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Heat Management in Advanced Aircraft Gas Turbine Engines

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ABSTRACT

This paper summarizes current and projected heat management techniques within aircraft gas turbines. Each of the primary heat sources is individually considered, including the thermodynamic cycle sources of fuel combustion and air compression, and the parasitic sources of friction, oil churning and rotor windage. For each source, the problem is presented, and solutions -- both present and future -- are offered. Heat management within low spool reduction gearing is also discussed because of the high probability of an advanced high speed turboprop and geared high bypass ratio turbofan propulsion system in the future.

INTRODUCTION

Today, the modern aircraft gas turbine is designed to contain up to the equivalent of 165,000 horsepower of energy released by the combustion of fuel within a flow annulus of only 452 cm^2 (70 in^2) at the combustor discharge. This is just one dramatic example of the challenge of heat management facing today's designers. Designers of tomorrow's engines face the spector of even greater overall heat loads generated within the components of the engine while, simultaneously, facing the additional requirement of even tighter thermal clearance control throughout the engine necessary for improved component efficiencies.

The goal of providing lower and lower fuel consumption in advanced engines on a cost-effective basis dictates increased overall compression ratios, higher combustor exit temperatures and increased rotor speeds. These factors, in addition to the parasitic sources of friction, oil churning and rotor windage, combine to present considerable challenges in managing the heat generated within the relatively small volume of the engine while utilizing only two primary heat sinks available: air and fuel.

To address the design challenges in heat management, development of more sophisticated computerized analytical design systems and thermal model simulations is being pursued. Advanced instrumentation capability is providing comprehensive design information from rig testing to calibrate and verify these new analytical tools.

DISCUSSION

Figure 1 illustrates two different fuel efficient propulsion concepts being studied for future transports: the high bypass ratio geared turbofan and the high speed geared Prop-Fan. Cycle studies have indicated that overall pressure ratios of 40:1 to over 60:1 and combustor exit temperatures in the range of 1427-1538°C (2600-2800°F) will be required to maximize fuel efficiency. Bypass ratio for the geared turbofan could be as much as 12:1 while the Prop-Fan bypass ratio could be 80:1. The result is smaller core engines to provide the required cooling air. The pusher Prop-Fan configuration has the additional heat management problem of the turboprop engine exhaust being ducted around the prop drive reduction gearbox and exhausted through the root of the Prop-Fan blades.

The turbofan configuration shown in Figure 2 illustrates in more detail the regions of the engine that provide the major heat management concerns. They are the reduction gearing for the fan and associated air/oil heat exchanger, the bearing compartments, the rear section of the high compressor, rotor windage, the combustor and the active clearance control systems in the high pressure compressor and the turbines. These

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regions are presently, and will continue to be, the major challenges in heat management.

