

NASA TM-83022-1458
11-22-82

NASA-TM-83022 19850002643

Turbine Engine Hot Section Technology (HOST)

Material presented at a government-industry
information exchange workshop held at
NASA Lewis Research Center, Cleveland, Ohio,
October 19-20, 1982

October 1982

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CONTENTS

	Page
AN OVERVIEW OF NASA'S TURBINE ENGINE HOT SECTION TECHNOLOGY (HOST) PROJECT Daniel J. Gauntner and C. Robert Ensign, NASA Lewis Research Center	1
<i>PRE-HOST ACTIVITIES</i>	
PRE-HOST ACTIVITIES FOR LIFE PREDICTION Gary R. Halford, NASA Lewis Research Center.	9
EQUIVALENT DAMAGE - A CRITICAL ASSESSMENT J. H. Laflen and T. S. Cook, General Electric Company.	11
BENCHMARK NOTCH TEST FOR LIFE PREDICTION P. A. Domas and J. Yau, General Electric Company, W. N. Sharp and M. Ward, Louisiana State University.	25
NONLINEAR STRUCTURAL AND LIFE ANALYSIS OF A TURBINE BLADE Albert Kaufman, NASA Lewis Research Center	39
NONLINEAR STRUCTURAL AND LIFE ANALYSIS OF A COMBUSTOR LINER Albert Kaufman, NASA Lewis Research Center	45
PRE-HOST HIGH TEMPERATURE CRACK PROPAGATION Thomas W. Orange, NASA Lewis Research Center	55
<i>INSTRUMENTATION</i>	
HOST INSTRUMENTATION R&D PROGRAM - OVERVIEW Norman C. Wenger, NASA Lewis Research Center	65
COMBUSTOR VIEWING SYSTEM William W. Morey, United Technologies Research Center	69
DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM Dennis Elmore, Pratt & Whitney Aircraft GPD.	75
HIGH TEMPERATURE STATIC STRAIN SENSOR DEVELOPMENT PROGRAM Charles Hulse, Frank Lemkey, Richard Bailey, and Howard Grant, United Technologies Research Center.	83
LASER SPECKLE TECHNIQUE FOR BURNER LINER STRAIN MEASUREMENTS Karl A. Stetson, United Technologies Research Center	93
HEAT FLUX SENSORS FOR BURNER LINERS AND TURBINE BLADES AND VANES W. G. Alwang, Pratt & Whitney Aircraft	101
HOT SECTION LASER ANEMOMETRY William C. Nieberding, NASA Lewis Research Center.	109

LASER ANEMOMETER OPTIMIZATION Robert V. Edwards, Case Western Reserve University	113
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TURBINE HEAT TRANSFER

TURBINE HEAT TRANSFER John E. Rohde, NASA Lewis Research Center.	119
GAS SIDE HEAT TRANSFER - 2-D FLOW Ronald York, Detroit Diesel Allison.	137
GAS FLOW ENVIRONMENT AND HEAT TRANSFER NONROTATING 3-D PROGRAM Roy J. Schulz, The University of Tennessee Space Institute	149
JET ARRAY IMPINGEMENT FLOW DISTRIBUTIONS AND HEAT TRANSFER CHARACTERISTICS - EFFECTS OF INITIAL CROSS FLOW AND NONUNIFORM ARRAY GEOMETRY L. W. Florschuetz and D. E. Metzger, Arizona State University.	161
COOLANT PASSAGE HEAT TRANSFER WITH ROTATION John E. Rohde, NASA Lewis Research Center.	175

STRUCTURES

STRUCTURAL ANALYSIS Robert H. Johns, NASA Lewis Research Center.	181
BURNER LINER THERMAL/STRUCTURAL LOAD MODELING R. J. Maffeo, General Electric Company	185
COMPONENT SPECIFIC MODELING Murray S. Hirschbein, NASA Lewis Research Center	197
3-D INELASTIC ANALYSES METHODS FOR HOT SECTION COMPONENTS - BRIEF DESCRIPTION Christos C. Chamis, NASA Lewis Research Center	203

FATIGUE

LIFE PREDICTION AND CONSTITUTIVE BEHAVIOR - OVERVIEW Gary R. Halford, NASA Lewis Research Center.	209
CREEP-FATIGUE LIFE PREDICTION FOR ENGINE HOT SECTION MATERIALS (ISOTROPIC) Vito Moreno, Pratt & Whitney Aircraft.	213
CONSTITUTIVE MODEL DEVELOPMENT FOR ISOTROPIC MATERIALS Albert Kaufman, NASA Lewis Research Center	215
LIFE PREDICTION AND CONSTITUTIVE MODELS FOR ANISOTROPIC MATERIALS Robert C. Bill, Propulsion Laboratory, AVRADCOM Research and Technology Laboratories, NASA Lewis Research Center.	223

HOST HIGH TEMPERATURE CRACK PROPAGATION Thomas W. Orange, NASA Lewis Research Center	227
<i>SURFACE PROTECTION</i>	
SURFACE PROTECTION - OVERVIEW Stanley R. Levine, NASA Lewis Research Center.	231
AIRFOIL DEPOSITION MODEL Fred J. Kohl, NASA Lewis Research Center	241
COATING LIFE PREDICTION Michael A. Gedwill, NASA Lewis Research Center	259
EFFECTS OF SURFACE CHEMISTRY ON HOT CORROSION LIFE - OVERVIEW John Merutka, NASA Lewis Research Center	263
<i>COMBUSTION</i>	
COMBUSTION HOT SECTION TECHNOLOGY MODELING David B. Ercegovic, Propulsion Laboratory, AVRADCOM Research and Technology Laboratories, NASA Lewis Research Center.	269
AEROTHERMAL MODELING ASSESSMENT S. Srivatsa, Garrett Turbine Engine Company.	273
AEROTHERMAL MODELING - PHASE I: A PROGRESS REPORT G. J. Sturgess, Pratt & Whitney Aircraft	283
AEROTHERMAL MODELING PROGRAM Milton Kenworthy, General Electric Company	301
DILUTION ZONE MIXING STUDIES J. D. Holdeman, NASA Lewis Research Center	307
DILUTION JET MIXING - PHASE I AND II R. Srinivasan, Garrett Turbine Engine Company.	315
COMBUSTION SYSTEM FOR RADIATION INVESTIGATIONS J. D. Wear, NASA Lewis Research Center	331
VALIDATION OF STRUCTURAL ANALYSIS METHODS USING THE IN-HOUSE LINER CYCLIC RIGS Robert L. Thompson, NASA Lewis Research Center	335
HOST LINER CYCLIC FACILITIES - FACILITY DESCRIPTION Donald Schultz, NASA Lewis Research Center	345
<i>CONCLUDING REMARKS</i>	
THE FUTURE OF HOST Daniel J. Gauntner, NASA Lewis Research Center	357

AN OVERVIEW OF NASA'S
TURBINE ENGINE HOT SECTION TECHNOLOGY (HOST) PROJECT

by

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and
C. Robert Ensign

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

EXTENDED ABSTRACT

The NASA Lewis Research Center is currently involved in a research project to investigate the physical phenomena affecting the durability of turbine engine hot section components. This effort, entitled the "Turbine Engine Hot Section Technology (HOST) Project", is investigating aerodynamic, thermal, structural, material behavior, and damage initiation phenomena to better understand the problem of durability, and to construct improved analysis methods for the more accurate prediction of component life.

Current turbine engine hot section components are exposed to higher temperature, higher stresses, and more severe thermal gradients than ever before. These components are typically made from high temperature superalloys and utilize protective coatings. As future turbine engines tend towards higher thrust to weight ratios and lower levels of specific fuel consumption, the needed technology will be more sophisticated and the operating environments will become more severe. The durability and efficiency goals of the hot section components operating in these adverse environments will be more difficult to achieve. Any shortfalls in achieving these goals could have significant effects on the overall availability and operating costs of America's military and civil turbine engine fleets.

The striving for improved hot section parts has traditionally had its basis in improved performance, fuel, and maintenance (durability) costs. NASA's Aircraft Energy Efficiency (ACEE) program achieved technology for improving all these areas, while aiming at the reduction of fuel consumption for commercial air transports (E³, ECI). Several review articles have shown the continuing importance of fuel prices and maintenance cost. Earlier work reviewed several Air Force engines, between 1958 and 1976, and found that the maintenance cost/fuel cost ratio increased from almost two to a level near six

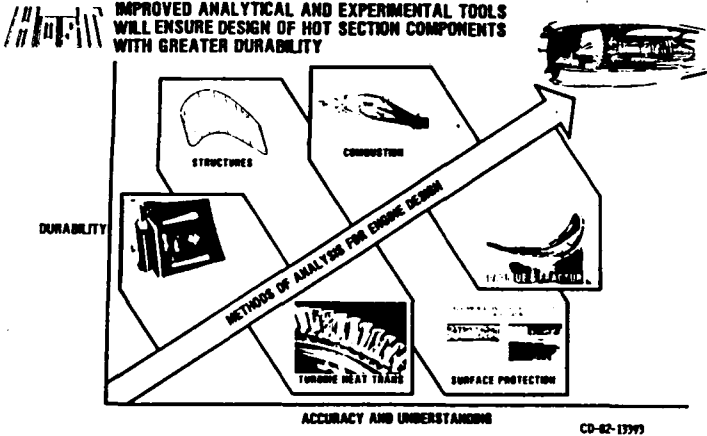
in that period. Maintenance costs were increasing much faster than were fuel costs. Later reviews compared Direct Operating Cost elements over the years 1968 and 1979. The period 1973-1979 suffered the well-known increase in fuel prices. Accordingly, as a part of DOC, maintenance costs decreased from near 25% in 1973 to near 15% in 1979, while fuel went from 25% to over 50%, for U.S. scheduled airlines. But on a cents per available seat mile basis, maintenance costs themselves still increased almost 25% between 1973 and 1979. NASA is addressing the technology requirements for reduction of fuel consumption in its Energy Efficient Engine and Advanced Turboprop efforts. The Turbine Engine Hot Section Technology Project addresses the durability aspects of maintenance costs.

The research efforts under the HOST Project banner are addressed to hot section component problems. The covered disciplines include Structural Analysis, Fatigue/Fracture, Surface Protective Coatings, Combustion, Turbine Heat Transfer, and High Temperature Instrumentation. Structural Analysis includes research into thermal mechanical load models, component geometry specific models, and 3-D inelastic analysis methods development. Fatigue and Fracture includes constitutive model development for both isotropic and anisotropic materials, including single crystal and directionally solidified forms. It also includes research in life prediction methods for creep-fatigue interactions, and elastoplastic crack propagation. The Surface Protection research includes studies of corrosion and oxidation phenomena, environmental mechanics models, and metallic and thermal barrier coating analysis method developments. The Combustion work includes aerothermal model assessment and development, dilution jet modeling, high pressure flame radiation/heat flux testing, and development of a thermal structural cyclic test facility. The Turbine Heat Transfer area is studying 2-D and 3-D flow and heat transfer on airfoil external boundaries emphasizing boundary layer transition and viscous modeling. It also investigates coolant passage heat transfer, including midchord jet impingement cooling and rotational passage effects. Instrumentation is being developed to obtain high temperature benchmark quality data to develop and verify the analysis methods. These include flow sensors (LDV), heat flux sensors (thin film), strain sensors (1800°F static thin film), gas temperature sensors (frequency compensated) and hot section optical viewing systems.

The HOST Project began in 1981 and is a continuing program in the NASA Headquarters Office of Aeronautics and Space Technologies (OAST) Materials and Structures Program Office. New research efforts into hot section durability problems are developed and will be continuously developed in concert with NASA, Department of Defense, and aircraft turbine engine industry long-range goals, objectives, and needs. By December, 1982, two dozen major contracts and selected interdisciplinary grants will be in place and supported with HOST resources. An approximately equal amount of HOST resources will be expended over the next five-year period to support in-house research, facility updates, and other support.

