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SMALL GAS TURBINE ENGINE TECHNOLOGY

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ABSTRACT

Performance of small gas turbine engines in the 250 to 1,000 horsepower size range is significantly lower than that of large engines. Engines of this size are typically used in rotorcraft, commutercraft, general aviation and cruise missile applications. Principal reasons for the lower efficiencies of smaller engines are well known: Component efficiencies are lower by as much as 8 to 10 percentage points because of size effects. Small engines are designed for lower cycle pressures and temperatures because of smaller blading and cooling limitations. The highly developed analytical and manufacturing techniques evolved for large engines are not directly transferrable to small engines. Thus, it has been recognized that a focused effort addressing technologies for small engines was needed and could significantly impact their performance. Recently, in-house and contract studies were undertaken at the NASA Lewis Research Center, under joint NASA/Army-AVSCOM sponsorship to identify advanced engine cycle and component requirements for substantial performance improvement of small gas turbines for projected year 2000 applications. This paper presents the results of both in-house research and contract studies, conducted with Allison, AVCO Lycoming, Garrett, Teledyne CAE and Williams International. Rotorcraft results are emphasized. In summary, projected fuel savings of 22-percent to 42-percent could be attained. Accompanying direct operating cost reductions of 11-percent to 17-percent, depending on fuel cost, were also estimated. High payoff technologies are identified for all engine applications, and recent results of experimental research to evolve the high payoff technologies are described.

INTRODUCTION

Small gas turbine engine performance in the 250 to 1,000 horsepower size class is significantly lower than that of large engines. The major reasons for this are generally well known and consist of lower component efficiencies, poorer cycle efficiencies due to lower operating pressures and turbine inlet temperatures, and the difficulty of transferring large engine derived improvements to small engine sizes. The purpose of this paper is to define those technologies required to significantly improve small engine performance by the year 2000, emphasizing rotorcraft applications. Source materials for defining these technologies were obtained from NASA Lewis Research Center in-house research and contract studies conducted under the Small Engine Technology (SECT) Program under joint NASA/Army-AVSCOM sponsorship.

Small turbine engines are used in a broad spectrum of aeronautical applications including rotorcraft, commuters, general aviation and cruise missiles. Rotorcraft are used extensively by the military to rapidly deploy troops and supplies in ground battles and perform attack, scout, search and rescue missions. Civil helicopter uses include commuters, traffic/police observation, off-shore oil rig transport, airborne rescue and ambulance, agricultural, and other utilitarian roles. These applications require 350 - 1500 SHP turboshaft powerplants.

Turboprop commuters serve to efficiently transport small groups of people over short distances. They frequently interconnect with long-range commercial air transport carriers and thus provide an essential link in our national air transportation system. Commuter engines generally range from about 800 SHP upwards to about 2000 SHP.

General aviation aircraft provide a variety of public services: Flying people and freight; surveying and mapping natural resources; seeding and treating crops; patrolling pipelines, forests and fisheries; mineral prospecting; rescue and ambulance services; flight training and other such business and commercial purposes. These fixed-wing airplanes employ piston and turboprop engines in the 150 to 1000 SHP class.

Cruise missiles deliver a military warhead to a target at supersonic/subsonic speeds over a range of less than a hundred to several thousand miles, depending on the mission. These generally use a single turbofan/turbojet engine in the 200 - 1000 lb thrust class.

These broad ranges of applications and mission requirements lead to a variety of technology needs. However, there is considerable commonality in terms of needs between the various applications: These are:

1. Expanded vehicle capability in terms of range, speed and/or payload.

2. Reduced costs, both initial and operational.

3. Increased logistic flexibility relating to reduced demands for logistic support such as consumables.

4. Improved survivability, primarily a military concern.

Approaches for satisfying these technology needs are described in this paper, with special emphasis on rotorcraft applications.

MULTIPLE MISSION NEEDS

As shown in Figure 1, small turbine engines are used in a broad spectrum of aeronautical applications including rotorcraft, commuters, general aviation and cruise missiles. The broad range of applications and mission requirements has led to a variety of technology needs. However, there is considerable commonality between the various applications. These mission needs are summarized in the following table:

MULTIPLE MISSION NEEDS

ROTORCRAFT	COMMUTER AIRCRAFT
° EXPAND VEHICLE CAPABILITY ° INCREASE LOGISTIC FLEXIBILITY ° REDUCE OPERATING COST ° IMPROVE SURVIVABILITY	° EXPAND VEHICLE CAPABILITY ° REDUCE OPERATING COST
GENERAL AVIATION	**CRUISE MISSILES**
° EXPAND VEHICLE CAPABILITY ° REDUCE INITIAL AND OPERATING COSTS	° EXPAND VEHICLE CAPABILITY ° IMPROVE SURVIVABILITY

Expanded Vehicle Capability:

This refers to an extension of vehicle operating limits in terms of range, speed and/or payload. An improvement in the propulsion system can be used to maximize any one of these parameters depending on mission requirements. An important feature of rotorcraft propulsion systems, unlike the other propulsion systems, is that they are required to operate at power levels near 50-percent of rated power for as much as 75-percent of the time for typical missions. Thus, for these systems, it is of major importance that the efficiency at part power be significantly increased to realize large reductions in fuel burned and, therefore, increased range and/or payload. Accomplishing this will depend to a large extent on the application of variable geometry in small size turbomachinery, particularly in the power turbine. Present practice is to minimize the amount of variable geometry required in the compression system by limiting the cycle pressure ratio, and to provide no variability within the turbine. This is a result of trying to keep cost and complexity to a minimum. Highly advanced small turbine engines will require novel and innovative concepts that minimize the cost and complexity of variable geometry if full benefit is to be achieved in the advanced cycles.

Reduced Cost:

This includes both initial and operating cost. While all users are interested in reduced costs, generally system users with high utilization rates tend to be more concerned with operating costs as opposed to users with low utilization rates, who are more concerned about initial cost. Users with high utilization rates may be more willing to absorb higher initial costs with the potential of reduced operating costs.

Increased Logistic Flexibility:

This relates to a reduced demand for logistic support, such as reduced consumables. This is of special significance to military missions. Presented below is a battlefield scenario which reflects on the impact that a 50-percent fuel savings can have on reducing the amount of tonnage shipped across the battlefield and its effect on mission capability and flexibility.

IMPACT OF FUEL ON MILITARY MISSION FLEXIBILITY
-FROM BATTLEFIELD SCENARIOS-

- FUEL IS 70% OF TONNAGE SHIPPED
  - ARMOR/MECH/INF DIVISIONS
    - AVIATION: 100,000 GAL/DAY PER DIV
    - GROUND: 50,000 GAL/DAY PER DIV

    FOR 15,000 MAN DIVISION, 10 GAL/DAY/MAN PER DIV
  - AIR ASSAULT DIVISION
    - AVIATION: 320,000 GAL/DAY PER DIV
    - GROUND: 20,000 GAL/DAY PER DIV

    FOR 15,000 MAN DIVISION, 22 GAL/DAY/MAN PER DIV

- A 50% SAVINGS IN AVIATION FUEL CAN SIGNIFICANTLY REDUCE TOTAL TONNAGE SHIPPED, THUS INCREASING MISSION CAPABILITY AND FLEXIBILITY.

Improved Survivability:

This is primarily a military concern, although increased safety of operation is always sought for civil aircraft.

Needs, Goals and Approach:

The mission needs identified above can be translated into the goals contained in the following table:

NEEDS	GOALS	APPROACH
° EXPAND VEHICLE CAPABILITY	° DECREASE SFC	° ADVANCED CYCLES
° REDUCE OPERATING COST	° IMPROVE RELIABILITY AND DURABILITY	
° INCREASE LOGISTIC FLEXIBILITY	° SIMPLER, FEWER PARTS	° ADVANCED COMPONENTS (IFM)
° REDUCE INITIAL COST	° SPECIAL CONSIDERATIONS	
° IMPROVE SURVIVABILITY		° ADVANCED MATERIALS

Reduced fuel consumption can directly benefit the first three needs: Vehicle capability, operating costs, and logistic flexibility. Reliability and durability improvements can reduce operating costs, while simpler, fewer parts can cut both initial and operting costs. Special considerations refer to military survivability.

The approach column identifies the major thrusts to be addressed to achieve overall goals - efficient engines. These are further illustrated in Figure 2 as technology opportunities. Advanced cycles offer the potential of large reductions in specific fuel consumption (SFC). This will be further illustrated below. Enhanced computational tools development, verification and application to advanced concepts are required to provide highly advanced, efficient, durable components. Advanced materials such as ceramics and composites are required to achieve the maximum performance and life from the advanced engine concepts.

EFFICIENT ENGINE CYCLES

Small vs. Large:

Previous investments in technology programs by government and industry have led to significant efficiency gains for large engines. These gains have resulted from improved cycles, components and materials. However, much of these technologies have not been transferrable to smaller engines. As engine power size decreases, performance decreases due to the combined effects of increased relative clearances, lower Reynolds number, increased relative surface roughness, etc. The result of these adverse effects, illustrated in Figure 3, are particularly noticeable below 2000 SHP where the benefits of high cycle pressure ratio are unobtainable for small engines due to the severity of adverse scaling penalties. Small engines employ different component configurations such as centrifugal compressors, reverse flow combustors, and radial turbines to minimize these effects.

In the future, problems associated with size effects will no longer be limited to small engines. Future large turbofan engines designed for high by pass ratios and higher cycle pressure ratios (greater than 50:1) will be limited in performance by some of the same size related problems which presently limit the performance of small engines. This is the result of the inherent reduction in core engine flow size associated with the higher bypass ratio and the higher core pressure ratio, all of which reduces the turbomachinery size and the combustor length and volume. To counter the losses associated with the small turbomachinery blading, such things as replacing the back stages of the typically all axial compressors with a centrifugal stage are now being considered, following the same trends as for small engines.

State-of-the-Art Cycle Performance:

Figure 4 presents one example of the performance of a state-of-the-art 800 SHP uninstalled, simple-cycle gas turbine. It is used as a reference for comparison to advanced cycles. A turbine inlet temperature of 2200°F and a pressure ratio of 14:1 were assumed as being representative. In addition, correct state-of-the-art component efficiencies and combustor/turbine cooling requirements were assumed. The Brake Specific Fuel Consumption (BSFC) for these operating conditions is approximately 0.43 lbs/shp-hr and the specific power is approximately 180 SHP/lb/sec.

Advanced Simple Cycle:

Figure 5 presents the performance of an advanced simple cycle in comparison to the state-of-the-art simple cycle. For the advanced simple cycle, advanced component efficiencies were assumed, along with higher operating pressures to 24:1, higher temperature to 2600°F, and uncooled ceramic hot section components. The advanced simple cycle BSFC of 0.36 lb/hp-hr is 17-percent less than the state-of-the-art engine, and its specific power is 55-percent higher. Of the 17-percent BSFC improvement, 8-percent is attributable to advanced component efficiencies, 4-percent is due to the higher cycle pressure, temperature and reduced coolant penalty, and 5-percent is a result of entirely eliminating the coolant penalty by using uncooled ceramics.

Regenerative Cycle:

Figure 6 presents the performance of an advanced regenerative cycle in comparison to the state-of-the-art simple cycle and the advanced simple cycle. Again, advanced component efficiencies were assumed, along with the high turbine inlet temperature of 2600°F, and uncooled ceramics. In terms of BSFC, the advanced regenerated cycle provides the potential for a very significant 37-percent reduction over the state-of-the-art engine, along with some increases in specific power. In addition, the optimum cycle pressure ratio for the regenerated cycle is much lower than that for the advanced simple cycle, thus resulting in fewer compression states. Regenerative cycles could utilize either a rotary regenerator in a stationary recuperator.

While these potential performance gains are quite large, they must be examined in a representative mission model, taking into consideration projected changes in engine size, weight, cost and other factors before the real benefits of advanced cycles can be assessed. Because of the diversity of small engine applications, representative missions for each application must be individually examined. A typical rotorcraft application is shown in Figure 7. As can be seen, advanced technology could deliver a potential fuel savings of 30 to 55-percent compared to today's state-of-the-art- technology. The reduction in fuel requirements would result in a 12-percent decrease in airframe weight for a new airframe and could lead to a 21-percent to 25-percent decrease in Direct Operating Cost (DOC), depending on the assumed acquisition and maintenance costs for an advanced new regenerative engine.

Using similar calcualtions for missions other than rotorcraft, the potential payoffs contained below were calculated:

PAYOFFS

ROTORCRAFT		COMMUTERS
- 50% FUEL SAVED - 20% LESS DOC		- 35% FUEL SAVED 12-15% LESS DOC
	ADVANCED ENGINES	
GENERAL AVIATION		**CRUISE MISSILES**
20-40% FUEL SAVED 0-15% LESS DOC		60% RANGE INCREASE

The mission analyses conducted for the various applications focused on fuel savings and resulting economic benefits. On this basis, fuel savings would be at least 20-percent for advanced simple cycles and as much as 55-percent for advanced regenerative cycles for rotorcraft, assuming a "rubber" aircraft sized to accomplish a fixed mission. However, the higher performance of the advanced engines could alternatively be exploited to expand the range or payload capability of fixed gross weight aircraft.

To achieve these payoffs, the technology challenges listed below would have to be successfully met.