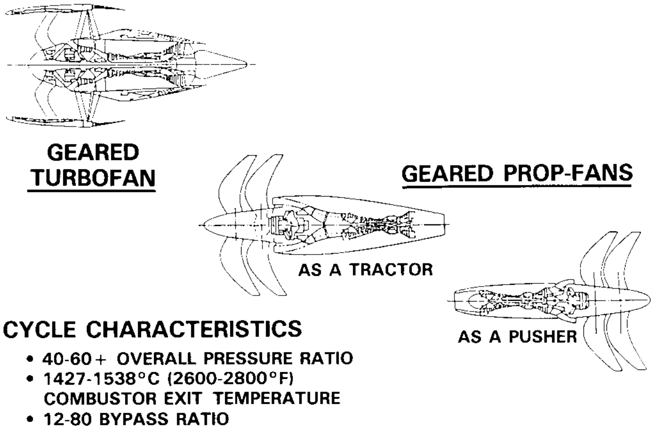

FIG. 1 FUTURE FUEL EFFICIENT PROPULSION SYSTEMS

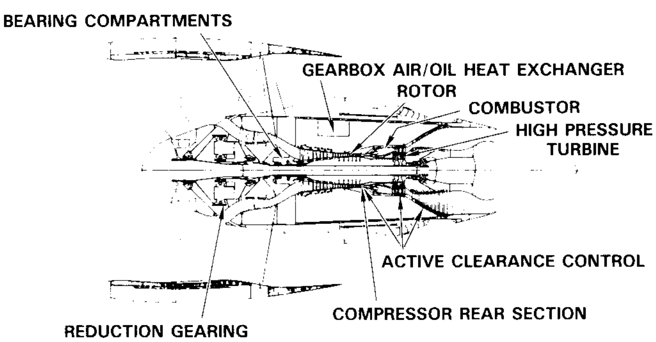

FIG. 2 REGIONS OF HEAT MANAGEMENT

The sources and sinks for heat flow in a gas turbine are listed in Figure 3. The engine regions creating the major sources of heat flow were illustrated in Figure 2. The compression system and combustor combine into the thermodynamic sources of air compression and fuel combustion. The reduction gearing, oil cooler, active clearance control and rotor geometry make up the major parasitic loss sources due to friction, oil churning, core oversizing and windage. The heat sinks available are primarily the airflow passing through the engine, the fuel used in combustion as well as fuel recirculated back to the fuel tanks and the engine parts, themselves. In each section of the engine, the operating environment must be controlled considering parts life requirements and failure modes. Decisions must be made among material advancement, or better cooling or improved thermal barrier coatings.

SOURCES

- **THERMODYNAMICS**
  - FUEL COMBUSTION
  - AIR COMPRESSION
- **PARASITICS**
  - FRICTION
  - OIL CHURNING
  - WINDAGE
  - OVERSIZED CORE

SINKS

- **AIR**
  - AIR/OIL COOLER DRAG
- **FUEL**
  - THERMAL STABILITY
  - FUEL COKING POTENTIAL
- **ENGINE PARTS (TRANSIENTLY)**
  - LIFE
  - DURABILITY/RELIABILITY
  - CORROSION/OXIDATION

FIG. 3 HEAT SOURCES AND SINKS

The performance losses associated with heat generation are minimized by consideration of the overall heat management problem in the earliest phases of the preliminary design process. Design variables that must be considered include oversizing the compressors to provide necessary cooling air at various temperatures and pressures, location of air/oil coolers to minimize drag, maximum fuel use as a heat sink with careful consideration of the thermal stability and coking limitations, and the selection of materials and coatings for engine parts that balances allowable gaspath temperatures with increased parts cost and parts life.

The thermal environment is controlled with extensive secondary airflow systems in current aircraft engines. The secondary airflow system is comprised of any flow of air in a gas turbine engine outside of the main flowpath of the engine. Examples are airfoil cooling, disk cooling, bearing compartment buffer air and leakage air. As an example, Figure 4 illustrates the extensive secondary airflow system to control the thermal environment of the PW2037 engine. In the PW2037, air is taken from the flowpath at a number of locations and used in heat management as well as a number of other tasks.

Fifth stage bleed air is introduced into the center of the low spool shaft and ducted rearward through the entire length of the shaft, passing under the last two low pressure turbine bores prior to mixing with the turbine exhaust.

Air is injected radially inward into the high pressure compressor rotor assembly at the 13th stage. A major task for this air is to flow rearward to cool the low pressure turbine first stage disk bore. This airflow also increases the thermal response of the high pressure compressor (HPC) rotor during transients.

Air is bled from the 14th stage of the high pressure compressor and provides cooling of the high pressure turbine second vanes, second outer airseal and the high pressure turbine labyrinth seal assembly.

PW2037 SECONDARY FLOW SYSTEM

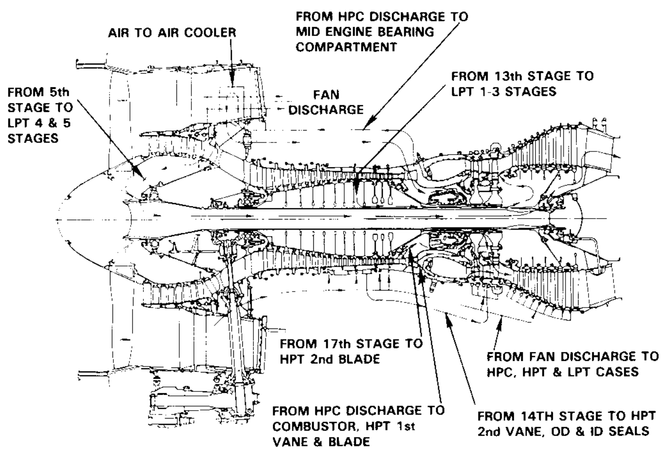

FIG. 4 PW2037 SECONDARY FLOW SYSTEM

Air taken from the 17th stage of the high pressure compressor is injected radially inward and flows rearward to provide cooling for the high pressure turbine's second stage blades and also provides improved thermal responsiveness of the high pressure turbine disk bores during transients.