The HOST Project is operated under the Matrix Management approach, utilizing a small project office staff to manage a much larger group of professional and support personnel to conduct research and accomplish the technical objectives. The HOST Project has been subjected to the NASA Headquarters Technology Control clause "For Early Domestic Dissemination (FEDD)." The intent of the clause is to encourage early transfer of NASA sponsored and developed technology to the U.S. domestic aerospace community, while simultaneously restricting the access by non-domestic sources to the benefits of the technology for up to two years after first publication.

Overall management and control over project progress is maintained through monthly technical and financial reporting vehicles. These reports are only available to government personnel. More formal and more comprehensive reviews of progress are contained in annual reports, final reports and in presentations at the HOST Annual Contractor Workshop held at NASA Lewis in the fall of each year. These formal reports and attendance at the workshop are only available to representatives of the U.S. Government and to U.S. domestic companies.



CD-62-17393

SEMINAR ON TURBINE ENGINE HOT SECTION TECHNOLOGY

- JANUARY 22, 1981
- GOVERNMENT-INDUSTRY-UNIVERSITIES
- OVER 100 ATTENDEES
- FROM 50 ORGANIZATIONS
- RESULT: GREATER AWARENESS OF HOST WITH SEVERAL JOINT-BID PROPOSALS.



HOST AGENCY

- INDUSTRY AWARENESS AND IMPROVEMENT OF PRIME IMPORTANCE
- ENGINE MANUFACTURERS, ASSEMBLERS, SMALL BUSINESSES
- A NEVER-ENDING NEED

"DEVELOP ANALYTICAL MODEL AND PREDICTIVE TOOLS FOR THE DURABILITY AND LIFE ASSURANCE DESIGN OF HOT SECTION COMPONENTS OF ADVANCED AIRCRAFT TURBINES"

*** PASO DOCUMENT

AF WRIGHT AERONAUTICAL LABS.	AUGUST 28, 1980	GARRETT TURBINE ENGINE	MAY 12, 1981
GARRETT RESEARCH	SEPTEMBER 12, 1980	AVCO-LYCORING	MAY 19, 1981
PRATT AND WHITNEY	SEPTEMBER 12, 1980	AF WRIGHT AERONAUTICAL LABS.	MAY 21, 1981
GENERAL ELECTRIC	SEPTEMBER 15, 1980	B AND M TECHNOLOGICAL SERVICES, INC.	JULY 1, 1981
GENERAL ELECTRIC	SEPTEMBER 23, 1980	PRATT AND WHITNEY	JULY 21, 1981
GARRETT RESEARCH	OCTOBER 1, 1980	PRATT AND WHITNEY	AUGUST 11, 1981
INDUSTRY/UNIV. JOINT. WORKSHOP	OCTOBER 8-9, 1980	GENERAL ELECTRIC	AUGUST 27, 1981
AF WRIGHT AERONAUTICAL LABS	OCTOBER 16, 1980	PRATT AND WHITNEY	SEPTEMBER 11, 1981
IN AND S HEADQUARTERS REVIEW	OCTOBER 20, 1980	IN AND S HEADQUARTERS REVIEW	SEPTEMBER 14-15, 1981
AERONAUTICAL ADVISORY COMMITTEE	OCTOBER 30, 1980	GENERAL ELECTRIC	OCTOBER 7, 1981
PRATT AND WHITNEY	NOVEMBER 6, 1980	SCIENCE APPLICATION INC.	OCTOBER 8, 1981
PRATT AND WHITNEY	NOVEMBER 14, 1980	INT - PROFESSOR J. MAHR	OCTOBER 16, 1981
DOUGLAS AIRCRAFT	NOVEMBER 19, 1980	AIR FORCE WRIGHT AERONAUTICAL LABS.	NOVEMBER 30, 1981
GENERAL ELECTRIC	DECEMBER 3, 1980	MANC, INC.	DECEMBER 3, 1981
GARRETT RESEARCH	DECEMBER 9, 1980	TOW	DECEMBER 3, 1981
BOEING	DECEMBER 12, 1980	AIR FORCE WRIGHT AERONAUTICAL LABS.	DECEMBER 9, 1981
GENERAL ELECTRIC	DECEMBER 15, 1980	DETROIT DIESEL ALLISON	DECEMBER 10, 1981
DETROIT DIESEL ALLISON	DECEMBER 16, 1980	PRATT AND WHITNEY	DECEMBER 10, 1981
IN AND S HEADQUARTERS REVIEW	DECEMBER 17, 1980	YALE	DECEMBER 11, 1981
S AND T SENIOR STEERING COMMITTEE	DECEMBER 22, 1980	AIR FORCE WRIGHT AERONAUTICAL LABS.	DECEMBER 17-18, 1981
AFWAL/NASA "LOVE HP"	JANUARY 21, 1981	SVERDRUP TECH. INC.	JANUARY 20, 1982
HOST SEMINAR	JANUARY 22, 1981	DETROIT DIESEL ALLISON	JANUARY 20, 1982
LOCKHEED	JANUARY 23, 1981	SCIENTIFIC RESEARCH ASSOCIATES	JANUARY 26, 1982
GARRETT RESEARCH	JANUARY 23, 1981	AIR FORCE WRIGHT AERONAUTICAL LABS.	JANUARY 29, 1982
SHAWER RESEARCH	FEBRUARY 4, 1981	PRATT AND WHITNEY	FEBRUARY 11, 1982
PRATT AND WHITNEY	FEBRUARY 11, 1981	NASA HEADQUARTERS	FEBRUARY 12, 1982
LOCKHEED	FEBRUARY 12, 1981	PRATT AND WHITNEY	FEBRUARY 16, 1982
GARRETT TURBINE ENGINE	FEBRUARY 13, 1981	DETROIT DIESEL ALLISON	FEBRUARY 16, 1982
FRANKLIN INSTITUTE	MARCH 1, 1981	AIR FORCE WRIGHT AERONAUTICAL LABS.	MARCH 3, 1982
AVCO-LYCORING	MARCH 26, 1981	PRATT AND WHITNEY	MARCH 11, 1982
DETROIT DIESEL ALLISON	APRIL 15, 1981	PRATT AND WHITNEY	MARCH 16, 1982
GENERAL ELECTRIC	APRIL 22, 1981	TELEDYNE C&I	MAY 6, 1982
PRATT AND WHITNEY	APRIL 23, 1981	AVCO-LYCORING	MAY 12, 1982
GARRETT TURBINE ENGINE	APRIL 27, 1981	UNIVERSITY OF TENNESSEE SPACE INSTITUTE	MAY 20, 1982
AIRCRAFT ENGINE DIAGNOSTICS CONFERENCE	MAY 6-7, 1981		

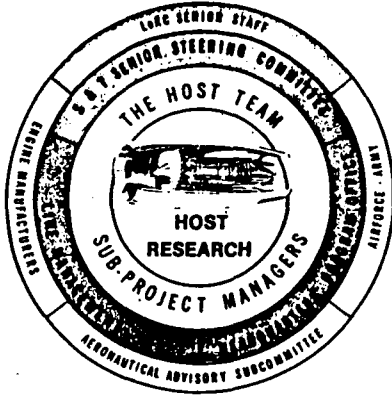
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AERONAUTICS LONG RANGE PLAN

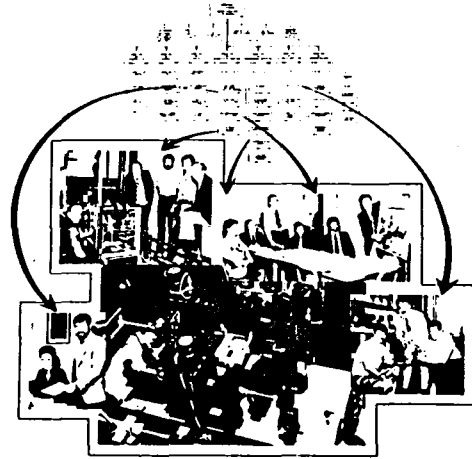
OBJECTIVES

- 0 ESTABLISH RECOGNITION OF IMPORTANCE OF NASA AERONAUTICS TO BOTH CIVIL AND MILITARY AVIATION.
- 0 PROVIDE U.S. R&T CAPABILITY BY MAINTAINING RESEARCH CENTERS IN POSITIONS OF UNDISPUTED EXCELLENCE IN FACILITIES, COMPUTATIONAL CAPABILITY AND TECHNICAL STAFF.
- 0 RESTORE A BALANCED AERONAUTICS PROGRAM CONSISTING OF DISCIPLINE RESEARCH, SYSTEMS RESEARCH AND SELECTED PROOF OF CONCEPT ACTIVITIES WITH EMPHASIS ON COMMON APPLICABILITY TO BOTH CIVIL AND MILITARY AVIATION.
- 0 STRENGTHEN NASA UNIVERSITY PARTNERSHIP IN AERONAUTICS R&T.
- 0 STRENGTHEN USER INTERFACES TO PROMOTE TECHNOLOGY TRANSFER.

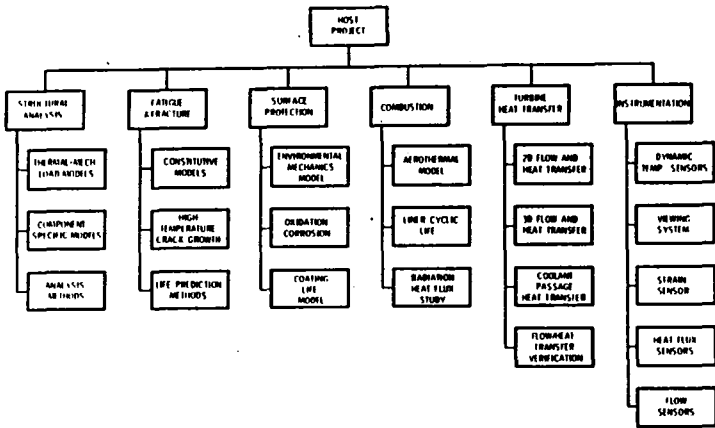
THE LEVELS OF REVIEW



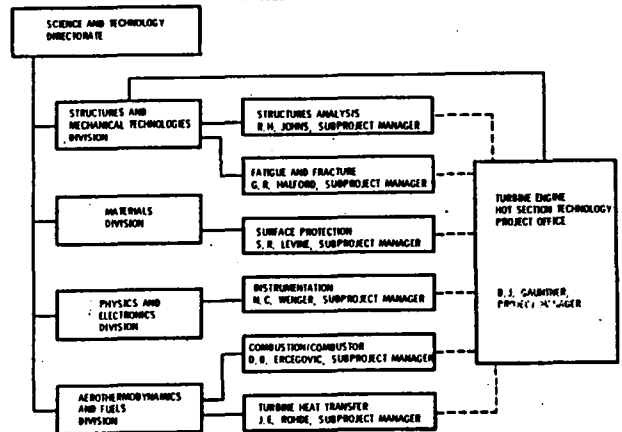
LEWIS MATRIX MANAGEMENT OF MULTIDISCIPLINARY PROGRAMS



WORK BREAKDOWN STRUCTURE



PROJECT OFFICE ORGANIZATION



RELATIONSHIP TO DOD PROGRAMS

- AFVAL AFOSR ARMY NAVY
- COOPERATE
- COORDINATE
- JOINT REVIEWS
- PROPOSAL TEAMS
- COMPATIBLE

PROCUREMENT STRATEGY

- COMPETITIVE CONTRACTS (CPFF)
- THREE, FOUR AND FIVE YEAR CONTRACTS, WITH OPTIONS
- SOURCE EVALUATION BOARD PROCEDURES FOR A GUIDELINE
- INCREMENTAL FUNDING
- SELECTED DUAL AWARDS