TECHNOLOGY CHALLENGES

SIMPLE CYCLES

- IMPROVE TURBOMACHINERY PERFORMANCE SIGNIFICANTLY
- ELIMINATE COOLING PENALTY AT HIGH CYCLE TEMP. AND PRESSURE
- IMPROVE 'ILITIES AND COST

REGENERATIVE CYCLES

- REDUCE HEAT EXCHANGER WEIGHT, SIZE, AND COST PENALTIES
- ESTABLISH HIGH EFFICIENCY/RELIABILITY VARIABLE AREA POWER TURBINE TECHNOLOGY
- IDENTIFY USEFUL MULTI-PURPOSE ENGINE

To achieve approximately 20-percent fuel savings and approximately a 10-percent DOC reduction with simple cycles requires: Significant turbomachinery performance improvements; establishing approximately 2600°F uncooled hot section technology; and novel design/manufacturing approaches to improve the economics and reliability without significantly sacrificing performance potential.

Regenerative and recuperative cycles provide the potential for doubling the preceeding fuel and DOC payoffs. However, major reductions in heat exchanger weight, size, and cost penalties are required. Technology for highly efficient, yet reliable variable area power-turbines is also necessary for rotorcraft applications. The DOC and fuel savings gains cited above are merely potential until applied to specific missions. This is done in the next section.

Application of Efficient Engine Technologies to Specific Missions:

Contract studies aimed at identifying high payoff technologies for year 2000 small gas turbine applications were conducted under the Small Engine Component Technology (SECT) Program under joint NASA/Army-AVSCOM sponsorship. A list of contractors and the applications studied follows:

SMALL ENGINE COMPONENT TECHNOLOGY (SECT) STUDIES
CONTRACTOR SELECTED APPLICATIONS

CONTRACTOR	ROTORCRAFT 500-1,000 HP	GENERAL AVIATION/ COMMUTER 500-1,000 HP	APU 300-500 HP	CRUISE MISSILE 200-1,000 LB T
ALLISON	X			
AVCO LYCOMING		X		
GARRETT	X	X	X	X
TELEDYNE CAE				X
WILLIAMS INTERNAT'L				X

Details describing the studies, conditions imposed, procedures, etc. are contained in references 1 through 6. Guidelines are essentially the same as those described previously; i.e. year 2000 technology, etc. The discussion below will focus on the rotorcraft missions studied.

Allison:

Allison selected for study the eight passenger tilt rotor executive/commercial aircraft shown in Figure 8. The mission for the twin turboshaft powered tilt rotor aircraft is 350 nautical miles with a cruise speed of 250 knots at an altitude of 20,000 feet. The reference current technology engine has a turbine rotor inlet temperature of 2200°F, a pressure ratio of 14:1 and 1,000 SHP at take-off. The advanced cycles studies and the optimum cycle parameters for each advanced cycle considered follow:

	Concentric	Non-Concentric	Recup.	Regen.	Wave Rotor
Pressure Ratio	25	30	14	10	38
TRIT, °F	2800	2800	2800	2800	2800
Effectiveness			.6	.7	
P/P			.10	.13	
LP Compressor	Axial	Axi-Centrif.	-	-	Axial/ Centrif.
HP Compressor	Centrif.	Centrif.	Axial/ Centrif.	Axial/ Centrif.	Wave Rotor
HP Turbine	Radial	Radial	Axial	Axial	Wave Rotor
LP Turbine	Axial	Radial	-	-	Axial
Power Turbine	Axial	Axial	Axial	Axial	Axial

Turbine inlet temperatures of 2200° to 2800°F were studied. Uncooled turbines were assumed. The 2800°F temperature was selected on the basis of hot spot temperature capability of ceramic composite material and projected year 2000 combustor pattern factors of less than 0.2. Regenerator and recuperator effectiveness values of 0.6 to 0.8 and pressure drops of 6-percent to 14-percent were used.

During preliminary analysis of the five configurations, two designs were eliminated: The wave rotor cycle because of a lack of analytical capability to evaluate the cycle; and the simple concentric cycle because analysis indicated that the bore stresses in the ceramic HP turbine were prohibitive.

The remaining advanced engine concepts were installed (analytically) in the tilt rotor aircraft and flown over the reference mission. Reductions in fuel burned were 30.5-percent for the regenerative; 30.7-percent for the recuperative; and 16.5-percent for the non-concentric engines. DOC reductions achieved are shown in Figure 9. All configurations achieved the program goal of 10-percent DOC reduction. However, the non-concentric engine produced the greatest reduction and was selected for further study.

The general arrangement of the non-concentric engine is shown in Figure 10. The configuration incorporates the following: Turbine rotor inlet temperature of 2800°F, pressure ratio of 30, axial/centrifugal high pressure compressor, radial high pressure turbine, radial low pressure turbine, and an axial power turbine. Required advanced technologies for bringing this concept to fruition are ceramic/ceramic composites for the combustor and turbine; highly efficient components and three-dimensional codes; and advanced bearings for reliability and durability. Assuming $1/gal fuel cost, advanced ceramic/ceramic composites provide 58-percent of the DOC reduction, with advanced aerodynamics providing approximately 40-percent.

Garrett:

The utility helicopter shown in Figure 11 was selected for the military/civil mission shown in Figure 12. The mission consists of five segments with four hover periods and total mission length of 130.4 nautical miles. The reference current state-of-the-art engine has a turbine inlet temperature of 2100°F, a pressure ratio of 13.5, and produces 1,000 HP at take-off. An advanced simple cycle and a recuperated cycle were studied. A summary of their characteristics follows:

	Advanced Simple Cycle	Recuperated Cycle
TRIT	2600°F	2600°F
Pressure Ratio	22.1	10:1
Compressor	2-Stage Centrifugal	1-Stage Centrifugal
HP Turbine	Axial, Uncooled	Axial, Uncooled
LP Turbine	Axial, Uncooled	Axial, Variable Geometry, Uncooled
Recuperator Effectiveness	-	.8

Selection of 2600°F temperature was based on hot spot temperature capability of ceramic/ceramic composite material and year 2000 combustor pattern factors of less than 0.2. Regenerator and recuperator effectiveness values of 0.6 to 0.8 and pressure drops of 6 to 14-percent were investigated.

Analysis results, based on DOC and fuel burned, are contained in Figure 13. Reduction in fuel burned is 21.9-pecent for the simple cycle and 41.6-percent for the recuperated cycle, compared to the reference engine. At $1/gal fuel cost, both cycles produce approximately 7-percent reduction in DOC. Recuperator weight and cost offset the additional reduction in fuel burned of the recuperated cycle. However, at $2/gal, the recuperated cycle has 11.4-percent reduction in DOC - 2.7-percent more than the simple cycle. The increased fuel cost more than offsets the recuperator weight and cost.

Required technologies for bringing these concepts to fruition are primarily: Advanced materials - Ceramic/ceramic composites and Ni_3Al disk; advanced aerodynamics; and system technologies - metal matrix shafts, bearings, and seals.

Technology Rankings:

SECT contractor technology rankings for all of the missions studied are summarized:

CONTRACTOR TECHNOLOGY RANKINGS BASED ON DOC OR LCC

	ROTORCRAFT/COMMUTER						CRUISE MISSILE	
	ALLISON		AVCO LYCOMING		GARRETT		TELEDYNE	WILLIAMS
	$1/GAL	$2/GAL	$1/GAL	$2/GAL	$1/GAL	$2/GAL		
MATERIALS								
CERAMICS	58%	57%	65%	55%	45%	40%	31%	23%
ADVANCED METALLIC DISK					20%	3%		
ADVANCED AERO-DYNAMICS	40%	42%	29%	25%	30%	34%	22%	24%
3-D CODES AND COMPONENTS								
TURBINE	17%	18%	11%	10%	17%	17%	15%	12%
COMPRESSOR	14%	15%	18%	15%	10%	14%	7%	12%
COMBUSTOR	9%	9%			3%	3%		
RECUPERATOR			6%	20%		22%		
SYSTEMS TECH-NOLOGIES								
BEARINGS	°	°			°	°	°	45%
SHAFTS					°	°	°	
SEALS					°	°	°	
GEARBOX							40%	
SLURRY FUEL COMBUSTOR							°°	°°

° NECESSARY TO ACHIEVE GAINS FROM MATERIALS AND AERODYNAMICS
°° CRUISE MISSILE RANGE BENEFIT

The technologies are ranked on cost benefit based on direct operating cost or life cycle costs. Ranking the technologies by fuel savings instead, would lead to somewhat different conclusions.

For the rotorcraft/commuter applications, materials and advanced aerodynamics provide significant cost benefit irrespective of the fuel cost. The recuperator, however, pays off at $2/gal for the cycles selected. The system technologies, while not displaying cost benefits, are necessary to achieve the gains from the advanced materials and aerodynamics.

For subsonic strategic cruise missile applications, the high payoff technologies in order of priority are reversed. System technologies provide approximately 40-percent of the cost benefits; advanced materials provide 23 to 31-percent; and advanced aerodynamics provide approximately 23-percent.

SMALL GAS TURBINE TRENDS AND CURRENT RESEARCH ACTIVITIES

Recent trends in small gas turbine design and performance are described in this section. Also described are research activities aimed at improving the efficiency of engine components.

Compressors:

Compressor design trends during the last 5 to 10 years have been towards increasing blade speed and aerodynamic blade loading to reduce the number of stages and to increase the overall compression system pressure ratio; implementing tailored blade shapes for axial machines; and evolving highly swept-back, high tip-speed centrifugal blade designs. The highly swept-back designs have demonstrated higher efficiencies and increased stall margin at peak efficiency point operation. These performance improvements have been achieved through the evolution and extensive use of advanced aerodynamic and structural codes. State-of-the-art efficiencies of current compressors are shown in Figure 14. This figure also shows the fall-off in efficiency with size. The NASA/Army advanced compressor technology goal is also shown on the figure.

Major thrusts of the current NASA/Army-AVSCOM compressor research program are shown in Figure 15. They include higher pressure ratios, increased efficiency and reduced number of stages. A recent major accomplishment was to quantify factors degrading the performance of centrifugal compressors as flow size is reduced. In the near future, the detailed flow measurements within the centrifugal blading will be obtained for use in developing and validating 3-D inviscid and viscous codes. These codes will then be available for designing compressors which minimize the losses associated with flow size.

The program to quantify the degradation in performance as flow size is reduced is depicted in Figure 16. As shown, a 25 lb/sec centrifugal compressor, described in references 7 and 8, was directly scaled-down to a 10 lb/sec size. The design was also scaled-down to a 2 lb/sec compressor. However, in scaling to the 2 lb/sec size, the blade thickness had to be increased in order to maintain a structurally sound design. Thus, to maintain a link to the original compressor, the 2 lb/sec compressor was then directly scaled-up to a 10 lb/sec size, incorporating the thicker blades. The three compressors were then experimentally evaluated. Impellers with two relative blade surface roughnesses were also evaluated for the 2 lb/sec size.

Results are summarized in Figure 17. Defined are effects resulting from Reynolds Number, shroud thickness, blade thickness, tip clearance and surface finish. For reference purposes, the design Reynolds Number for each compressor is indicated along the abscissa.

The data presented in Figure 17(a) reflect that for the directly scaled 2 and 10 lb/sec compressors, essentially the same efficiencies were achieved when operated at the same Reynolds Number. In fact, when factoring in the impact of different relative shroud thicknesses between the two machines, the efficiencies are identical. The effect of shroud thickness is reflected in Figure 17(e) for the 10 lb/sec thin-bladed configuration. Shroud thickness affects the amount of interheating which occurs within the compression process, which is inherently inefficient. Thus, this data indicates that the scaling laws hold for directly scaled machines operated at the same Reynolds Number. The problem is that the Reynolds Number changes with changes in characteristic length such as blade chord; and seldom can the relative surface finish, tip clearance, blade thickness and even shroud thickness be maintained as flow size is reduced. The impact of these parameters on performance is reflected in Figures 17(b) to 17(e), respectively.

Further analysis of the data is currently in process, and reports are being written. In addition to quantifying the scaling effects, the experimental data will provide a basis for validating advanced mathematical codes. These codes will then be used to minimize these losses. It is also assumed that advanced manufacturing technology will minimize the adjustments now required when scaling down to the smaller sizes.

Turbines:

The current performance of small turbines and future goals are shown in figure 18. The turbines are separated into gas generator and power turbines due to their contrasting characteristics. Gas generator turbines are smaller, operate at high temperature and pressure, develop high work per stage, are unshrouded and usually cooled. Power turbines are larger, uncooled, shrouded, and develop less work per stage. The current technology is indicated by dashed curves and include only axial machines, the current in-service type. The solid curve, figure 18(b), represents the goal for future axial power turbines. Axial machines are expected to remain the preferred type because of ease of packaging behind the gas generator turbine. The goal for gas generator turbines, figure 18(a), is represented by a rather broad band and includes both radial and axial machines. The bottom part of the band represents the goal for axial turbines and the upper part the near term performance of radial turbines. The chief obstacles to the use of radial turbines in rotorcraft engines is not their performance level, but rather developing the technology to make them compact, lightweight, and able to operate at much higher temperatures. To accomplish this requires improved high-temperature alloys, cooled radial rotors and, ultimately, ceramic rotors.

