analysis work into areas of particular interest. Specifically, an actual industrial design gas turbine blade shape has been analyzed for the purpose of corroborating the one-dimensional temperature drop predictions and for evaluating the extent of temperature pattern redistribution.

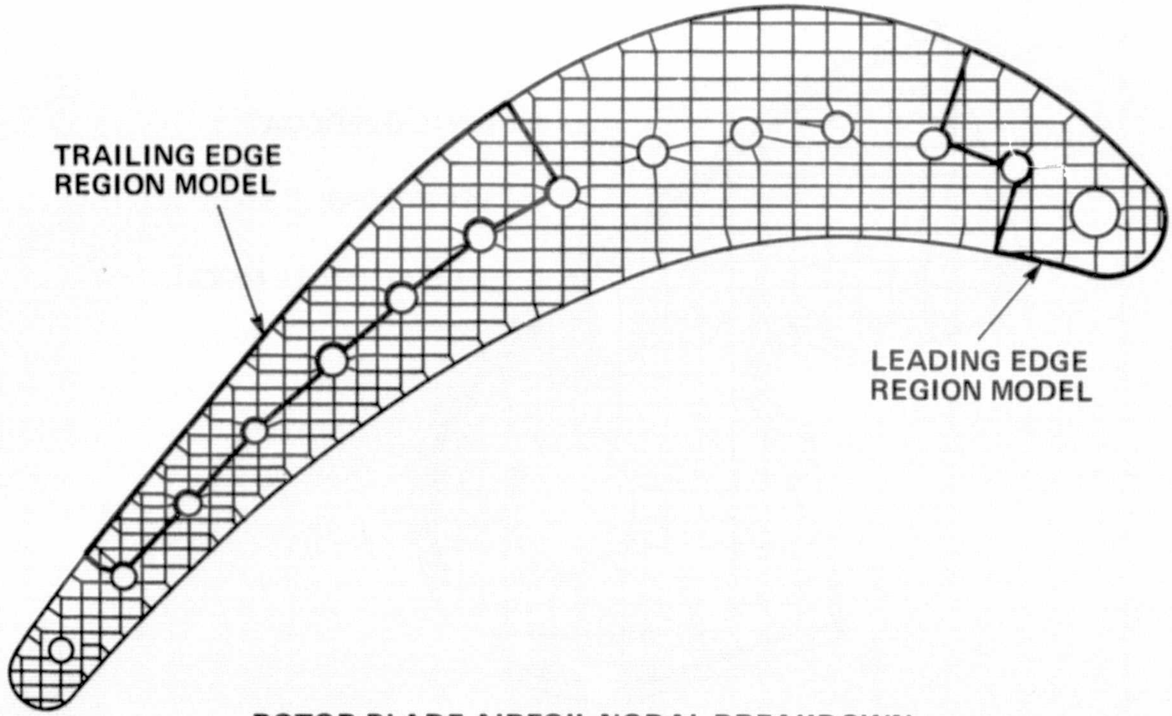


COOLING FLOW REDUCTION WITH THERMAL BARRIER VS COOLING EFFECTIVENESS
Figure 14

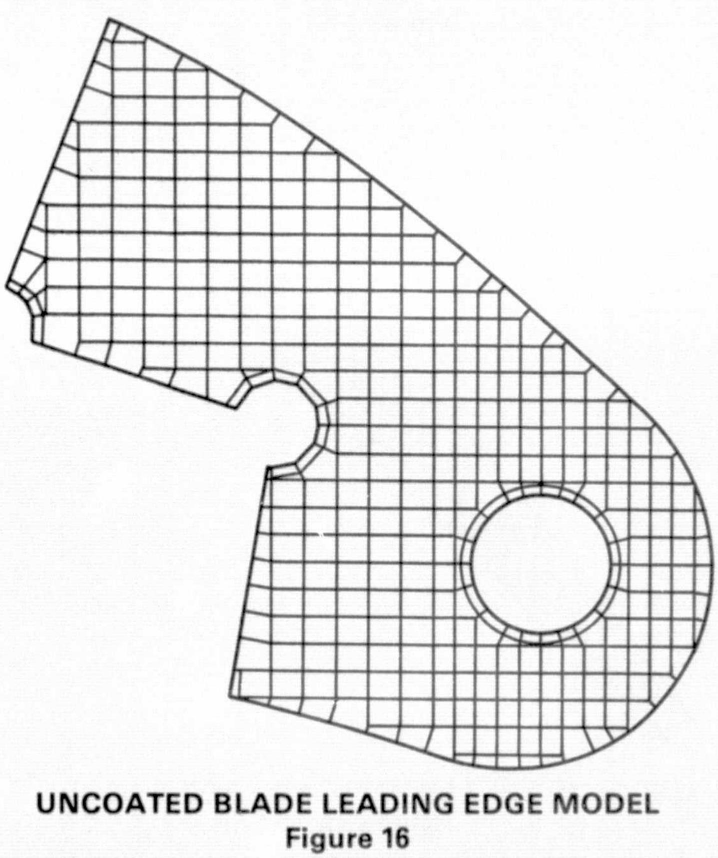
Selected for analysis was a mean height cross section from a current design industrial gas turbine rotor blade. Figure 15 illustrates a nodal breakdown of this particular airfoil. The leading edge and trailing edge regions were selected for detailed examination. The leading edge region in particular encompasses the area of highest heat flux on the airfoil and therefore is the region to be affected most by the addition of the thermal barrier. The trailing edge region is representative of several one-dimensional heat transfer areas of the gas turbine as well as the particular airfoil problem at hand.

3.2.2 Description of Two-Dimensional Calculation Method

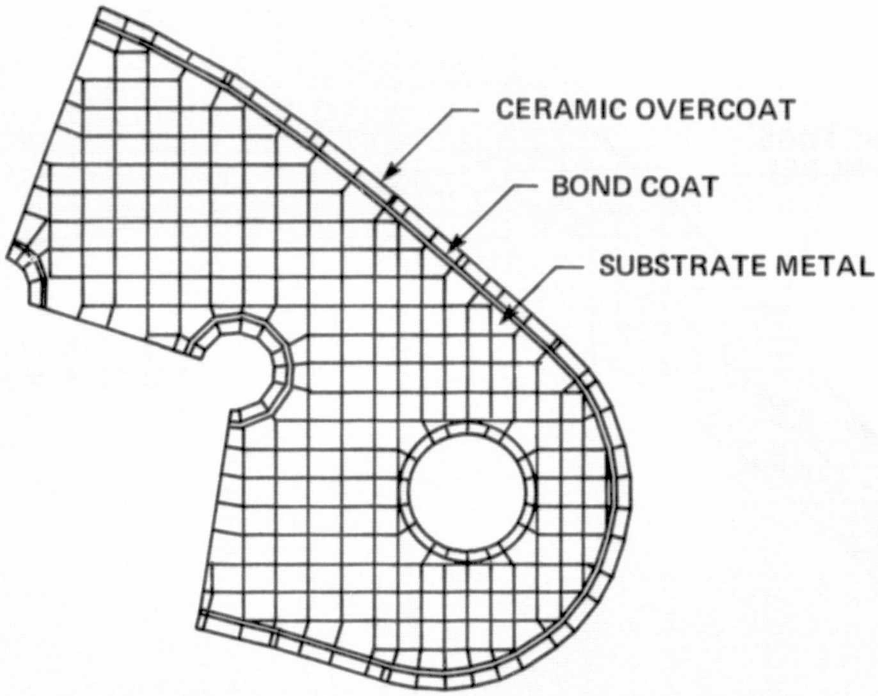
The individual leading edge and trailing edge calculation models are illustrated by Figures 16 through 19. The figures illustrate the finite difference meshes utilized for solution of the convection/conduction heat transfer problem for both coated and uncoated cases. For all cases, the meshes were generated automatically and they incorporate the use of small nodes in the region of cooling holes for finer temperature resolution.



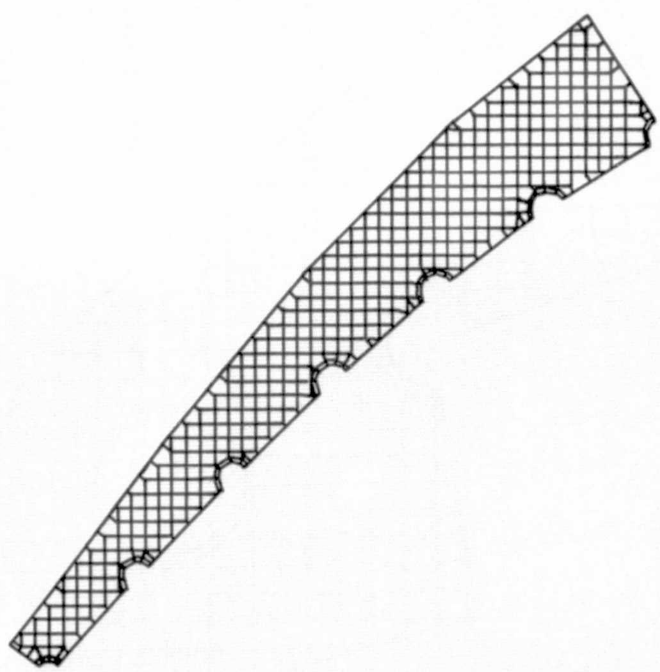
ROTOR BLADE AIRFOIL NODAL BREAKDOWN
Figure 15



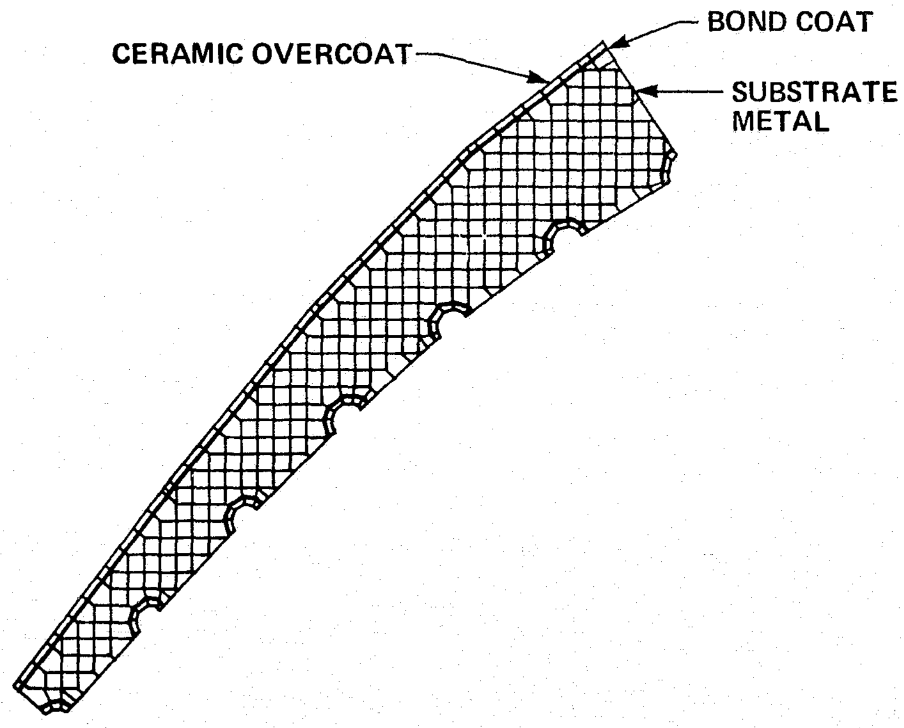
UNCOATED BLADE LEADING EDGE MODEL
Figure 16



THERMAL BARRIER COATED BLADE LEADING EDGE MODEL
Figure 17



UNCOATED BLADE TRAILING EDGE MODEL
Figure 18



THERMAL BARRIER COATED BLADE TRAILING EDGE MODEL
Figure 19

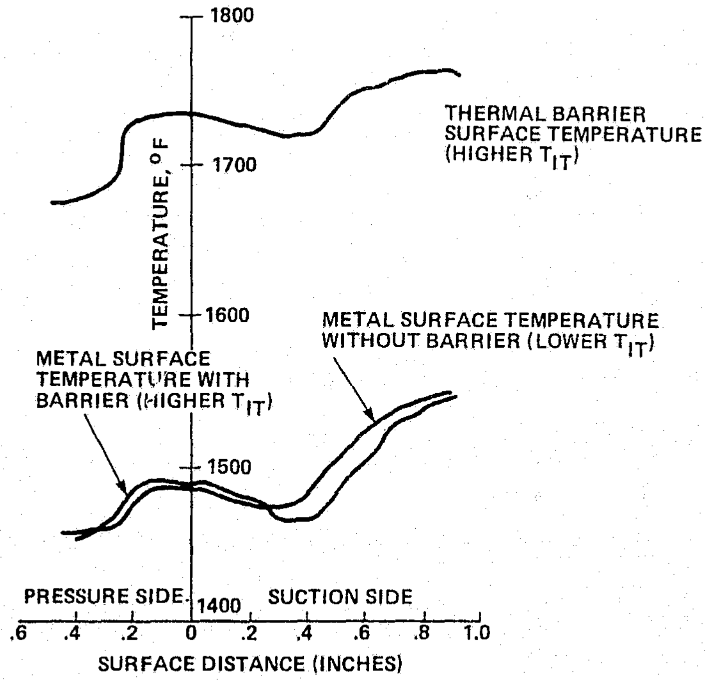
The two-dimensional model heat transfer calculations were executed by first computing velocity distributions and external heat transfer coefficients. These external boundary conditions and cooling air boundary conditions were then utilized in a two-dimensional implicit conduction analysis program in conjunction with the above described meshes to calculate temperature distributions.

3.2.3 Turbine Blade Temperature Distribution Results

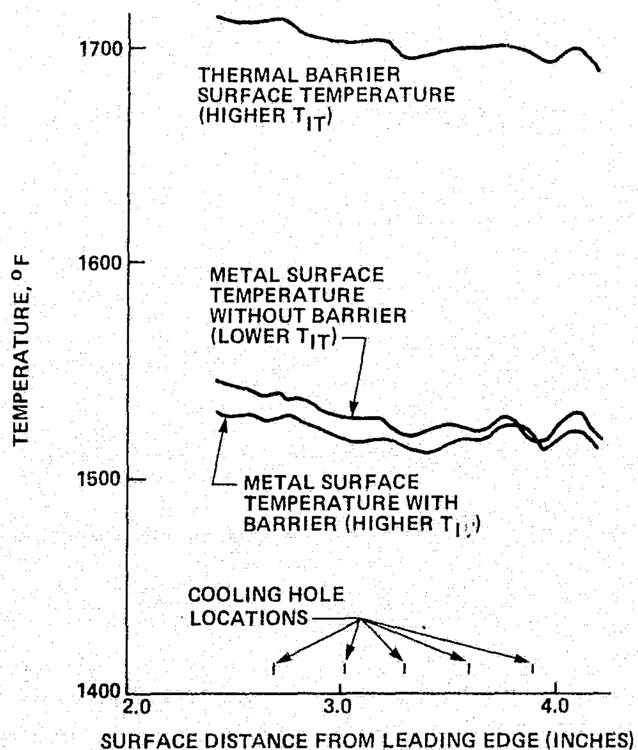
The results of the two-dimensional temperature calculations are illustrated in Figures 20 and 21.

It is seen that comparing coated and uncoated blade temperatures for essentially the same metal surface temperatures, a drop in temperature of approximately 133°K (240°F) is experienced through the leading edge thermal barrier coating system while approximately 100°K (180°F) is experienced due to the lower local flux in the trailing edge region.

The shape of surface temperature distribution between coated and uncoated cases is not greatly changed by the addition of the coating.



BLADE LEADING EDGE SURFACE TEMPERATURE DISTRIBUTION WITH AND WITHOUT THERMAL BARRIER
Figure 20



BLADE TRAILING EDGE SURFACE TEMPERATURE DISTRIBUTION WITH AND WITHOUT THERMAL BARRIER
Figure 21

SECTION 4
ENGINE ANALYSIS

4.1 SELECTION OF GAS TURBINE ENGINES FOR ANALYSIS

The following four gas turbine engine categories have been identified for study:

1. Current Production Type (CPT) Large Industrial Engine — Simple and Combined Cycle

2. Near Term Advanced (NTA) Version of Current Production Engine Upgraded With Thermal Barrier Application — Simple and Combined Cycle (Including Both Existing and Some Redesigned Hardware)

3. Near Term Advanced (NTA) Version of Current Production Engine Upgraded With Thermal Barrier Application — Recuperated Cycle

4. Advanced Design High Temperature Engine — Simple and Combined Cycle

The first three of these options are essentially current production or near production derivatives which could be commercially introduced with modest or minimal development lead time. The fourth engine is a new, advanced design which will entail a much longer development cycle and higher development cost.

CURRENT PRODUCTION TYPE ENGINE

The current production type engine selected for study is the Westinghouse W501D gas turbine. This unit has been described in Section 2.4. The turbine element for this unit utilizes air-cooled turbine hardware by means of convection and impingement/convection techniques.

NEAR TERM ADVANCED VERSION OF CURRENT PRODUCTION TYPE ENGINE UPGRADED WITH THERMAL BARRIER APPLICATION

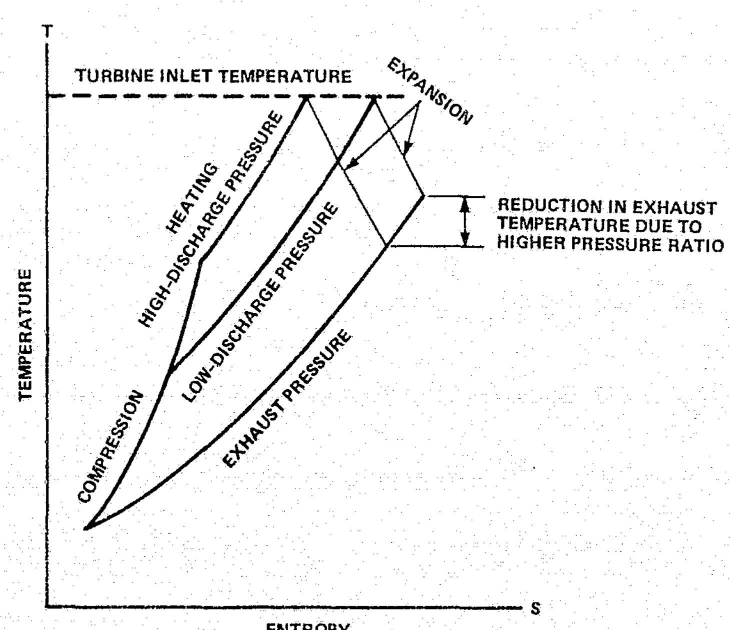
The intent underlying the specification of this alternative was to define an engine derived from a current production configuration without major redesign and which utilized the properties of the thermal barrier coating to the maximum extent possible.

That is, the basic configuration of the above-described Current Production Type Engine including compressor and essential turbine aerodynamic design are to be retained in the upgraded engine.

In this engine, the thermal barrier coating would be applied to cooled and potentially-cooled hardware, and the fundamental turbine inlet temperature limiting item would be hardware not readily amenable to cooling. For this configuration, such an item is the last row turbine rotor blade.

NEAR TERM ADVANCED VERSION OF RECUPERATED CYCLE CURRENT PRODUCTION TYPE ENGINE UPGRADED WITH THERMAL BARRIER APPLICATION

The engine under consideration for this application is essentially the same as that described immediately above. This engine, however, is intended for recuperated cycle application as opposed to simple and combined cycle application. Since the recuperated cycle efficiency generally optimizes at lower compressor pressure ratios for a given turbine inlet temperature than the simple and combined cycles as shown by Figure 5, the turbine element of this engine would be suitably modified by blading angle changes to operate at the required lower pressure ratio. In addition it can be shown thermodynamically (see Figure 22) that for a given turbine inlet temperature, lowered compressor pressure ratio results in higher turbine exhaust temperature (as well as higher temperatures in the latter turbine element sections). This change would in turn dictate that a slightly lower turbine inlet temperature be used in accordance with uncooled last-row turbine blade limitations.



THERMODYNAMIC RELATIONSHIP OF TURBINE INLET TEMPERATURE, TURBINE EXHAUST TEMPERATURE, AND CYCLE PRESSURE RATIO
Figure 22

NEW ADVANCED DESIGN HIGH TEMPERATURE ENGINE

During the ECAS Study (Task II), the conceptual design for an advanced air-cooled, high temperature, gas-turbine engine was prepared. This engine configuration, illustrated by Figure 23, employs a 16:1 pressure ratio single shaft compressor, and advanced turbine section cooling techniques including transpiration methods for certain blading rows. At the outset of the present study, it was determined that this engine would be a candidate for application of the thermal barrier coating in place of the transpiration cooled components. Although the ultimate level of turbine inlet temperature obtainable may be somewhat diminished by such an application, the thermal barrier application may lead to earlier implementation due to simpler and more conventional manufacturing support requirements, and, in addition, may have greater dirty fuel tolerance. For the ECAS study, this engine was configured in a coal gasification combined cycle. The present study considers as well the implementation of this engine in the simple cycle configuration.