1982 CONTRACTS/GRANTS

GE - THERMAL DATA TRANSFER MODULE
PIA - ISOTROPIC CREEP FATIGUE LIFE
YALE - MASS TRANSFER THEORY
GARRETT - DILUTION JET MIXING
DUA - 2-D FLOW/HEAT TRANSFER
ARIZONA STATE - MULTIPLE JET IMPINGEMENT
UTSI - 3-D VISCOUS FLOW
PWA - GAS TEMPERATURE SENSOR
UTRC - HOT SECTION VIEWING SYSTEM
UTRC - STATIC STRAIN SENSOR
PIA }
GARRETT } - AEROTHERMAL MODEL ASSESSMENT
GE }

1982 CONTRACTS/GRANTS

- COMPONENT-SPECIFIC MODELING
SYRACUSE - FATIGUE CRACK GROWTH MECHANISMS
PIA - COOLANT PASSAGE HEAT TRANSFER
- 3-D INELASTIC METHODS
PIA - HEAT FLUX SENSORS

STRUCTURAL ANALYSIS

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
THERMAL DATA TRANSFER								STREAMLINED TECHNOLOGY LINKING THERMAL AND STRUCTURAL METHODS
COMPONENT SPECIFIC MODELING								SOFTWARE TECHNOLOGY FOR COMPONENT RELATED, TIME-DEPENDENT, THERMAL-MECHANICAL LOAD HISTORY
3D INELASTIC ANALYSES								ADVANCED 3-D INELASTIC STRUCTURAL/STRESS ANALYSIS METHODS AND SOLUTION STRATEGY
LINER CYCLIC RIG		(M)						BURNER STRUCTURAL/LIFE ANALYSIS
HIGH TEMPERATURE STRUCTURES LAB		(M)						INTEGRATED EXPERIMENTAL/ANALYSIS RESEARCH
MATERIAL CONSTITUTIVE MODELING			(M)					MATERIAL BEHAVIOR TECHNOLOGY

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FATIGUE & FRACTURE

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
ISOTROPIC MATERIAL LIFE PREDICTION								CREEP-FATIGUE CRACK INITIATION MODELS
ANISOTROPIC LIFE/CONSTITUTIVE MODEL								CREEP-FATIGUE CRACK INITIATION MODELS
ISOTROPIC MATERIAL CONSTITUTIVE BEHAVIOR								UNIFIED CONSTITUTIVE MATERIALS MODEL
HIGH TEMPERATURE CRACK GROWTH								SMALL CRACK PROPAGATION MODEL
FATIGUE LAB VITALIZATION		(M)						COMPUTER CONTROLLED HIGH TEMPERATURE THERMAL MECHANICAL CYCLIC TESTING
FRACTURE MECHANICS LAB			(M)					

CR 87-1186

SURFACE PROTECTION

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
MASS TRANSFER PARTICLE DEPOSITION THEORY								THEORETICAL FRAMEWORK FOR MASS TRANSFER FROM COMBUSTION GASES
CORROSION LIFE PREDICTION								ROLE OF OXIDE SCALE AND COATING COMPOSITION IN HOT CORROSION
THERMAL BARRIER COATING ANALYSIS								TECHNOLOGY MODEL FOR TBC LIFE ANALYSIS
LIFE PREDICTION VERIFICATION								VERIFIED PREDICTION CAPABILITY
OXIDATION LIFE PREDICTION		(M)						TECHNOLOGY TO PREDICT CYCLIC OXIDATION/COATING DEGRADATION
DUAL CYCLE ATTACK			(M)					COMBINED CORROSION/OXIDATION EFFECTS
RIG/ENGINE CORRELATION VERIFICATION				(M)				VERIFIED CORRELATIONS

CR 87-1186

COMBUSTION

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
AEROTHERMAL MODELING ASSESSMENT								KEY MODEL AND DATA DEFICIENCIES IDENTIFIED
COMBUSTION MODELING DEVELOPMENT								NEW PHYSICAL MODELS AND COMPUTING METHODS
MULTIPLE JET DILUTION MIXING								EXIT TEMPERATURE PROFILE PREDICTION TECHNOLOGY
FLAME RADIATION/HEAT FLUX		(M)						HIGH PRESSURE FLAME RADIATION AND HEAT FLUX
DILUTION JET ANALYSIS			(M)					JET MIXING MODEL
LINER CYCLIC RIG								CYCLIC TEST FACILITY

CR 87-1186

TURBINE HEAT TRANSFER

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
2D FLOW AND HEAT TRANSFER - TRANSITION								FLOW AND FILM INFLUENCE ON TRANSITION AND DURATION
- AIRFOIL INTERACTION								MODEL 2D ROTATION HEAT TRANSFER AIRFOIL INTERACTION EFFECTS
3D FLOW AND HEAT TRANSFER - NON ROTATING								VISCOUS FLOW MODELING OF 3D VELOCITIES
- ROTATING								VISCOUS MODELING OF 3D ROTATION EFFECTS
MICRODROPPING								MULTIPLE JET IMPINGEMENT MODEL
COOLANT PASSAGE HEAT TRANSFER								ROTATION EFFECTS ON INTERNAL HEAT TRANSFER
LDV TURBULENCE/VELOCITY STUDY		(M)						WARM TURBINE LDV SURVEY TECHNOLOGY
VERIFICATION (CASCADE, HPF)				(M)				FLOW, HEAT TRANSFER, STRAIN PREDICTION VERIFICATION

CR 87-1186

INSTRUMENTATION

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
GAS TEMPERATURE SENSOR								1 KHz RESPONSE DYNAMIC GAS TEMPERATURE PROBE
HOT SECTION VIEWING SYSTEM								STEREOSCOPIC FIBER OPTIC VIEWING OF OPERATING COMPONENT
STATIC STRAIN SENSOR								MINIATURE STRAIN GAGE 1800° F CAPABILITY
STRAIN SYSTEM TECHNOLOGY								GAGE APPLICATION TECHNOLOGY
HEAT FLUX SENSORS								TOTAL FLUX SENSORS
ADVANCED STRAIN ALLOYS								MULTIPLE ELEMENT ALLOYS FOR STRAIN GAGES
STRAIN GAGE LAB		(M)						HOUSE GAGE EVALUATIONS
FLOW MEASUREMENTS			(M)					LASER ANEMOMETER FOR LeRC SYSTEM

CR 87-1186

PRE-HOST ACTIVITIES FOR LIFE PREDICTION

Gary R. Halford

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In late 1978, it became clear that a Hot Section Durability Program would be inevitable, although direct funding would not be available for nearly two years. Therefore, existing R&D monies were redirected to fund some of the ground work for defining the needs of the intended HOST Program. As shown in Figure 1, contract programs were initiated in the general area of life prediction, and summaries of each will be presented:

Blade Tip Durability Analysis	GE	} - A. Kaufman
Combustor Liner Durability Analysis	P&WA	
Benchmark Notch Program	GE	} - J. Laflen
Equivalent Damage Program	GE	
High Temperature Crack Propagation	P&WA	- T. Orange

Contractor reports have been prepared for each program, and the results of each have been extremely helpful in charting the HOST efforts in the general area of life prediction.

FIG. 1 OVERVIEW PRE-HOST ACTIVITIES IN LIFE PREDICTION

G. R. Halford, LeRC

- BLADE TIP DURABILITY
- COMBUSTOR LINER DURABILITY

- BENCHMARK NOTCH
- EQUIVALENT DAMAGE

- HIGH TEMPERATURE CRACK GROWTH

Equivalent Damage - A Critical Assessment

J. H. Laflen and T. S. Cook
General Electric Co.
Cincinnati, Ohio

ABSTRACT

The overall goal of achieving improved life cycle management of aircraft engine, gas turbine components is a major industry thrust. Low Cycle Fatigue (LCF) crack initiation prediction, an important element of life cycle management as traditionally applied, may be overly conservative in estimating total cyclic life capability. Consequently, there is increasing pressure to improve predictive methods both for crack initiation and for subsequent crack propagation. This increased emphasis is the result of significantly higher component replacement costs as a consequence of more complex designs coupled with advanced materials and processing techniques. Moreover, despite added strength, the increased performance demands placed on engine components to achieve higher engine thrust-to-weight ratios have resulted in decreased cyclic lives. It is apparent, therefore, that significant cost savings can be realized through improved accuracy in high temperature, LCF crack initiation prediction.

In practical applications, engine components generally undergo very complex cycles of multiaxial strain, temperature, and dwell time, all of which add uncertainty to the problem of life prediction. During the process of designing and analytically evaluating the lifetime of gas turbine engine components, it is necessary to simplify many of these complexities to make the problem tractable. Nevertheless, there remain several important questions which can be clarified through the study of life prediction models: among these are how to address the problems of multiaxial loading, cumulative damage, and mean stress effects, and how they influence fatigue crack initiation life.

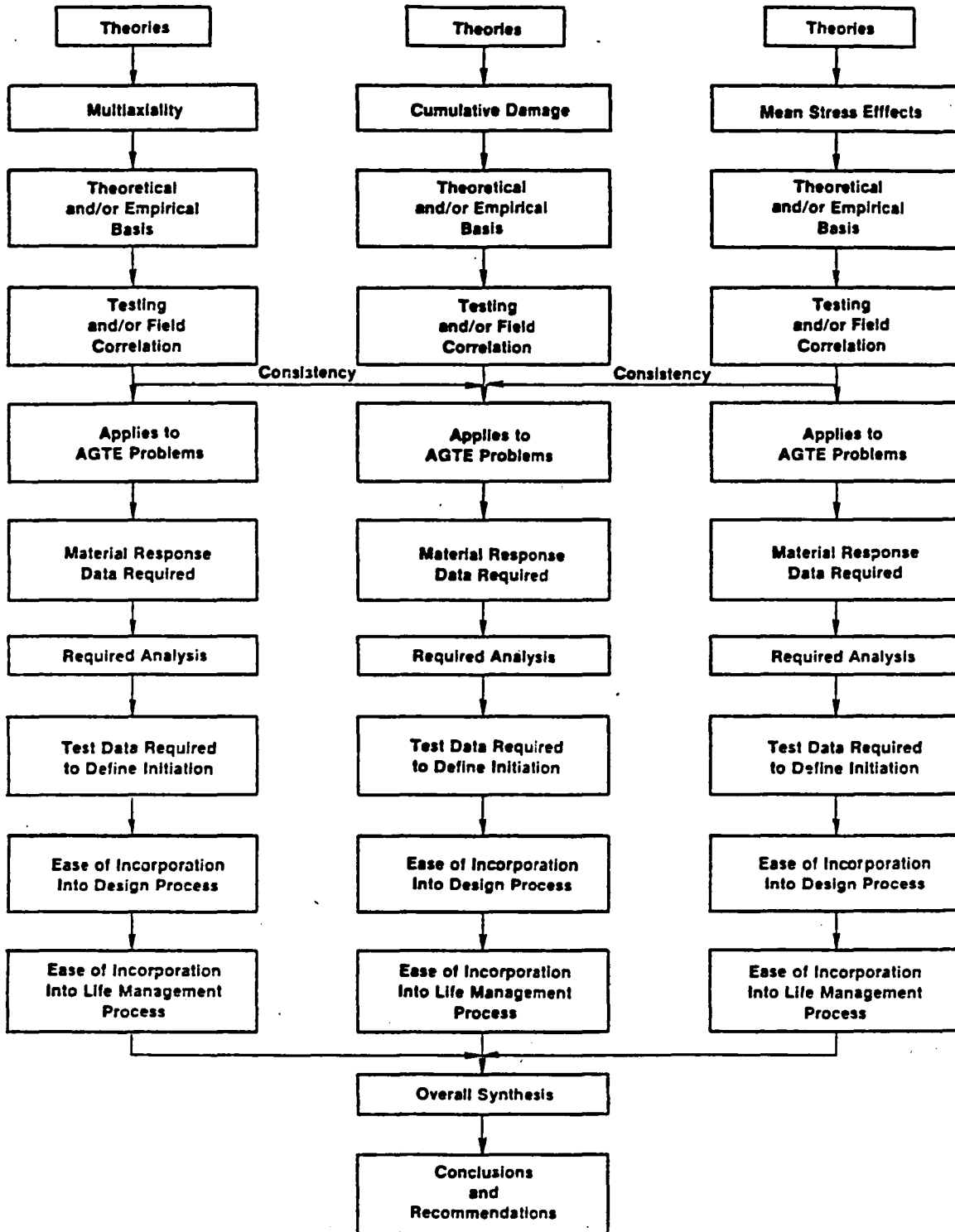
Consequently, an 18-month study was undertaken to determine the utility of equivalent damage concepts for application to hot section components of aircraft engines. Specifically, the topics studied were mean stress, cumulative damage, and multiaxiality. Other factors inherently linked to this study were the basic formulation of damage parameters at elevated temperatures and the fact that hot section components experience severe temperature fluctuations throughout their service lifetime. Both of these latter considerations placed constraints on the level of confidence with which recommendations regarding

specific equivalent damage criteria could be made since most such criteria were developed for use at lower temperatures. Despite this, the study yielded useful results, both from the point of view of data consolidation techniques under isothermal conditions and in producing concepts that will be useful in future studies.