High pressure compressor discharge air provides all the cooling to the combustor as well as the high pressure turbine inlet nozzle guide vanes and the high pressure turbine first stage blades. In addition, this air is used to "buffer" the bearing compartment located below the combustor from the high temperature, high pressure air surrounding the compartment. Since the HPC discharge air temperature is in excess of the auto-ignition temperature of the air/oil environment of the bearing compartment, it must be cooled prior to its introduction into the buffer system. An air-to-air cooler located in the fan discharge airstream within a fan exit case strut is used to reduce the HPC air temperature for use in the bearing compartment.

Fan discharge air bleed is also collected in a manifold in the fan exit case and ducted to the active clearance control system used to control blade tip clearances in the last half of the high pressure compressor and in both the high and low pressure turbines.

The overall objective is to establish a system that maximizes the efficiency of the airfoil cooling air supply, minimizes disk rim and cavity cooling requirements and leakage, and reduces airfoil tip leakage. In the state of the art for high temperature, high pressure ratio engines, the secondary flow system draws approximately 25% of engine core mass flow. Approximately two-thirds of this flow is required for the cooling of turbine airfoils. Internal cooling and leakage account for the remaining airflow.

Significant leakage also exists past compressor and turbine airfoil tips due to clearance increases as engine power conditions deviate from design conditions. Reduction of these leakage flows results in direct compressor and turbine efficiency and engine cycle benefits.

Advanced fuel efficient cycles now under study indicate a more severe thermal environment with which the designer will have to cope. Higher operating temperatures throughout the engine will result from higher overall compressor pressure ratios and higher combustor exit temperatures. In addition, the need to drive down the engine price by reducing parts count will require up to a 30% increase in high spool rotor speed. Increased rotor speeds will increase bearing compartment temperatures, rotor windage losses and the propensity for oil churning.

Figure 5 shows the projected increase in the Mach 0.8 cruise thermal efficiency as overall cycle pressure ratio is increased for engines targeted for the 1990's and beyond. Projections show increases in overall cycle pressure ratio as much as 100% over engines in service today. As overall pressure ratio is increased, compressor exit temperatures are projected to steadily increase to a level of over 704°C (1300°F) as illustrated in Figure 5.

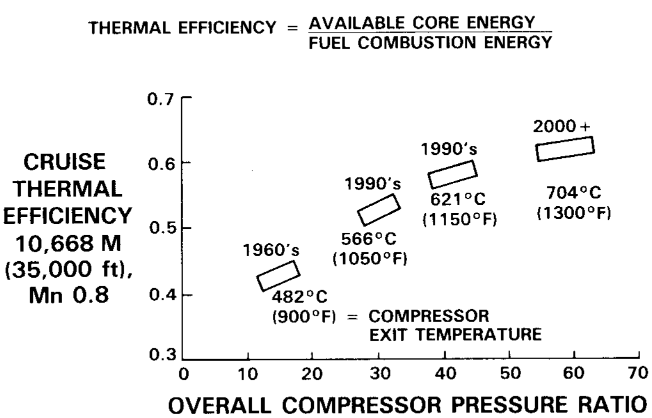

FIG. 5 THERMAL EFFICIENCY AND OVERALL PRESSURE RATIO TRENDS

High pressure turbine vane cooling air is bled from the high compressor exit since air at the highest possible pressure is required for showerhead cooling. It is apparent, that as compressor exit air temperature increases, the turbine cooling task becomes more difficult. To put this in perspective, compressor discharge temperatures will begin to approach the combustor exit temperature of the early commercial turbojets.