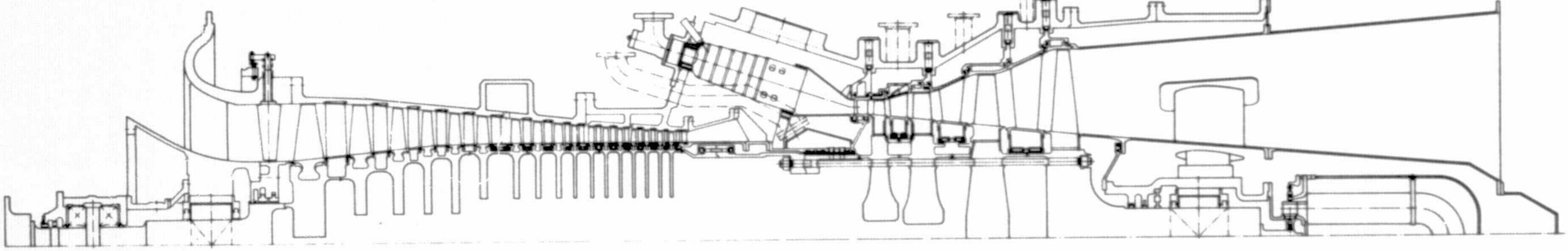
4.2 TURBINE INLET TEMPERATURE DETERMINATION CONSIDERATIONS

After having determined the selection of engines for analysis, the remaining task becomes one of identifying turbine inlet temperature levels compatible with the use of thermal barriers in these engine classes.

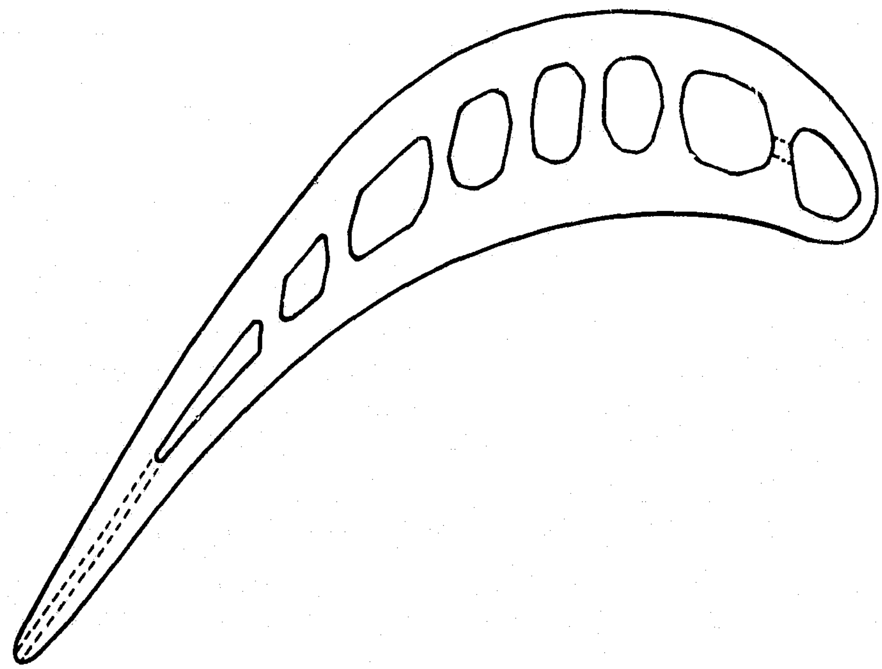
Determination of precise temperature levels and component lifetime consequences is, of course, a function of detailed design and analysis tasks generally associated with engine final design. For the purposes of the present study, however, the scope of preliminary design estimating techniques is more appropriate.

In this context, the procedure has been adopted to compute turbine element performance on a row by row basis, and by this means to determine relative total temperatures at each blade row. This information, in conjunction with known blade stress levels, materials properties and accepted design practice, then is used to compute turbine blade lifetimes and cooling requirements.

After examination of the heat transfer results and the turbine element calculation, it was determined that one of the most effective — and immediate — applications of the thermal barrier coatings would be to apply the coating in the prescribed thickness over current type cooled blading designs and increasing turbine inlet temperature to approximately 1477°K (2200°F). In this manner lengthy redesign of blade cooling systems is minimized and straightforward material substitutions for the required higher temperature capability of uncooled components is applicable. For higher temperature applications, blading cooling system redesigns to incorporate advanced impingement/convection cooling schemes such as that shown in Figure 24, is indicated. Introduction of such systems at turbine inlet temperatures of 1477°K (2200°F) and higher, both with and without thermal barrier coatings have been investigated and the consequent performance improvements are reported in Section 5.



ADVANCED AIR-COOLED HIGH TEMPERATURE TURBINE CONCEPTUAL DESIGN
Figure 23



CONCEPT ADVANCED IMPINGEMENT/CONVECTION-COOLED ROTOR BLADE DESIGN CONCEPT
Figure 24

For the purpose of establishing maximum temperature limits to each of the engine categories, the ground rule has been adopted, as utilized in the earlier ECAS work, that the principal limit to turbine inlet temperature exists with the last rotating turbine blade. This blade row is characterized by large taper ratio and high twist, and is not readily amenable to application of conventional impingement/convection cooling techniques. Therefore, the turbine inlet temperature at which the last stage turbine blade row becomes stress limited defines a basic limit to growth which has been adopted for this work.

The last row blade life working design criterion of 100,000 hours utilizing conventional practice design rules has been utilized in the present analysis. In addition, the properties of several production and near production blade materials has been considered. In particular, the strength-temperature properties of an advanced version of the cast nickel base alloy, IN792, has been selected for use in rotating blade stress analyses. (Although MAR M509, a cobalt base typical stationary vane alloy, was used in the heat transfer analysis, the thermal properties are similar to the nickel based IN792 alloy family.)

These types of calculations have been used to estimate limits on turbine inlet temperature for the upgraded current production engine cases. Specifically, the limit of 1588^{o}K (2400^{o}F) has been determined for the simple-cycle (and combined cycle) version of the upgraded engine, while 1560°K (2350°F) has been determined for the recuperated cycle upgrade engine as a result of lower thermodynamic optimum compressor pressure ratio for the recuperated cycle.

The 1644°K (2500°F) turbine inlet temperature limit for the advanced-design, gas-turbine engine by contrast has been specified by NASA, so that a direct comparison could be made with the transpiration cooled version.

Subsequently, evaluation of the stress levels associated with this design and the local temperature values indicated that with the 16:1 compressor pressure ratio of this engine, the last row blade life would satisfy the 100,000 hour life criterion. Additional turbine inlet temperature growth capability is inherent in this design.

SECTION 5
PERFORMANCE ANALYSIS

5.1 PERFORMANCE CALCULATION APPROACH

After preparing cooling requirement correlations for convective surface cooling, both with and without thermal barrier coatings, this information was subsequently applied to develop estimates for actual turbine element blade cooling requirements. In this endeavor, both current and near-term advanced impingement/convection designs have been considered. The procedure has included: computation of relative total temperature by means of turbine row by row aerodynamic analysis, computation of required blade row cooling effectiveness values and corresponding uncoated blading cooling flow requirements, estimation of cooling air reduction due to thermal barrier addition and preparation of simplified cooling flow networks for the candidate engines. The results of the engine cooling flow analysis indicate that application of the thermal barrier coatings would provide a reduction in overall engine cooling air requirements of from 13 to 14% referenced to uncoated convection/impingement cooling schemes over the range of turbine inlet temperatures from 1366 degrees K (2000 degrees F) to 1477 degrees K (2200 degrees F).

Having cooling flow network tabulations for the candidate engines, performance calculations were then executed for each of the cycle configurations of interest. For these calculations, proprietary Westinghouse Gas Turbine Engineering computer programs have been utilized.

In performing the efficiency calculations, the impact of thermal barrier coating thickness has been treated as follows. The effect of adding the nominal 0.38 mm (0.015 in) coating thickness to blade surfaces, specifically in the trailing edge regions, has a non-negligible effect on the net flow areas through the bladed passages of the turbine element. All other configuration features left unchanged, the overall gas turbine compressor pressure ratio is increased slightly by the addition of the thermal barrier coatings. For example, an uncoated turbine having a nominal compressor pressure ratio of 16.0 would develop a compressor pressure ratio of approximately 16.6 when thermal barrier coatings of the nominal thickness are added. Such a pressure ratio increase could be accommodated by the surge margins available in current type compressor designs and could be incorporated in new advanced turbine designs. In the computations performed for the present study, then, the thickened trailing edges have been accounted for with remaining turbine geometry unchanged, and consequently slight increases in compressor pressure ratio are embodied in the coated calculation cases as compared to the uncoated cases. The results of these comparisons are summarized in the following sections.

5.2 SIMPLE CYCLE PERFORMANCE

5.2.1 Current Production Type (CPT) Gas Turbine Performance Considerations

Current production simple-cycle, peaking gas turbines are characterized by operation with base load or continuous duty turbine inlet temperatures of approximately 1366-1422^{o}K (2000-2100^{o}F). These engines utilize air cooling of vane and blade airfoils in the hottest sections of the turbine element. Computations have been made with current design gas turbines to investigate the effect on performance of applying the thermal barrier coating while retaining the same current turbine inlet temperature level. This work has been performed for two levels of coating thickness. The first zirconia thickness value 0.38 mm (0.015 in) has been analyzed and tested previously by NASA on certain aircraft gas turbine engines. The second value, arbitrarily selected at 0.76 mm (0.030 in), has had limited experimental evaluation.

As shown in Table 3 the improvements have been calculated to be approximately 3/4% improvement in HHV heat rate, and approximately 1-3/4% increase in power output corresponding with the 13% reduction in cooling air usage. For the thicker coating application, these results are summarized in Table 4.

TABLE 3
CPT DESIGN GAS TURBINE PERFORMANCE IMPROVEMENT WITH THERMAL BARRIER COATING (TBC)*, CURRENT TURBINE INLET TEMPERATURE

Percent Change In Cooling Air Usage	Percent Change In Heat Rate	Percent Change In Power Output
ΔG_c	ΔHR	ΔSP
-13.3%	-0.78%	+1.8%

*In this and subsequent tables, Thermal Barrier Coating (TBC) refers to the nominal 0.38 mm (0.015 in) thickness coating. The only exception, a double thickness coating is referred to explicitly in Table 4.

TABLE 4
CPT DESIGN GAS TURBINE PERFORMANCE IMPROVEMENT WITH 0.76 MM (0.030 IN) THERMAL BARRIER COATING, CURRENT TURBINE INLET TEMPERATURE

Pecent Change In Cooling Air Usage	Percent Change In Heat Rate	Percent Change In Power Output
ΔG_c	ΔHR	ΔSP
-18.6%	-1.80%	+3.1%

5.2.2 Near Term Advanced (NTA) Engine Performance Consideration

In this and the following sections the descriptive term, near term advanced (NTA) engine, has been used to describe that engine updated from current type designs by the application of thermal barrier coatings, advanced blade cooling, and materials improvements. This is used to distinguish such a gas turbine from an essentially new design approach high temperature gas turbine.

As shown by the heat transfer analysis results, application of the nominal 0.38 mm (0.015 in) thermal barrier coating has indicated a coating temperature drop of from 100°K (180°F) to 133°K (240°F) when used in conjunction with a particular example of a highly cooled current design gas turbine blade row. In accordance, for the near term advanced gas turbine engine upgraded by the application of thermal barrier coatings a nominal turbine inlet temperature of 1477°K (2200°F) has been selected. Performance calculations have been made which compare the results of current production type design with uncoated blading operating at current turbine inlet temperatures to the same design with coated blading operating at 1477°K (2200°F). These results, shown in Table 5, indicate that by application of the barrier coating and increasing turbine inlet temperature accordingly simple cycle heat rate can be improved by approximately 1-1/2% (equivalent to approximately 175 Btu/kW-Hr higher heating value, or 1/2 point on overall thermodynamic efficiency), and output can be increased by approximately 10%.

TABLE 5
SIMPLE CYCLE PERFORMANCE IMPROVEMENT OBTAINED BY ADDING TBC AND UPRATING TO 1477°K (2200°F) TURBINE INLET TEMPERATURE

ΔG_c	ΔHR	ΔSP
+9.8%	-1.52%	+9.74%

Calculations have also been made of the effect of applying thermal barrier coatings to a cooling design suitable for 1477°K (2200°F) turbine inlet temperature. For this case, it is necessary to introduce advanced cooling system design changes, including advanced impingement/convection blade cooling schemes, for satisfactory operation with uncoated blading at 1477°K (2200°F). The performance improvement associated with introduction of thermal barriers for this case is shown in Table 6.

TABLE 6
NTA SIMPLE CYCLE GAS TURBINE PERFORMANCE IMPROVEMENT WITH TBC, TURBINE INLET TEMPERATURE 1477°K (2200°F)

ΔG_c	ΔHR	ΔSP
-13.9%	-0.72%	+2.1%

Thus, the percentage gains in efficiency and power due to simple thermal barrier addition and cooling flow reduction are nearly the same at the 1477°K (2200°F) level as at the 1366°K (2000°F) level turbine inlet temperature.

The case of performance improvement potential achievable with the near term advanced engine when operated at higher turbine inlet temperature has also been evaluated. In this case the turbine inlet temperature has been set at that level, 1588°K (2400°F), for which the uncooled last row turbine blade is the temperature limiting element. The most significant change in performance is the additional gas turbine specific power output obtained with the higher turbine inlet temperature as indicated below.

TABLE 7
NTA SIMPLE CYCLE GAS TURBINE PERFORMANCE IMPROVEMENT WITH TBC AT TURBINE INLET TEMPERATURE 1588°K (2400°F) COMPARED WITH UNCOATED CURRENT TYPE DESIGN ENGINE AT CURRENT TURBINE INLET TEMPERATURE LEVEL

ΔG_c	ΔHR	ΔSP
+45%	-0.61%	+19.0%

Much more cooling air is required for this turbine inlet temperature resulting in about the same heat rate and higher specific power compared to the current temperature.

Although the advantage in simple-cycle heat rate is small, the improvement in specific power output, a parameter closely related to combined cycle efficiency, is more important as will be shown in a subsequent discussion of combined cycle performance.

5.2.3 Summary Simple Cycle Performance Comparison

An overall comparison of the impact of thermal barrier coatings on simple cycle gas turbine performance is indicated by Figure 25. Four distinct steps in a potential application of thermal barrier coatings are illustrated, and they include both addition of the thermal barrier coatings to a given design as well as uprating blading cooling designs and substitution of improved materials. Specifically, the steps include:

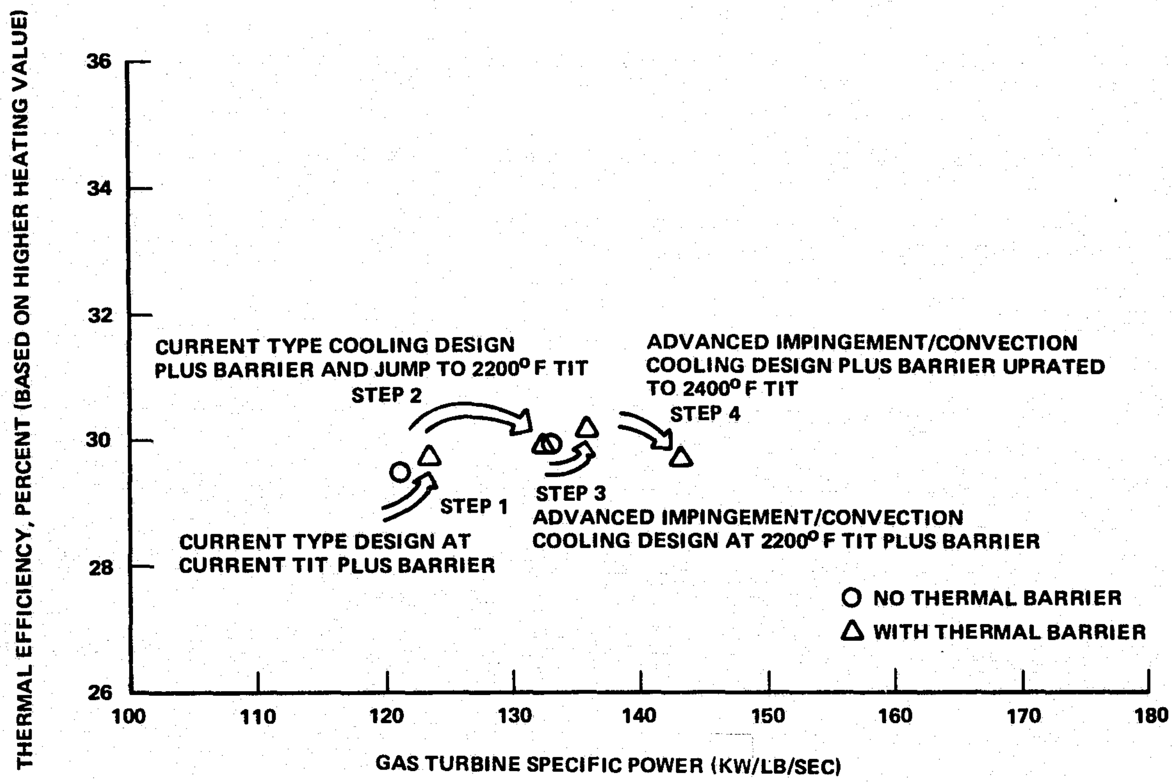
- Step 1 — Addition of Thermal Barrier to Current Type Cooling Design at Current T_{IT}.
- Step 2 — Addition of Thermal Barrier to Current Type Cooling Design and Jump to 1477°K (2200°F) T_{IT}.
- Step 3 — Addition of Thermal Barrier to Advanced Impingement/Convection Cooling Design at 1477°K (2200°F) T_{IT}.
- Step 4 — Addition of Thermal Barrier to Advanced Impingement/Convection Cooling Design and Jump to 1588°K (2400°F).

The first step reflects the addition of thermal barrier coatings to a gas turbine with current type cooling technology while maintaining a given turbine inlet temperature. Most of the improvement in efficiency comes from the reduction in cooling air usage and a slight increase in cycle pressure ratio due to thicker trailing edges on coated blades.

The second step reflects the current type of cooling with the addition of the thermal barrier and a jump in turbine inlet temperature to 1477°K (2200°F). The most significant change is seen to occur in gas turbine specific output. This results primarily from the increased turbine inlet temperature and it can be shown this is a major contributor to the improvement of combined cycle performance.

The third step is similar to step 1 in that the thermal barrier is applied while turbine inlet temperature is held constant at 1477°K (2200°F). For the uncoated case at 1477°K (2200°F) turbine inlet temperature an advanced impingement/convection blade cooling system will be required, and it is important to note that the resultant efficiency for this case is almost identical to that for the case with current type cooling system design plus the thermal barrier and operating at the 1477°K (2200°F) turbine inlet temperature level.

The fourth step reflects the further temperature uprate of the 0.38 mm (0.015 in) thermal barrier coated engine case incorporating the advanced impingement/convection cooling approach. The uprate turbine inlet temperature has been set at 1588°K (2400°F) and is reflected in the significant gain in specific power output although simple-cycle efficiency is somewhat diminished due to the amount of cooling air



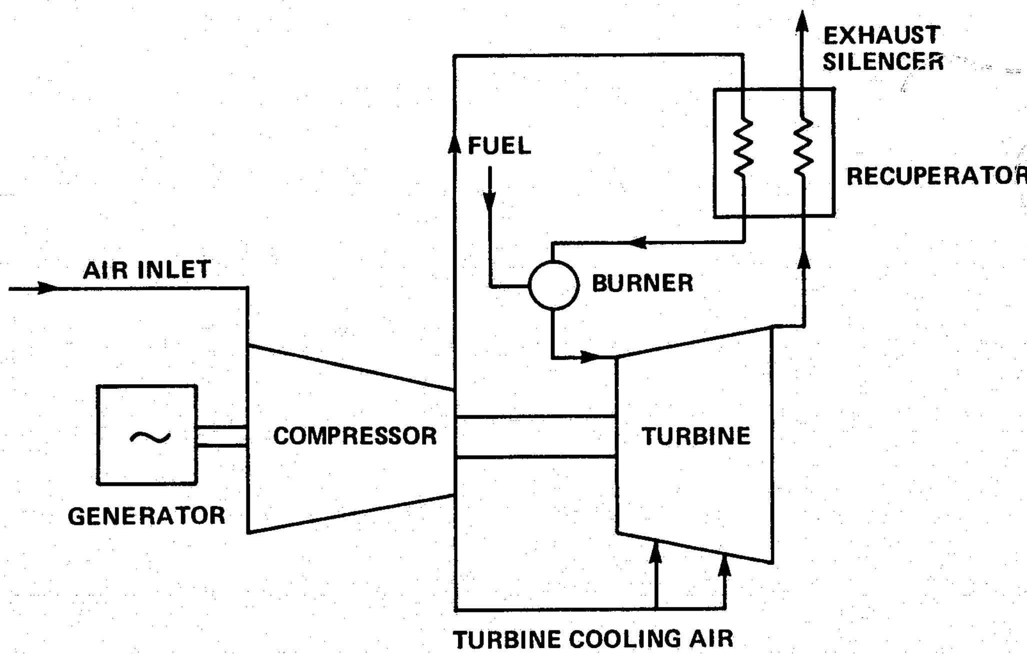
EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON SIMPLE CYCLE EFFICIENCY
Figure 25

requirements. It is this gain in specific power output that will prove to be most important in achieving highest combined cycle efficiency. Improved specific power output also helps reduce capital cost for both the simple and combined cycle power plants.