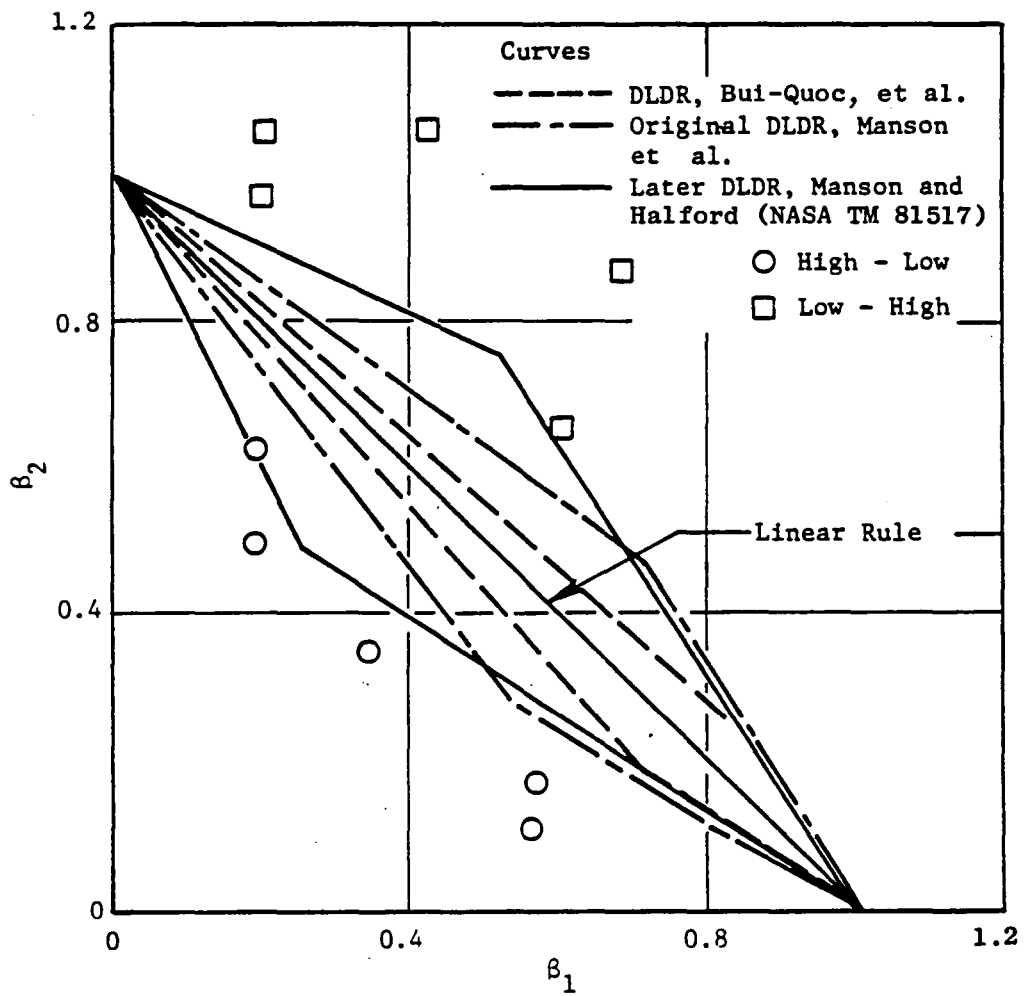
Through a literature review of the three areas of interest, the most promising techniques were extracted for further study. In the case of mean stress techniques, statistical evaluations of suggested approaches were made by comparing each technique to various isothermal data sets. Similarly, a combined literature review and isothermal data analysis technique was used to determine the most appropriate cumulative damage approach. In the case of multiaxiality, this decision process rested solely on the basis of the literature review. Following the initial screening, both the mean stress techniques and the cumulative damage concepts were tested against data sets involving either time dependent aspects of damage and/or varying temperature. The following conclusions were suggested by this study:

1. The equivalent strain relationship is the best mean stress criteria for low homologous temperatures and aircraft gas turbine engine alloys if the appropriate isothermal data are available. The Leis technique appears to be the best predictive mean stress parameter when data are not available to determine the exponent in the equivalent strain technique.
2. Thermal mechanical fatigue (TMF) experiments are required to verify the mean stress criteria. However, an elevated temperature mean stress criterion should be more conservative than normal approaches when applied to the out-of-phase TMF cycles which are normally encountered in hot path components.
3. Isothermal mean stress criteria should be verified in the longer life (design) regime. Most experimental results are obtained in the shorter cycle life range where more inelastic strain is present. A specific series of experiments was described.
4. The double linear damage rule is the best isothermal cumulative damage technique currently available. However, the technique was not consistent (it was conservative) in predicting a series of two-step tests where a temperature change was introduced into the second block of loading.

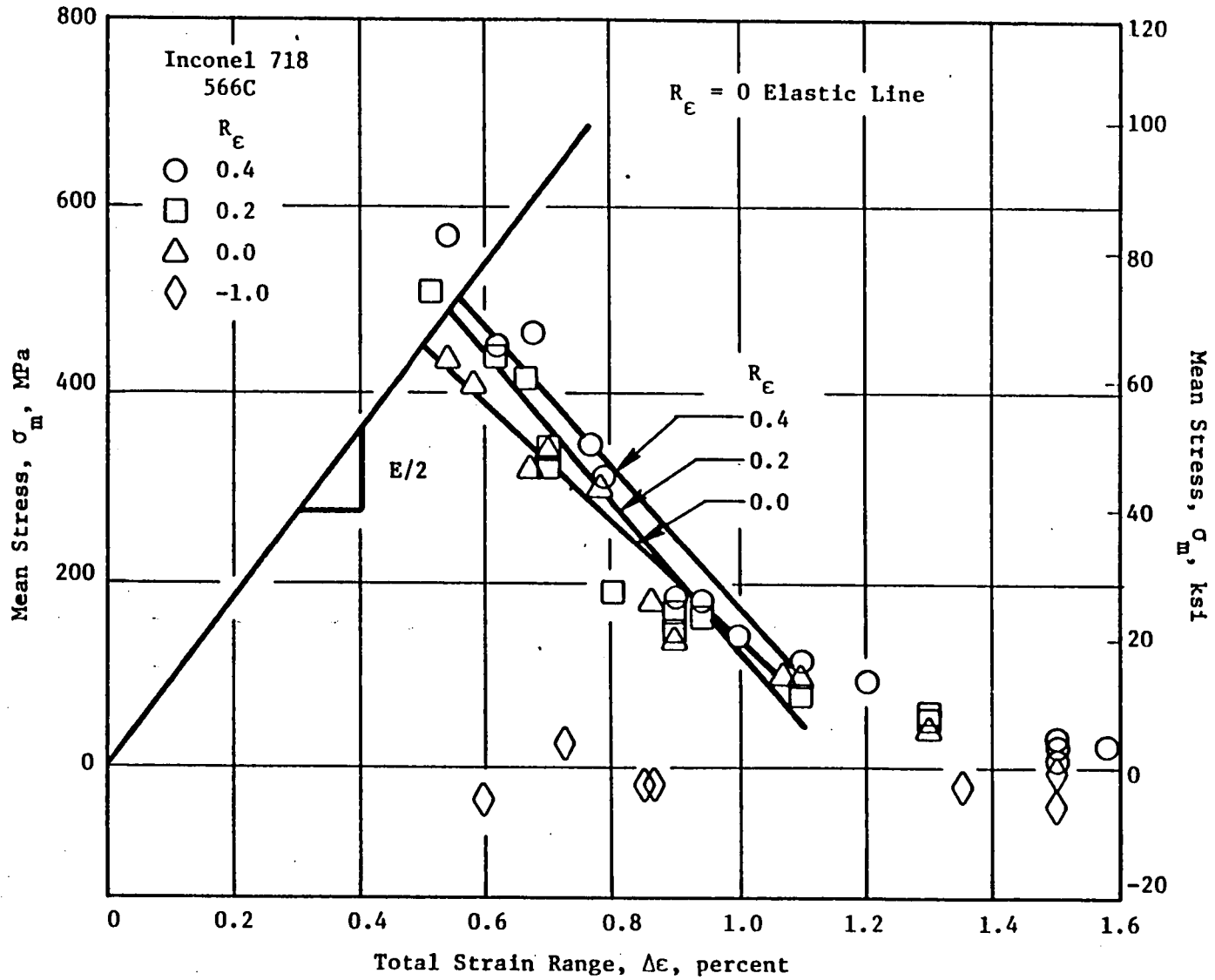
5. A test series was described for developing a consistent damage methodology. These tests used a so-called equal life technique to aid in the formulation of uniaxial equivalent damage criteria, and involved changing temperature during the experiments. Such a test series was viewed as a first step in developing a damage methodology for TMF.
6. A multiaxial equivalence criterion was developed based on a literature review. The criteria deemed important were the use of a triaxiality factor function and a consistent mean stress formulation.
7. A need exists for multiaxial test data on aircraft engine industry alloys. These experiments should concentrate on positive biaxial stress ratios, and should study the effect of mean stress. In general, multiaxial relationships for elevated temperature applications will remain an open research area for quite a while. Current research should concentrate on lower temperature phenomena which would suggest criteria at elevated temperatures. Uniaxial damage considerations at elevated temperatures appear complicated enough for the present.