The turbine cooling task becomes even more difficult due to the higher combustor exit temperatures required for improved cycle performance. Figure 6 shows the trend of maximum combustor exit temperature with time for both commercial and military engines. The commercial engine trend in the future shows a steady increase in combustor exit temperature that is optimum for maximum cycle efficiency. A more rapid military engine trend is projected to maximize specific power (i.e., thrust per pound of airflow to minimize engine weight).

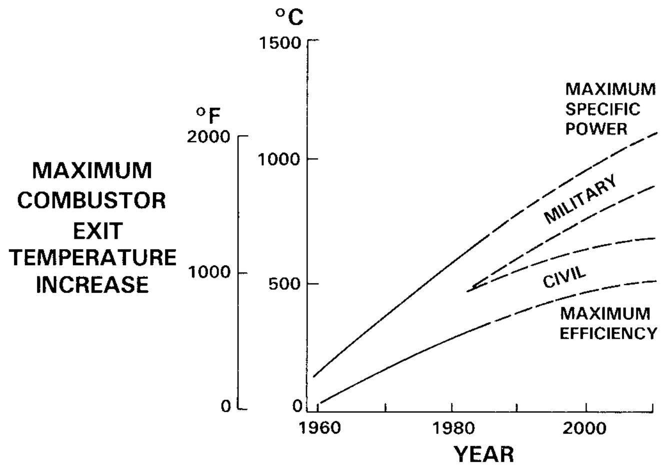

FIG. 6 COMBUSTOR EXIT TEMPERATURE TRENDS

A typical increase in gaspath air temperature through the engine is illustrated in Figure 7 for the high bypass ratio geared turbofan. It is interesting to note that nearly one-half of the maximum temperature increase occurs during the compression process. Therefore, as compressor pressure ratio increases, it is necessary to substitute higher temperature materials for the compressor blades, vanes, disks and cases. Higher strength nickel base alloys and higher temperature capability titanium alloys are being developed. Figure 8 shows the increase in temperature capability of compressor rotor disk materials with time. In addition, in order to minimize thermal mismatch (clearance) between the rotating blades and the compressor case, active clearance control is employed by impinging low pressure compressor bleed air on the outside of the rear potion of the high compressor case to maintain high aerodynamic component efficiencies.

The major design challenge for future combustors is the substantially higher pressure and higher temperature operating environment. At these conditions, the combustor must be durable and also meet emissions standards. In the definition of the Energy Efficient Engine (E^3) combustor, the high cycle pressure ratio of 38:1, coupled with a requirement to limit liner cooling air to 30% and maintain a life goal of 8000 hours, precluded use of current liner designs and cooling systems. A major improvement in combustor liner design resulted from this challenge.