Current turbine aerodynamic design methods consist largely of the use of two and three-dimensional inviscid computer codes with empirically determined loss correlations based on stage loading, blading aerodynamic parameters, and endwall effects. Heat transfer calculations are generally made with two-dimensional boundary layer codes based primarily on flat plate data and empirical correlations. This design approach has provided efficient aerodynamic blade shapes for many large turbines that perform as predicted. However, many advanced designs and small low-aspect ratio turbines have highly three-dimensional blade shapes with strong secondary flows. These flows cannot be predicted accurately with current computer codes and consequently increased aerodynamic losses and inaccurate heat transfer calculations result. At present, there are uncertainties as high as 35/25-percent in gas-side and coolant-side heat transfer coefficients, respectively. This, in turn, can result in life prediction uncertainty factors greater than ten.

Recently, radial turbines have been given renewed emphasis for small engine applications due to several inherent advantages they possess over axial turbines. Among these are: Improved packaging when mated with reverse flow combustors; higher work extraction per stage; greater aerodynamic efficiency; less sensitivity to tip-clearances; and increased ruggedness. Technology challenges for radial turbines are also formidable, and include: Higher rotor inertia; less understanding of the rotor flow-field; and an almost total absence of existing rotor cooling technology, including cooled rotor fabrication techniques.

Major thrusts of the NASA/Army-AVSCOM turbine research program are contained in Figure 19. Both axial and radial turbines are being studied: Major emphasis, however, is being placed on radial turbine research. Emphasis is also placed on high temperature operation. Highly efficient engine cycles require that the turbine operate efficiently up to 2800°F average temperatures with minimal coolant. Achieving these capabilities will come largely from: Application of high temperature materials, especially composites and ceramics; a more thorough understanding of fluid behavior in turbine passages, and the concomitant ability to predict this behavior; rigorous computer codes for hot gas convective coefficient prediction arrived at through fundamental experiments; and experiments in near-engine environments to validate the advanced technology.

An example of turbine research currently underway is the aero and structural design of an advanced mixed-flow ceramic turbine. The mixed-flow configuration allows blading to be used, which increases the aerodynamic efficiency, while maintaining radial blade elements, thus eliminating blade bending stresses. A new, advanced structural analysis code was developed and used to predict fast fracture failure probability of the monolithic ceramic rotor. To date, the aerodynamic and structural analyses of the rotor are complete and indicate high efficiency and a probability of survival of 0.985. The turbine is shown in Figure 20. Future work will focus on verifying the design results.

Other turbine research activities in progress or recently completed include: An experimental program to determine the performance of variable geometry radial turbines; small axial turbine scaling studies; performance and code verification for high-work, high-temperature radial turbines; and a comprehensive investigation of the performance and heat transfer of a cooled radial rotor.

Combustors:

Major combustion concerns of the 1970's and early 1980's aimed at gaseous emissions control and usage of alternative fuels have abated considerably by the mid-1980's. However, many combustion problems remain, especially for small combustors operating at aggressive, fuel-efficient cycles. Problems areas are summarized in Figure 21. An especially serious problem is liner cooling and durability. Small combustor liners require proportionately more coolant due to their increased surface to volume ratios, compared to large combustors. State-of-the-art small combustors, for example, can require up to 30-percent of the combustor airflow for cooling. At advanced cycle conditions of increased temperature and pressure, heat loads to the liners will increase, coolant temperatures will also increase and the amount of coolant available will decrease. More air will be required for combustion and mixing. Recuperative/regenerative cycles impose their own unique combustion problems in that coolant air temperatures are further increased. Another significant size related combustion problem is fuel injection, due to proportionately smaller passages which are subject to clogging. Current trends are to replace pressure atomizing nozzles with air blast designs. These contain larger passages but impose additional problems of poorer spray quality at low power and off-design conditions.

Major thrusts of the NASA/Army-AVSCOM combustion research program are contained in Figure 22. The overall objective of evolving the combustor technology required for advanced, fuel-efficient cycle operation can be realized by: Increasing the temperature and pressure operating capability of the combustor, including achieving higher temperature rises; reducing or eliminating air coolant requirements; and, simultaneously improving combustor durability and reliability.

Small gas turbines incorporate a variety of combustor types. Several of these are shown on Figure 22. Axial flow designs are large combustors scaled down in size. As their name implies, flow is directly through the combustion system. This design minimizes combustor surface area but can produce somewhat higher pressure loss and somewhat less uniform exit temperature distributions than the other designs shown. Reverse flow, or flowback combustors, are unique to small engine designs. They are often selected for application because they package well in small engines. However, they incorporate large combustor surface areas as well as a reverse turn, which must also be cooled. Radial outflow combustors approximate the reverse flow type. These combustors often eliminate the need for fuel injectors by introducing fuel through the engine shaft.

Research activities currently underway are applicable to all three combustor types. An example of combustion research in progress is contained in Figure 23. Advanced ceramic matrix liner concepts were evolved, applied to a full-size combustor and performance evaluated at simulated pressure ratio conditions to 22:1. The liner incorporated a thick yttria-stablized zirconia coating on a pliable metallic surface. Details are contained in reference 9. Only backside convective cooling was supplied. Direct injection of film/transpiration coolant into the combustor, mandatory for current technology designs, was not required. Good short-term durability was demonstrated to outlet temperatures exceeding 2600°F. This is at least 300°F hotter than current cycle requirements. Measured liner temperatures were less than 1600°F, which is well within design limits. For comparative purposes, liner wall temperature for the ceramic matrix liner, as well as conventional liner design, are included on the figure.

REFERENCES

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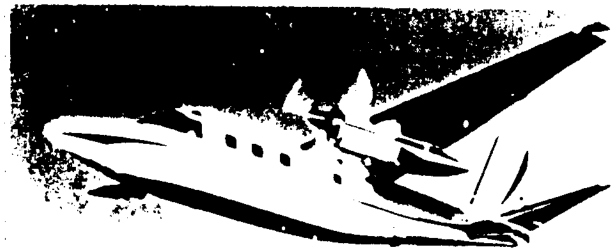
ADVANCED SMALL ENGINE TECHNOLOGY

NASA

BROAD CIVIL/MILITARY APPLICATIONS

HELICOPTER

COMMUTER AIRCRAFT



GENERAL AVIATION

CRUISE MISSILES

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FIGURE 1

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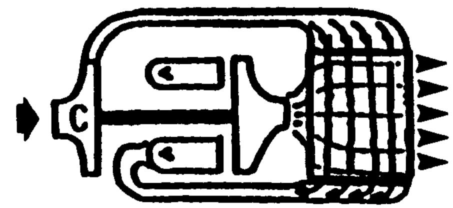
C-3

SMALL ENGINE TECHNOLOGY OPPORTUNITIES

ADVANCED CYCLES

REGENERATIVE CYCLE

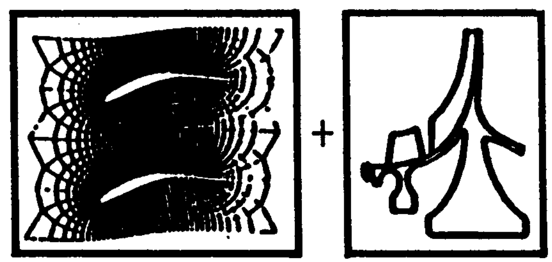


VERY EFFICIENT ENGINES

ADVANCED COMPONENTS

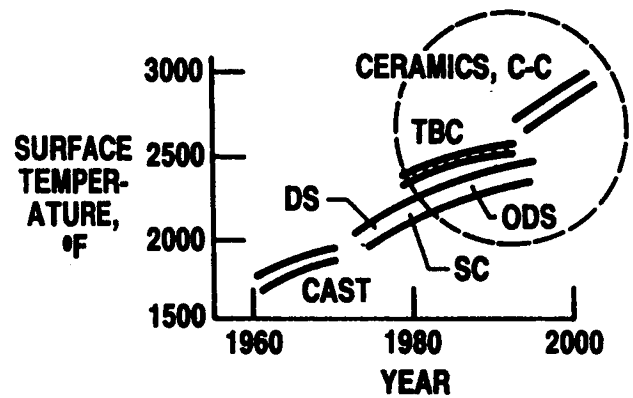
IFM

ADV. CONCEPTS



SUPERIOR TURBOMACHINARY

ADVANCED MATERIALS



>2600°F TIT UNCOOLED
- HIGH EFFICIENCY
- LOW WEIGHT

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FIGURE 2

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ADVANCED SMALL ENGINE TECHNOLOGY

SMALL ENGINES ARE NOT AS EFFICIENT

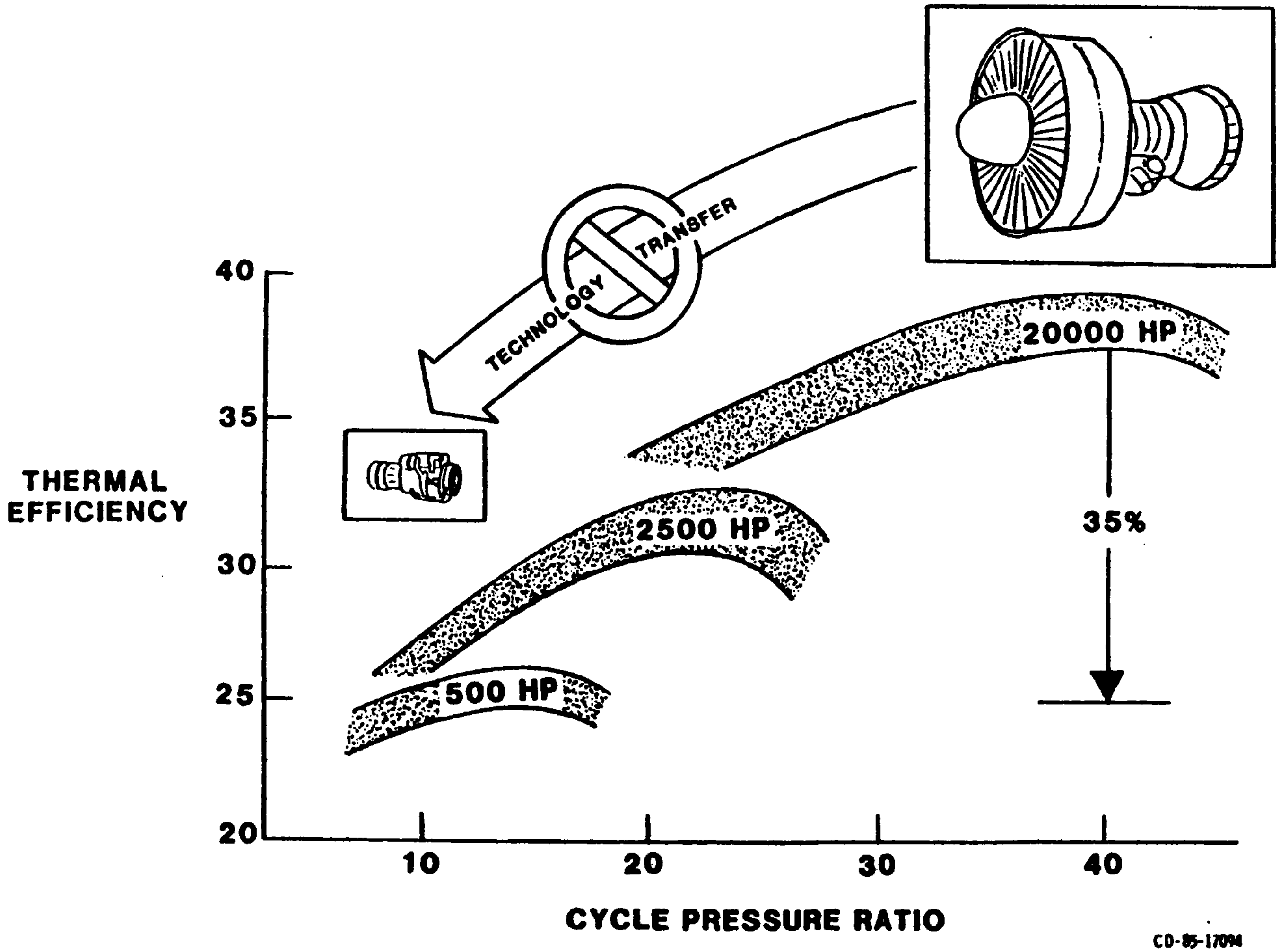

FIGURE 3

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ADVANCED SMALL ENGINE TECHNOLOGY