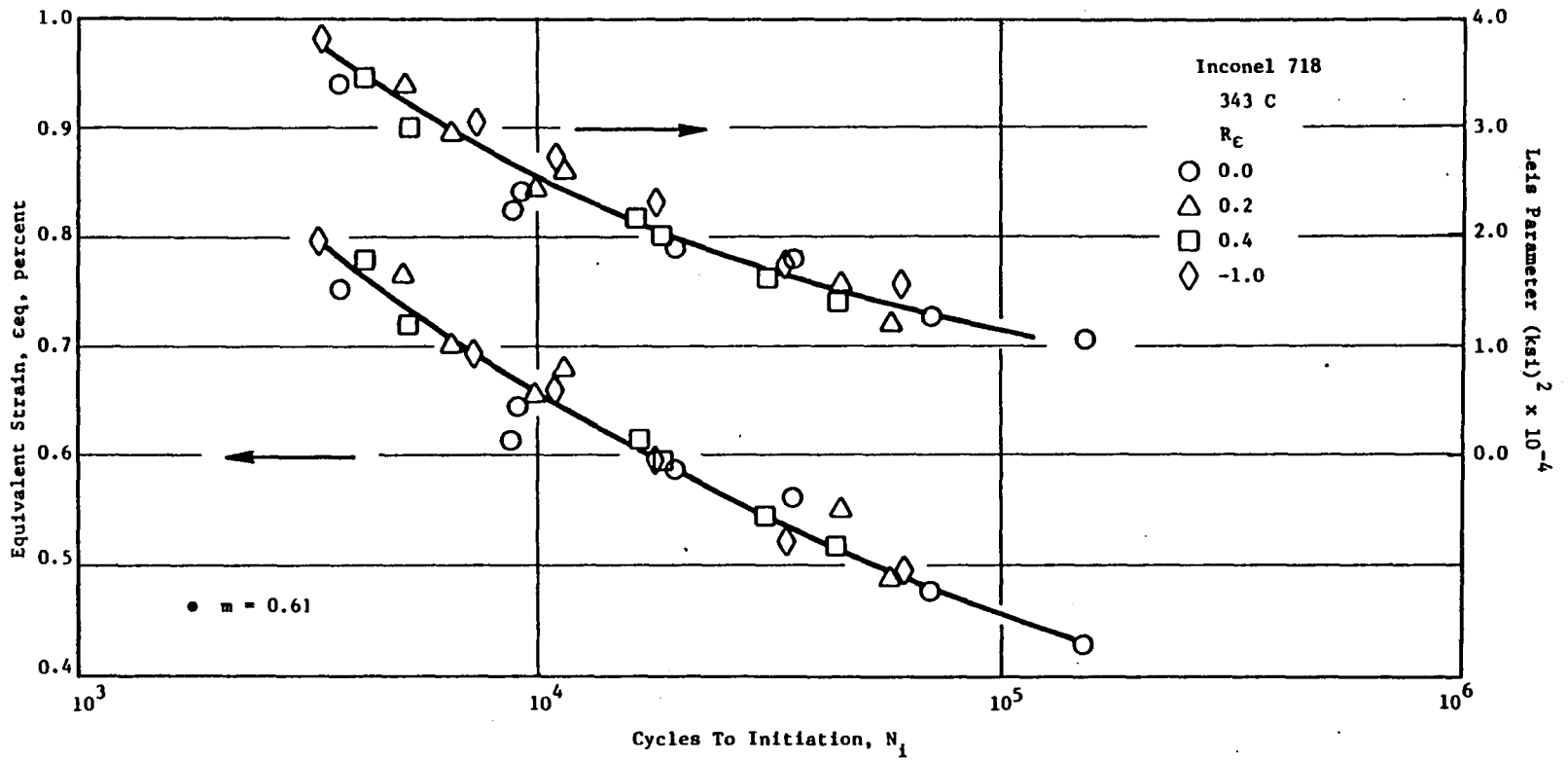
Criteria Diagram.



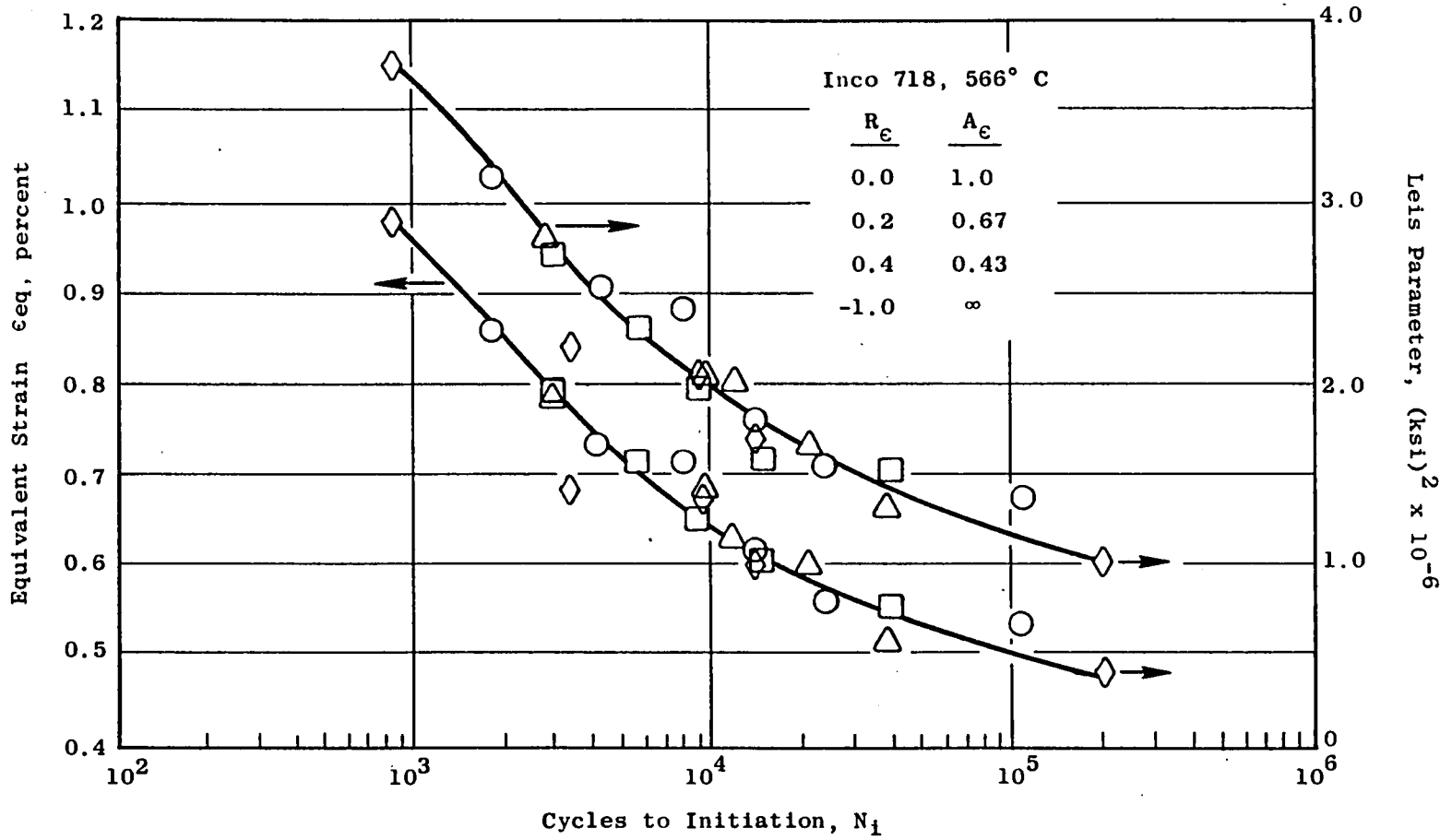
Sequence Test Results Versus Theories; Inconel 718 566° C
Three DLDR Theories.



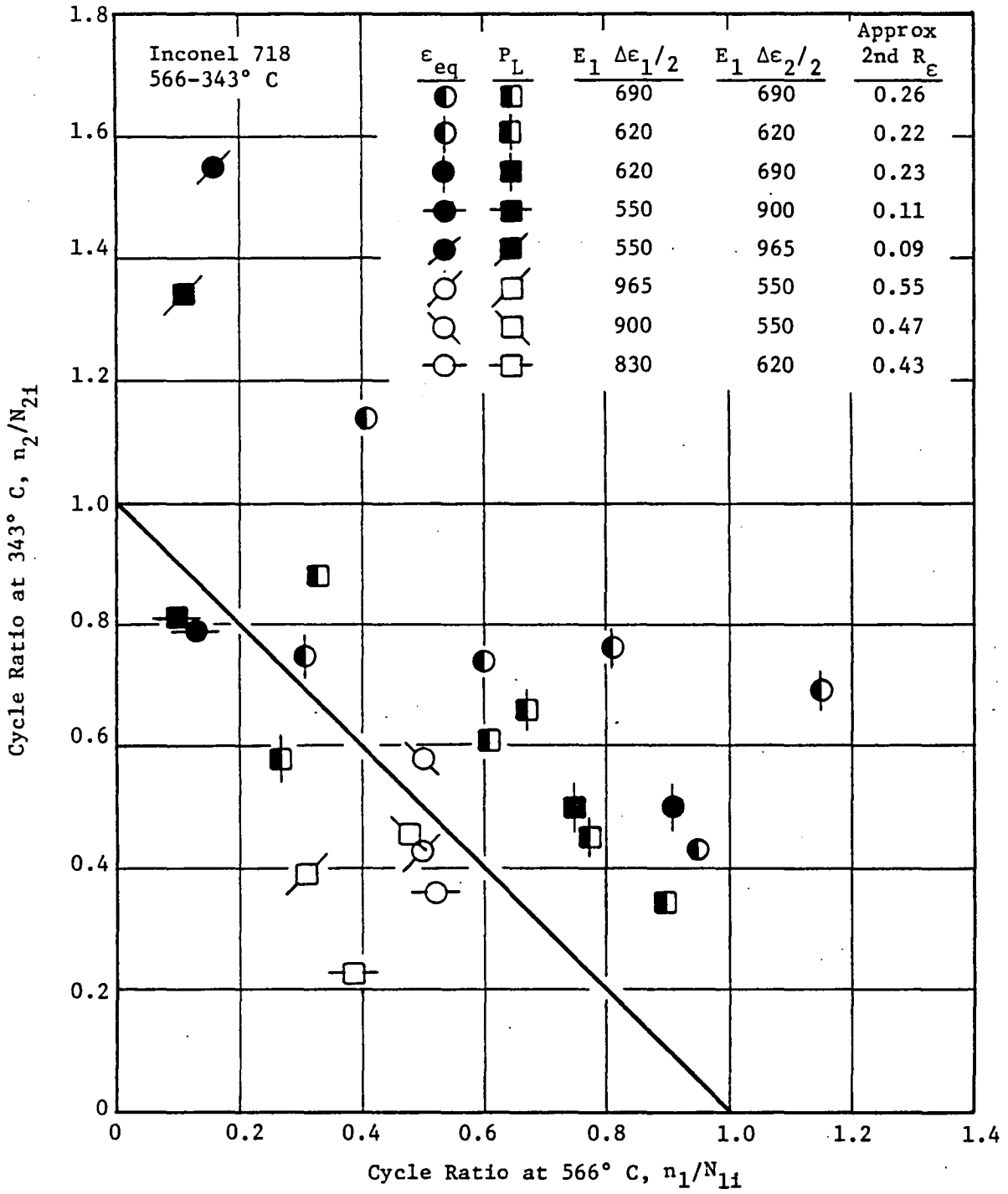
Mean Stress as a Function of Total Strain Range for Inconel 718 at 566° C, 20 CPM.