5.3 RECUPERATED CYCLE PERFORMANCE

5.3.1 Near Term Advanced Recuperated Cycle Gas Turbine Performance Considerations

For the recuperated cycle version of the near term advanced gas turbine engine, a turbine inlet temperature of 1477°K (2200°F) has been selected as in the case of the simple cycle analysis. The recuperated version as well is intended to utilize the same basic turbomachinery hardware including compressor, combustion system, and turbine element. The schematic cycle arrangement for this case is shown in Figure 26. The cycle has been calculated with a recuperator effectiveness of 0.827 as in the Task I baseline case of the ECAS study. Compressor intercooling has not been utilized. A small reduction of overall compressor pressure ratio (from approximately 13:1 to approximately 12:1) has been achieved relative to the simple cycle version by means of minor turbine geometry adjustments to more nearly optimize cycle efficiency while still maintaining satisfactory temperatures relative to critical turbine blading.



RECUPERATED CYCLE SCHEMATIC ARRANGEMENT
Figure 26

Calculations have been run for comparison of performance with and without the thermal barrier at the 1477°K (2200°F) temperature level. The results are shown in Table 8.

TABLE 8
RECUPERATED CYCLE NTA GAS TURBINE PERFORMANCE IMPROVEMENT WITH TBC, TURBINE INLET TEMPERATURE 1477°K (2200°F)

ΔG_c	ΔHR	ΔSP
-13.9%	-0.74%	+2.1%

It is seen that the performance improvement results are thus quite similar for the recuperated cycle as in the case of the simple-cycle barrier addition at the 1477°K (2200°F) level. The effect of adding the thermal barrier coating to a recuperated cycle gas turbine with current type cooling design and uprating from current turbine inlet temperatures to the 1477°K (2200°F) level has been evaluated as in the case of the simple-cycle engine. The results of this calculation are shown below.

TABLE 9

RECUPERATED CYCLE PERFORMANCE IMPROVEMENT OBTAINED BY ADDING TBC AND UPRATING TO 1477°K (2200°F) TURBINE INLET TEMPERATURE (COMPARED WITH NON-BARRIER COATED RECUPERATED CURRENT TYPE DESIGN AT CURRENT T_{IT})

ΔG_c	ΔHR	ΔSP
+9.8%	-2.3%	+9.4%

Thus the performance improvements obtained by adding the thermal barrier to a current type cooling system and uprating to the 1477°K (2200°F) turbine inlet temperature are more significant than those obtained by adding the thermal barrier to the near term engine operating at the 1477°K (2200°F) level.

The potential for performance improvement by further turbine inlet temperature increase has also been considered. In this case the engine analysis indicated a last-row blade limit of approximately 1560°K (2350°F) on turbine inlet temperature (slightly lower than the comparable simple-cycle case figure due to slightly lower compressor pressure ratio). This case was compared with a recuperated cycle version of a current type design, uncoated and operating at current level turbine inlet temperature, and the following comparison results.

TABLE 10

RECUPERATED CYCLE NTA GAS TURBINE PERFORMANCE IMPROVEMENT WITH TBC AT TURBINE INLET TEMPERATURE 1560°K (2350°F) COMPARED WITH RECUPERATED CYCLE UNCOATED CURRENT TYPE DESIGN ENGINE AT CURRENT TURBINE INLET TEMPERATURE LEVEL

ΔG_c	ΔHR	ΔSP
+32.6%	-3.4%	+17.3%

5.3.2 Summary Recuperated Cycle Performance Comparison

Figure 27 illustrates the overall comparison of the impact of thermal barrier coating application and increased turbine inlet temperature upon the performance of the recuperated cycle version of the near-term advanced gas turbine engine. As with the simple-cycle cases a stepwise progression for a potential thermal barrier application and turbine inlet temperature uprating program is indicated. Based on the ECAS Task 1 parametric analysis results, however, it was elected to concentrate evaluation of the recuperated cycle with thermal barrier coatings at the 1477°K (2200°F) and higher turbine inlet temperature levels. Therefore the stepwise progression is not exactly the same as that of the simple cycle case. The steps are instead:

- Step 1 - Addition of Thermal Barrier to Current Type Cooling Design and Jump to 1477°K (2200°F) T_{IT}

- Step 2 - Addition of Thermal Barrier to Advanced Impingement/Convection Cooling Design at 1477°K (2200°F) T_{IT}

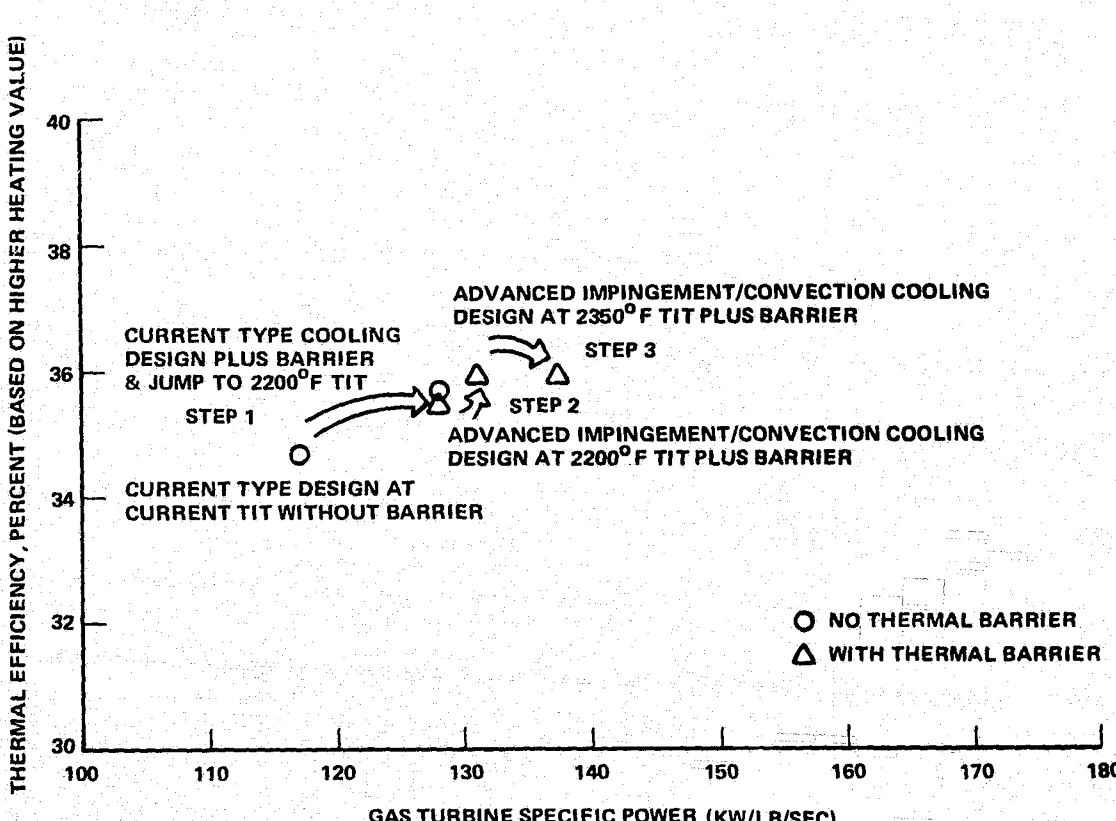
- Step 3 - Addition of Thermal Barrier to Advanced Impingement/Convection Cooling Design and Jump to 1560°K (2350°F) T_{IT}

In the context of the overall progression, the first step appears to be quite attractive as shown by Figure 27.

5.4 COMBINED CYCLE PERFORMANCE

5.4.1 Current Production Gas Turbine Performance Considerations

Combined-cycle systems currently being produced are characterized by the application of the low-cost, highly-packaged approach successfully developed for the simple-cycle peaking gas turbine market. These cycles were introduced about 1970 in response to the need for liquid and gas fueled intermediate duty (intermediate between base load and peaking duty) power generation capability. The typical cycle utilizes current



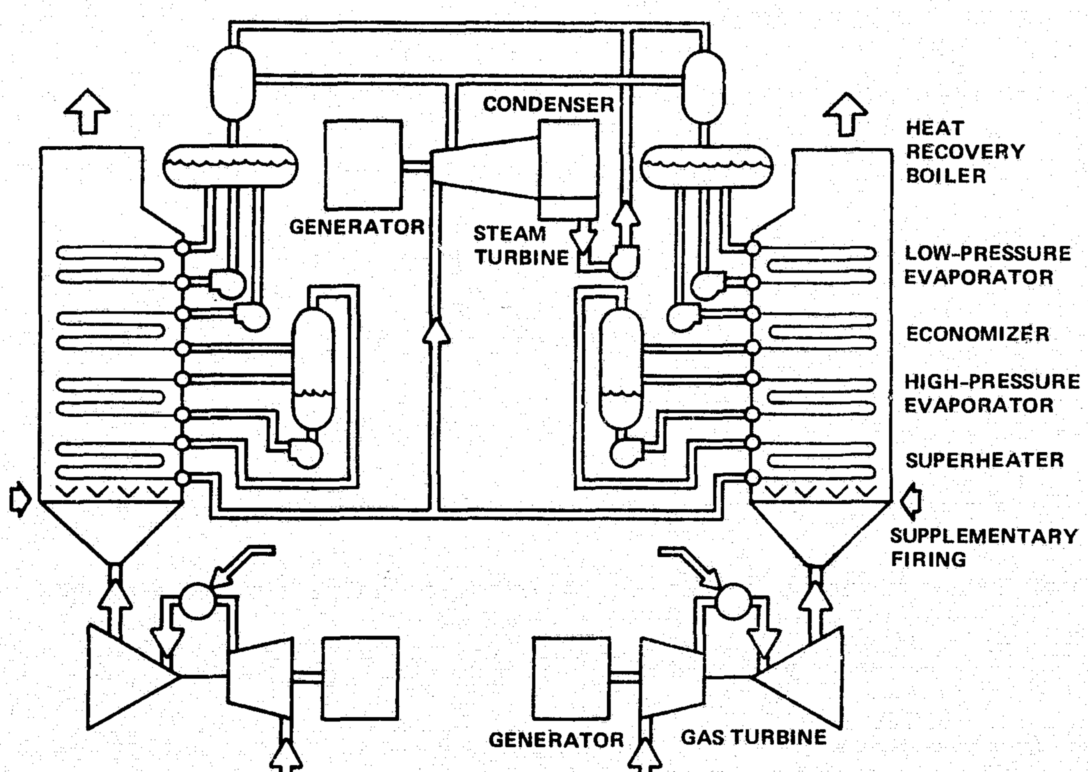
EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON RECUPERATED CYCLE EFFICIENCY
Figure 27

design simple-cycle peaking gas turbines of the type described in Section 4.1. Modularized heat recovery steam generators are utilized to recover exhaust heat and generate steam for a non-reheat steam turbine, usually with 8620 KPa, 783°K (1250° PSIG, 950°F) throttle steam conditions. A typical cycle arrangement diagram for such a combined cycle is shown in Figure 28. This particular arrangement is used in the current production and near-term advanced, combined cycles considered below.

When the gas turbine cooling air savings associated with the application of thermal barriers to current production gas turbines are considered, the resultant improvement in combined-cycle performance is obtained as summarized in Table 11.

TABLE 11
COMBINED CYCLE PERFORMANCE IMPROVEMENT; CPT GAS TURBINE WITH TBC AT CURRENT TURBINE INLET TEMPERATURE LEVEL

ΔG_c	ΔHR	ΔSP
-13.3%	-0.57%	+1.1%



CURRENT TYPE DESIGN COMBINED CYCLE SCHEMATIC ARRANGEMENT
Figure 28

5.4.2 Near Term Advanced Engine Combined Cycle Performance Considerations

The gas turbine inlet temperature, 1477°K (2200°F), has been identified for the near-term advanced combined-cycle performance as in the cases of the near-term advanced simple and recuperated cycles. The combined-cycle configuration has been maintained the same as in the current production combined cycle cases. This configuration employs supplementary heat recovery steam generator firing. Therefore, with the elevated gas turbine inlet temperature of the near term advanced gas turbine engine, turbine exhaust temperature is correspondingly increased and the need for supplementary firing is reduced.

Calculations have been made comparing the performance of the present design combined cycle configuration at current turbine inlet temperature and the same design operating, with the addition of the thermal barriers, at the turbine inlet temperature of 1477°K (2200°F). The performance improvement results are summarized in Table 12.

TABLE 12
COMBINED CYCLE PERFORMANCE IMPROVEMENT OBTAINED BY ADDING TBC AND UPRATING TO 1477°K (2200°F) TURBINE INLET TEMPERATURE

ΔG_c	ΔHR	ΔSP
+9.8%	-2.80%	+5.85%

The performance improvement is thus seen to be significantly greater with respect to heat rate in the combined-cycle uprate as compared to the simple-cycle uprate. (Refer to Table 5.) Specifically, the combined-cycle performance has been improved by 2.80% as compared to a corresponding improvement of 1.52% in heat rate for the simple-cycle case.

Independent studies at Westinghouse have identified the importance of simple-cycle gas turbine specific power (unit output divided by unit airflow input) as it relates to combined cycle efficiency. It has been determined that gas turbine specific power is the most important single parameter affecting combined cycle efficiency with combined cycle efficiency highest when gas turbine specific power is maximized. As shown by Figure 5, simple-cycle specific power is increased dramatically as turbine inlet temperature is increased, and most importantly, optimum combined cycle efficiency is obtained at specific power maxima. Thus, for a given increase in turbine inlet temperature and concommitant increase in specific power, even though simple cycle efficiency may not be greatly improved, combined cycle efficiency is significantly increased. From another viewpoint, it follows also that at any given turbine inlet temperature, optimizing simple cycle efficiency by increasing compressor pressure ratio does not necessarily maximize combined cycle efficiency.

Calculations have also been run with the gas turbine inlet temperature set at 1477°K (2200°F) with and without the addition of the thermal barrier coating. This requires, of course, that the gas turbine cooling system design be suitably uprated from current type design practice.

An advanced impingement/convection cooling system has been incorporated (as in previous simple and recuperated near-term advanced cycles) for the following comparison.

TABLE 13
COMBINED CYCLE NTA GAS TURBINE PERFORMANCE IMPROVEMENT WITH ADDITION OF TBC AT TURBINE INLET TEMPERATURE 1477°K (2200°F)

ΔG_c	ΔHR	ΔSP
+13.9%	-0.76%	+1.41%

In comparison with the corresponding simple cycle tabulation (Table 6), the gain in heat rate is slightly larger for the combined cycle case, but power output gain is lower. (Approximately proportional to the ratio of gas turbine power to steam turbine power.)

To assess further performance improvement potential, the combined cycle case directly corresponding to the temperature uprated near-term simple-cycle gas turbine at 1588°K (2400°F) turbine inlet temperature has been evaluated. As with the previous current production and near-term advanced combined-cycle cases, the current type 8620 kPa 783°K (1250 PSIG, 950°F) non-reheat bottoming steam turbine cycle has been retained. The performance improvement potential as compared with the current production type combined-cycle with uncoated engine components is shown below.

TABLE 14
COMBINED CYCLE PERFORMANCE IMPROVEMENT UTILIZING NTA GAS TURBINE WITH TBC AT TURBINE INLET TEMPERATURE 1588°K (2400°F) COMPARED WITH UNCOATED CURRENT TYPE DESIGN ENGINE AT CURRENT TURBINE INLET TEMPERATURE LEVEL

ΔG_c	ΔHR	ΔSP
+45.1	-5.40%	+11.8%

The heat rate improvement as indicated above represents a most significant change. In terms of efficiency it represents an advancement of nearly 2.5 efficiency points (or almost 450 Btu/kW-Hr) betterment over current combined-cycle designs (offered at HHV heat rates of about 8200 Btu/kW-Hr.)

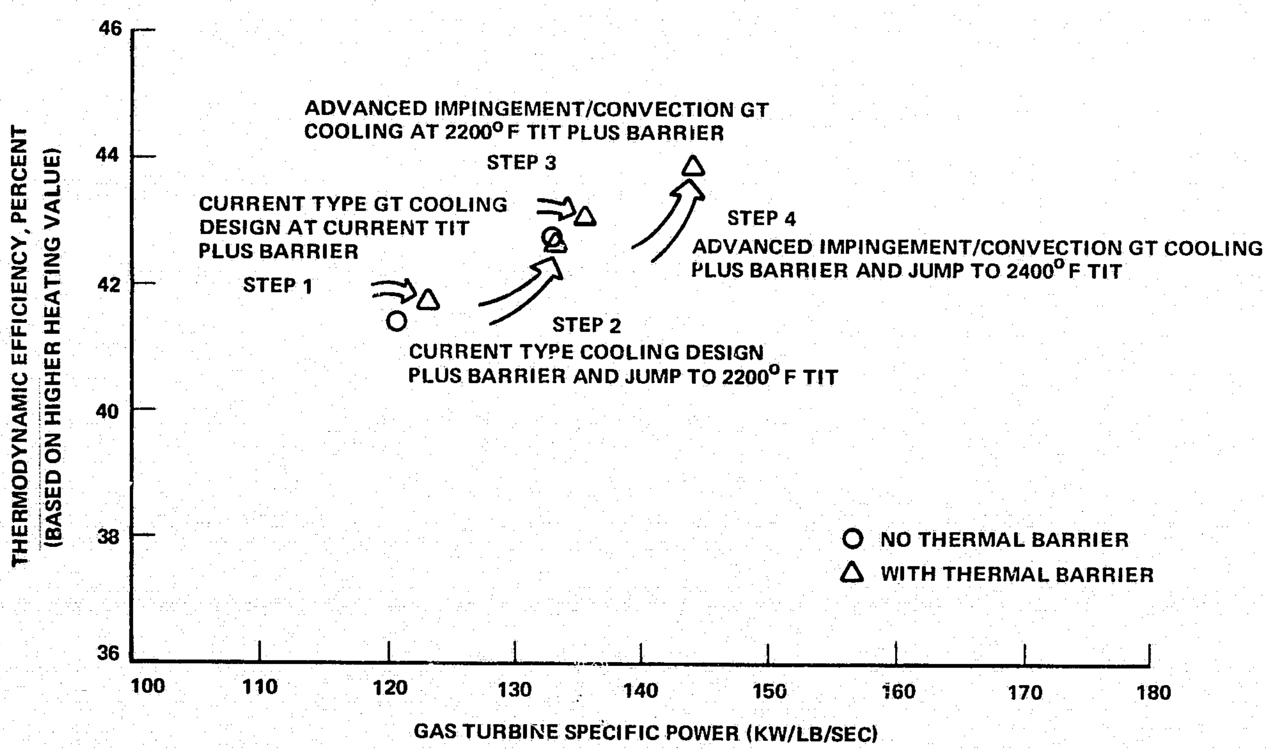
5.4.3 Summary Combined Cycle Performance Comparison

A graphical comparison of the combined cycle performance results including current production, and near-term advanced engine designs is given in Figure 29.

The basic relationship of the combined cycle improvements with respect to turbine inlet temperature increments and introduction of thermal barrier coatings closely parallels that for the simple-cycle cases. The four steps highlighted in the figure correspond to:

- Step 1 - Application of thermal barrier coatings to current design engines in combined-cycle systems.

- Step 2 - Application of thermal barrier coatings to current type gas turbine cooling system designs and jumping turbine inlet temperature to 1477°K (2200°F). (Same combined cycle arrangement as with current production combined cycle plants.)



EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON COMBINED CYCLE EFFICIENCY
Figure 29

- Step 3 - Application of thermal barrier coatings to near-term advanced gas turbine engines with cooling systems capable of uncoated operation to 1477°K (2200°F). (Same combined cycle arrangement as with current production type combined cycle plants.)

- Step 4 - Temperature uprate to 1588°K (2400°F) of the thermal barrier coated near-term advanced gas turbine engine combined cycle. (Same combined-cycle arrangement as with current production type combined-cycle plants.)

A foremost point to note from Figure 29 in comparison with corresponding temperature increment uprates of simple and recuperated cycles is that the gains in cycle efficiency are greatest in the case of the combined cycles.

5.5 ADVANCED ENGINE CYCLE PERFORMANCE

5.5.1 Advanced Gas Turbine Engine Simple-Cycle Performance Considerations

The advanced gas turbine engine design considered in this study represents a new, conceptual approach to high-temperature power generation gas turbine technology. The design was prepared during the Concept Design Task II work of the ECAS study by the Westinghouse Electric Corporation. This design differs from the current production, and near-term advanced gas turbine engines in that compressor pressure ratio has been nominally set at 16:1 and as originally conceived the design utilized transpiration cooling of the initial turbine element stage.

As presently formulated and applied, the thermal barrier coating is not recommended for application to transpiration-cooled blading. Therefore, for the present study, the use of advanced impingement/convection blade cooling schemes has been selected to be used in conjunction with the thermal barrier coating.