STATE-OF-THE-ART CYCLE PERFORMANCE

800 HP SLS

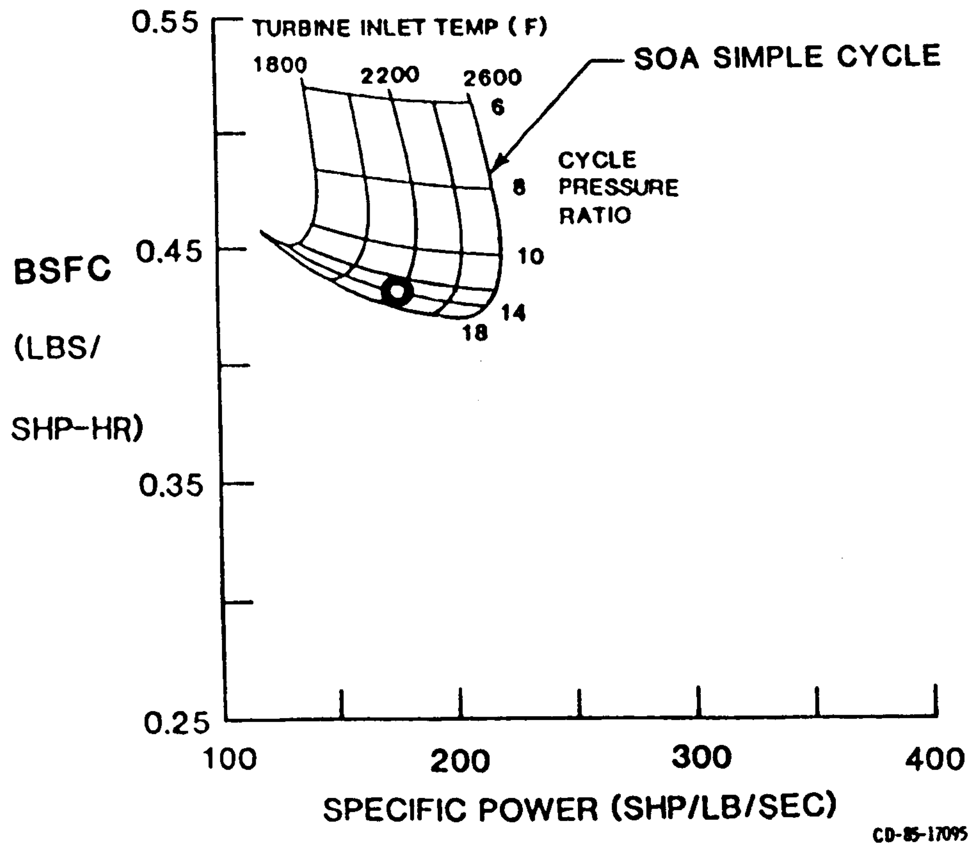

FIGURE 4

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ADVANCED SMALL ENGINE TECHNOLOGY

IMPACT OF ADVANCED TECHNOLOGY ON CYCLE PERFORMANCE

800 HP SLS

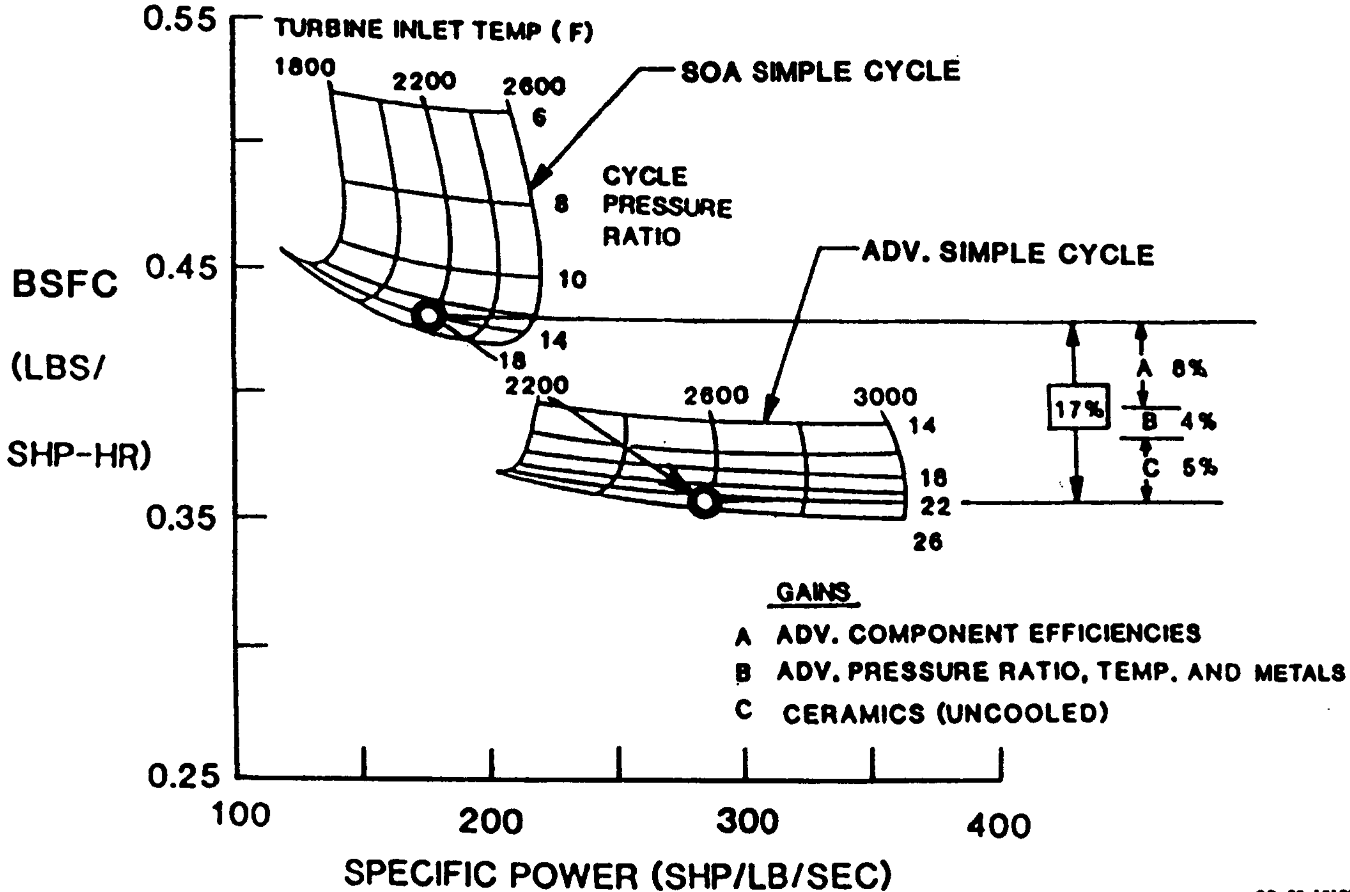

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FIGURE 5

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Advanced Small Engine Technology

IMPACT OF ADVANCED TECHNOLOGY ON CYCLE PERFORMANCE

800 HP SLS

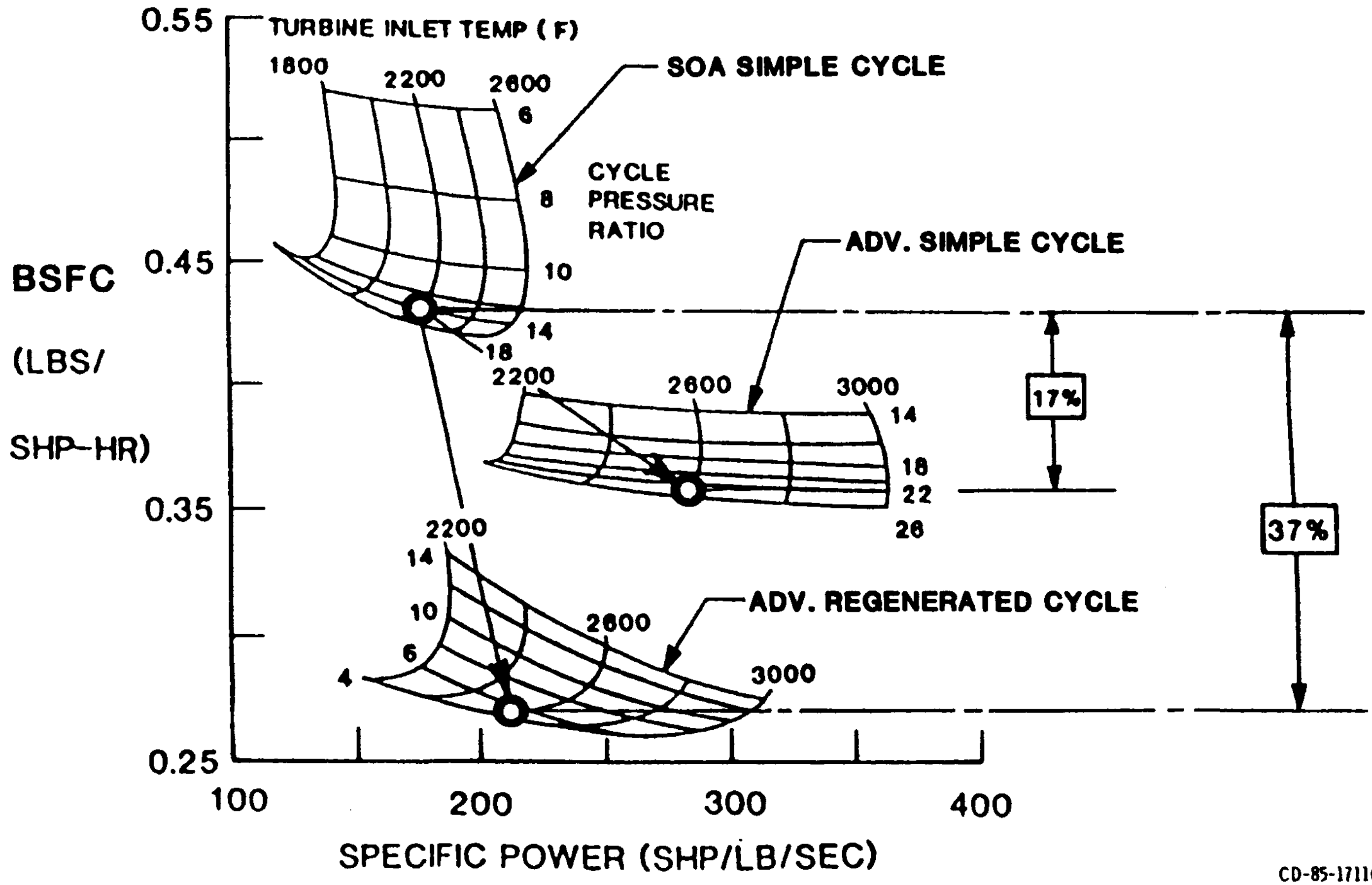

FIGURE 6

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ADVANCED SMALL ENGINE TECHNOLOGY