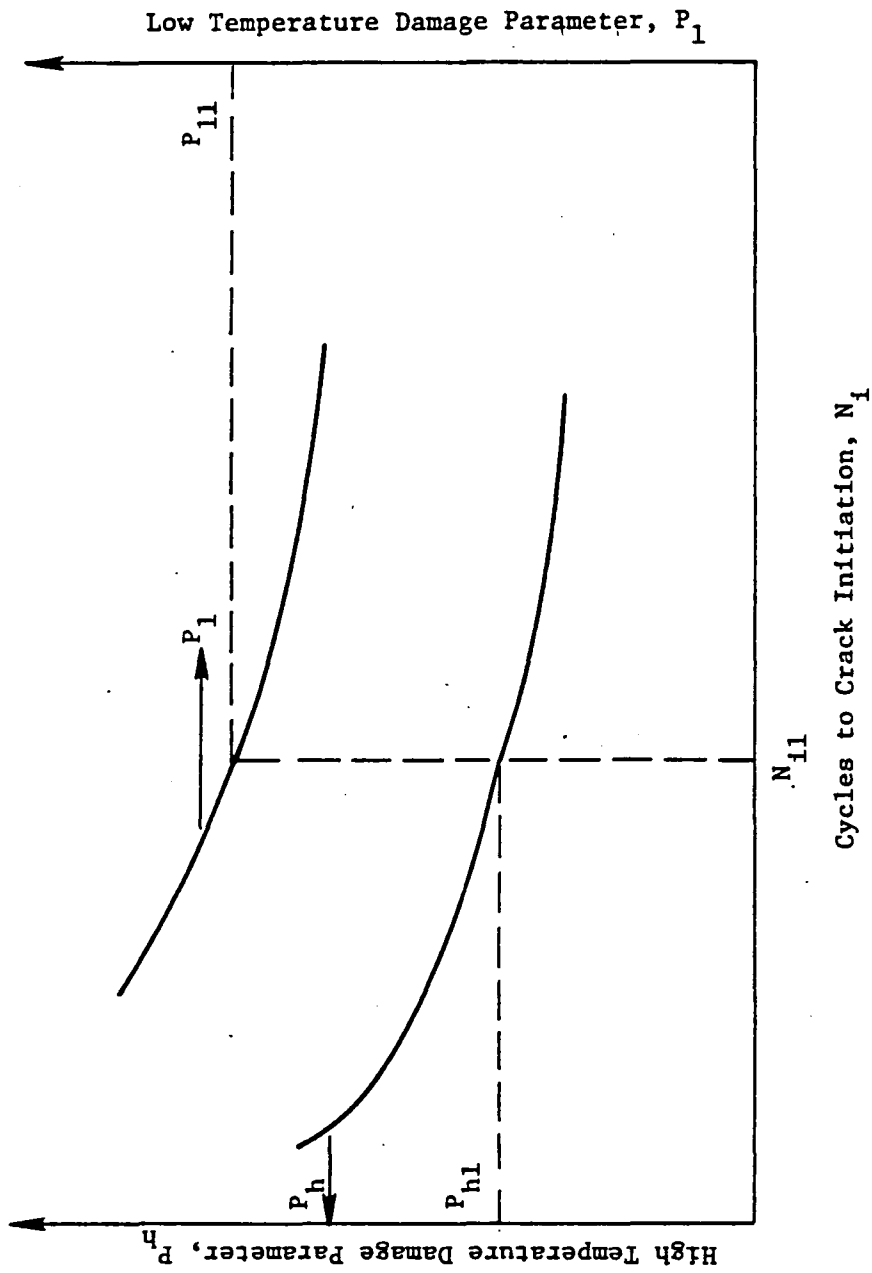
Baseline Data at 343° C Analyzed by the Equivalent Strain and Leis Parameter Methods.



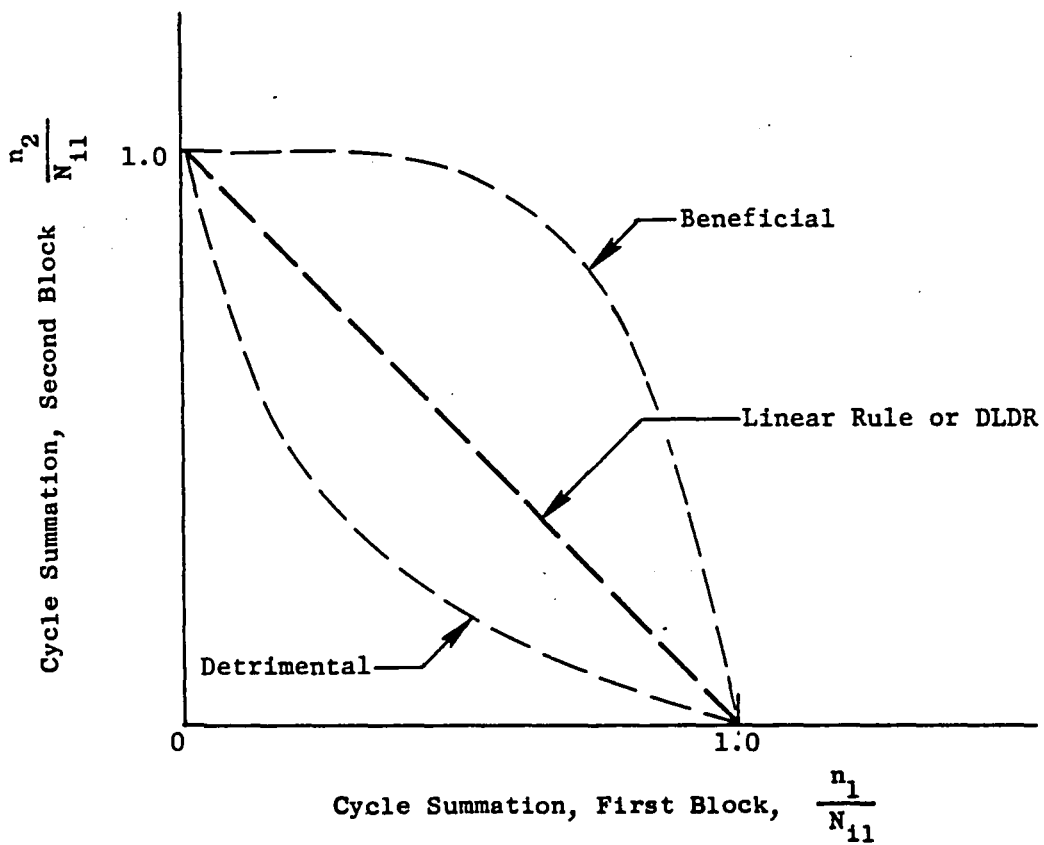
Baseline Data at 566° C Analyzed by the Equivalent Strain and Leis Parameter Methods.



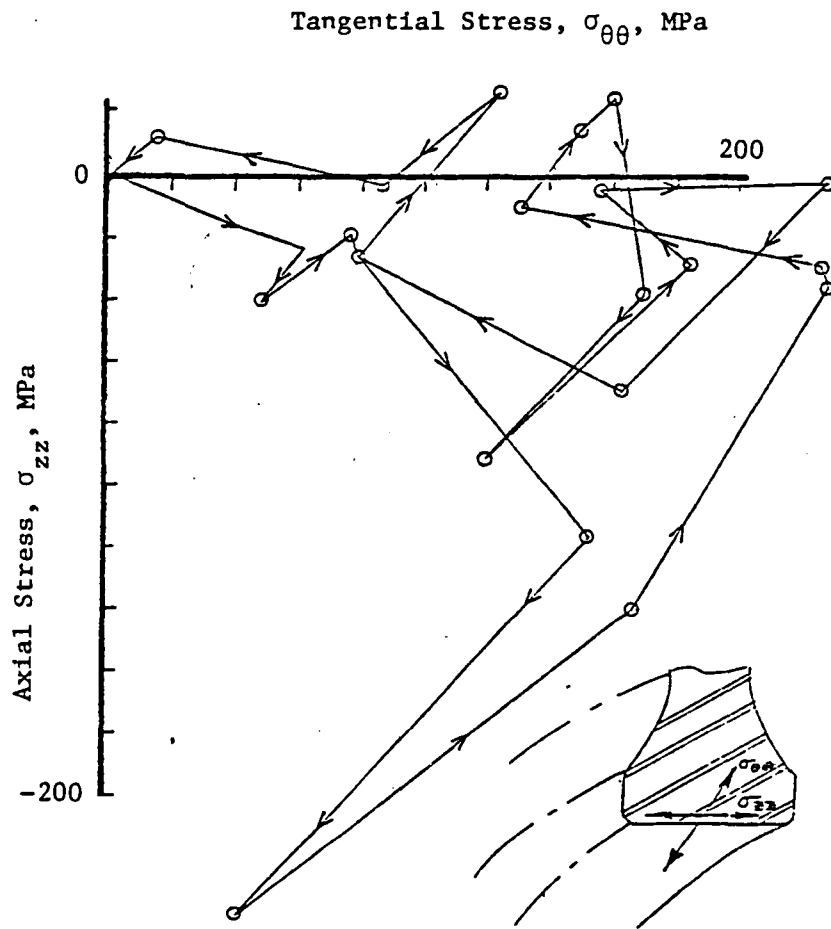
Results of Two-Step Load/Temperature Tests: R_ϵ Variable in Second Step.



Generalized Equal-Life Test.



Potential Results of Equal-Life Tests.



Example of the Complex States of Stress That Can Evolve During a Simulated Mission in a Disk Bore.

Suggested Multiaxial Formulation

$$\frac{\Delta \epsilon_e}{2} = \frac{\epsilon_f}{f_2(TF)} (2 N_f)^{-\alpha} + \left(\frac{\sigma_f - k^* I_{\sigma m}}{E^*} \right) (2 N_f)^{-\beta}$$

Where:

$$\Delta \epsilon_e = \sqrt{\frac{2}{3} \Delta e_{ij} \Delta e_{ij}}$$

$$e_{ij} = \epsilon_{ij} - \frac{1}{3} \epsilon_{kk} \delta_{ij}$$

$$TF = \frac{\sigma_1 + \sigma_2 + \sigma_3}{\frac{1}{\sqrt{2}} \sqrt{(\sigma_1 - \sigma_2)^2 + (\sigma_2 - \sigma_3)^2 + (\sigma_1 - \sigma_3)^2}}$$

$$I_{\sigma m} = \sigma_{1m} + \sigma_{2m} + \sigma_{3m}$$

$$E^* = \frac{3E}{2(1+\mu)}$$

ϵ_f , α , σ_f , k^* , β are material parameters

N_f = Cycles to Failure

Benchmark Notch Test For Life Prediction

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ABSTRACT

Aircraft gas turbine engine components are subjected to severe stress, temperature, and environmental conditions. In these components, local stress raisers (e.g., notches, boltholes, welds, fillet radii) are very often life limiting areas in that low cycle fatigue failures generally initiate in these critical regions. Economic and reliability demands have prompted inordinate effort in development of analytic methods to first predict stresses and strains in these complex geometry regions and, ultimately, predict the low cycle fatigue life for components containing these necessary design features. These analytical developments have apparently been successful since numerous techniques (most notably in the form of finite-element computer models) have evolved. There remains, however, the need to check or verify these analytical methodologies against actual experimental data measurements. This is not a simple task. Most stress concentration regions in gas turbine engines are geometrically very small, eliminating many conventional extensometry methods for strain measurement. Further, conditions of interest include long times at elevated temperatures (near 649° C), eliminating conventional strain gage measurement methods. The laser Interferometric Strain Displacement Gage was recognized as having the potential to accomplish this demanding task and was employed in this program.

The overall objective of this program was the generation, measurement, and documentation of the actual strains incurred at the root of a discontinuity in cyclically loaded test samples subjected to inelastic deformation at high temperature where creep deformations readily occur. A secondary objective was to perform an analysis of the steady-state cyclic stress-strain response at the root of the discontinuity in the tested samples for comparison to the measured results.

A comprehensive set of local notch root strain measurements for a variety of load patterns in an Inconel 718 notch specimen at 649° C (1200° F) was obtained and documented using the laser Interferometric Strain Displacement

Gage (ISDG). The ISDG was successfully adapted to the high-temperature measurements in this typical Ni-base superalloy and was shown to have a relative uncertainty of $\pm 3\%$ of the measured strain with an additional uncertainty of ± 150 microstrain.

Measurements were made for six load patterns including continuous cyclic, creep and cyclic with tensile and compressive hold periods on flat, double-notch bars with an elastic stress concentration factor of 1.9.