One approach to the evaluation of this application of thermal barrier coatings is to compare the performance of this advanced design gas turbine engine (with advanced impingement/convection cooling blading designs) against that of near-term advanced gas turbine designs with thermal barrier coatings. In each case, the approach to blade cooling will be similar although the basic engine aerodynamic and cycle parameter designs were originally optimized for different temperature and pressure ratio levels. Table 15 illustrates the performance level of near-term gas turbine engines with thermal barrier compared to that of the advanced design engine utilizing the thermal barrier coatings.

TABLE 15

ADVANCED DESIGN (ECAS II) SIMPLE CYCLE GAS TURBINE PERFORMANCE WITH TBC COMPARED WITH NEAR-TERM GAS TURBINE ENGINES WITH TBC

Engine	Turbine Inlet Temperature (°F)	Compressor Pressure Ratio	ΔHR	ΔSP
Near-Term Advanced with Current Type Cooling	2200	13.0	Base Value	Base Value
Near-Term Advanced with Adv. Imp./Conv. Cooling	2200	13.1	0.8%	2.2%
Near-Term Advanced with Adv. Imp./Conv. Cooling	2400	13.1	-0.9%	8.4%
Advanced Design (ECAS II)	2500	16.6	9.1%	27.6%

(A qualitative analysis of the relative positions of the near-term advanced engine results has been given in Section 5.2.3 earlier.)

It is most important to interpret these comparisons as a reflection of the potential for application of the thermal barrier to advanced design engines. While the reported impracticality of coating transpiration-cooled hardware precludes a direct one-on-one comparison of non-coated and coated designs at the 1644°K (2500°F) level, the above comparisons indicate that significant gains in performance are potentially available in an advanced engine development program which would incorporate thermal barrier coatings. It should be cautioned, however, that the ECAS II design is a conceptual design (drafting board engine) while the other engines in the comparison are based on an existing design. Therefore, some uncertainty exists in the ECAS II results by comparison.

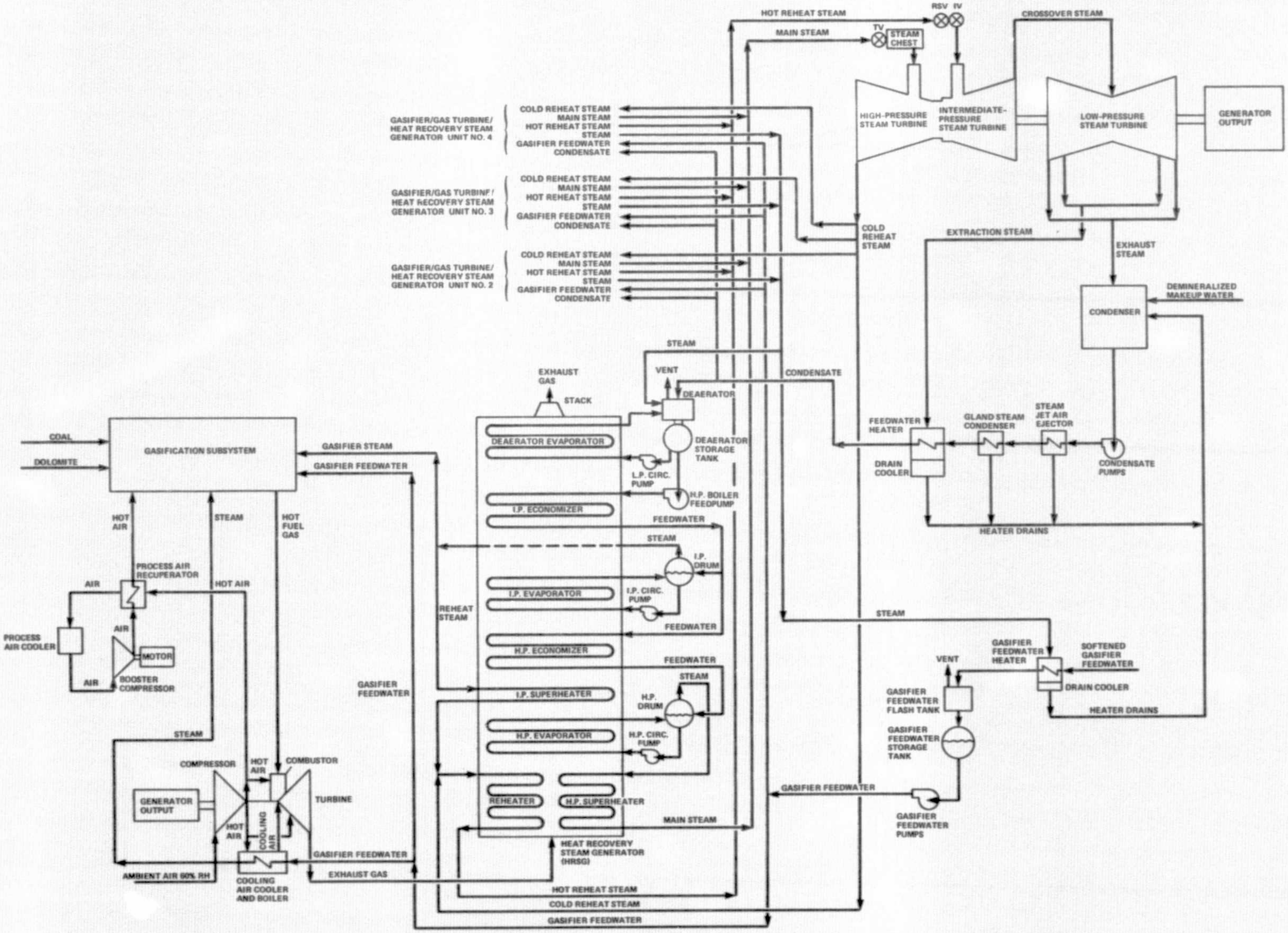
5.5.2 Advanced Gas Turbine Engine Combined Cycle Performance Considerations

In addition to the current production and near-term advanced engine uprated gas turbines, one combined-cycle case was considered with the new design advanced gas turbine engine, operating with the use of thermal barriers at high temperature. This new design, essentially the same engine considered for the advanced engine simple-cycle case, is utilized in the present combined-cycle application at a turbine inlet temperature of 1644°K (2500°F) and a nominal compressor pressure ratio of 16:1.

The combined-cycle arrangement for this case differs in two important ways from the cycles evaluated in the current production and advanced near-term engine cases. First, this cycle utilizes gasified coal fuel as opposed to the liquid petroleum-based fuels considered earlier. This arrangement, which is identical to that developed for the ECAS Task II gasification combined-cycle conceptual design, includes a pressurized fluidized bed gasifier with in-bed desulfurization (Hot Gas Clean Up). Second, the combined-cycle arrangement is built around the use of a single reheat bottoming steam turbine with 16,550 kPa, 810^{o}K (2400 PSIG, 1000^{o}F) throttle steam conditions and a single reheat to 810^{o}K (1000^{o}F). This particular cycle arrangement is shown in Figure 30.

In comparison of the present cycle with the ECAS Task II conceptual design cycle, however, only one essential difference is evident: the approach to gas turbine blade cooling technology. For the ECAS study a necessarily concise estimate of the cooling air requirements with the transpiration approach had been made. For the present study, a similar approach to estimating the cooling air requirements for the same engine at the same conditions but using thermal barrier coating advanced impingement/convection cooling techniques has been made. The results have shown the cooling total air requirements to be within 3% of each other; essentially equivalent within the context of the advanced engine concept design status at the present stage of development. As a result the efficiency values for the two advanced design gasification combined cycles are nearly identical (to within 50 Btu/kW-Hr). It has been estimated that this cycle utilizing thermal barrier coated gas turbine blading would generate electricity at a net thermal efficiency of 47% (7250 Btu/kW-Hr) coal pile to bus bar.

The results are most significant in evaluating the overall position of the thermal barrier vis-a-vis transpiration cooling for the advanced air-cooled combined cycle at the turbine inlet temperature level under consideration. The thermal barrier would appear to offer an alternative method to transpiration cooling for achieving such turbine inlet temperatures with little or no sacrifice in overall efficiency and with probably greater tolerance to particulate carryover from the pressurized fluid bed gasifier.



HEAT BALANCE DIAGRAM FOR THE ADVANCED COMBINED CYCLE GAS TURBINE SYSTEM WITH AN INTEGRATED LOW-BTU GASIFIER

Figure 30

SECTION 6
CAPITAL COST AND COST OF ELECTRICITY ANALYSIS AND RESULTS

6.1 CAPITAL COST ESTIMATION AND COST OF ELECTRICITY CALCULATION APPROACH

The basic approach to capital cost estimation and cost of electricity calculation has been patterned after the procedure developed for the ECAS Task I study (Reference 9.9). That is, the complete conceptual design power plants including site configuration details have been adopted and modified where necessary to reflect the addition of the thermal barrier coatings to gas turbine blading.

Briefly reviewed, the major features of the plant configurations are as follows:

1. The Middletown, USA site described in WASH 1230 (Reference 9.14) modified for a coal fired fossil plant (UEC-AEC-720630) is being used for all combined cycle plants.

2. As in the ECAS work, an industrial site located on the outskirts of a large city near a commercial or industrial area has been selected for the simple and recuperated cycle gas turbine systems.

3. A four-gas turbine configuration has been utilized for the simple and recuperated gas turbine cycles giving a nominal plant output of approximately 400 MW. A conceptual plot plan arrangement for this plant is shown in Figure 31.

4. All liquid fuel combined cycle plants employ a configuration of two gas turbines bottomed by a simple non-reheat steam turbine with 8620 kPa, 783^{o}K (1250 psig, 950^{o}F) throttle steam conditions. Figure 32 illustrates a conceptual plant island arrangement for this type combined cycle plant which for most of the cases investigated would have a nominal output of about 300 MW.

5. For the coal gasification combined cycle plant under consideration, a four-gas turbine, single-reheat steam turbine configuration — 16550 kPa, 810^{o}K/810^{o}K (2400 psig, 1000^{o}F/1000^{o}F) — has been used. Nominal plant output approximates 800 MW.

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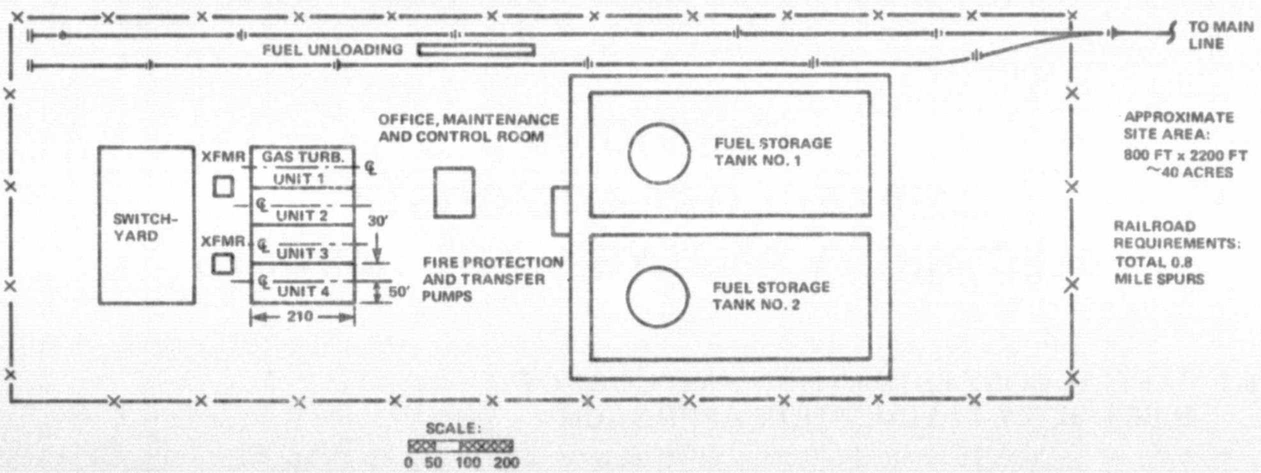
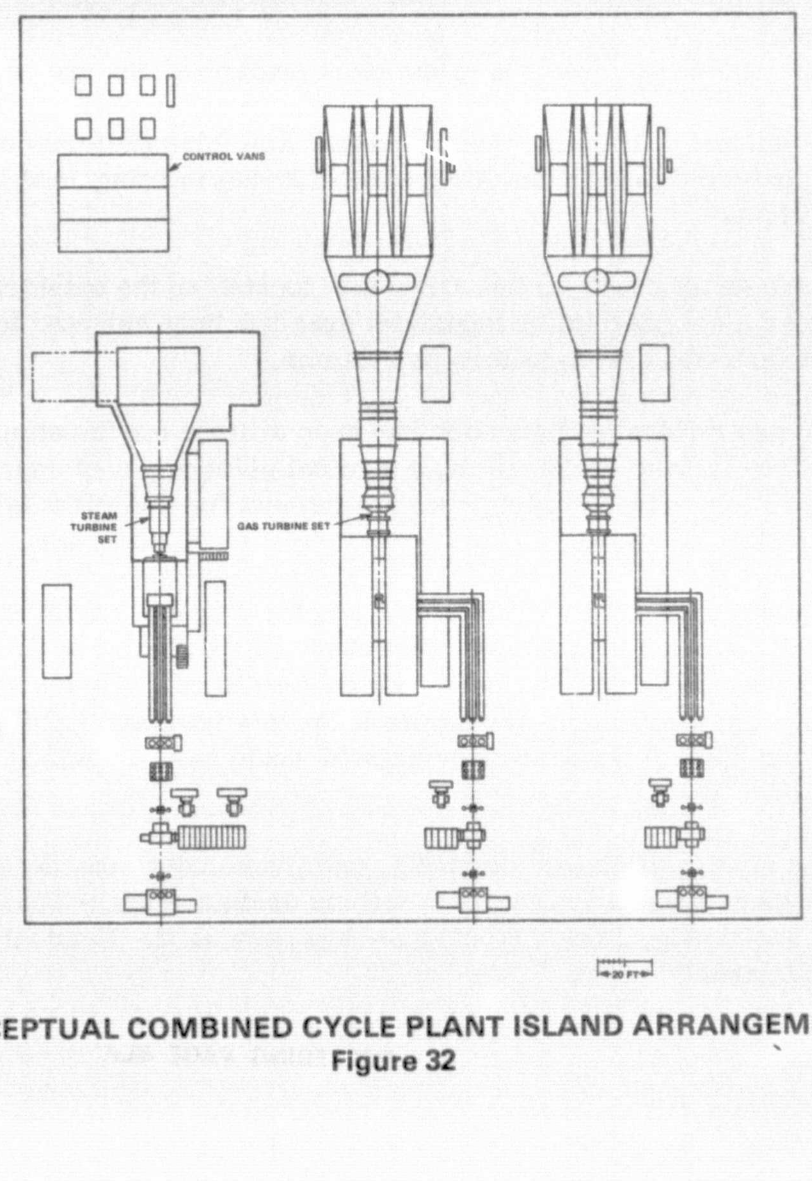


CONCEPTUAL PLOT PLAN ARRANGEMENT FOR 400 MW SIMPLE CYCLE PLANT
Figure 31



CONCEPTUAL COMBINED CYCLE PLANT ISLAND ARRANGEMENT
Figure 32

6. Economic parameters employed in this study include:

   - Annual escalation rate-6.5%
   - Base interest during construction rate-10%
   - Fixed charge rate - 18%

7. Balance of Plant costs, indirect construction costs, professional and ownership costs, and contingency costs are based on the approach developed by C. T. Main, Inc. for the ECAS Task I study.

8. Fuel Costs:

   \$2.46/MJ (\$2.60/10^6 Btu) for distillate fuel oil

   \$2.03/MJ (\$2.15/10^6 Btu) for residual fuel oil

   \$0.95/MJ (\$1.00/10^6 Btu) for Illinois No. 6 bituminous coal

   have been specified by NASA.

9. Thermal barrier stripping and coating costs:

   \$100/individual combustor liner

   \$200/individual combustor-turbine transition piece

   \$150/vane or blade airfoil

   have been supplied by NASA.

10. Mid 1975 estimates of turbomachinery prices have been developed by Westinghouse Generation Systems Division.

Having defined plant capital costs, fuel costs, and related indirect costs, etc., the overall cost of electricity has been computed by the method described in the ECAS Task I report Volume I Section 2.7 (Reference 9.9).

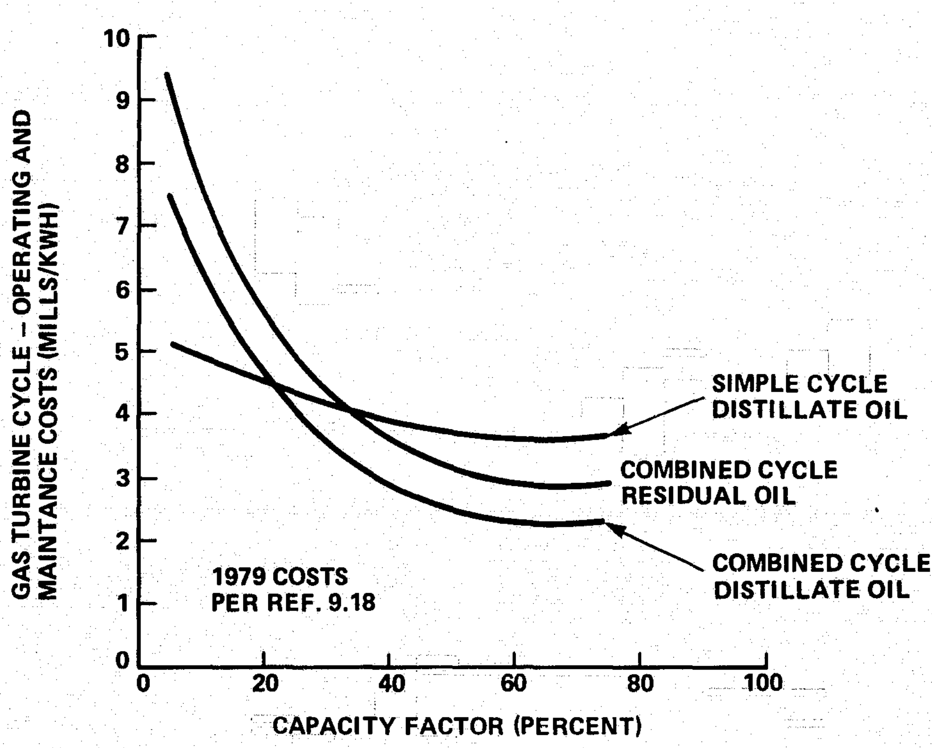
6.2 OPERATING AND MAINTENANCE COST CONSIDERATIONS

6.2.1 Overall Approach

The basic thrust in the calculation of cost of electricity for this study has been to determine the impact of improved performance made possible by the introduction of thermal barrier coatings and the attendant increase in turbine inlet temperature made possible thereby. This change is primarily associated with a reduction in the fuel cost portion of overall cost of electricity. For gas turbine type plants the fuel cost portion is usually the largest constituent of the three comprising overall cost of electricity: capital cost, fuel cost, and operating and maintenance cost. The operating and maintenance cost portion, although the smallest of the three, is unfortunately the least well known.

Gas turbine powerplant operating and maintenance costs, although of considerable interest to utilities, regulatory agencies and manufacturers, have only recently been compiled and presented in comprehensive form. The first annual Federal Power Commission report, "Gas Turbine Electric Plant Construction Cost and Annual Production Expenses" covers the year 1972 (Reference 9.15). In this and subsequent widely referenced FPC reports the vast majority of gas turbine power plants are operated in the simple cycle mode primarily for peak power generation duty. Interpretations of the data vary, from the position that the heavy duty industrial type gas turbines exhibit markedly lower operating and maintenance costs than aircraft derivative type (Reference 9.16) to the contention that no such conclusion can be drawn from the published data (Reference 9.17).

For the present study the operating and maintenance cost model presented in a comparative evaluation of several advanced generation systems has been adopted (Reference 9.18). The basic reason for selecting this particular study for reference is that it represents a viewpoint on advanced cycles of a power generation equipment <u>user</u> with considerable gas turbine experience and therefore should reflect an overall balanced viewpoint on operating and maintenance costs. The operating and maintenance cost values used in this referenced study, which incorporated gas turbine systems burning both distillate and residual petroleum fuels, are shown graphically in Figure 33. These values reflect escalation for 1979 introduction of the powerplants. With subsequent de-escalation to 1975 values by the annual rate of 6.5% the operating and maintenance costs become those stated below in Table 16.



SIMPLE AND COMBINED CYCLE OPERATING AND MAINTENANCE COST AS A FUNCTION OF CAPACITY FACTOR
Figure 33

TABLE 16
OPERATING AND MAINTENANCE COST ESTIMATES (MILLS/KW-HR)

Capacity Factor	Simple Cycle Gas Turbine - Dist. Fuel	Combined Cycle Dist. Fuel	Combined Cycle Residual Fuel
0.12	3.81	4.63	5.56
0.65	2.84	1.75	2.26

For simplicity, these O&M values have been rounded to a value of 3.80 mills/Kw-Hr for the simple and recuperated cycles at .12 capacity factor and 1.75 mills/Kw-Hr and 2.25 mills/Kw-Hr for the distillate and residual fuel combined cycles respectively at the 0.65 capacity factor. The 0.5 mill/Kw-Hr differential O&M cost penalty for residual fuel operation with the combined cycle has been applied to the simple and recuperated cycles as well.