ASET TECHNOLOGY BENEFITS--HELICOPTER APPLICATION

4 PASSENGER, 0.15 MACH, 4000 FT. ALT., 300 N.M. DES. RANGE, 400 SHP/ENG.

REGENERATOR PENALTY: 50% ENGINE WEIGHT

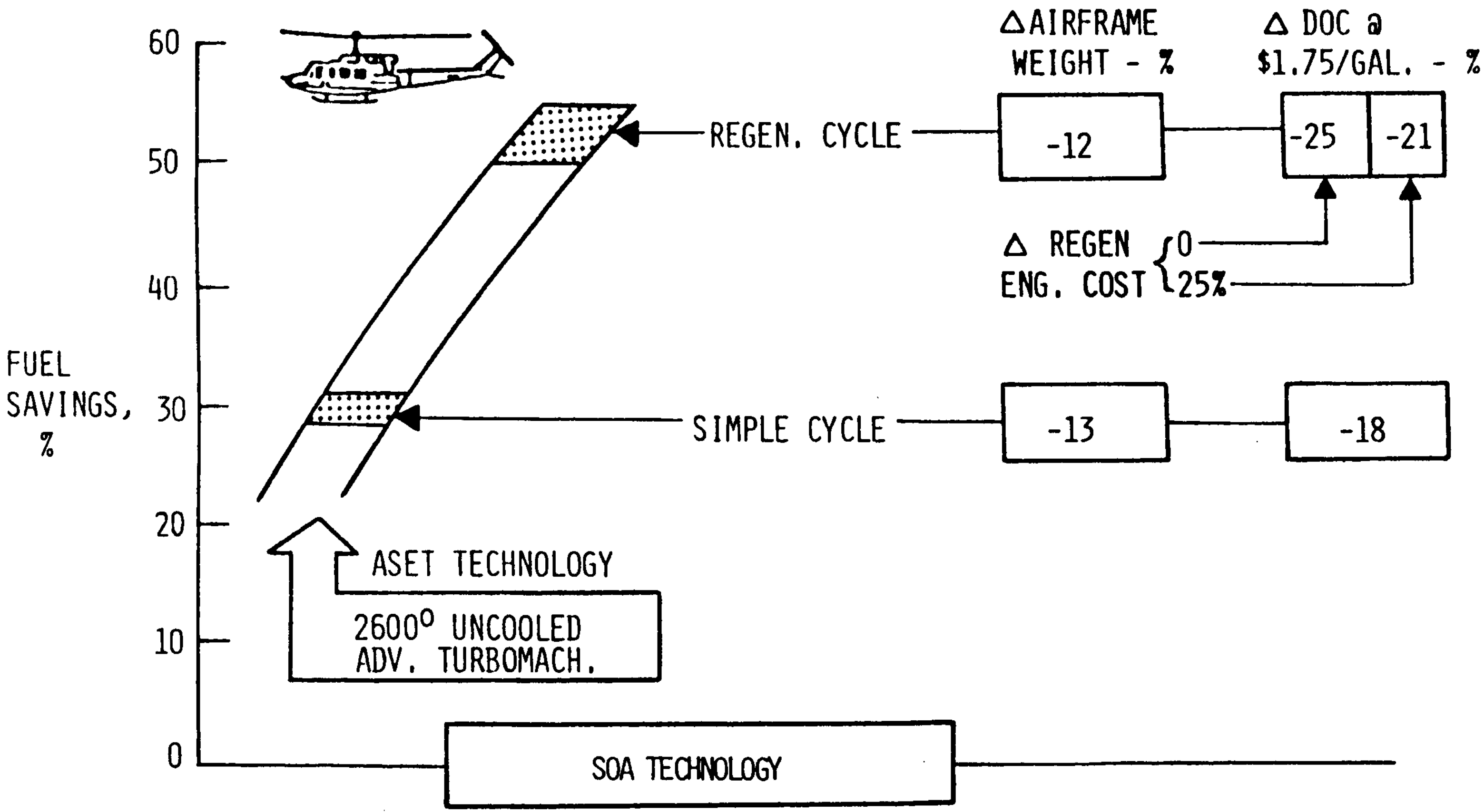

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FIGURE 7

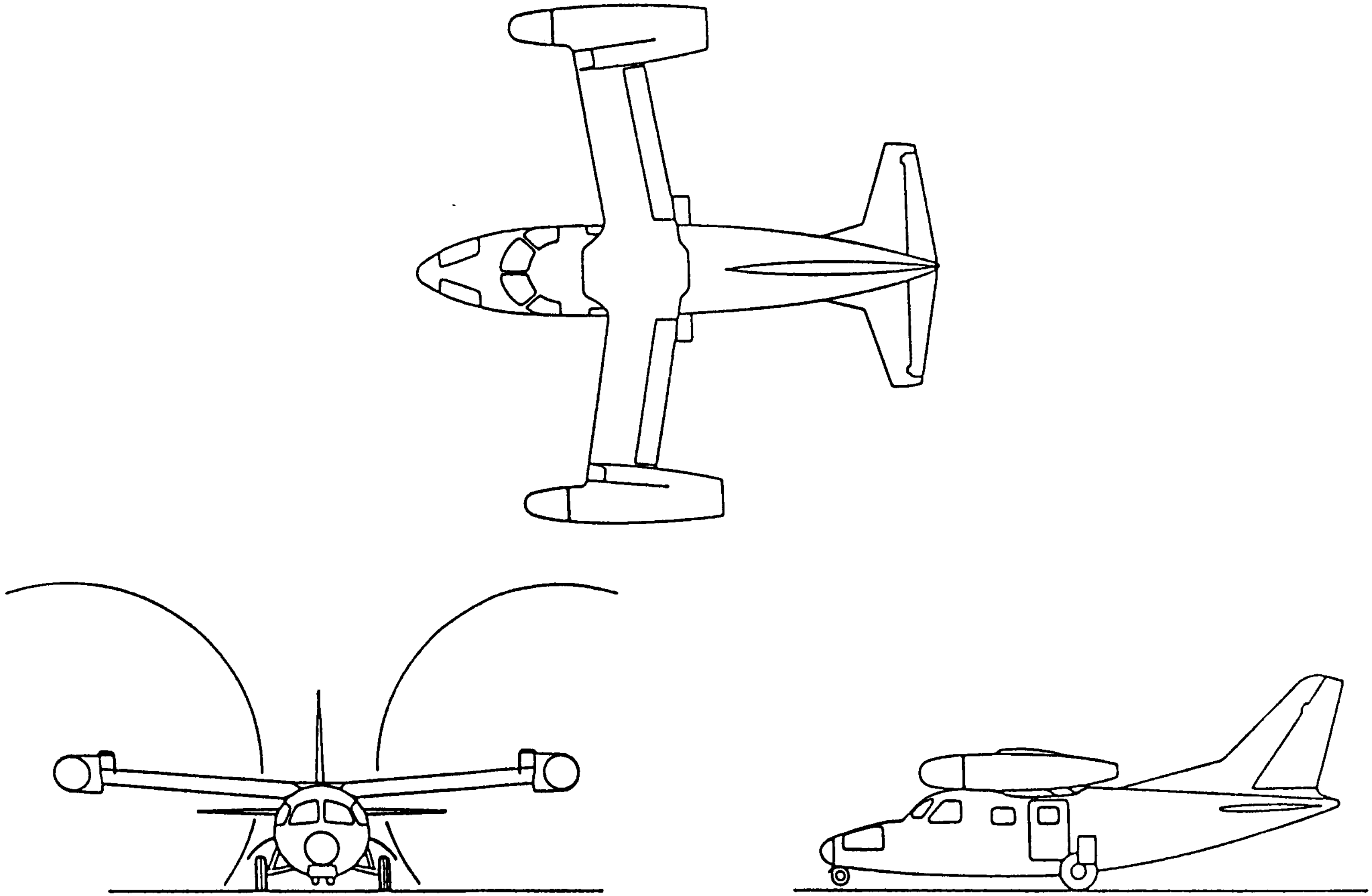


FIGURE 8: ALLISON TILT ROTOR AIRCRAFT

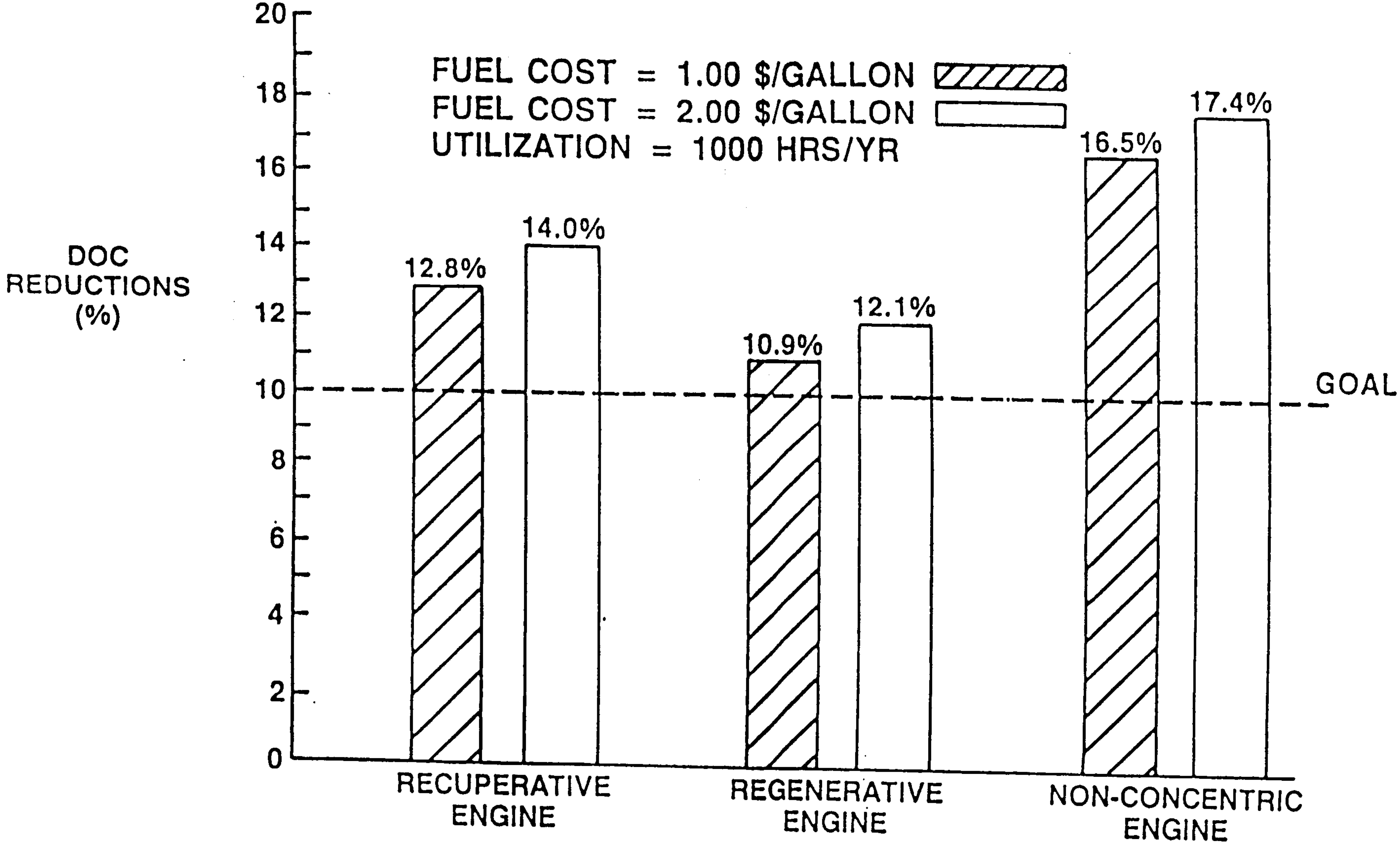
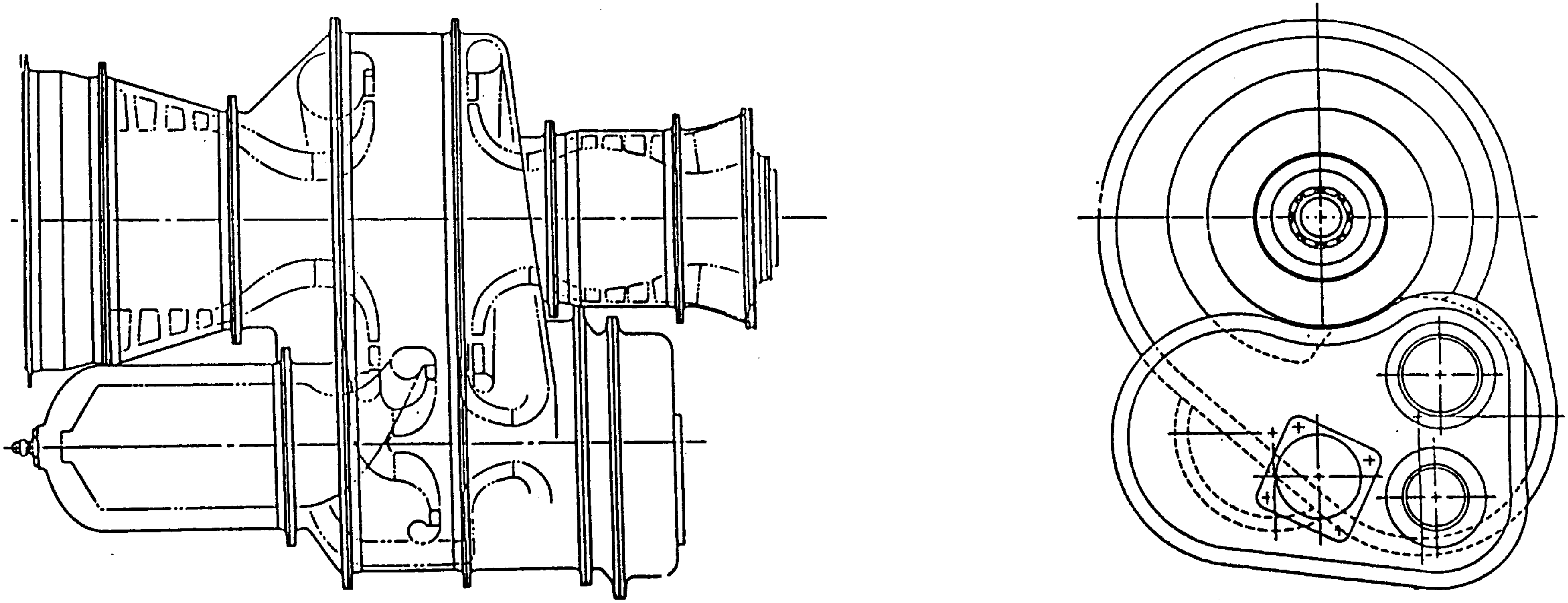
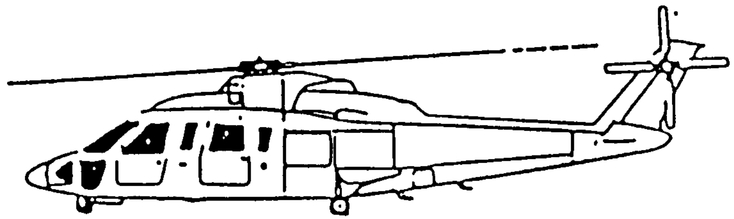


FIGURE 9: ALLISON MISSION ANALYSIS RESULTS

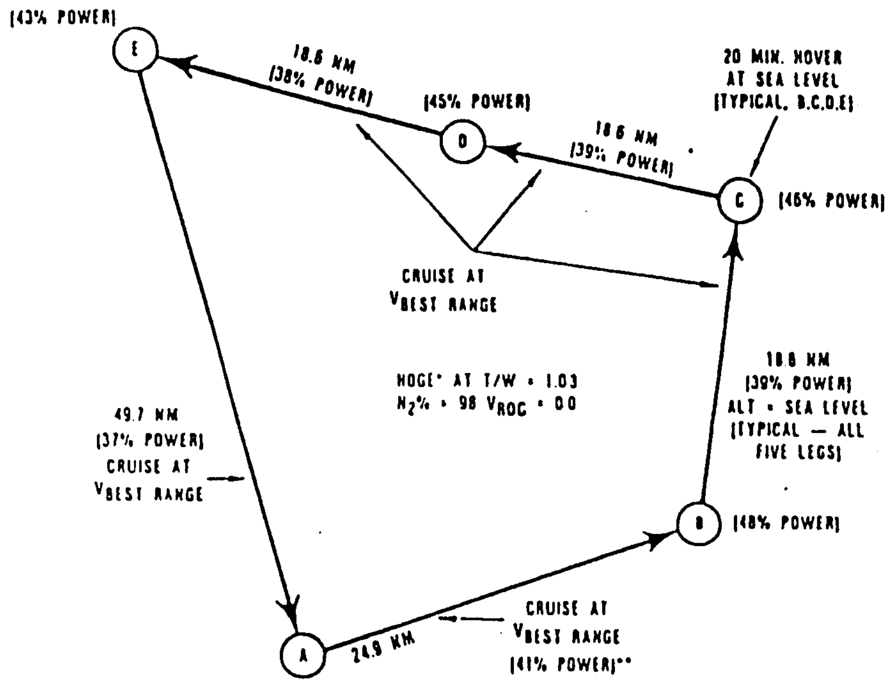
FIGURE 10: ALLISON NON-CONCENTRIC ENGINE