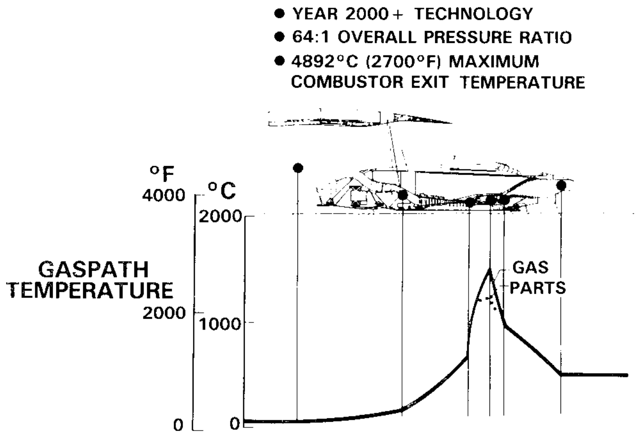

FIG. 7 FUTURE COMMERCIAL ENGINE GASPATH TEMPERATURES

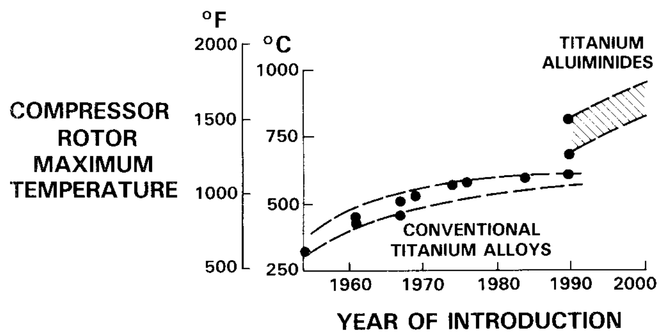

FIG. 8 COMPRESSOR ROTOR MATERIAL IMPROVEMENTS

The combustor liner was designed as a segmented structure with a unique Counter-Parallel FINWALL® cooling scheme. In this concept, compressor discharge air first convectively cools the back of the liner, reverses direction and then film cools the hot side. The segmented design was chosen to prevent cracking by allowing unrestrained thermal expansion, both axially and circumferentially.

This work has also provided the foundation for continued research and development of nonmetallics for future combustor applications. The history of the increase in liner temperature capability is shown in Figure 9. This projected increase in capability is absolutely essential to permit the projected increase in overall cycle pressure ratios.