6.2.2 Operating And Maintenance Cost Differential Considerations

A secondary objective in the present study is to assess the impact of thermal barrier introduction on operating and maintenance costs. It is convenient to consider differentials in O&M costs associated with specific aspects of thermal barrier introduction. In this regard, three cases are of interest:

Case 1 - Thermal barrier vs. no thermal barrier.
Same turbine inlet temperature, same metal temperature, reduced cooling flow, improved performance.

Case 2 - Thermal barrier vs. thermal barrier.
Same turbine inlet temperature, same cooling flow, reduced metal temperature, increased life.

Case 3 - Thermal barrier vs. thermal barrier.
Same turbine inlet temperature, same cooling flow, differing replacement intervals.

Case 1 — Application of Thermal Barrier with Turbine Inlet Temperature Held Constant and Cooling Flow Reduced to Facilitate Improved Performance.

In this consideration, it is the objective to estimate, at least on an approximate basis, the differential O&M cost which would accrue by simply adding the thermal barrier coating to an existing design gas turbine engine while maintaining the same turbine inlet temperature. For this case, it is further assumed that the turbine cooling air usage would be reduced in conjunction with the coating in such a manner as to maintain equivalent metal temperatures with the coated components.

NASA has arbitrarily specified for this study that the coating will be renewed at two-year intervals by means of component removal, inspection, stripping, and reapplication in a specified manner.* For this aspect of the study, it will be assumed that the biennial renewal process will require a routine turbine service inspection in which the turbine casing would be opened. Further, for simplicity, the biennial service inspection and casing removal would be equally applied to the non-thermal barrier coated turbine. As mentioned above, the component metal temperatures are specified equivalent in each case and therefore the component lifetimes will be taken as equivalent. This approach may indeed understate the benefits of the barrier coating in that thermal gradients may be reduced by the coating therefore enhancing fatigue life even though average metal temperatures may be unchanged.

With these ground rules in mind, the differential operating and maintenance costs can be stated:

$$\Delta O\&M = O\&M_{Barrier} - O\&M_{Non\text{-}barrier}$$

$$O\&M_{Barrier} = \frac{\text{Stripping \& Coating Costs + Turbine Service Cost + Component Replacement Cost}}{(2)\ (8766\ \frac{HRS}{YR})\ (\text{Capacity Factor})\ (\text{Unit Output})}$$

$$O\&M_{Non\text{-}Barrier} = \frac{\text{Turbine Service Cost + Component Replacement Cost}}{(2)\ (8766\ \frac{HRS}{YR})\ (\text{Capacity Factor})\ (\text{Unit Output})}$$

So that:

$$\Delta O\&M = \frac{\text{Stripping and Coating Costs}}{(2)\ (8766\ \frac{HRS}{YR})\ (\text{Capacity Factor})\ (\text{Unit Output})}$$

In support of this computation, the thermal barrier stripping and coating costs have been specified by NASA as follows:

TABLE 17
THERMAL BARRIER STRIPPING AND COATING COSTS

Component	Stripping And Coating Cost
Combustor Liner	\$100/each
Transition Piece	\$200/each
Stationary Vanes	\$150/airfoil
Rotating Blades	\$150/airfoil

*NASA currently is managing the development of such processes under Contract #NAS 3-20112, "Automated Plasma Spray Process Feasibility Study". At the conclusion of this work, additional information will be available to update the arbitrary assumptions made for this study.

As an additional assumption, the unit output for the calculation cases will be set at 100 MW for simple and recuperated cycle cases all of which are evaluated at .12 capacity factor and at 150 MW for combined cycle cases which are all evaluated at .65 capacity factor* the approximate power output of the current design engine under consideration at current turbine inlet temperature levels.

Capacity factor, the ratio of actual hours operated per year to the number of hours per year (8766), had been set at values of 0.12 (peaking), 0.45 (intermediate), 0.50 (intermediate), 0.65 (base) and 0.80 (base) for the ECAS Task I parametric analysis study. For the present differential O&M analysis, the values of 0.12, and 0.65 will be considered.

When the above data are taken into account, the following operating and maintenance cost differentials are obtained.

TABLE 18
CASE 1 DIFFERENTIAL O&M COSTS

Capacity Factor	ΔO&M (Mills/Kw-Hr)
0.12 Peaking	+ .391
0.65 Base	+ .048

Case 2 — Application of Thermal Barrier with Same Turbine Inlet Temperature and Cooling Flow (current production levels) Resulting in Reduced Metal Temperature for Coated Components

Unlike Case 1, the objective of this case is to compare the operating and maintenance cost of two sets of coated components, the difference being that one set utilizes the no-coat cooling flow as would be the case in a direct thermal barrier retrofit, to achieve lower metal temperatures.

Assuming the biennial turbine inspection interval as in Case 1, the differential operating and maintenance costs can be expressed as:

$$\Delta O\&M = \frac{(\text{Component Replacement Cost})_{\text{Coated (Nominal Temp)}} - (\text{Component Replacement Costs})_{\text{Coated (Lower Temp)}}}{2\ (8766)\ (CF)\ (\text{Unit Output})}$$

*Total output per gas turbine is roughly speaking 100 MW for the simple and recuperated cycle plants and 150 MW for the combined cycle plants studied.

In this consideration, it is necessary to assess the difference in component replacement costs due to the reduced metal temperature and shielded status of the coated components contributing to anticipated longer life.

It has been elected to represent the annual costs chargeable to component replacement as a sinking fund series payment:

$$\text{Annual component replacement cost} = S \frac{i}{(1+i)^n - 1}$$

S - Component Renewal Parts Price (mid '75 dollars)

i - Cost of money, (7.5% as used in ECAS Task I)

n - Number of years of component life

where

$$n = \frac{\text{Hours of Component Life}}{(\text{Capacity Factor})\ (8766\ \text{Hrs/Yr}))}$$

The biennial component replacement costs are then:

$$2S \frac{i}{(1+i)^n - 1}$$

Since each component may have a different calculated lifetime, then the total biennial renewal parts sinking fund charge would be the summation:

$$2 \sum_j S_j \frac{i}{(1+i)^{n_j} - 1}$$

j - Designated component under consideration

The subject of component lifetimes, particularly those which are candidate for the application of thermal barrier coatings, is a complex topic. Many variables, in addition to average metal temperature, including such parameters as startup time, (affecting low-cycle fatigue life), fuel delivery system cleanliness and fuel nozzle cleanliness (affecting hot spot temperatures), and utility preventive maintenance procedures have important effects on overall component lifetime. These factors make it technically impractical to relate lifetime predictions for all components of interest to a single parameter such as average metal temperatures. Tabulations of component lifetime which are in current use to predict nominal target service lifetimes for hot parts are therefore based on a combination of factors; not the least important of which is the more than 25 years manufacturing experience embodied in current designs.

The two tabulations below relate to continuous duty and peaking duty hot part component target service lifetimes. Table 19 displays hot part nominal lifetimes for continuous duty operation at base load or continuous duty design turbine inlet temperature and utilizing distillate oil fuel.

TABLE 19
CONTINUOUS DUTY COMPONENT TARGET SERVICE LIFETIMES

Component	Lifetime, Hours	Lifetime, Years (Assuming .65 Capacity Factor)
Combustor Liner	16000	2.808
Transition Piece	16000	2.808
Vane Row 1	30000	5.265
2	40000	7.020
3	60000	10.530
4	100000	17.550
Rotating Blade Row 1	30000	5.265
2	50000	8.775
3	75000	13.163
4	75000	13.163

Table 20 lists corresponding information for distillate oil fired peaking duty operation assuming normal start times (nominally totaling 30 minutes from "cold iron" to full load as opposed to emergency starts which may total half or less of this amount).

TABLE 20
CYCLIC DUTY COMPONENT TARGET SERVICE LIFETIMES

Component	Life Cycles	Life, Hours (Assuming 4 Hrs. Per Start)	Life, Years (Assuming .12 Capacity Factor)
Combustor Liner	800	3200	3.042
Transition Piece	800	3200	3.042
Vane Row 1	1000	4000	3.803
2	1000	4000	3.803
3	2500	10000	9.506
4	5000	20000	19.013
Rotating Blade Row 1	1000	4000	3.803
2	2500	10000	9.506
3	5000	20000	19.013
4	10000	40000	38.026

In interpreting these figures it should be noted that the predicted lifetime for cyclic duty is based entirely upon cyclic life as listed in the first column. The lifetimes shown as hours and years are based on two arbitrary assumptions:

- 4 hours operation per start is obtained
- a capacity factor of .12 is maintained (1050 hours/year)

In addition, these two tabulations assume "pure" duty operation. That is, continuous duty operation is predominantly continuous with minimal shutdowns and peaking duty is predominantly cyclic. Much operation is in fact mixed in that certain portions are representative of continuous and others likewise representative of peaking duty. Predicted lifetimes for this type of operation are based on combinations of life fractions utilized in peaking and continuous duty. These considerations, however, are beyond the scope of this portion of the study.

The above tabulated lifetimes are, of course, based upon the use of non-thermal barrier coated components. The task at hand now becomes one of estimating the improvement in component life as a result of the application of the thermal barrier. The most directly related task performed in this study has been the heat transfer analysis in which a temperature drop of approximately 110^{o}K (200^{o}F) through the barrier coating has been computed for current type turbine cooled blade designs representative of stage one (or most severe service) with the nominal coating thickness. Perhaps the simplest application of this information regarding life prediction is to the airfoil sections of rotating blades in which the stress patterns are predominantly uniaxial (important untwist effects occur in latter row blades, however). Blade life predictions for rotating blades are commonly made on the basis of calculated stress levels, blade metal temperatures, and materials stress rupture properties. Due to the single shaft constant speed nature of the turbine designs under consideration, resonant frequencies can be avoided by design thus reducing the importance of high cycle fatigue life limiting factors. When the estimated metal temperature reductions are accounted for, blade lifetimes for the air-cooled rotating blades (rows 1 and 2 in current designs) based on creep rupture criteria for continuous duty operation can be shown to jump from the values shown in Table 19 to over 100,000 hours.

Roughly speaking, this would represent more than a doubling of blade life using present materials. Lifetimes of combustor liners, stationary vanes, and to a lesser extent, transition pieces, are based on a combination of factors such as peak "hot spot" temperatures, and corrosion as well as average metal temperatures. The information available at this time is not sufficient to make a similar prediction on component lifetime for these parts. Therefore, arbitrarily, it has been elected to regard air-cooled component lifetimes as being doubled for continuous duty operation by the addition of the thermal barrier coating.

The assessment of increases in component life from the purely cyclic operation or low cycle fatigue consideration standpoint is likewise an arbitrary topic at this point in the evaluation of thermal barriers. In light of the insulating effect of the thermal barrier as well as the specific heat capacity* it can be expected that the thermal barrier coated components will experience less severe thermal stress transients than uncoated components. For this cyclic portion of the life comparison, it has been arbitrarily estimated that the coated components will have a cyclic life equal to 1.5 times that of the uncoated components.

Using the component lifetimes shown in Tables 19 and 20 and the arbitrary life increases assigned to the thermal barrier coated lower temperature components in addition to published mid-1975 renewal parts prices (Reference 9.20) for a comparable large industrial turbine (W501) the differential operating and maintenance costs are found to be:

TABLE 21
CASE 2 DIFFERENTIAL OPERATING AND MAINTENANCE COSTS

Capacity Factor	ΔO&M (Mills/Kw-Hr)
0.12 Peaking	- 1.137
0.65 Base Load	- 0.172

where

$$\Delta O\&M = \frac{\left[2\Sigma_j S_j \dfrac{i}{(1+i)^{n_j}-1}\right]_{\text{Coated Nominal Temp}} - \left[2\Sigma_j S_j \dfrac{i}{(1+i)^{n_j}-1}\right]_{\text{Coated Lower Temp}}}{(2)\,(8766)\,(CF)\,(\text{Unit Output})}$$

and

where unit output has been set at 100000 KW single and recuperated cycle (.12 CF)
150000 KW combined cycle (.65 CF).

and

i = .075

S_j = renewal parts price

*A typical thermal barrier material, calcia stabilized ZrO_2, has a specific heat of approximately 700 J Kg^{-1} K^{-1} (0.16 Btu lb^{-1} $^oF^{-1}$) which is approximately 35% higher than parent metal in the temperature range of interest (Reference 9.19).

Case 3 — Comparison of Thermal Barrier Renewal Time Interval on Operating and Maintenance Costs

Given that the specified nominal thermal barrier renewal interval is two years, it is desired to know what is the effect on O&M costs by altering the interval to one year and to five years.

Assuming that equivalent turbine inlet temperature and turbine cooling flow rates are used in each case, the differential operating and maintenance cost between the one-year and two-year intervals can be expressed as:

$$\Delta O\&M_{1-2} = O\&M_{1\ Yr} - O\&M_{2\ Yr}$$

where:

$$O\&M_{1\ Yr} = \frac{\begin{matrix}\text{Stripping and} \\ \text{Coating Costs}\end{matrix} + \begin{matrix}\text{Turbine} \\ \text{Service Costs}\end{matrix} + \begin{matrix}\text{Annual Component} \\ \text{Replacement O\&M Cost}\end{matrix}}{(1)\ (CF)\ (8766\ Hrs/Yr)\ (Unit\ Output)}$$

and

$$O\&M_{2\ Yr} = \frac{\begin{matrix}\text{Stripping and} \\ \text{Coating Costs}\end{matrix} + \begin{matrix}\text{Turbine} \\ \text{Service Costs}\end{matrix} + \begin{matrix}\text{Annual Component} \\ \text{Replacement O\&M Cost}\end{matrix}}{(2)\ (CF)\ (8766\ Hrs/Yr)\ (Unit\ Output)}$$

so that

$$\Delta O\&M_{1-2} = \frac{1}{2}\ \frac{\text{Stripping \& Coating Costs + Turbine Service Costs}}{(CF)\ (8766\ Hrs/Yr)\ (Unit\ Output)}$$

Likewise the differential operating and maintenance cost between two-year and five-year replacement intervals is:

$$\Delta O\&M_{2-5} = \frac{3}{10}\ \frac{\text{Stripping \& Coating Costs + Turbine Service Costs}}{(CF)\ (8766\ Hrs/Yr)\ (Unit\ Output)}$$

The differential O&M costs have been calculated and are displayed in Table 22.

where

arbitrarily unit output = 100000 KW, capacity factor = .12

= 150000 KW, capacity factor = .65

TABLE 22
CASE 3 DIFFERENTIAL O&M COSTS FOR VARYING THERMAL BARRIER RENEWAL INTERVALS

Capacity Factor	Δ O&M$_{1\text{-}2}$
0.12 0.65	+.985 +.121 Mill/Kw-Hr
Capacity Factor	**Δ O&M$_{2\text{-}5}$**
0.12 0.65	-.657 -.081 Mill/Kw-Hr

6.3 COST OF ELECTRICITY RESULTS

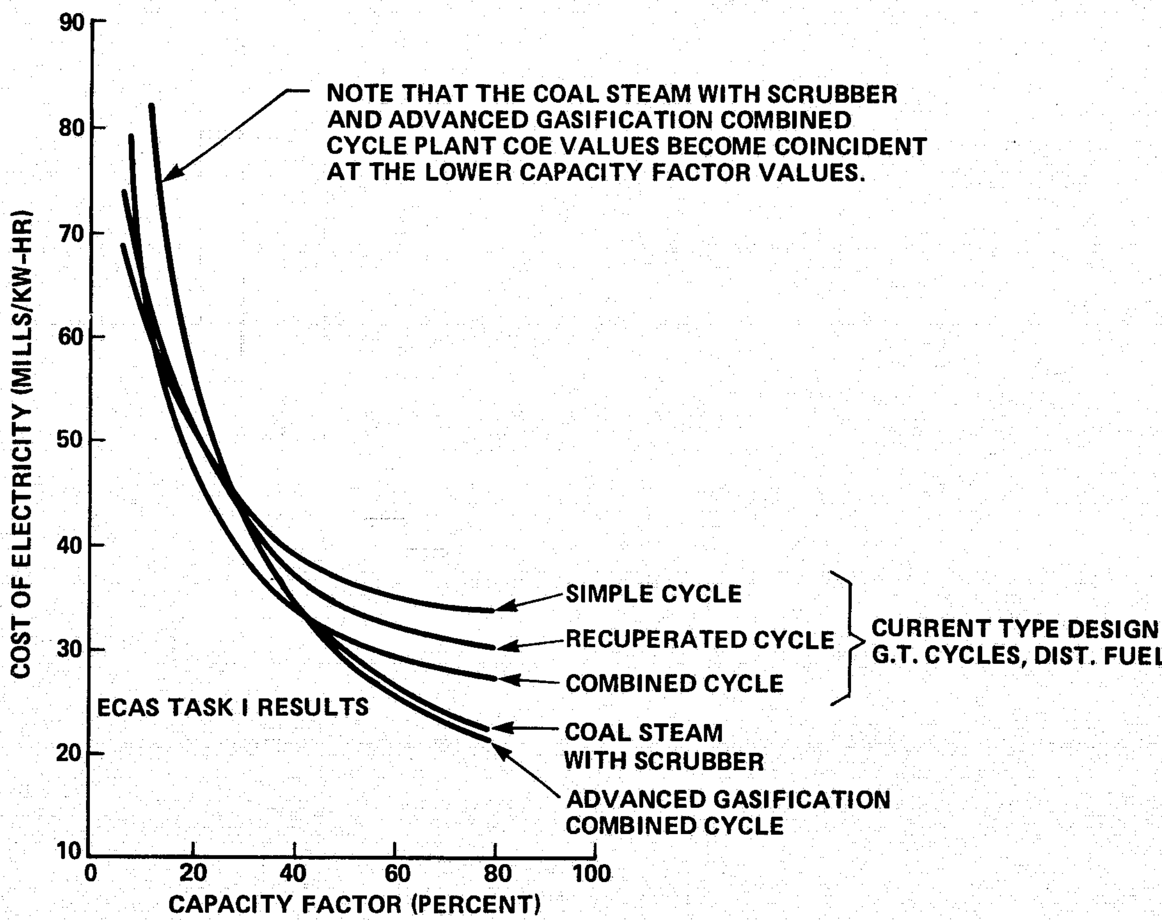
6.3.1 Cost Of Electricity Perspective

The cost of electricity associated with various power plants is strongly dependent upon capacity factor, or equivalently the number of hours operated per year. Indeed, the selection of new generating capacity is based almost exclusively upon the type of generating duty for which the plants are intended. Traditionally simple cycle gas turbines have been found to be the economic choice for peaking duty with capacity factors of about 0.15 and lower while for base load operation of about .50 capacity factor and higher the use of coal fired and nuclear steam plants has been selected. For intermediate duty capacity factor operation, the gas turbine-steam turbine combined cycle plant has gained increasing acceptance.

In order to gain a better perspective of the importance of capacity factor in interpreting the findings of the present study, it is useful to review some results of the ECAS Task I study (Reference 9.9). Figure 34 illustrates cost of electricity of five hypothetical power plants examined in that study as a function of capacity factor. These results pertain to parametric point power plants particularly in the case of the three gas turbine cycles and therefore are only generally representative of current design products, and all power plants differ from those of the present study with regard to fuel type, and baseline cost year.* The basic results, however, remain valid. For example, at a capacity factor of 0.12, the simple cycle gas turbine shows a COE advantage over the coal fired atmospheric pressure furnace steam plant of about 30 mills/Kw-Hr while at .80 capacity the steam plant enjoys a 12 mill/Kw-Hr advantage over the gas turbine plant. In the intermediate range, the combined gas-steam cycle plant shows marked advantage over all others. The nominal cycle parameters corresponding to these cycles are as follows.

*The ECAS Task I study used coal derived liquid fuel at $2.60/10^6 Btu for clean distillate, Bituminous coal at $.85/10^6 Btu, and capital costs tabulated in mid-'74 dollars.

GT simple and combined cycle	$1366^{o}K$ ($2000^{o}F$) T_{IT}, 12:1 compressor pressure ratio
GT recuperated cycle	$1366^{o}K$ ($2000^{o}F$) T_{IT}, 10:1 compressor pressure ratio
GT advanced gasification combined cycle	$1644^{o}K$ ($2200^{o}F$) T_{IT}, 12:1 compressor pressure ratio
Coal steam with scrubber	Steam conditions 24130 kPa $811^{o}K/811^{o}K$ (3500 psig, $1000^{o}F/1000^{o}F$)



COST OF ELECTRICITY AS A FUNCTION OF CAPACITY FACTOR
Figure 34

6.3.2 Simple Cycle Cost Of Electricity

Several comparisons and individual cases of simple cycle cost of electricity are examined below. In each case the capacity factor 0.12 (corresponding to 1050 hours of operation per year) has been utilized.

First, the changes in cost of electricity resulting from the addition of the thermal barrier coating at both current design turbine inlet temperature and at 1477°K (2200°F) are displayed by Table 23.