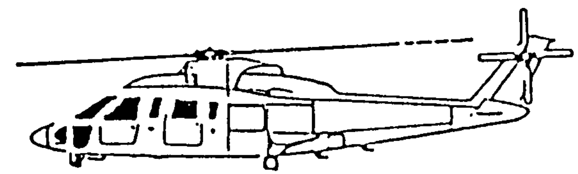




- TWIN ENGINE (1000 SHP EACH)
- WEIGHTS
  - EMPTY - 4375 LB
  - FUEL - 1303 LB
  - PAYLOAD - 3676 LB
  - TOGW (INCLUDES PILOT WEIGHT OF 210 LB) - 9564 LB

FIGURE 11: GARRETT REFERENCE ROTORCRAFT





MISSION LENGTH: 130.4 NM

MISSION TIMES:

HOVER — 1 HOUR, 20 MINUTES

CRUISE — 59 MINUTES

BLOCK — 2 HOURS, 19 MINUTES

*HOVER OUT OF GROUND EFFECT

**PERCENTAGE OF SLS T/O RATING (1000 SHP)

FIGURE 12: GARRETT ROTORCRAFT MISSION

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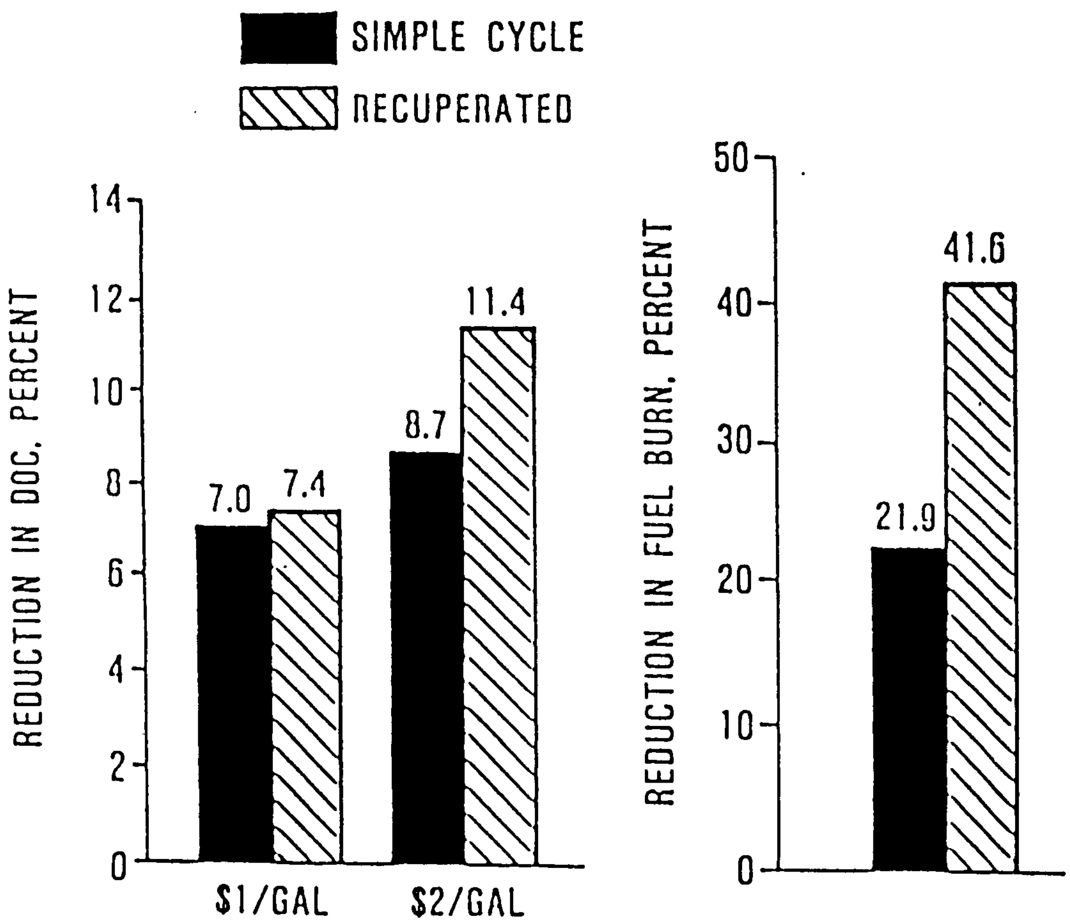
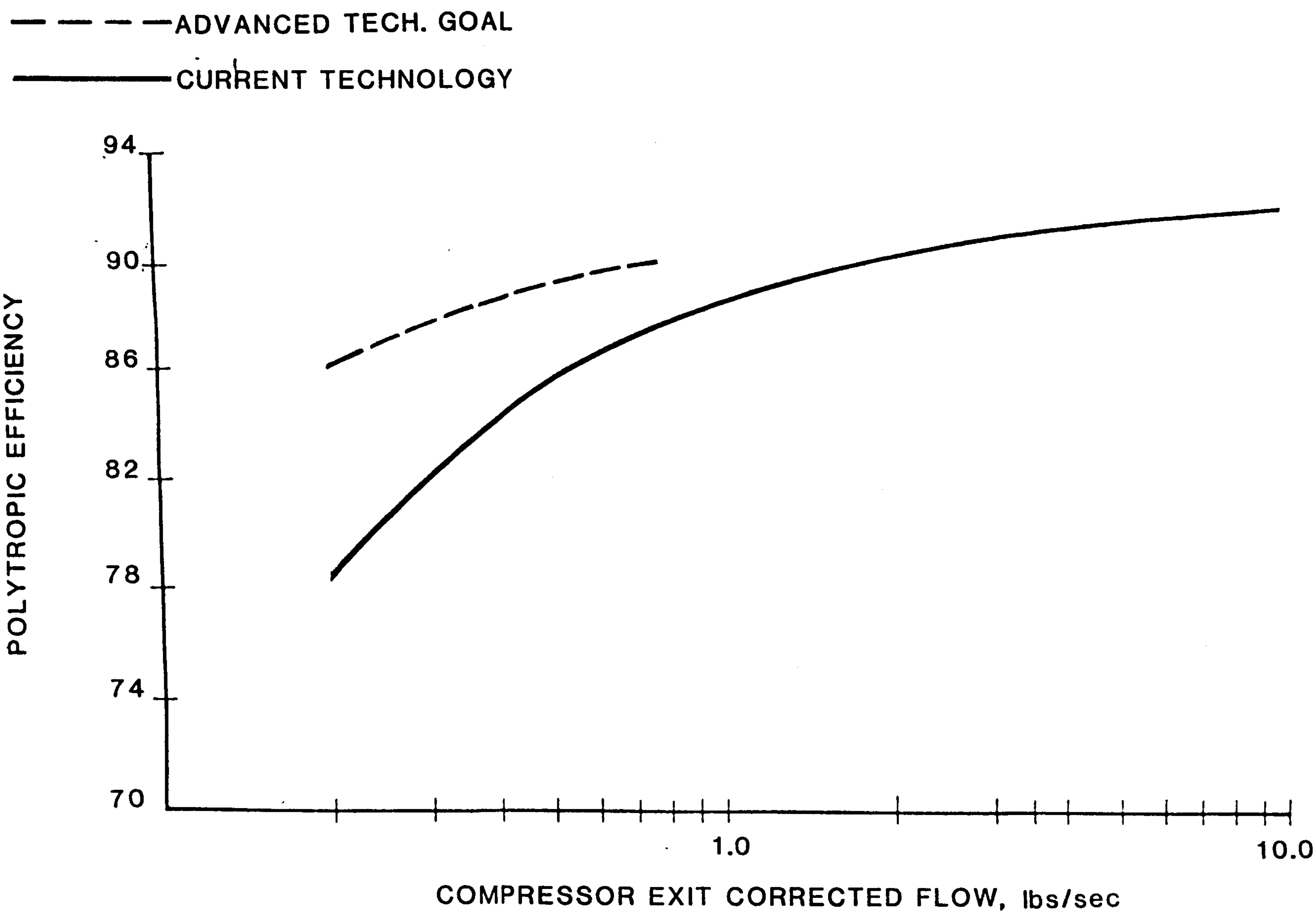


KEY TECHNOLOGIES

- ADVANCED MATERIALS (CERAMICS, Ni_3Al, CERAMIC RECUPERATOR)
- IMPROVED COMPONENT PERFORMANCE (TURBINE, COMBUSTOR, COMPRESSOR AERO)
- SYSTEM TECHNOLOGIES (METAL MATRIX SHAFTS, SEALS)

FIGURE 13: GARRETT MISSION ANALYSIS RESULTS

FIGURE 14: COMPRESSOR EFFICIENCY



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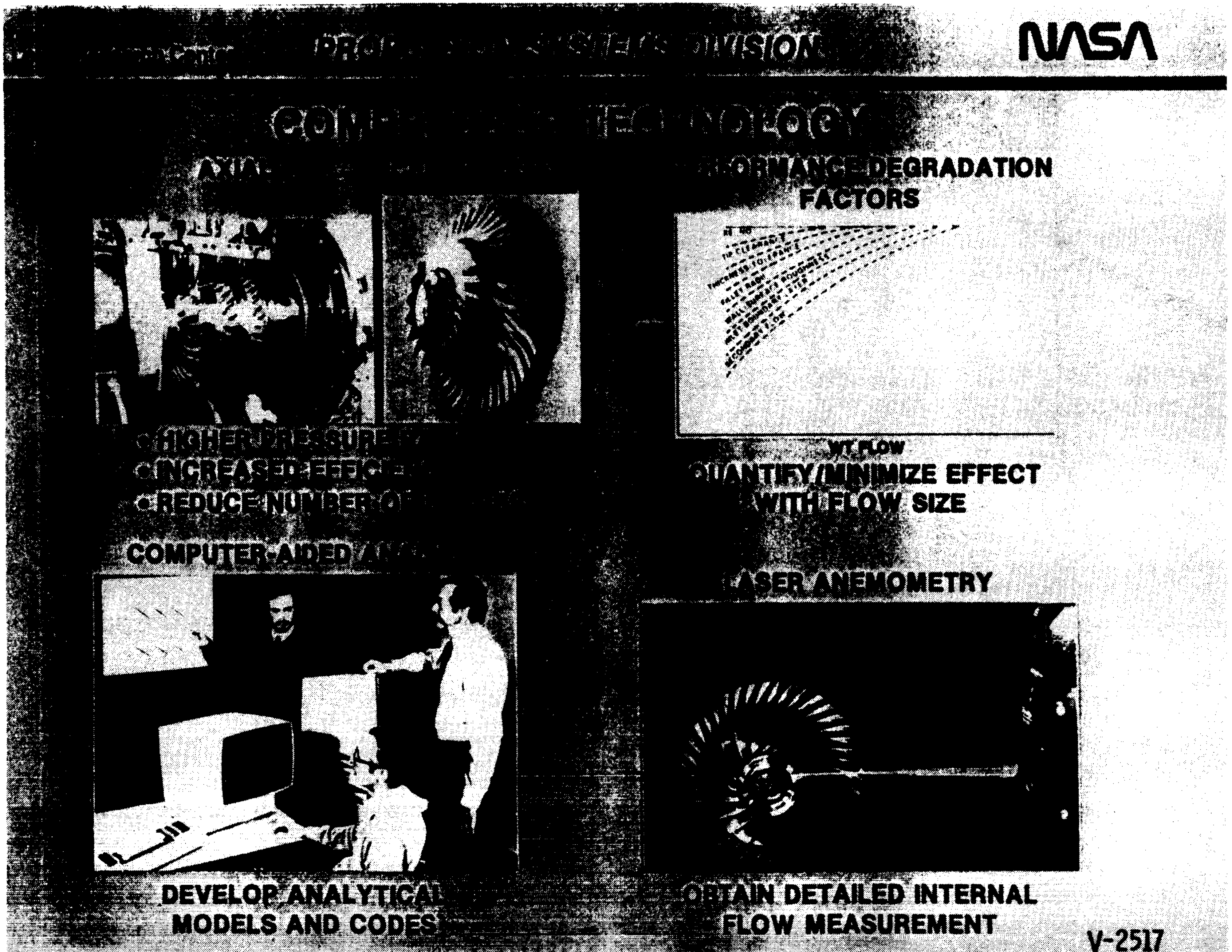