Pedigree tensile and cyclic stress-strain data were also generated at 649° C and employed in a simple Neuber analysis to obtain analytic predictions for comparisons to test results. A modified Neuber approach and a limited finite-element study were also compared to the data.

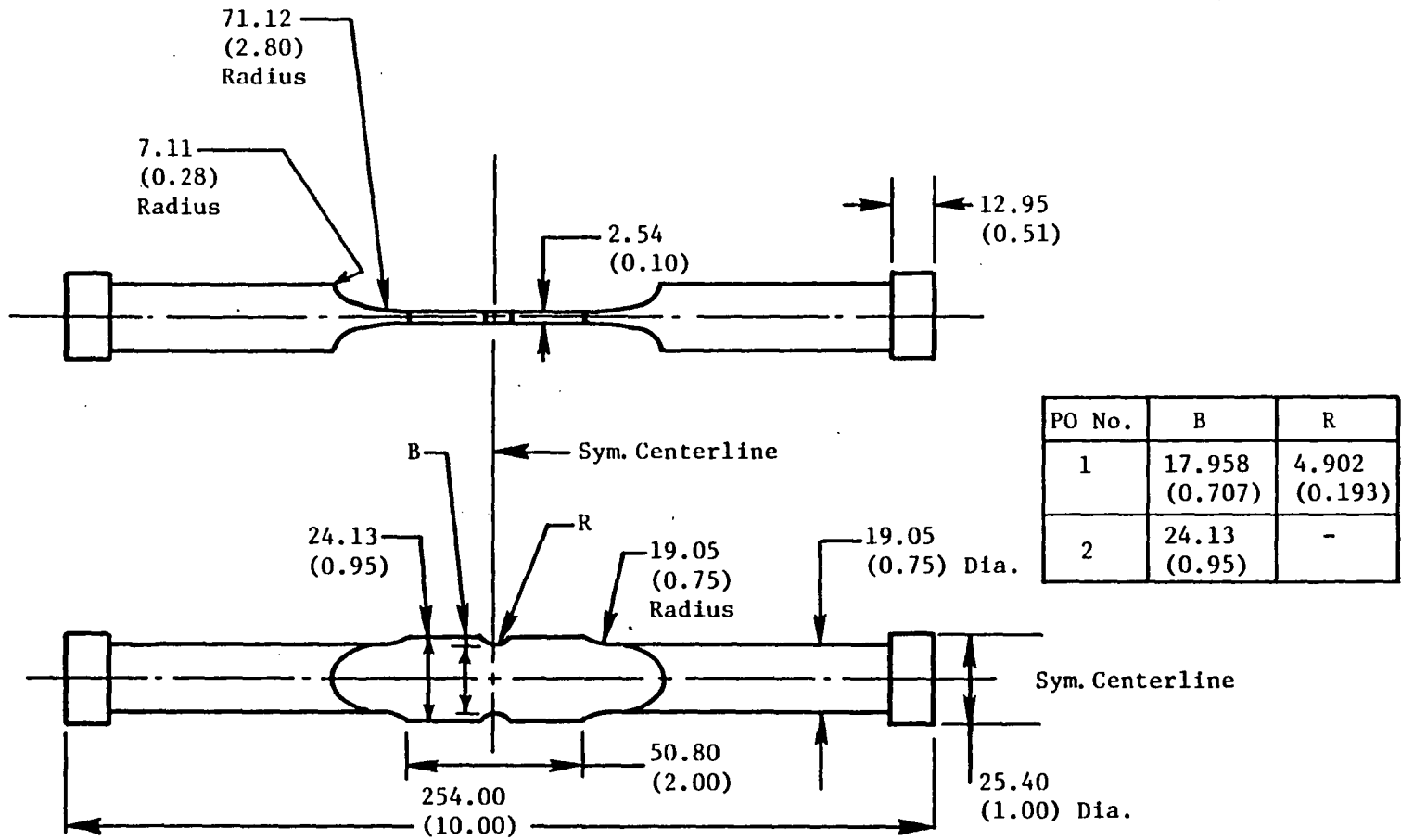
A smooth bar specimen subjected to the notch root strain history recorded from a continuously cycled notch bar was also used to obtain stress behavior data for comparison to the predictions.

The Neuber analysis predicted the first cycle notch root behavior very well on the basis of hysteresis loop comparison when the notch root strain rate stress-strain response curve was used. The stabilized cyclic loops were not well predicted even when cyclic softening had stabilized. The modified Neuber equation (corrected for stress redistribution due to plasticity) and the finite-element analysis improved the cyclic correlation but did not totally resolve the problem.

The smooth bar test using the notch root strain history control shed additional light on this in that the maximum monotonic and stable cyclic stresses were well predicted by Neuber (as was the minimum stable cyclic stress) while the minimum monotonic (first cycle unload) was not. This suggests that kinematic-hardening assumptions may be incorrect for the early cyclic transition period.

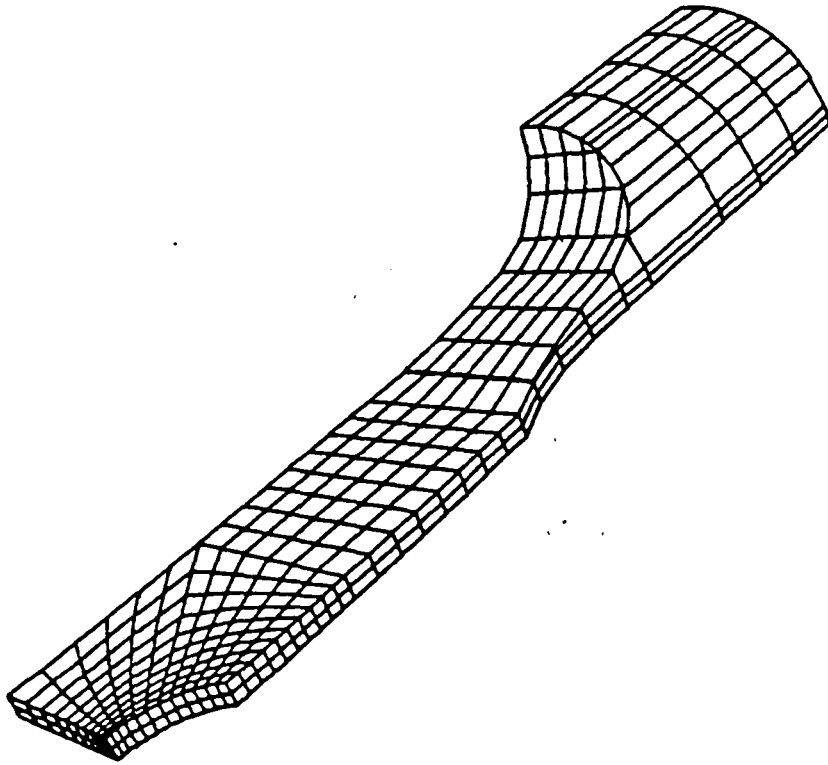
The utility of the computerized data acquisition and storage system associated with the ISDG was demonstrated by examining cyclic history dependent parameters (e.g., loop area) on a cycle-by-cycle basis. These parameters were used to assess data quality as well as behavioral trends, and have potential for extension to life prediction application. The hysteresis loop elastic loading slope was used to predict notch root crack initiation which was confirmed by posttest microscopy.

The program objectives of generation of benchmark notch data in a turbine disk alloy at elevated temperature and comparison to a Neuber analysis were met. Significant implications for future model development were determined.

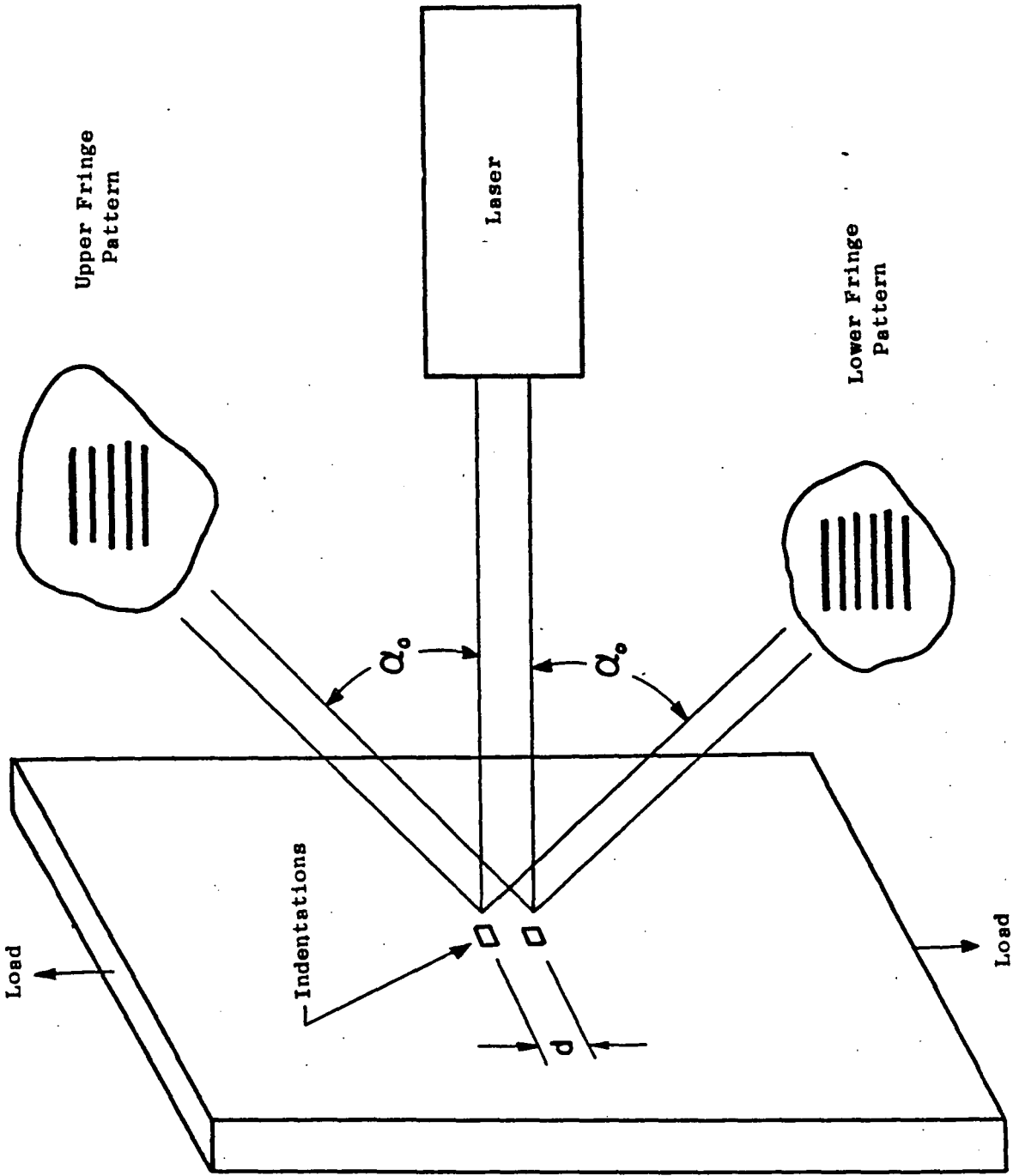


• Dimensions in mm (inches)