TABLE 23
COE FOR SIMPLE CYCLE CPT GAS TURBINES WITHOUT AND WITH TBC AT CURRENT T_{IT} LEVEL AND WITH TBC AT 1477°K (2200°F) T_{IT}

Cost Of Electricity (Mills/Kw-Hr)	Current T_{IT} Without TBC	Current T_{IT} With TBC	1477°K (2200°F) T_{IT} With TBC
Capital	28.454	28.222	26.678
Fuel	30.160	29.926	29.702
O&M	3.800	3.800 + .391	3.800 + .391
Total	62.414	62.339	60.571

At the current T_{IT} level, the improved performance made possible by cooling air usage reduction in the case of the barrier coated turbine is reflected in reduced fuel cost of electricity. It is also seen from the capital cost portion that the additional output from the thermal barrier coated engine case more than offsets the added cost of the thermal barrier coating and thus reduces the COE (cost of electricity) attributed to capital. This is particularly so at the higher turbine inlet temperature in which the markedly lower capital COE results in a substantial overall reduction in total COE.

The effect of thermal barrier addition and turbine inlet temperature increase in the case of the near-term advanced gas turbine simple cycle plant is illustrated by Table 24.

TABLE 24
COE FOR SIMPLE CYCLE NTA GAS TURBINE WITHOUT AND WITH TBC AT 1477°K (2200°F) AND WITH TBC AT 1588°K (2400°F) T_{IT}

Cost Of Electricity (Mills/Kw-Hr)	1477°K (2200°F) T_{IT}		1588°K (2400°F) T_{IT} With TBC
	Without TBC	With TBC	
Capital	26.510	26.345	25.316
Fuel	29.686	29.479	29.967
O&M	3.800	3.800 + .391	3.800 + .391
Total	59.996	60.015	59.474

In these cases the gas turbine engine utilizes advanced impingement/convection cooling technology for compatibility with uncoated operation at the 1477°K (2200°F) turbine inlet temperature level. A temperature uprated version of the near-term advanced engine has been evaluated including thermal barrier coating and with a turbine inlet temperature of 1588°K (2400°F). The overall cost of electricity results for this case are shown by the third column of Table 24.

Cost of electricity has been computed for the simple cycle plant incorporating the concept advanced design (ECAS II) gas turbine engine described in Sections 4.1 and 5.5.1. The results are shown below in Table 25.

TABLE 25
COE FOR SIMPLE CYCLE ADVANCED GAS TURBINE ENGINE DESIGN (ECAS II) WITH TBC AT 1644°K (2500°F) T_{IT}

Cost Of Electricity (Mills/Kw-Hr)	
Capital	31.153
Fuel	27.011
O&M	3.800 + .391
Total	62.355

The results indicate a significant decrease in fuel COE reflecting the higher efficiency obtainable with the advanced design in the simple cycle mode. However, the estimated capital costs of the advanced design have been found to be higher and are reflected by the increased capital COE. It should be borne in mind that this design is in the concept development stage and therefore estimated costs are subject to more uncertainty than, say, the near-term advanced version of current production type hardware.

For a greater perspective on the CPT and NTA gas turbine results disclosed in side-by-side comparisons above, a composite tabulation of simple cycle COE results has been prepared and is given in Table 26. The overall tabulation illustrates both efficiency and COE changes as a function of turbine inlet temperature and thermal barrier addition. The engines on this tabulation are all similar in that they represent current type design and near-term advanced developments of the current type designs. The turbine inlet temperature progression arrangement indicates COE to be a steadily declining quantity through the 1588°K (2400°F) level. It is interesting to note that the highest temperature case actually has a slightly lower efficiency and consequent higher fuel cost of electricity than the preceeding near-term advanced case with barrier and operating at the 1477°K (2200°F) level. The basic reason for the continued decline in total COE is that the increased turbine inlet temperature has increased gas turbine output sufficiently to reduce capital COE enough to more than offset increased fuel COE. The peaking of thermal efficiency between the 1477°K (2200°F) and 1588°K (2400°F) indicates that for improved efficiency at even higher temperature, an improved cooling configuration or thicker thermal barrier coating would be required.

TABLE 26

EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON SIMPLE CYCLE COST OF ELECTRICITY

SUMMARY RESULTS						
Engine Type	T_{IT}(°F)	η_{HHV} (%)	Cost Of Electricity (Mills/Kw-Hr) Capacity Factor 0.12			
			Capital	Fuel	O&M	Total
Current Production Type without Barrier	2000-2100	29.42	28.454	30.160	3.800	62.414
Current Production Type with Barrier	2000-2100	29.65	28.222	29.926	3.800 + .391	62.339
Current Production Type with Barrier	2200	29.88	26.678	29.702	3.800 + .391	60.571
Near-Term Advanced without Barrier	2200	29.89	26.510	29.686	3.800	59.996
Near-Term Advanced with Barrier	2200	30.11	26.345	29.479	3.800 + .391	60.015
Near-Term Advanced with Barrier	2400	29.60	25.316	29.967	3.800 + .391	59.474

6.3.3 Recuperated Cycle Cost Of Electricity

In the section which follows the recuperated cycle plant cost of electricity comparisons are made on the basis of a capacity factor of 0.12.

In order to establish a baseline for further comparison, the case of adding the thermal barrier coating to a recuperated cycle gas turbine with current type cooling system design configuration and increasing turbine inlet temperature to the 1477°K (2200°F) level has been considered. The cost of electricity results for this calculation are shown below in Table 27.

TABLE 27

COE FOR CURRENT TYPE DESIGN RECUPERATED CYCLE GAS TURBINE WITH ADDITION OF TBC AND T_{IT} INCREASE TO 1477°K (2200°F)

Cost Of Electricity (Mills/Kw-Hr.)	
Capital	31.177
Fuel	25.044
O&M	3.800 + .391
Total	60.412

COE comparisons made for near-term advanced recuperated cycle gas turbines with and without thermal barrier coatings at the 1477°K (2200°F) and higher turbine inlet temperature level are shown in Table 28.

TABLE 28
COE FOR NTA GAS TURBINE WITH ADDITION OF TBC AND INCREASE IN T_{IT}

Cost Of Electricity (Mills/Kw-Hr)	1477°K (2200°F) T_{IT} Without TBC	1477°K (2200°F) T_{IT} With TBC	1560°K (2350°F) T_{IT} With TBC
Capital	30.915	30.711	30.034
Fuel	24.897	24.716	24.758
O&M	3.800	3.800 + .391	3.800 + .391
Total	59.612	59.618	58.983

Shown in the first two columns is a comparison made without and with the thermal barrier coating at a turbine inlet temperature of 1477°K (2200°F). Advanced impingement/convection blade cooling techniques are utilized to provide capability of operation with non-coated components at this temperature level. A comparable uprating has been evaluated and is displayed in the third column of Table 28, in which the thermal barrier coating has been added to the near-term advanced gas turbine design (capable of 1477°K (2200°F) turbine inlet temperature) and the turbine inlet temperature level has been increased to the limit of the last row uncooled turbine blading. This, of course, involves design advancements including blade cooling design improvements, materials advancements, and the expenditure of additional cooling air.

A composite tabulation of the recuperated cycle cost of electricity results as a function of turbine inlet temperature and barrier addition is shown in Table 29. These engines all represent recuperated cycle versions of current type and near-term advanced simple cycle engines. (For completeness a recuperated cycle current type design at the current turbine inlet temperature level has been included in this tabulation.) The results show a steadily improving thermal efficiency up through the 1477°K (2200°F) turbine inlet temperature level with a very slight decline at the highest temperature considered, resulting from the increased coolant required. Capital cost of electricity is shown to be a steadily declining function of turbine inlet temperature while fuel cost of electricity of course, tracks thermal efficiency. Distillate fuel has been used for all these cases as with the comparable simple cycle results of Table 26. In comparison with the simple cycle results it is seen that total recuperated cycle COE is essentially equivalent to that for the simple cycle. It is important to remember, in addition, that both these cycles use the same fuel cost — a level compatible with today's fuel prices. For higher fuel costs, the higher thermal efficiency of the recuperated cycle would result in lowest cost of electricity.

TABLE 29
EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON RECUPERATED CYCLE COST OF ELECTRICITY

SUMMARY RESULTS						
Engine Type	T_{IT}(°F)	η_{HHV}(%)	Cost Of Electricity (Mills/Kw-Hr) Capacity Factor 0.12			
			Capital	Fuel	O&M	Total
Current Production Type Design without Barrier	2000-2100	34.63	33.106	25.623	3.800	62.529
Current Production Type Design with Barrier	2200	35.44	31.177	25.044	3.800 + .391	60.412
Near-Term Advanced without Barrier	2200	35.64	30.915	24.897	3.800	59.612
Near-Term Advanced with Barrier	2200	35.90	30.711	24.716	3.800 + .391	59.618
Near-Term Advanced with Barrier	2350	35.85	30.034	24.758	3.800 + .391	58.983

6.3.4 Combined Cycle Cost Of Electricity

As in the preceding simple and recuperated cycle cases, the objective of the combined cycle cost of electricity calculations is to assess the impact of the introduction of the thermal barrier coatings on present and advanced versions of such systems. As mentioned earlier, the present combined cycle systems have been introduced for the purpose of meeting the intermediate capacity factor or mid-range power generation duty. It is expected, however, that as the combined cycle plants become more efficient with the development of higher temperature turbine technology and as low-Btu gasification systems are developed, the combined cycle will be utilized largely for base load operations. For the cases under consideration below, the value of 0.65 capacity factor typical of base load operation has been used for all instances.

The first cases considered are those of the introduction of the thermal barrier coating to current type design gas turbine combined cycle systems burning distillate oil and operating at current level and at 1477°K (2200°F) T_{IT}. The results are shown in Table 30.

It is seen that the effect of the thermal barrier addition at current temperature level is a modest decline in both capital and fuel COE and these together more than offset the increased O&M costs associated with thermal barrier renewal to result in a net COE improvement. A much more dramatic overall COE improvement, however, is seen to result when the thermal barrier is added and turbine inlet temperature increased to 1477°K (2200°F). This COE change, over 6 times that described above, represents approximately a 3% improvement in the bottom line cost of electricity generation by such a plant.

TABLE 30

COE FOR COMBINED CYCLE CPT GAS TURBINE WITHOUT AND WITH TBC AT CURRENT T_{IT} LEVEL AND WITH TBC AT 1477°K (2200°F) T_{IT}

Cost Of Electricity (Mills/Kw-Hr)	Current T_{IT} Without TBC	Current T_{IT} With TBC	1477°K (2200°F) T_{IT} With TBC
Capital	10.499	10.417	10.058
Fuel	21.492	21.370	20.892
O&M	1.750	1.750 + .048	1.750 + .048
Total	33.741	33.585	32.748

Similarly, the cost of electricity has been calculated for the case of the near-term advanced gas turbine engine fitted with advanced impingement/convection cooling technology capable of uncoated operation at 1477°K (2200°F). In this case, the turbine inlet temperature has first been held constant at the 1477°K (2200°F) level while the thermal barrier has been added to facilitate reduced cooling flow and improved performance and secondly COE has been calculated at the 1588°K (2400°F) level with the coating. The results are shown in Table 31.

TABLE 31

COE FOR COMBINED CYCLE NTA GAS TURBINE WITHOUT AND WITH TBC AT 1477°K (2200°F) And With TBC At 1588°K (2400°F) T_{IT}

Cost Of Electricity (Mills/Kw-Hr)	1477°K (2200°F) T_{IT} Without TBC	1477°K (2200°F) T_{IT} With TBC	1588°K (2400°F) T_{IT} With TBC
Capital	10.015	9.930	9.649
Fuel	20.911	20.711	20.344
O&M	1.750	1.750 + .048	1.750 + .048
Total	32.676	32.439	31.791

Cost of electricity has been computed for the advanced coal gasification combined cycle plant based on the ECAS II engine design and combined cycle arrangement described earlier in Sections 4.1 and 5.5.2. The COE results, computed using the same methods employed in the other cases in this thermal barrier evaluation*, are shown below in Table 32.

*O&M cost, however, 2.448 mills/Kw-Hr, accounts for cost of dolomite, permanent station personnel, regular maintenance and special maintenance costs and is taken from the Westinghouse ECAS Task II Final Report (Reference 9.21).

TABLE 32
COE FOR GASIFICATION COMBINED CYCLE USING ADVANCED GAS TURBINE (ECAS II) WITH TBC AT 1644°K (2500°F) T_{IT}

Cost Of Electricity (Mills/Kw-Hr)	
Capital	20.033
Fuel	7.255
O&M	2.448 + .048
Total	29.784

The important thing to note is that the relative capital and fuel COE costs have reversed their positions in comparison with the combined cycle burning distillate fuel. That is, for both types of plants evaluated at the .65 capacity factor base load condition, the capital cost of the gasification combined cycle plant is approximately twice that of the distillate fuel plant, thus reflecting the additional capital cost of the gasification subsystem. The fuel cost portion of the gasification plant total, however, is much lower reflecting primarily the lower price of the input fuel. Fuel prices were \$2.46/MJ (\$2.60/10^6 Btu) for distillate fuel oil, and \$0.95/MJ (\$1.00/10^6 Btu) for Illinois No. 6 bituminous coal. In comparison with the total COE shown in Table 30, for example, it follows that the use of the gasification combined cycle in conjunction with the advanced design gas turbine with TBC, has the potential for significant reduction in the overall cost of electricity below the near-term advanced combined cycle burning distillate oil. In interpreting the comparison of COE, it is important to bear in mind the status of development of the advanced design combined cycle plant, for example, as compared with the near-term advanced version of current production type gas turbines. The advanced design plant is in the concept design stage of development and therefore, associated cost estimates necessarily incorporate a greater degree of uncertainty than do the near-term advanced engine and plant designs which are extrapolations of current production hardware.

In summary, tabulation of the combined cycle efficiency and COE results as a function of turbine inlet temperature increase and thermal barrier addition is given in Table 33. All cases shown are based on the current type non-reheat combined cycle arrangement and current production type and near-term advanced versions of the current production type gas turbine engine. Each case utilizes distillate fuel. The tabulation shows a steady increase in overall efficiency through the highest temperature, 1588°K (2400°F) considered. From the first case, that of a current production type engine without thermal barrier, to the near-term advanced design at the 1588°K (2400°F) level, an increase of almost 2.5 efficiency percentage points is realized. Concurrently, the cost of electricity exhibits a steady decline throughout the range of temperatures considered, with an overall reduction in total cost of electricity of approximately 6% indicated between lowest and highest turbine inlet temperatures.

TABLE 33
EFFECT OF THERMAL BARRIER ADDITION AND T_{IT} INCREASE ON
COMBINED CYCLE COST OF ELECTRICITY

SUMMARY RESULTS						
			Cost Of Electricity (Mills/Kw-Hr)			
			Capacity Factor 0.65			
Engine Type	T_{IT}(°F)	η_{HHV}(%)	Capital	Fuel	O&M	Total
Current Production Type without Barrier	2000-2100	41.49	10.499	21.492	1.750	33.741
Current Production Type with Barrier	2000-2100	41.73	10.417	21.370	1.750 + .048	33.585
Current Production Type with Barrier	2200	42.68	10.058	20.891	1.750 + .048	32.747
Near-Term Advanced without Barrier	2200	42.73	10.015	20.911	1.750	32.676
Near-Term Advanced with Barrier	2200	43.06	9.930	20.711	1.750 + .048	32.439
Near-Term Advanced with Barrier	2400	43.86	9.649	20.344	1.750 + .048	31.791

6.3.5 Effect Of Residual Oil Firing On Cost Of Electricity

Current and near-term advanced versions of the simple, recuperated and combined cycles under consideration have been evaluated with respect to the effect of residual oil firing on overall COE. The essential question to be examined is, does reduced fuel cost of the residual oil fuel compensate for increased capital and O&M cost associated with treating and burning the fuel? As shown by Table 34, which summarizes the results obtained for all three types of cycles, the use of residual fuel with the price as given will result in significantly lower total cost of electricity at both current temperature levels and at the near-term advanced level of 1477°K (2200°F). Each case shown in Table 34 includes the thermal barrier coating and the residual oil firing cases include a 0.5 mill/kw-hr cost adder, from Figure 33 (Section 6.2.1) to account for the higher scheduled maintenance activity associated with residual fuel.

6.3.6 Effect Of Direct Substitution Of Thermal Barrier On Cost Of Electricity For Current Type Designs

Two cases have been considered which would simulate the process of retrofitting the thermal barrier coating to current type gas turbine engines with no other changes. The cases, consisting of simple and combined cycles, have been calculated at current turbine inlet temperature levels and they utilize distillate fuels. In the calculation, of

TABLE 34
EFFECT OF RESIDUAL OIL FIRING AND THERMAL BARRIER ADDITION ON COE

Engine Type/Fuel (All Engines With Barrier)	T_{IT}(°F)	η_{HHV}(%)	Capacity Factor	Cost Of Electricity (Mills/Kw-Hr)			
				Capital	Fuel	O&M	Total
Simple Cycle							
Current Production Type, Distillate	2000-2100	29.65	.12	28.222	29.926	3.800 + .391	62.339
Current Production Type, Residual	2000-2100	29.25	.12	30.736	25.085	3.800 + .391 + .500	60.512
Near-Term Advanced, Distillate	2200	30.11	.12	26.345	29.479	3.800 + .391	60.015
Near-Term Advanced, Residual	2200	29.73	.12	28.745	24.680	3.800 + .391 + .500	58.116
Recuperated Cycle							
Near-Term Advanced, Distillate	2200	35.90	.12	30.711	24.716	3.800 + .391	59.618
Near-Term Advanced, Residual	2200	35.49	.12	33.166	20.680	3.800 + .391 + .500	58.537
Combined Cycle							
Current Production Type, Distillate	2000-2100	41.73	.65	10.417	21.370	1.750 + .048	33.585
Current Production Type, Residual	2000-2100	41.30	.65	10.960	17.869	1.750 + .048 + .500	31.127
Near-Term Advanced, Distillate	2200	43.06	.65	9.930	20.711	1.750 + .048	32.439
Near-Term Advanced, Residual	2200	42.63	.65	10.718	17.302	1.750 + .048 + .500	30.318
	Distillate Fuel Cost			$2.60/10^6 Btu			
	Residual Fuel Cost			$2.15/10^6 Btu			

course, cooling flows have been left unchanged and the consequent reduced turbine blading temperatures are accounted for by the estimation of reduced O&M expense in Case 2 of Section 6.2.2. Table 35, summarizing the results of this calculation, indicates that the simple cycle application would benefit more significantly from the retrofit than the combined cycle. (In this case, the .391 mill/Kw-Hr differential O&M cost is for TBC application and the 1.137 mill/kw-hr reflects credit for increased life.) Compared with the COE differential determined in this study, the simple cycle advantage would be considered small, about 1%; however, in view of the large numbers of simple cycle machines currently in service, the potentials for increased reliability and availability are attractive.

6.3.7 Effect of Thermal Barrier Renewal Interval on Cost of Electricity

The purpose of this portion of the overall evaluation is to determine the impact on total cost of electricity of varying thermal barrier renewal intervals. As the nature of the thermal barrier coating is now understood, it is anticipated that due to erosion and potential foreign object damage, the coating will require periodic renewal by stripping and recoating of components. It has been artibtarily specified by NASA that the baseline renewal interval be two years, a figure not incompatible with routine turbine inspection intervals.

TABLE 35
EFFECT OF DIRECT ADDITION OF THERMAL BARRIER TO CURRENT TYPE DESIGNS
(NO COOLING AIR REDUCTION WITH BARRIER ADDITION)

Engine Type	T_{IT}(°F)	η_{HHV} (%)	Cost Of Electricity (Mills/Kw-Hr)				
			Capital	Fuel	O&M Base	ΔO&M	Total
Simple Cycle							
Current Production Type without Barrier	2000-2100	29.42	28.454	30.160	3.800	—	62.414
Current Production Type with Barrier	2000-2100	29.40	28.739	30.181	3.800	+ .391-1.137 .12 Capacity Factor	61.974
Combined Cycle							
Current Production Type without Barrier	2000-2100	41.49	10.499	21.492	1.750	—	33.741
Current Production Type with Barrier	2000-2100	41.37	10.553	21.554	1.750	+ .048-.172 .65 Capacity Factor	33.733

The specific question to be answered then, at this time, is what COE changes would occur were the interval changed to 1 year and to 5 years? The answer may be found by examination of differential O&M changes and again a very simple model has been used as described in Section 6.2.2. The results of the overall COE totals are shown in Table 36. As indicated, the differences for the simple cycle case are much larger than those for the combined cycle case. In comparison with other total COE differentials identified in this study, the simple cycle case COE spread between 1-year and 5-year replacement are significant whereas the combined cycle differentials are not.