FIGURE 15

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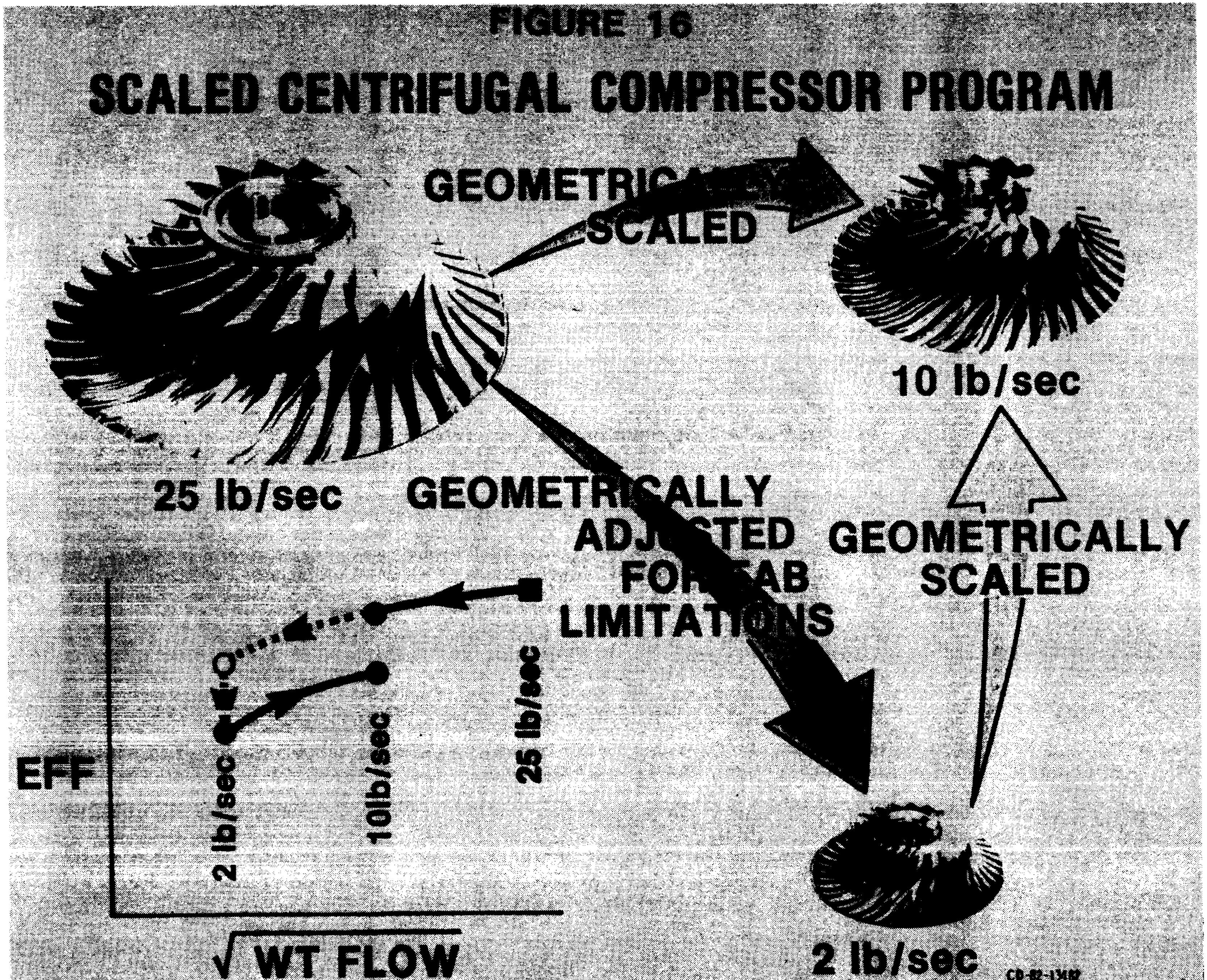
FIGURE 16
SCALED CENTRIFUGAL COMPRESSOR PROGRAM
GEOMETRICALLY SCALED
10 lb/sec
25 lb/sec
GEOMETRICALLY ADJUSTED FOR FAB LIMITATIONS
GEOMETRICALLY SCALED
EFF
2 lb/sec
10lb/sec
25 lb/sec
√ WT FLOW
2 lb/sec
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FIGURE 17 SCALED CENTRIFUGAL COMPRESSOR PROGRAM RESULTS

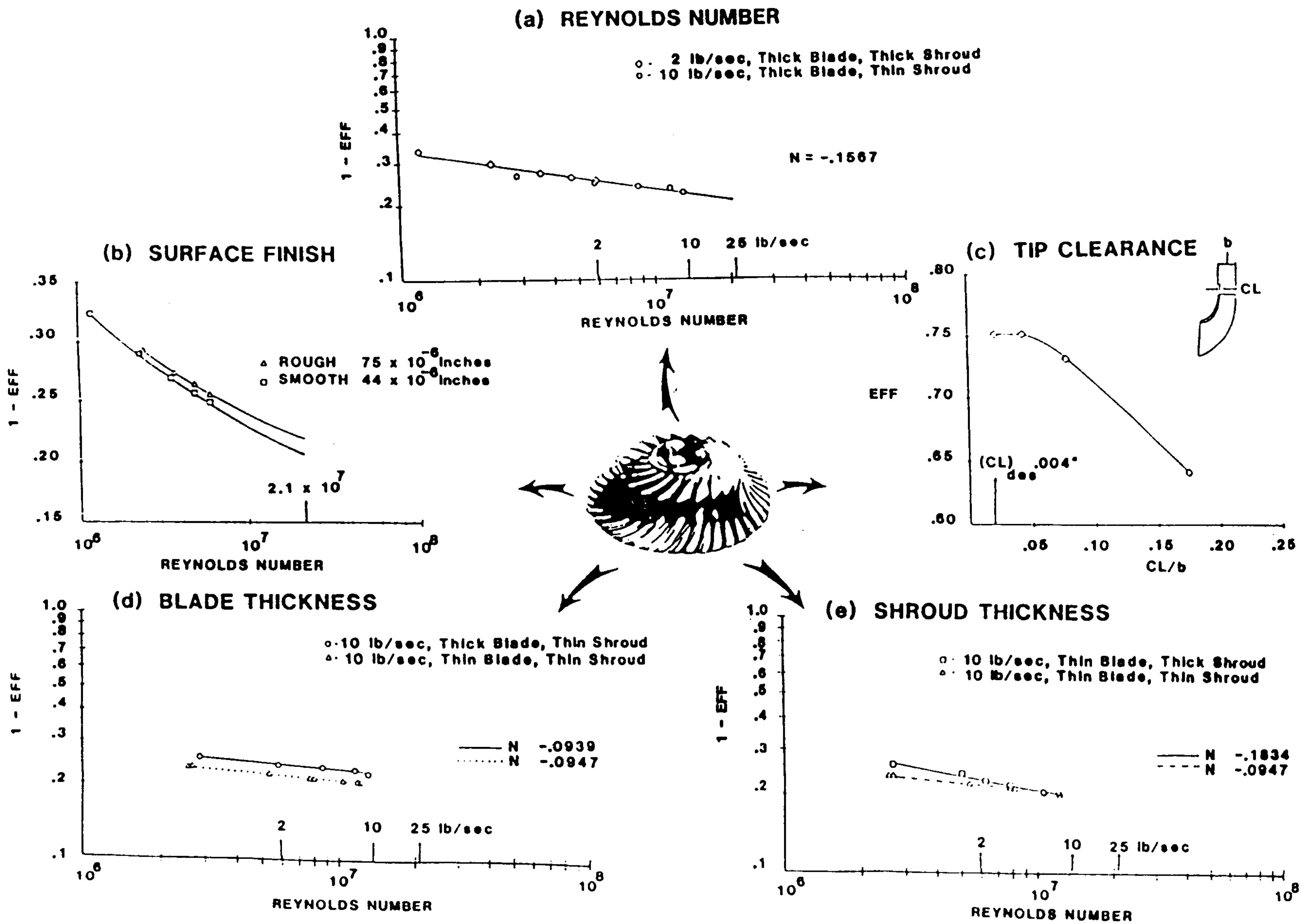

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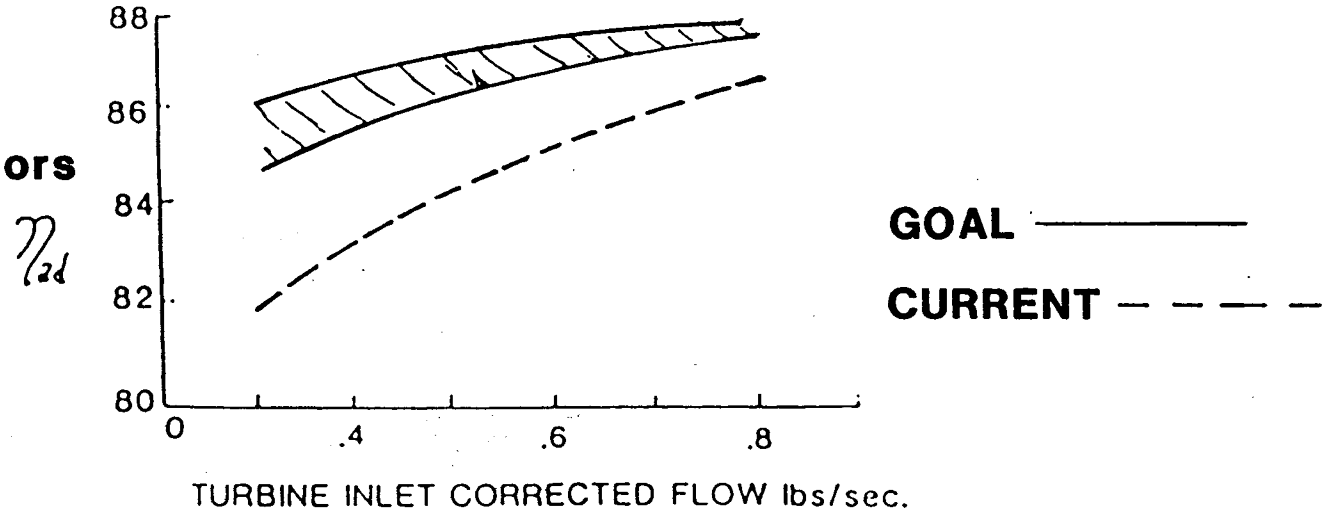
FIGURE 18: TURBINE EFFICIENCY

(a) Gas Generators



(b) Axial

Power Turbines

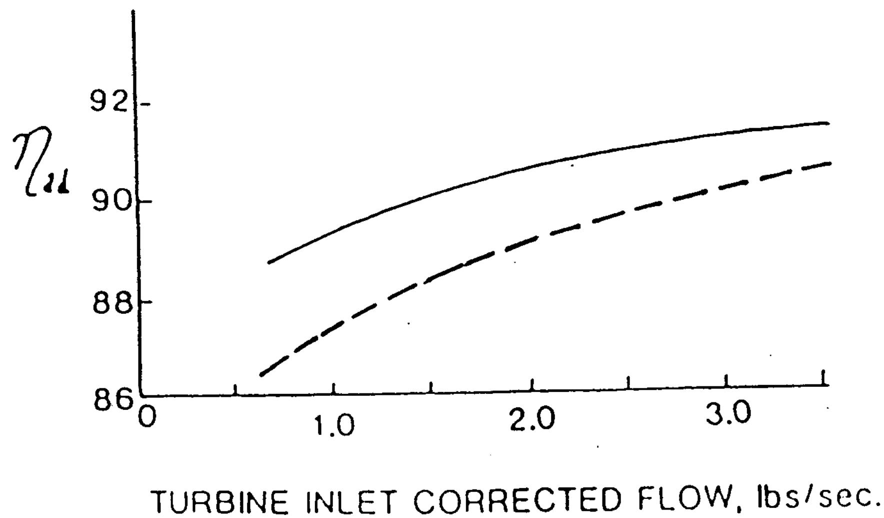
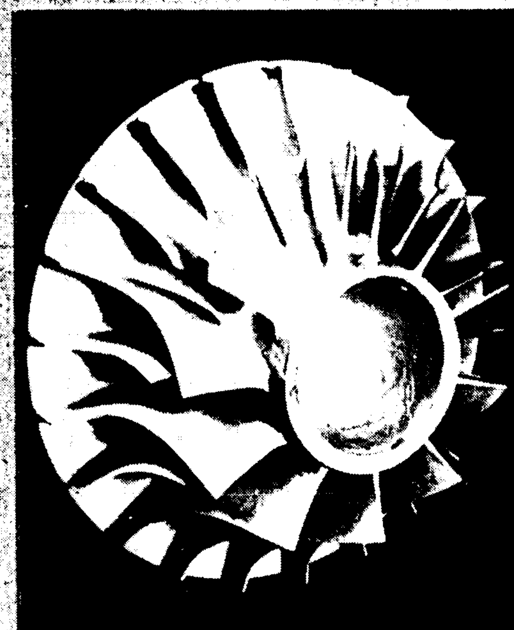