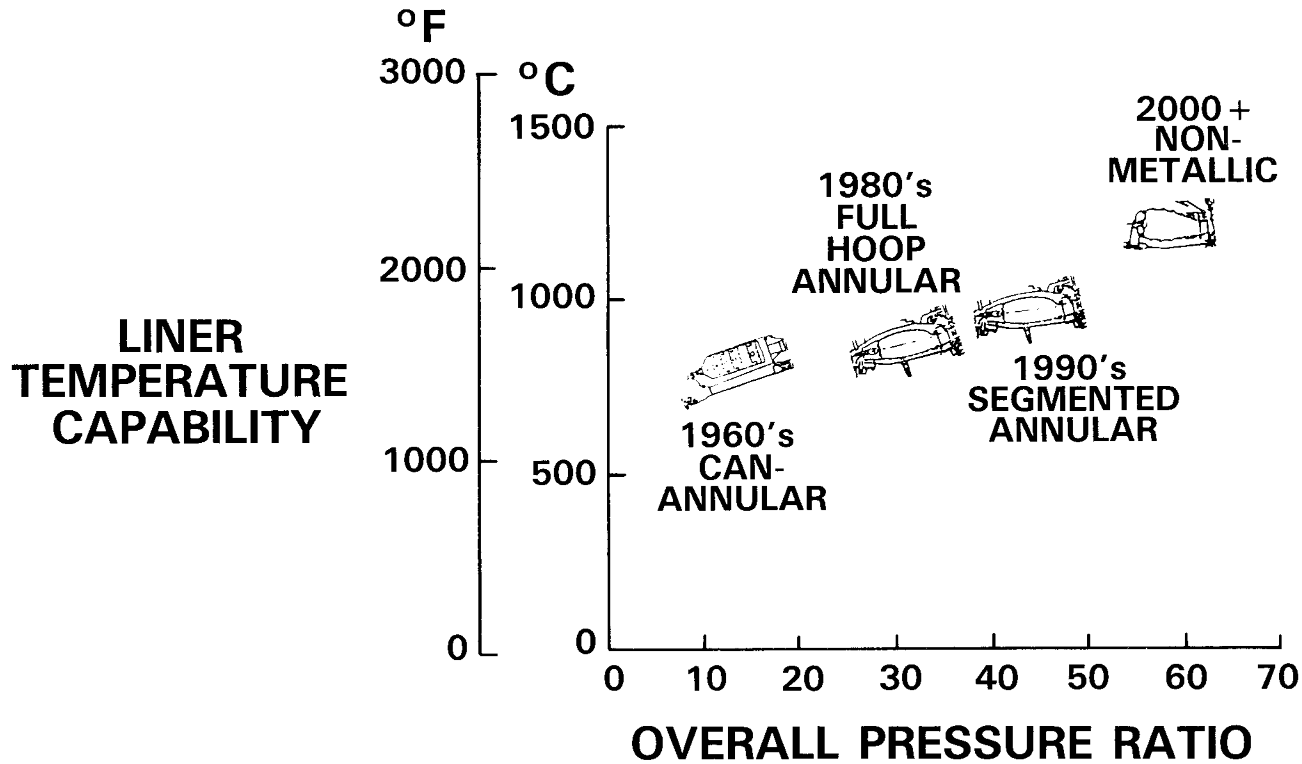

FIG. 9 COMBUSTOR LINER TEMPERATURE AND OVERALL PRESSURE RATIO TRENDS

A key ingredient in the ability of the turbine to withstand increased combustor exit temperatures is the uniformity of the combustor exit temperature profile or pattern factor. The more uniform the combustor exit temperature profile is, the lower the temperature spikes are that can cause hot spots in the turbine. In order to keep pattern factors acceptable as combustor exit temperature increase, full annular combustors with multiple fuel nozzles were developed and they provided a significant advancement over the earlier can-annular designs.

Turbine materials in modern aircraft engines reflect the need for higher strength and higher temperature capabilities to meet the continuously increasing operational demands of gas turbine engines.

The introduction of single crystal material for turbine airfoils has provided a major step in achieving higher temperature capabilities without an increase in cooling requirements. A single crystal alloy eliminates grain boundaries to enhance the basic creep and stress rupture strength. The material has nearly a 55°C (100°F) advantage over earlier directionally solidified materials, and the potential is far from exhausted.

Figure 10 shows the increase in temperature capability of turbine airfoil material. It is obvious that ceramics offer a dramatic increase in temperature capability. However, the material properties of ceramics present significant challenges that must be solved before ceramic airfoils can be used in aircraft gas turbine engines.

Corrosion- and oxidation-resistant coatings for airfoils extend airfoil life and, ultimately, decrease engine performance degradation. Experimental endurance evaluations confirm that the latest coatings are at least twice as durable as earlier coatings.

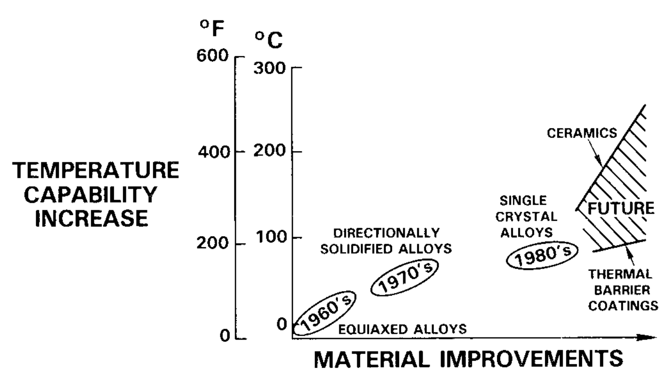

FIG. 10 TURBINE AIRFOIL MATERIAL IMPROVEMENTS

Despite material and coating improvements, the gas temperature is increasing at a faster rate than the capability of the basic turbine blade material to withstand it. Figure 11 compares the increase in high pressure turbine blade material temperature capability with that of the temperature of the gas passing through the turbine. This increasing gap is closed by higher cooling effectiveness designs for the blades and vanes as well as thermal barrier coatings applied to the surface of the airfoils. Nonmetallics, such as ceramics, are being investigated to permit higher combustor exit temperatures without the cost and complexity of complex castings of cooling passages in exotic metal alloys. Figure 12 shows the increase in turbine airfoil cooling effectiveness with time. The turbine airfoil cross section shown in Figure 12 illustrates the sophisticated cooling passages that can be cast in airfoils today. Future airfoils will require even more sophisticated cooling systems employing a high density film if cooling levels are to be controlled.