TABLE 36
EFFECT OF BARRIER REPLACEMENT INTERVAL ON COST OF ELECTRICITY

Engine Type	T_{IT}(°F)	Replacement Interval	Cost Of Electricity (Mills/Kw-Hr)			
			Base Value	O&M	ΔO&M	Total
Simple Cycle						
Current Production Type	2000-2100	1-Year	62.339	.12 Capacity Factor	+ .985	63.324
		2-Year			-	62.339
		5-Year			- .657	61.682
Combined Cycle						
Current Production Type	2000-2100	1-Year	33.585	.65 Capacity Factor	+ .121	33.706
		2-Year			-	33.585
		5-Year			- .081	33.504

SECTION 7
CONCLUSIONS AND RECOMMENDATIONS

7.1 CONCLUSIONS

In drawing and interpreting conclusions from this evaluation, it is useful to recapitulate important background information. As described in the Summary (Section 1), all cases under consideration apply to gas turbine applications for electric power generation in the context of simple, recuperated and combined cycles. In addition, the studies considered only the heavy duty industrial type of construction as opposed to aircraft derivative engine applications. The vast majority of the calculation cases have considered the application of thermal barrier coatings to current type designs and near term advanced design embodiments of such units.

1. The first conclusion which can be drawn from the evaluation is that the thermal barrier coating is potentially a valuable element in an overall program of graduated turbine inlet temperature increases for current type engine designs to increase efficiency and reduce overall cost of electricity.

2. In the context of simple, recuperated and combined cycles, the evaluation has shown that the largest payoff in terms of efficiency and cost of electricity for a given turbine inlet temperature increase occurs in the case of the combined cycle.

3. The largest single jump in efficiency and decrease in COE (cost of electricity) of those steps considered occurs with all three cycle types as current type blade cooling systems are uprated by the addition of the thermal barrier and turbine inlet temperature is increased to 1477°K (2200°F).

4. Significant performance improvements and COE reductions can be achieved with near term advanced versions of today's combined cycle designs at turbine inlet temperature values to approximately 1588°K (2400°F). This result is particularly significant in that it indicates considerable potential yet available for fuel conservation and overall COE reduction with basically current type hardware. Since production and support facilities are in place for this type of hardware, the benefits for development should be available on a near term basis with lowest overall development cost.

5. With the given fuel prices, substitution of residual fuel results in significant COE decreases for all three types of cycles. Although in the recent past, the price differential between residual and distillate has not been great enough to justify extensive utility application of residual for gas turbines, it is felt that the future will see a restoration of price differential characteristic of that specified in this study.

6. With the specified coating costs and the simple O&M model used, application of the thermal barrier at a set turbine inlet temperature and reducing cooling flow to maintain equivalent metal temperatures produces a tradeoff in COE. The coating costs used in this study have been arbitrarily specified by NASA. At this time a thermal barrier coating development program has been initiated by NASA* for the purpose of determining the feasibility of automating the process. However, the program is still in the early stages and significant cost information is not available at this time.

7. Within the limits of the O&M model and component life assumptions utilized, small simple and combined cycle COE differences result from application of the thermal barrier while holding turbine inlet temperature and cooling flow constant and accounting for increased component life. Within the context of other differential COE values determined in this study the simple cycle COE differences appear to be significant while the combined cycle values are insignificant.

8. Using the given coating costs and simplified O&M model, what would appear to be significant differences in simple cycle COE and insignificant differences in combined cycle COE result from the variation in barrier replacement interval from 1 to 2 to 5 years. Barrier durability, of course, is a prime consideration which was not addressed directly in this portion of the analysis. Experimental results will be required for more in-depth analysis of the question of optimum barrier renewal interval. Ultimately, demonstrated durability will have to be the criterion for selection of renewal interval.

9. At the given fuel prices (roughly representative of those prevalent at the time of preparation of this report), uprating a current production type simple cycle gas turbine by the addition of recuperation and operating at a peaking capacity factor does not offer COE reductions compared to an uprated simple cycle version of that machine. Higher fuel prices, however, will act to the advantage of the recuperated cycle gas turbine.

*Contract NAS 3-20112, "Automated Plasma Spray Process Feasibility Study"

7.2 RECOMMENDATIONS

The recommendations as described below recognize that the thermal barrier approach to turbine component coating is in the early stages of development. Therefore, many questions must be answered including basic ones such as determination of long time barrier durability characteristics and optimum barrier composition formulation. Early testing including successful high temperature aircraft gas turbine engine operation has been most encouraging and at this time prospects look good that the basic questions mentioned above will be expeditiously resolved. The following recommendations are based on the premise that this will be the case.

1. The first goal in implementing thermal barrier development should be to apply the coating to current type cooling designs and uprate turbine inlet temperature to $1477^{o}K$ ($2200^{o}F$). As shown in the performance and cost of electricity results sections, this step would provide a significant improvement in fuel conservation and reduction in COE. By its nature it would permit early near term implementation.

2. Ultimate development of the present type combined cycle and simple cycle designs to $1588^{o}K$ ($2400^{o}F$) in conjunction with thermal barrier introduction should be pursued. As described earlier in the report the $1588^{o}K$ ($2400^{o}F$) target represents a near term limit target based on foreseeable developments in cooling, combustion and material technology. It is estimated that with proper development support these areas could be brought to production readiness in approximately 6 years. Therefore, the ultimate development of near term type engine technology to the $1588^{o}K$ ($2400^{o}F$) level could represent a second phase extension of the $1477^{o}K$ ($2200^{o}F$) step recommended in item 1 above. This type of two phase approach has been expanded in the recommended thermal barrier development program described in Section 8.

3. The potential for application of the near term advanced engine technology achieved in such development programs should be evaluated for use in coal and oil gasification cycles. Both current type and advanced engine designs have been evaluated in conjunction with coal gasification cycles (References 9.9, 9.10, 9.11, 9.22). It is equally important, and perhaps more so, to assess the impact of the near term engine on the gasification combined cycle. Certain current developments are aimed at demonstrating the operation of low-Btu gasifiers in conjunction with utility size current type gas turbines. Concurrent development of a near term gas turbine technology would permit early integration and verification of a near term advanced gas turbine engine utilizing thermal barrier coatings in the low-Btu gas environment.

For the ECAS Task II study (Reference 2.21), it was required that the combustion of a low quality coal derived liquid residual fuel be evaluated as a combined cycle fuel. The study results indicated a relatively lengthly development program for the high temperature turbine application under consideration. An alternative and perhaps shorter term development program may include the integration of a lower temperature near term engine technology incorporating thermal barriers with an oil gasification system. It is recommended that such a combination be evaluated for use of the coal derived residual liquid fuels.

4. Determination of the effects of typical residual fuel firing on the thermal barrier coating durability should be investigated with high priority.

5. The potential for retrofitability of the thermal barrier to existing utility machines for clean fuel and ultimately residual and coal derived liquids should be evaluated in more depth.

6. Determination of an optimum thickness of barrier coating for industrial turbine blading should be made. In the present study the base thickness of barrier coating has been taken as .38mm (.015 in.) as specified by NASA. This thickness has been successfully applied and tested by NASA on aircraft engine components. Industrial turbine blading, by contract, is characterized by both larger size and larger radii. Therefore the optimum coating thickness may not necessarily be the same as that for aircraft components. Barrier thickness, of course, affects metal temperature, cooling requirements and ultimately engine performance.

7. In the present development program to commercialize the thermal barrier application process particular attention should be focused on the procedure and price for thermal barrier renewal on components having already seen service.

8. In subsequent development programs, emphasis should be placed on coating foreign object damage considerations. Assessment of the susceptibility of the coating to foreign object damage as well as development of means for on-line monitoring of the coating integrity should be pursued in the overall development plan.

SECTION 8
RECOMMENDED THERMAL BARRIER DEVELOPMENT PROGRAM

8.1 SUMMARY

The results of the thermal barrier evaluation (follow-on work to the ECAS Program, Contract NAS 3-19407) recently completed at Westinghouse Generation Systems, Lester, Pa., indicate that thermal barrier coatings in the order of 0.48 mm (0.019 in)* thickness allows the turbine designer to raise his turbine inlet temperature or reduce the amount of coolant flow or reduce metal temperatures. These advantages result from the fact that the thermal barrier coating is a good heat insulator; thus, higher gas temperatures are not felt by the protected metal. A second important possibility relative to use of the thermal barrier coatings is the protection from metal corrosion and erosion offered by the coating. By protecting the metal surface from corrosive gas impingement, the metal is expected to be able to operate for substantially longer periods of time.

Westinghouse feels that thermal barrier coatings should be brought to a state of production readiness for blades, vanes, transition pieces, and combustor liners. The ability of the coating to insulate the metal skin on blades and vanes indicates turbine inlet gas temperatures can be ultimately raised to the 1533-1588^{o}K (2300-2400^{o}F) regime on derivatives of current type design machines. The increased efficiency of the engine and of a combined cycle application resulting from 1533/1588^{o}K (2300^{o}/2400^{o}F) turbine inlet temperatures more than justifies the development program and design effort required to optimize blade and vane designs.

The benefits to be derived from the potential erosion/corrosion protection offered by thermal barrier coatings cannot be over-emphasized. Hot gas stream components, such as blades, vanes, combustor liners, and transition pieces may suffer premature failure when subjected to dirty fuel combustion environments without protective coatings and adequate fuel pretreatment. Improved operation using dirty fuels would give utility companies greater fuel flexibility.

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*Consisting of 0.10 mm (0.004 in.) bond coat and 0.38 mm (0.015 in.) thermal barrier coat.

To maximize the benefits to be derived from the use of thermal barrier coatings with derivatives of current type design machines will require a 6-year development program as described herein. However, increased efficiency, extended duty cycles, and multiple fuel capability are needed now by the electric generation industry. Therefore, a two-step evolutionary development program featuring a 1477°K (2200°F) turbine inlet temperature application followed by the full potential 1588°K (2400°F) turbine which attacks the problem of conserving fuel in a rapid, direct manner has been prepared.

The two-step development program would start by utilizing coatings to upgrade existing gas turbine designs to 1477°K (2200°F) turbine inlet temperature. This step would not require major turbine redesign and would concentrate on coated blades/vanes, process development, and combustor development. It is felt that the 1477°K (2200°F) service may allow the retrofitting of existing 90 MW class machines presently in utility service.

In parallel with the 1477°K (2200°F) machine, design, process, and test work will proceed on the 1588°K (2400°F) turbine inlet temperature machine. By working jointly with utility companies, the advanced 1588°K (2400°F) machine, with full utilization of thermal barrier coatings, can be at a state of production readiness in 6-years total program time.

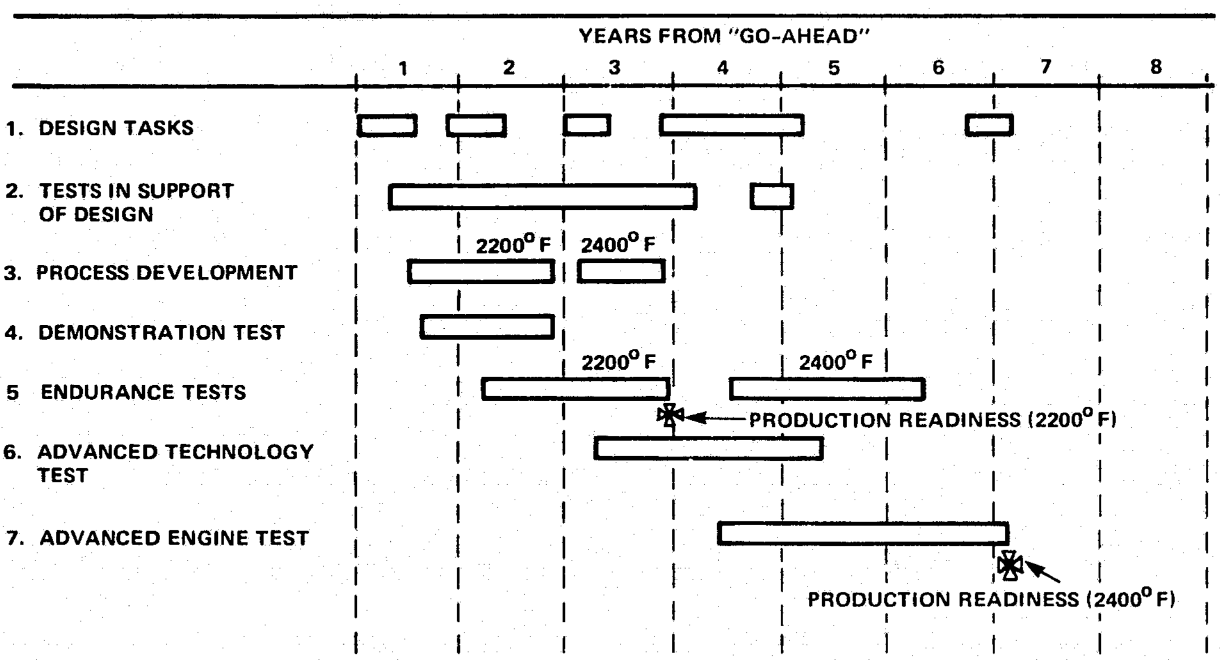
A summary schedule of the recommended development program is presented in Figure 35 with a brief description of the items shown in the schedule given in the following paragraphs.

The development program consists of the following phases:

- Design and design-support testing (Schedule items 1 and 2)
- Process Development (item 3)
- Full-Scale Engineering Tests (Schedule items 4 and 6)
- Full-Scale Field Tests at Utility Sites (Schedule items 5 and 7)

DESIGN AND DESIGN-SUPPORT TESTING

The work required in this phase of the program is to take the available technical data on thermal barrier coatings and produce a detailed design of an advanced version of a current type design (such as a W-501 described in Section 2.4) machine. In those areas where technical data is sparse or not applicable, material property tests and component tests would be conducted to verify critical design assumptions.



RECOMMENDED THERMAL BARRIER DEVELOPMENT PROGRAM SCHEDULE
Figure 35

PROCESS DEVELOPMENT

To bring the use of thermal barrier coatings to a state of production readiness, the application process must be reproducible and available to industry. Specifications, measurements, tests, and controls have to be developed to assure a consistent quality product.

FULL SCALE ENGINEERING TESTS

The Westinghouse full-scale W251 laboratory test engine would be used to verify that coated hardware can withstand the thermal shock and related enviornments associated with operation of the Near-Term 1477°K (2200°F) machine and the Full-Potential 1588°K (2400°F) machine.

FULL SCALE TESTS AT UTILITY SITES

To attain production readiness, successful endurance tests with normal commercial servicing and treatment would be demonstrated on the near-term and full-potential engines. Time periods of up to 8,000 operating hours are recommended, and, if thermal barrier coatings are in suitable operating condition, extended periods of operation would be included. As a final step for the full-potential engine, an advanced-engine test would be conducted at utility site.

8.2 GOALS AND OBJECTIVES

The goals of this development program are to:

(1) demonstrate in a three-year period that thermal barrier coatings can successfully perform in commercial service on stationary gas turbines;

(2) by judicious design practice, employ thermal barrier coatings on retrofit hardware suitable for upgrading existing commercial 90 MW class machines;

(3) demonstrate the full potential of thermal barrier coatings by operating an advanced stationary gas turbine in the 1588 $^{\circ}$K (2400°F) turbine inlet temperature regime with satisfactory performance.

In order to meet the above goals, it will be necessary to attain the following program objectives:

(a) determine the limits wherein coated components are able to satisfactorily operate in a stationary gas turbine under all normal operating environments.

(b) develop the necessary specifications and process controls to guarantee the manufacturing process.

(c) assure that the near-term usage of thermal barrier coatings meets the requirement of retrofitability for existing machines.

(d) commercially operate the near-term and full-potential gas turbines with thermal barrier coated components at 1477°K (2200°F) and 1588°K (2400°F) turbine inlet temperature, respectively.

8.3 DESIGN AND DESIGN-SUPPORT TESTING

A more thorough description and rationale for the major tasks of this proposed program is given in the following paragraphs.

8.3.1 Design

The use of thermal barriers on gas turbine components that are integral parts of the hot gas stream offers two attractive possibilities; namely, as a low conductivity overlay that results in lower metal temperatures under the coating and as a corrosion inhibitor. The first potential use, as a thermal barrier, is on blades, vanes, combustor liners, and transition pieces. The feature of potential corrosion protection could add useful life as well to combustor liners, transition pieces, blades, and vanes.

The design tasks for the two-step development program are shown in Figure 36. The near-term engine would be a modification of a current design such as the Westinghouse W501 which has a turbine inlet temperature of 1366-1422°K (2000-2100°F). It is felt that significant, though relatively straightforward, design changes are required in the turbine section to accommodate the 1477°K (2200°F) target.

The full-potential engine is basically an advanced engine and will require work in many aspects of engine design. The design work on the full-potential engine should start in parallel with the near-term work.

8.3.2 Design-Support Testing

To assist the design effort, it is necessary that tests be conducted to yield material property data and design data relative to a specific application of coatings on a component. In general, material property tests are conventional tests on standard test shapes, such as tensile specimens or coated pins. Tests to yield design data would utilize specific pieces of hardware (components) such as a vane segment or a blade. Figure 37 presents a listing of the tests recommended as part of the development program. A description of each test is given below.

8.3.2.1 MATERIAL PROPERTY TESTS

PURPOSE

- To provide engineering data and screening results for design.
- To augment published data for critical design applications.

DESIGN TASKS	NEAR-TERM 3 YRS - 2200° F	FULL POTENTIAL 6 YRS - 2400° F
- HEAT TRANSFER CALCULATIONS	X	X
- COOLING AIR MANAGEMENT		X
- AERODYNAMIC DESIGN AND ENGINE PERFORMANCE	X	X
- COMBUSTION SYSTEMS	X	X

TWO-STEP DEVELOPMENT PROGRAM DESIGN TASKS
Figure 36

TESTS IN SUPPORT OF DESIGN	NEAR-TERM 3 YRS - 2200° F	FULL POTENTIAL 6 YRS - 2400° F
- MATERIAL PROPERTY TESTS	X	X
- COMBUSTOR SCREENING TESTS	X	X
- VANE TEMPERATURE CYCLING TEST	X	?
- BLADE VIBRATION THERMAL TEST	X	?
- TRANSITION PIECE THERMAL CYCLE TEST	X	X
- FLYING OBJECT SIMULATED DAMAGE TEST	X	X
- LOCALIZED REPAIR TEMPERATURE CYCLING TEST	X	?
- COATING INTEGRITY ASSURANCE DEVELOPMENT TESTS		X
- SPALLATION DETECTOR DEMONSTRATION TESTS		X

TWO-STEP DEVELOPMENT PROGRAM TESTING REQUIREMENTS
Figure 37

DESCRIPTION

Because of the specialized nature of the erosion and corrosion testing, this portion of the material properties tests would be conducted at the Westinghouse Research and Development (R&D) Center, Churchill, Pa. The program would incorporate a hot shock tube facility and pressurized passages for erosion and corrosion testing. The shock tube would be used to obtain particle impact characteristics for predictions of component life as a function of gas cleanliness. The pressurized passage tests supplement the shock tube tests and address the determination of erosion and corrosion rates with respect to fuel constituents.

The following test conditions and parameters are of interest in the erosion/corrosion tests:

Erosion

- 922-1227°K (1200-1750°F) metal temperature range
- 3 particle sizes
- 2 candidate coatings
- Nickel and cobalt base materials

Corrosion
- $1061-1227^{\circ}K$ ($1450-1750^{\circ}F$) metal temperature range
- Clean fuel, fuel with vanadium, and fuel with high sulfur
- 2 candidate coatings
- Nickel and cobalt base materials

The tests to be conducted at the Materials Engineering Laboratory at Lester, Pennsylvania would consist of the following types and conditions:

Sample	Parameters
Tensile	Room Temperature to $1255^{\circ}K$ ($1800^{\circ}F$) Nickel and cobalt base
Creep Rupture	$1061^{\circ}K$ ($1450^{\circ}F$) to $1255^{\circ}K$ ($1800^{\circ}F$) Nickel and cobalt base
Bond Strength	Room temperature to $1255^{\circ}K$ ($1800^{\circ}F$) Nickel and cobalt base
High-Cycle Fatigue	$922^{\circ}K$ ($1200^{\circ}F$) to $1144^{\circ}K$ ($1600^{\circ}F$) Nickel base
Low-Cycle Fatigue	$922^{\circ}K$ ($1200^{\circ}F$) to $1255^{\circ}K$ ($1800^{\circ}F$) Nickel and cobalt base
Thermal Fatigue	$422^{\circ}K$ ($300^{\circ}F$) to $1255^{\circ}K$ ($1800^{\circ}F$) cycles Nickel and cobalt base
Thermal Conductivity	Room temperature to elevated temperature
Thermal Expansion	Room temperature to elevated temperature

PERIOD OF PERFORMANCE

The samples would be tested over an 11-12-month period with the analyses concluding in the 15th month.

8.3.2.2 COMBUSTOR SCREENING TESTS

PURPOSE

To verify that the selected high-temperature combustor designs can adequately withstand the thermal shock and running temperatures associated with the near-term and full-potential engine environment.

DESCRIPTION

Based upon analytical calculations and previous experience, a basic candidate combustor design together with several modifications will be tested for the near-term and full-potential applications. The goal is to develop combustors to produce exhaust temperatures of 1477°K (2200°F) and 1588°K (2400°F), respectively, with satisfactory service life and reduced emissions levels. In order to meet the near-term engine goal of 3 years to production readiness, it is planned that the 1477°K (2200°F) combustor development will proceed as a non-coated design. As the thermal barrier application process is perfected, the 1588°K (2400°F) combustor development can proceed to the higher temperature level.