FIGURE 19
SMALL TURBINE AERODYNAMICS

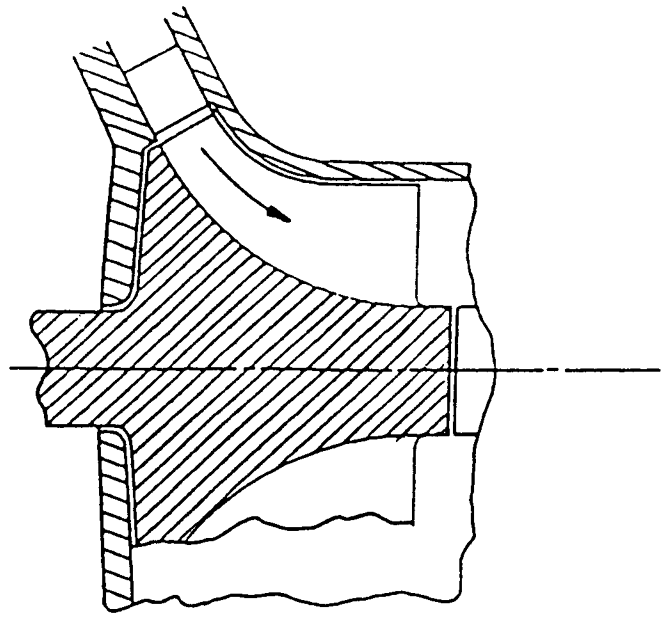
AXIAL TURBINE



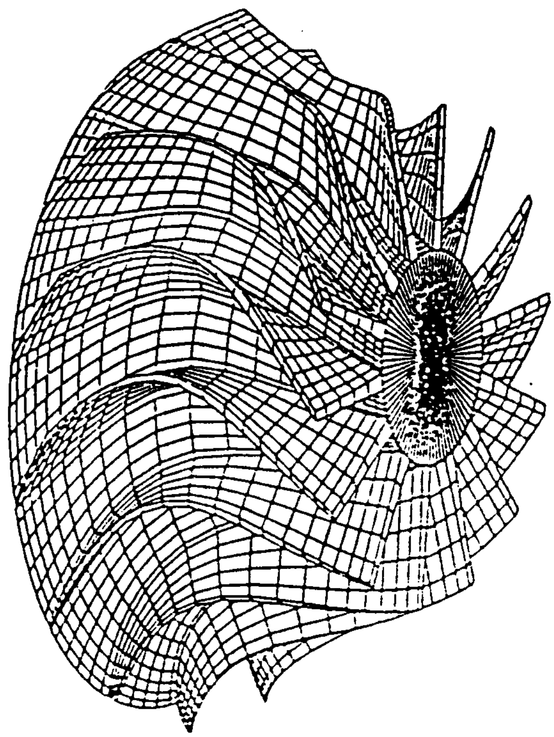
RADIAL TURBINE

- ENDWALL EFFICIENCY IMPROVEMENT
- LOW ASPECT RATIO AERODYNAMICS
- ENGINE ENVIRONMENT EFFECTS
- SMALL SIZE PENALTIES
- HIGHLY LOADED STAGES
- VARIABLE GEOMETRY
- MANIFOLDS & DUCTS
- COOLING AERODYNAMICS
- VALIDATION OF COMPUTATIONAL METHODS

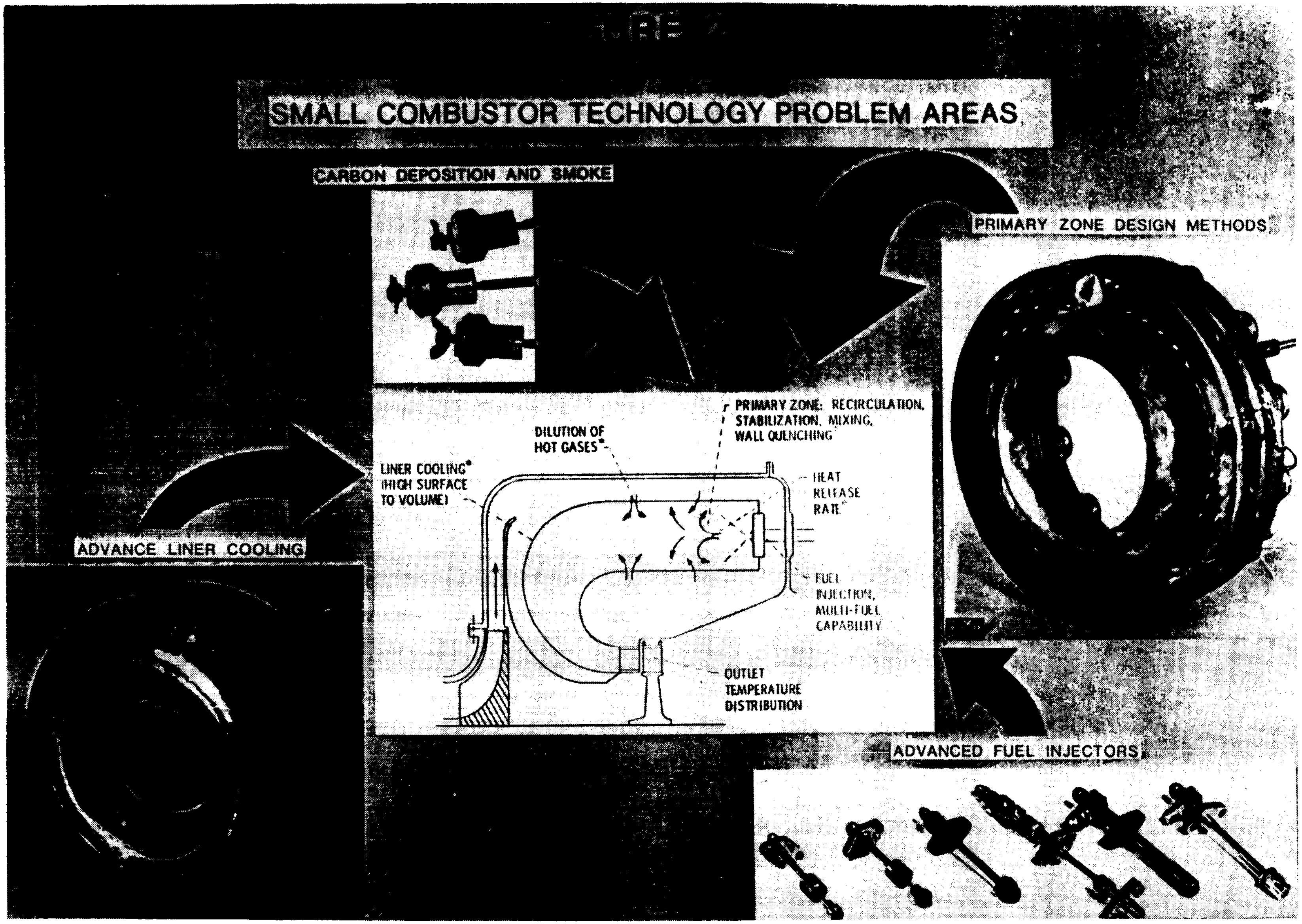
FIGURE 20: MIXED-FLOW TURBINE



(a) Flow Path

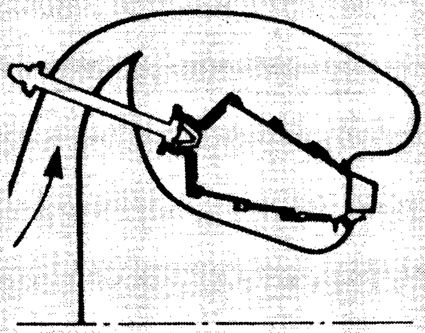


(b) Structural Analysis



SMALL COMBUSTOR TECHNOLOGY PROBLEM AREAS
CARBON DEPOSITION AND SMOKE
PRIMARY ZONE DESIGN METHODS
ADVANCE LINER COOLING
ADVANCED FUEL INJECTORS
PRIMARY ZONE: RECIRCULATION, STABILIZATION, MIXING, WALL QUENCHING
DILUTION OF HOT GASES*
LINER COOLING* (HIGH SURFACE TO VOLUME)
HEAT RELEASE RATE*
FUEL INJECTION, MULTI-FUEL CAPABILITY
OUTLET TEMPERATURE DISTRIBUTION

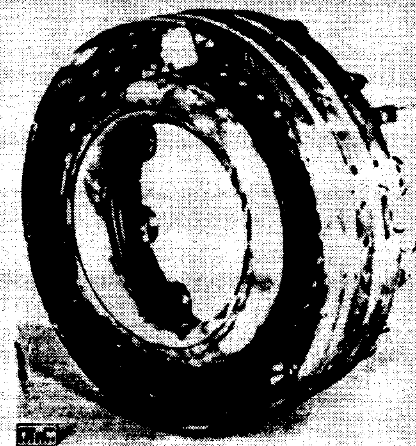
SMALL COMBUSTOR TECHNOLOGY



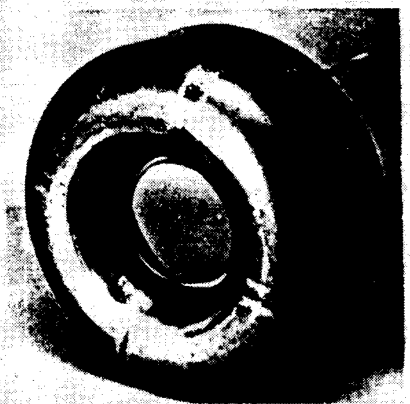
AXIAL FLOW

TECHNOLOGY NEEDS

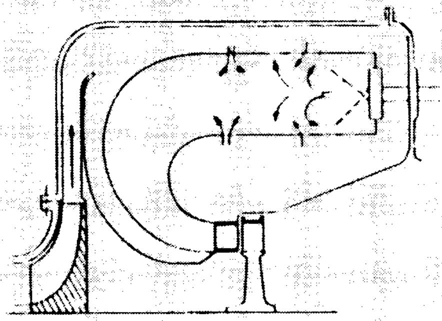
- INCREASED TEMP/PRESSURE CAPABILITY
- REDUCED AIR COOLANT REQUIREMENTS
- INCREASED DURABILITY/ RELIABILITY



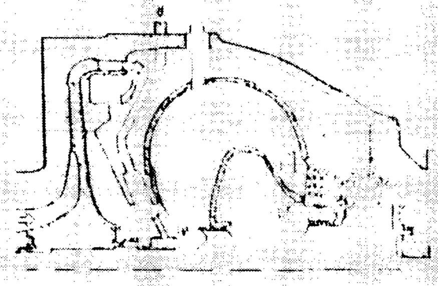
PRIMARY ZONE DESIGN METHODS



ADVANCED LINER COOLING



REVERSE FLOW



RADIAL OUTFLOW

FIGURE 22

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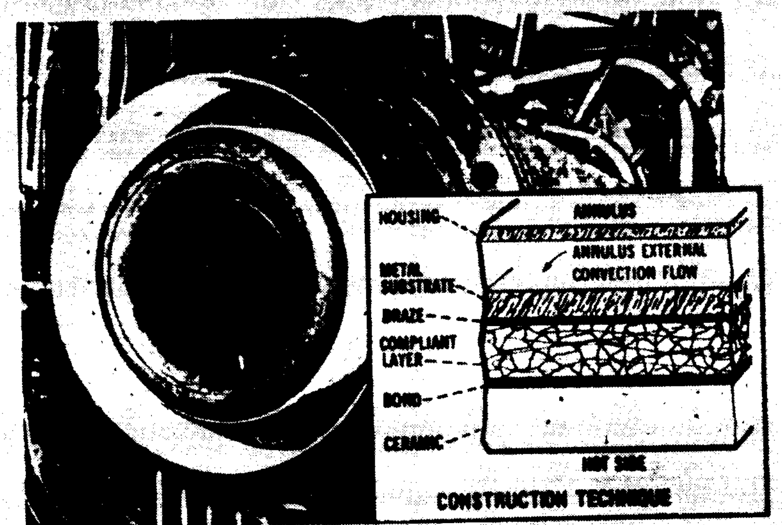
HIGH TEMPERATURE EVALUATION OF CERAMIC MATRIX LINER



SIGNIFICANT SFC REDUCTIONS REQUIRE

- APPLICATION OF ADVANCED CYCLES TO
  - SIMPLE CYCLES
  - REGENERATIVE CYCLES
- APPLICATION OF UNCOOLED CERAMICS
- EVOLUTION OF ADVANCED COMPONENTS

CERAMIC MATRIX LINER RESULTS

- OPERATION TO 2630 °F DEMONSTRATED
  - 300° HIGHER THAN CURRENT
  - FILM/TRANSPIRATION COOLING NEED ELIMINATED
- NON-STRATEGIC MATERIALS USED

COMPARISION OF COMBUSTOR LINER CONCEPTS

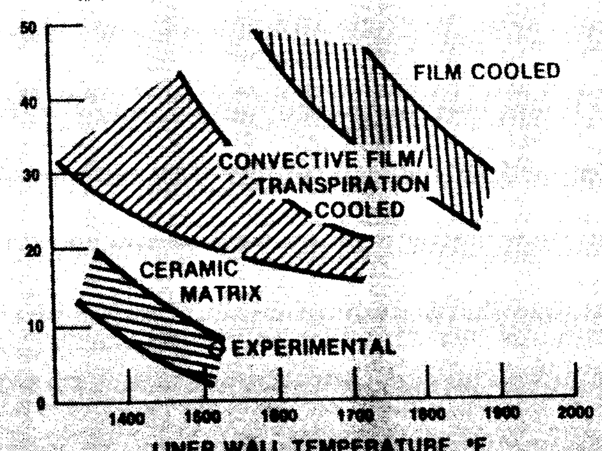


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FIGURE 23