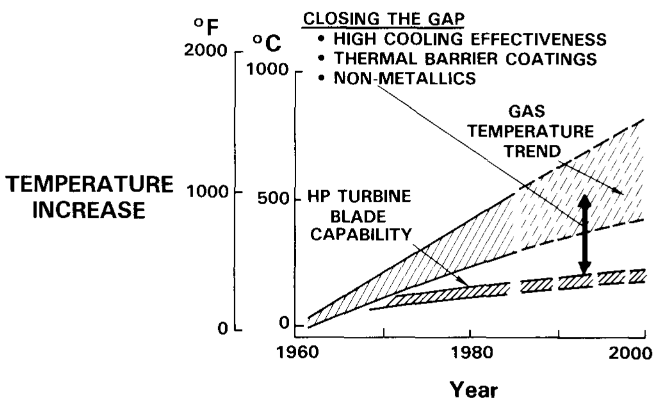

FIG. 11 GASPATH TEMPERATURES INCREASING FASTER THAN TURBINE BLADE MATERIAL CAPABILITY

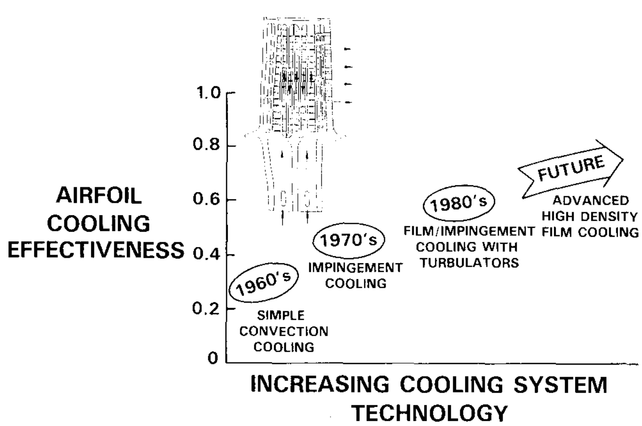

FIG. 12 TURBINE AIRFOIL COOLING EFFECTIVENSS TREND

PARASITIC HEAT GENERATORS

A much smaller, but still important, part of heat management is the parasitic heat generation which can add up to several hundred horsepower loss. Bearing compartments and accessory gearboxes generate heat by friction and oil churning. Rotor windage is also a source of heat generation. Main power transfer reduction gearboxes will also become increasingly important in future generations of aircraft gas turbines.

Figure 13 illustrates the heat management problem for a typical bearing compartment. Heat sources include bearing and seal friction, oil churning and radiation into the compartment from the surrounding high temperature engine parts. Techniques used to reduce the heat load are heatshielding and insulation, cold air buffering, improved bearing configurations, improved lubricity lubricants and improved oil scavenging.

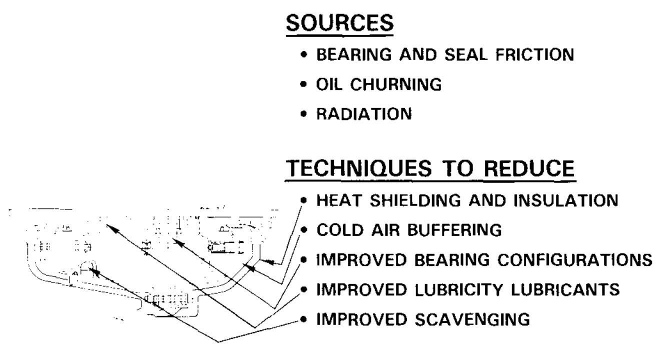

FIG. 13 BEARING COMPARTMENT HEAT GENERATION AND CONTROL

Advanced engines demand lightweight lubrication systems and minimal heat rejection as engine pressure ratios, operating temperatures, and rotational speeds are increased in the ongoing quest for improved thrust-to-weight ratios and engine performance. The simplification of buffer systems or the elimination of buffering will be required to reduce weight and improve fuel consumption. Reduced heat generation from oil churning in bearing compartments and gearboxes, and improved oil delivery systems will be needed.

These reductions offset the tendency for an exponential increase in bearing heat generation as bearing speeds increase. Improved shaft sealing, together with reduced seal pressure drop achieved by increased compartment pressure, is required.

Figure 14 illustrates the heat-generation problems associated with rotor windage for both compressors and turbines. The main sources of heat generation are cavities within the flowpath, protrusions and multiple rotating surfaces. The techniques being employed to reduce the heat generation include drum rotors that virtually eliminate the compressor disk joints, minimal cavities, reduced number of rotating parts, windage covers and improved air tangential velocity control for introducing secondary cooling air into the rotating turbine disks.