The near-term program will be structured to start testing with #2 oil and 1477°K (2200°F) firing temperature in such a manner as to screen out the inferior design concepts from an operating temperature standpoint.

The 1588°K (2400°F) combustor program would be structured in the same manner. However, it is recommended that testing with residual oil and #2 oil doped appropriately (vanadium and sodium) be incorporated in this program.

The combustors would be visually examined periodically during the tests to assess condition; and, in the case of equal visual appearances, sectioning would be used to adjudge the superior coating. Flaking, cracking, and crazing of the coating will be one of the evaluation criteria.

PERIOD OF PERFORMANCE

It is anticipated that the 1477°K (2200°F) combustor development will be completed within 2 years. The 1588°K (2400°F) combustor tests will start in the 20th month and continue for a 12-month period.

8.3.2.3 VANE TEMPERATURE CYCLING TEST

PURPOSE

To verify that the coating thickness and composition selection for use on the first stage vanes will endure the anticipated temperature cycling and thermal shock environments.

DESCRIPTION

Test vanes of a current type design would be vendor coated and instrumented per design engineering sketches. The test specimens would be placed in a high-temperature test rig and subjected to repeated thermal shocks in a sequence that simulates the starting and emergency shutdown of a gas turbine. At periodic intervals, the specimens would be visually examined for incipient failure. Cracking, peeling, or flaking of the coating will be considered as a failure. If the coating composition and thickness selected by design engineering is the same for the 1477°K (2200°F) application as the 1588°K (2400°F) and one test would suffice for both the near-term and full-potential engines. Obviously, two tests would be required if the 1477°K (2200°F) coating differs from the 1588°K (2400°F) application.

PERIOD OF PERFORMANCE

Testing can begin in the 13th month of the program and would continue for about two months. Evaluation of the results would take another one-half month.

8.3.2.4 BLADE VIBRATION/THERMAL TEST

PURPOSE

This test is to confirm the ability of the coating to withstand the high temperature/vibration environment associated with operation of a gas turbine over an extended life period.

DESCRIPTION

Coated second-stage current type blades (approximately 4) would be subjected to a elevated temperature/vibration environment for a sufficient number of cycles to simulate fatigue conditions on the specimen. It is planned that the blades would be heated to their bond operating temperature and vibrated at their fundamental natural frequency (approximately 500-600 Hertz). Tests would continue until the accrued cycles are felt to be a reasonable number for fatigue simulation, at least 10 million cycles, and enough blades will be tested to establish a S-N curve. Flaking, cracking, crazing, etc., will be considered as failure. Suitable instrumentation would be employed to record skin temperature. As is the case for the Vane Temperature Cycling Test, if the 1477°K (2200°F) coating and the 1588°K (2400°F) coating are different, both applications would require test verification.

8.3.2.5 TRANSITION PIECE THERMAL CYCLE TEST

PURPOSE

The purpose of this test is to confirm the ability of thermal coatings to withstand the high temperature cycling environment associated with the stopping and starting of gas turbines.

DESCRIPTION

Shortly after initiation of process development at candidate vendors' plants, a current design transition piece would be coated by the vendor showing greatest process/ control.

The transition piece would be installed in a combustion rig and subjected to 100 thermal cycles using #2 oil as fuel. A temperature cycle shall consist of flame ignition in the combustor and sustained burning at $1477^{\circ}K$ ($2200^{\circ}F$) exhaust gas temperature for approximately 15 minutes. At the end of this period the fuel flow would be shut off while airflow continues to provide a severe thermal shock.

A second transition test would be conducted approximately 6 months later wherein the coating under test would be applied using the process from a "qualified" vendor. This second test is to confirm adequacy of the final approved process. All test conditions and parameters would remain the same as the earlier test. Obviously, if the $1588^{\circ}K$ ($2400^{\circ}F$) coating is different from the $1477^{\circ}K$ ($2200^{\circ}F$) type, the tests would be repeated for the $1588^{\circ}K$ ($2400^{\circ}F$) case.

Instrumentation for the test will consist of approximately 10 skin temperature thermocouples, a downstream gas temperature rake, and thermal paint applied to the exterior of the transition piece.

8.3.2.6 FLYING OBJECT SIMULATED DAMAGE TEST

PURPOSE

To determine the ability of coated components to withstand impact from flying objects.

DESCRIPTION

Coated dummy blade shapes would be heated to an elevated temperature and struck with flying objects to assess the ability of the coating to withstand damage. Design studies will be required to establish object size, shape, and velocity plus the criteria for test temperature.

SCHEDULE

It is felt that early screening tests would be run within 6 months from go-ahead. Final verification tests involving vendors' coatings would be conducted in the 18-20-month time period.

8.3.2.7 LOCALIZED REPAIR TEMPERATURE CYCLING TEST

PURPOSE

To demonstrate that repaired coating surfaces can withstand thermal cycling.

DESCRIPTION

It is felt that damage to coated surfaces may occur during fabrication, assembly, and operation. On the assumption that a candidate repair technique will be developed, samples containing repaired areas would be subjected to thermal cycle tests such as described in subparagraph 8.3.2.3.

SCHEDULE

These tests would be conducted in the 20-24-month time period.

8.3.2.8 COATING INTEGRITY ASSURANCE DEVELOPMENT TESTS

PURPOSE

To develop the spallation detector system for use on 1588°K (2400°F) engine class blades/vanes.

DESCRIPTION

A means of detecting serious spallation on surfaces operating in a 1588°K (2400°F) turbine inlet temperature engine environment has to be perfected. Candidate detection schemes would be evaluated and tested as prototype hardware in simulated gas turbine operating conditions. The number and type of test(s) will depend on the candidate detection schemes.

SCHEDULE

This activity would occur in the 30-36-month time period.

8.3.2.9 SPALLATION DETECTION DEMONSTRATION TESTS

PURPOSE

To demonstrate that the Spallation Detection System will operate under actual gas turbine environments.

DESCRIPTION

The spall warning technique and the spallation detector developed under 8.3.2.8 would be combined in a full-scale test using the full-size Laboratory Test Gas Turbine Engine. The engine would be run with spalled components that have a vendor-installed spall warning system. It would be demonstrated that the spall detector can sense the presence of spalled hardware at full flow, temperature, acoustics, and vibrational environment.

C-2

8.4 PROCESS DEVELOPMENT

At this time, several commercial firms are available to apply thermal barrier coatings as per their respective processes and associated limitations. Depending on the specific coating types and configuration selected, work in the process area will be required at each candidate vendors' factory. In paragraph 8.1, it was mentioned that vanes, blades, combustor liners, and transition pieces are likely candidates for the application of thermal barrier material. It is felt that each of the components presents an area requiring process development, as discussed in the following subparagraphs.

8.4.1 Blades

Current design blades incorporate basic convection flow cooling systems with blade tip cooling air discharge. Advanced impingement/convection designs are anticipated to have in addition the use of coolant discharge holes along the trailing edge as well as on certain regions of the airfoil surface. The dimensions of these openings are critical. The materials and process technology developments would address the task of maintaining repeatable control of the opening dimensions. No extraordinary problems in achieving this goal are envisioned. A related problem and one which may pose considerably more difficulty is the question of developing an application process which maintains repeatable and accurate control of the coating thickness over the entire coated surface.

8.4.2 Vanes

The vanes are similar, in many respects, to the blades and would require control of spray thickness and contact area. However, a different problem exists in many current designs in that vanes are not individual shapes but are cast in a subassembly of multiple vanes joined at the hub and tip section with shrouds. This may prevent necessary access portions of the vane surface for spraying. One alternative would be to conduct a routine design modification of the vanes to provide for mounting of single vane elements. The necessity for such a move would have to be determined by the outcome of the process development program.

8.4.3 Transition Pieces

The transition pieces of current design are approximately three feet long and of compound curvature which may require process development to assure uniform coatings on the inside surfaces. This appears to be the major area of concern on the transition pieces.

8.4.4 Combustor Liners

It was mentioned earlier that a 1477°K (2200°F) combustor liner may not require the use of thermal barrier coatings while coating the 1588°K (2400°F) combustor liner seems a certainty. The process problems and limitations are as follows.

The inside surface of the combustor liner is the candidate area for coating application and is reasonably accessible. The process must guarantee that the numerous air holes are not blocked or reduced in size and that the coating is of uniform thickness. An automated process appears to be feasible on the combustors, but would be more difficult on the transition pieces.

8.4.5 Repair And Rework

It is anticipated that the coated surfaces may be damaged during assembly or handling and the extent to which localized repairs can be made, versus return to the vendors, has to be determined. As in the case of the prime coating process, it is necessary that the repairs be done in an approved, controlled manner.

8.4.5.1 COATING REMOVAL AND RE-APPLICATION

The problem of coating removal would have to be studied to determine:

(a) the process or processes to be employed;

(b) the criteria to be used in determining the necessity for coating removal;

(c) the criteria and/or standards for local (field) versus vendor removal.

In a similar manner, re-application of a coating would have to be studied to set the limits on vendor rework and, to what extent, if any, field re-application can be considered.

8.4.6 Specification Preparation

At the same time that gas turbine manufacturing technical people are working at vendors' facilities to develop controlled processes, the specifications to assure process compliance and process control have to be written. It appears that the following specifications would be needed:

- Raw Material Specification
- Coating Application and Base Material Preparation
- In-Process and Final Source Tests/Inspections
- Repair and Rework Techniques
- Coating Thickness Control and Measurement
- Vendor Selection and Process Qualification
- Coating Removal and Re-Application - Shop and Field

8.4.7 Vendor Selection And Qualification

Based upon the selected coatings, the process development, and the specifications, it would be necessary to work closely with several vendors and establish the means by which they (vendors) are initially qualified to apply thermal barrier coatings and the means by which they maintain qualification status. Batch sampling, random sampling and other compliance techniques will be explained and reviewed during the qualification procedure. Recommendations from the candidate vendors would be reviewed and, where approved, incorporated into applicable specifications.

8.5 FULL SCALE DEVELOPMENT AND FULL SCALE COMMERCIAL TESTS

The design and design support testing, together with the process development work, will result in a high level of confidence in the ability to fabricate coated components and for the components to survive stationary gas turbine operating environments. However, component level testing has limitations relative to the ability to create combined complex environments and operation of thermal barrier coated components in full scale gas turbines will significantly increase the confidence level. In Figure 38 is shown a listing of the full scale 1477°K (2200°F) near term and 1588°K (2400°) full potential tests recommended to be run in order to bring thermal barrier coatings to a state of production readiness. For reader clarity, the 1477°K (2200°F) and 1588°K (2400°F) tests will not be intermixed in the following subparagraphs.

NEAR-TERM	FULL POTENTIAL
• FULL-SCALE DEMONSTRATION - 2200° F - LIGHT PETROLEUM FUEL • ENDURANCE TEST - POSSIBLE 2200° F IF RETROFITABILITY PROVES TRUE - COMMERCIAL BASELOAD SERVICE WITH CLEAN, LIGHT PETROLEUM FUEL - RETROFIT TYPE HARDWARE	• ADVANCED TECHNOLOGY TEST - 2400° F - LIGHT FUEL AND HEAVY FUEL • ENDURANCE TEST - NOT TO EXCEED 2200° F - COMMERCIAL BASELOAD SERVICE WITH CLEAN OR HEAVY FUELS - NEW 2400° F HARDWARE • ADVANCED ENGINE TEST - 2400° F - NEW 501 TYPE ENGINE

RECOMMENDED FULL SCALE DEVELOPMENT AND FULL SCALE COMMERCIAL TESTS
Figure 38

8.5.1 Near Term Full Scale Tests

8.5.1.1 Full Scale Demonstration

A demonstration test, using a modified gas turbine test bed, is recommended as a necessary step in the development program. Such a facility, a W-251 gas turbine test bed, is located at the Westinghouse Development Center at Concordville, Pennsylvania, and has been used for engineering evaluation and final testing of developmental hardware. In order to obtain full scale data early, and expedite the 3 year program, it will be necessary to coat the blades and vanes as quickly as possible for this test. NASA-Lewis Research Center has the capability, knowledge, and facility to coat these parts well in advance of getting a commercial vendor qualified.

PURPOSE

The reason for conducting a full scale test in the W-251 engine at Concordville is to:

(a) demonstrate that the coated components and 1477^{o}K (2200^{o}F) combustors can function as a system when subjected to thermal shocks and limited operation in a full scale stationary gas turbine;

(b) obtain early performance data to substantiate the design assumptions to be used for fabrication of the endurance engine test components.

DESCRIPTION

Hardware - The hardware to be used for this test is as follows:

(80) first-stage blades

(40) first-stage vanes

(8) combustor liners primary and secondary

(8) transition pieces

TEST SEQUENCE

After the control circuits, control computer and data instrumentation have been checked out, the test would be started. The machine would be brought up to a predetermined idle speed and held for fifteen minutes while pre-run relationships for all instrumentation is established. Upon approval of the Test Director, the machine will be taken to the 4850 rpm nominal operating speed at 1477^{o}K (2200^{o}F) turbine inlet temperature and held for thirty minutes. After holding the machine the prescribed time, the power will be cut in a shutdown sequence, simulating operation in a utility application.

The test engine would then be allowed to cool and all data reviewed for approval to continue the test. The temperature relationship of coated blades and vanes, flows, and operating conditions will be part of the recorded data for each test sequence. Upon approval of the Test Director, the machine would again be started and brought to full speed/temperature conditions as rapidly as possible with, as described before, repeat the steady state shutdown sequence. A total of one hundred start/stop sequences would be performed and, after the final shutdown, the barrier coated components would be removed for final examination. It is expected that visual examinations would be adequate for confirmation of satisfactory performance during the course of accumulating the 100 sequences.

8.5.1.2 ENDURANCE TEST

With satisfactory performance in the demonstration test, the remaining environment not adequately simulated is the long term, or endurance, condition. A successful endurance test in a large machine such as the W501 at a utility site is felt to be necessary in order to assure production readiness. An endurance test could be run on the test machine in the laboratory environment, but it is felt considerable monetary savings will result if a utility machine is used plus the commercial operation of the utility machine removes all questions of performance in "real-life" conditions.

PURPOSE

(1) To demonstrate satisfactory long term operation of thermal barrier coated hardware in a commercial operation.

(2) Complete the conditions necessary to bring thermal barrier coated hardware to a state of "production readiness."

DESCRIPTION

In parallel with the design support tests (at about the16th month of the program), negotiations would begin with utility companies to obtain permission to install barrier coated hardware on one of their machines. Assuming an agreement can be reached with a utility, a complete set of hardware would be ordered, then shipped to the utility site, and installed during a regularly-scheduled down period.

It would be desirable to operate the machine with "dirty" fuel or at a site with local ambient air problems, viz. adjacent to a body of salt water. However, the exact parameters will depend on the selected site and interface agreements with the particular utility. In a similar fashion, the total number of hours on test, period of inspections, etc., will be subject to negotiation.

PERIOD OF PERFORMANCE

It is recommended that the endurance test be started as soon as the combustor tests are satisfactorily completed and candidate vendors are able to produce controlled, coated hardware. Hopefully, three to four thousand hours of operation could be accumulated within three years from "Go-Ahead." Visual examination and overhaul would be performed as part of the utility's normal sequence.

8.5.2 Full Potential Full Scale Tests

8.5.2.1 ADVANCED TECHNOLOGY TEST

PURPOSE

(a) To demonstrate that the 1588°K (2400°F) coated components can perform satisfactorily in an actual gas turbine environment.

(b) To confirm that the specifications and quality procedures used in the manufacture of components for this test are complete and adequate for Technology Readiness consideration.

DESCRIPTION

Hardware - The coated hardware to be used for this test is as follows:

(80) first-stage blades

(70) second-stage blades

(48) second-stage vanes

(40) first-stage vanes

(8) combustor liners - primary and secondary

(8) transition pieces

TEST SEQUENCE

The test would be performed in the Engineering Test Engine and consist of 100 thermal cycles from room temperature to 1588°K (2400°F) and return. Engine starts and shutdowns would utilize #2 fuel with operation on heavy fuel for normal cycle running. Temperature profiles and thermal gradients would be measured to verify design calculations. Emissions levels would be measured for combustion design data. All coated components would be visually examined before and after the test to confirm satisfactory performance.

SCHEDULE

The hardware and test bed engine can be ready to begin testing in the 48th month. It is expected that the test would continue for approximately 4 additional months.

8.5.2.2 FULL POTENTIAL ENDURANCE TEST

PURPOSE

(a) To demonstrate satisfactory long time operation of thermal barrier coated hardware in commercial operation.

(b) Complete the conditions necessary to bring thermal barrier coated hardware to a state of "technology readiness" for 1588°K (2400°F) operation.

DESCRIPTION

To complete the conditions required for technology readiness, the coating composition and usage selected for the 1588°K (2400°F) engine will require long term, or endurance testing.

Even though existing machines cannot be operated at 1588°K (2400°F) turbine inlet temperature without major design changes (for cooling purposes), it is recommended that a target of 6-8000 hours be accumulated in commercial base load operation at approximately 1477°K (2200°F). A study would be made to determine the highest safe inlet operating temperature utilizing coated hot gas stream components. It is felt 6-8000 hours will reveal any potential long term problems.

To accomplish this endurance goal, a full set of coated hardware would be installed in a utility company's machine and operated by utility personnel. The hardware would be shipped to the utility site and installed during a regularly scheduled down period.

PERIOD OF PERFORMANCE

It is recommended that the endurance test be started in parallel with the technology advancement test in order to accumulate hours in the most expeditious manner. Six to eight thousand hours of operation in a year as a target dictates base load operation. Visual exams and overhaul would be performed as part of the utility's normal sequence. The start of testing would be near the 52nd month of the program and yield representative results by the 64th month. At that time, a program decision will be made to continue operating the hardware.

8.5.2.3 ADVANCED ENGINE TEST

Completion of the endurance test (paragraph 8.5.2.2 above) places the use of thermal barrier coatings in a state of technology readiness. To demonstrate production readiness, for the 1588°K (2400°F) engine, a new machine, such as an advanced W501 incorporating thermal barriers in the 100MW plus class, should be installed and placed in operation at a utility site. This advanced engine, (not to be confused with the new advanced design high temperature engine described in Section 4.1), should be operated at 1533-1588°K (2300-2400°F) and periodically visually inspected by engineering personnel at regular intervals.

PURPOSE

(1) To demonstrate production readiness for thermal barrier coatings on stationary gas turbines.

(2) To operate a 100MW class gas turbine with thermal barrier coated components at turbine inlet temperatures at or near 1588°K (2400°F).

DESCRIPTION

After the endurance test (paragraph 8.5.2.2) has successfully operated for 2-3000 hours, negotiations would be started with a utility on the basis that a new type machine, such as an advanced W501, equipped with thermal barrier coated components, would be placed in operation at their site. Assuming agreement is reached, hardware would be fabricated and delivered to the utility site.

PERIOD OF PERFORMANCE

For the summary schedule on page 4, it can be seen that the advanced engine would go into service about month 60, or 5 years, after program go-ahead. It is expected that a significant number of operating hours will be accrued in the first year of operation and production readiness will be an established fact by the end of the sixth year.

SECTION 9
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APPENDIX—SYMBOLS AND ABBREVIATIONS

CF - Capacity Factor, Hours of Operation/Hours per year (8760)

COE - Cost of Electricity, mills/Kw-Hr

Cooling Effectiveness - $\dfrac{T_{GAS} - T_{AVG\ WALL}}{T_{GAS} - T_{COOLANT}}$

where T_{GAS} - Relative gas stagnation temperature, ^{o}K (^{o}F)

$T_{AVG\ WALL}$ - Average metal wall temperature, ^{o}K (^{o}F)

$T_{COOLANT}$ - Temperature at which cooling air enters blade, ^{o}K (^{o}F)

CPT - Current Production Type Gas Turbine Engine

h - Convective heat transfer coefficient

HHV - Fuel higher heating value

i - Cost of money, percent

ISO - Standard ambient air conditions, $288^{o}K$ ($59^{o}F$), 1 atmosphere, 60% relative humidity

k - Thermal conductivity, W/m-K, (Btu/hr ft ^{o}F)

n - Number of years of component life

NTA - Near-Term Advanced Type Gas Turbine Engine

O&M - Operating and Maintenance cost

Q/A - Heat Flux, W/m^2, (Btu/hr ft^2)

S - Component renewal parts price, $

TBC - Thermal Barrier coating

T_{IT} - Turbine Inlet Temperature, ^{o}K (^{o}F), - In this report T_{IT} is that temperature immediately upstream of the first stationary vane row.

ΔG_c - Percent reduction in cooling airflow usage for turbine element blades and vanes

ΔHR - Percent improvement in powerplant heat rate. Customary units for heat rate are Btu/Kw-Hr

ΔSP - Percent improvement in powerplant specific output. Customary units for gas turbine specific output are Kw/Lb/Sec

ξ_R - Recuperator effectiveness

$$\xi_R = \frac{T_{COMBUSTOR\ INLET} - T_{COMPRESSOR\ DISCHARGE}}{T_{TURBINE\ EXHAUST} - T_{COMPRESSOR\ DISCHARGE}}$$

η_{HHV} - Higher heating value thermal efficiency

Subscript

j - Specific turbine element component under consideration

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