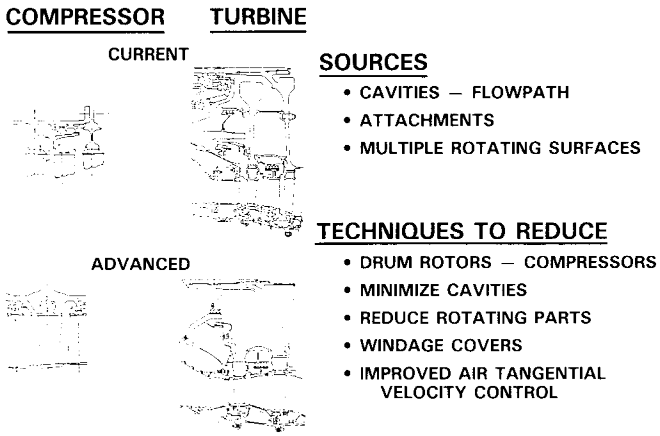

FIG. 14 ROTOR WINDAGE HEAT GENERATION AND CONTROL

Higher bypass ratios provide improved propulsive efficiency as shown in Figure 15. Beyond the region of a bypass ratio of ten, it will be necessary to drive the propulsor unit (Prop-Fan or geared turbofan) through a reduction gearbox in order to operate both the propulsor and the drive turbines at their most efficient speeds. Reduction gearing heat sources and techniques to reduce heat generation are listed in Figure 16. Major heat sources are gear tooth friction, bearing friction, oil churning and windage.

High-strength/high-contact ratio tooth forms will be used to reduce heat generation from tooth friction. Other techniques include high efficiency aerodynamic oil scavenging, a modulated lubrication supply system, simplified gear elements and oil baffling and improved high temperature lubricants.

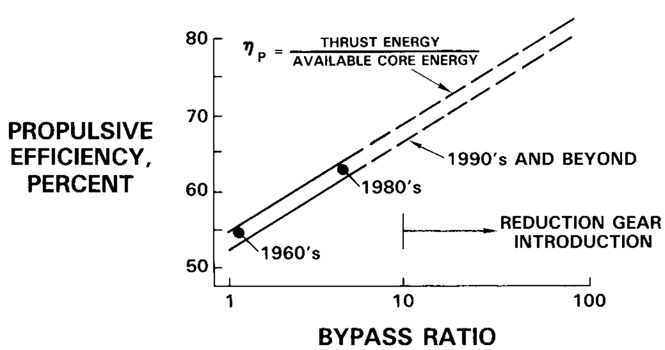

FIG. 15 PROPULSIVE EFFICIENCY AND BYPASS RATIO TRENDS

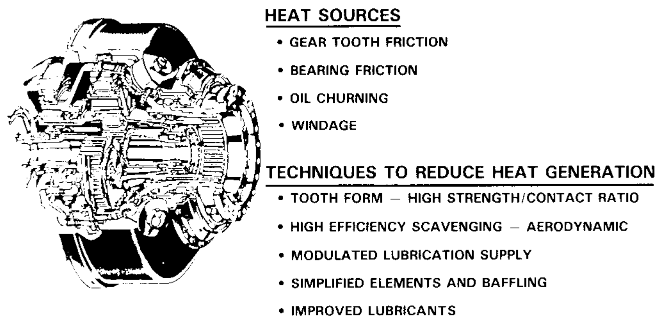

FIG. 16 REDUCTION GEAR HEAT GENERATION AND CONTROL

One example of the heat load in a reduction gearbox is illustrated by the numbers associated with the 12,000 horsepower gearbox for a 150-passenger Prop-Fan transport. There are 89,658 J/second (5,100 Btu/minute) of heat transferred to the oil. This requires an oil flow rate of 95 kg/min (210 lb/min). The oil must be piped to a remote air/oil heat exchanger. A very efficient heat exchanger is required to reduce the overall frontal area and length and, therefore, the aerodynamic drag of the cooling air.

SUMMARY

From this brief description, it is obvious that the entire propulsion system package is involved in the heat management problem. As the drive for lower and lower fuel consumption continues, it is apparent that successful heat management will be the key to improved efficiency engines.

To accomplish this, advanced technology will be required in innovative design techniques utilizing advanced computer design systems, high temperature materials, casting techniques and improved cooling effectiveness methods.