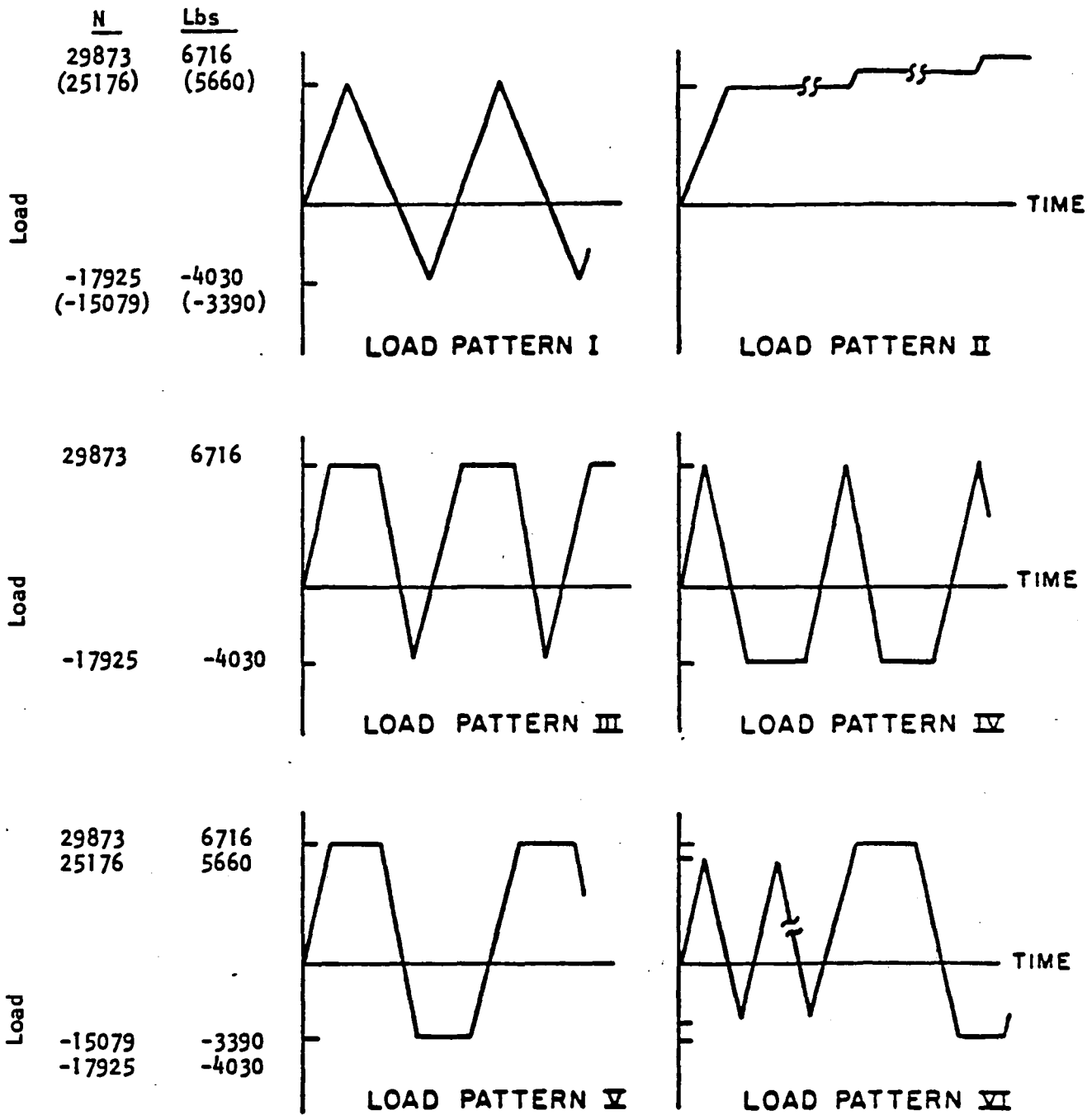
Benchmark Notch Specimen CAP111280 ($K_t = 1.9$).



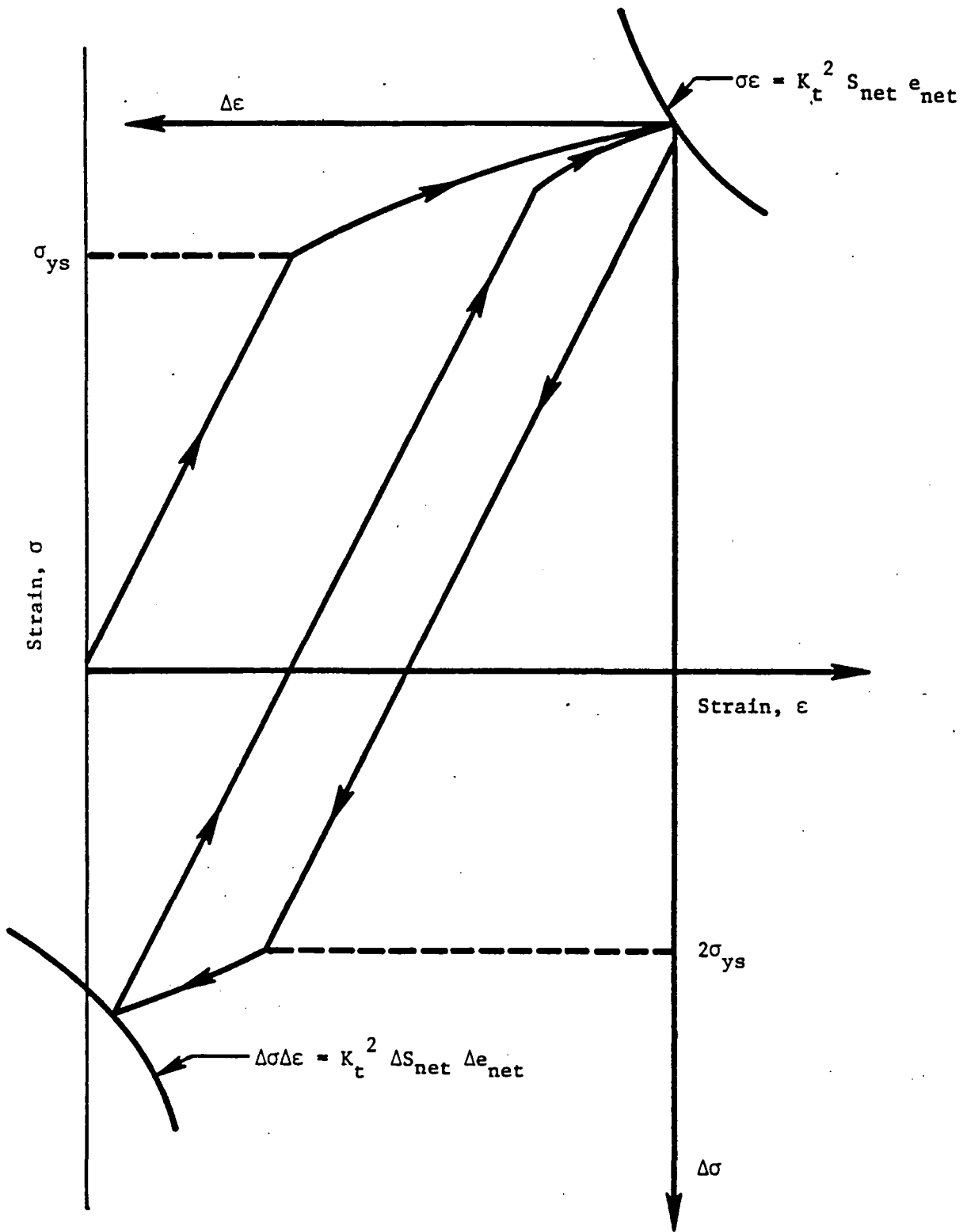
Three-Dimensional Finite-Element Mesh Pattern
of 1/8 of the Benchmark Notch Fatigue Specimen.



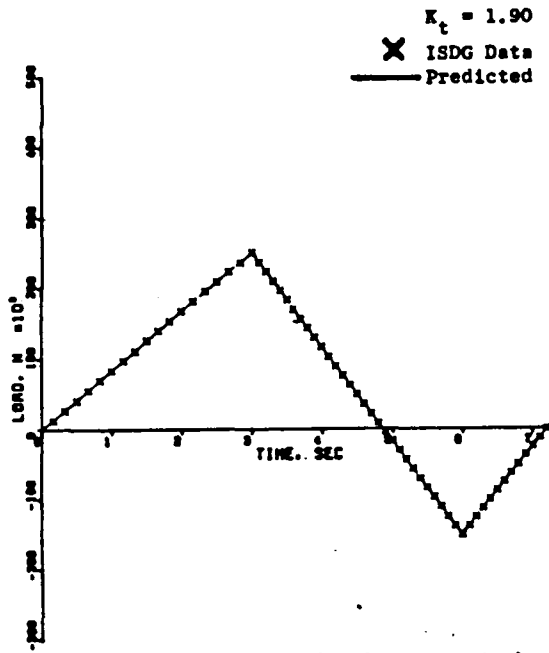
Schematic of the ISDG.



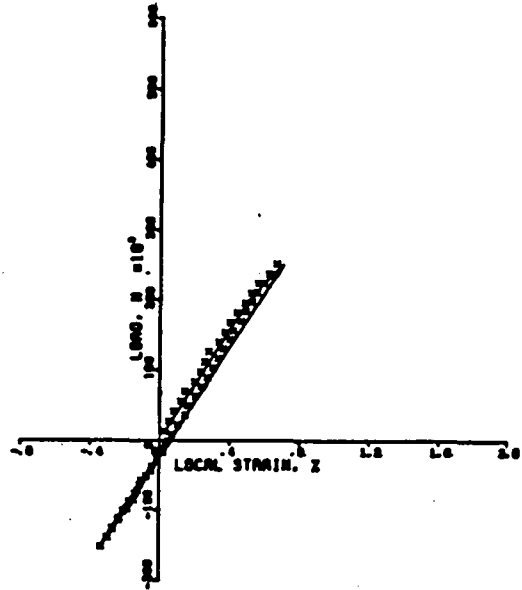
Load Spectra.



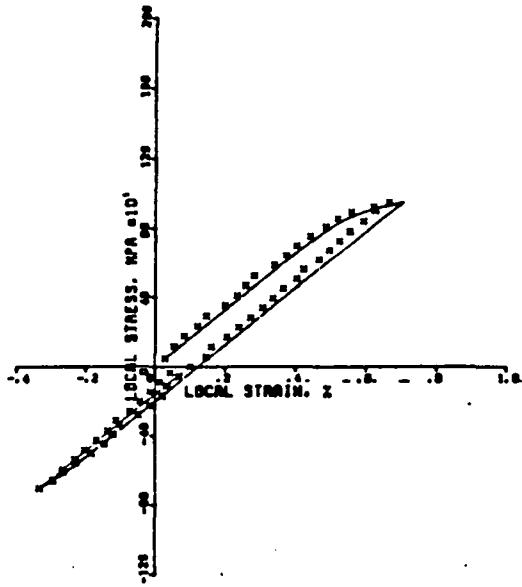
Schematic Illustration of the Application of Neuber's Rule to Cyclic Loading.



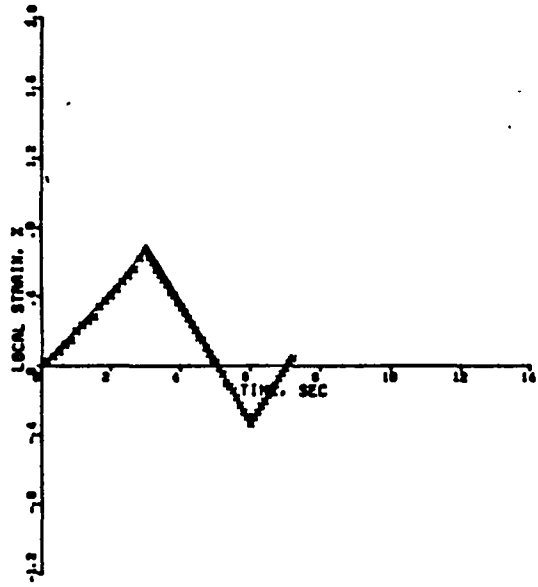
(a) Load Pattern I (Load Versus Time)



(b) Load Versus Local Strain



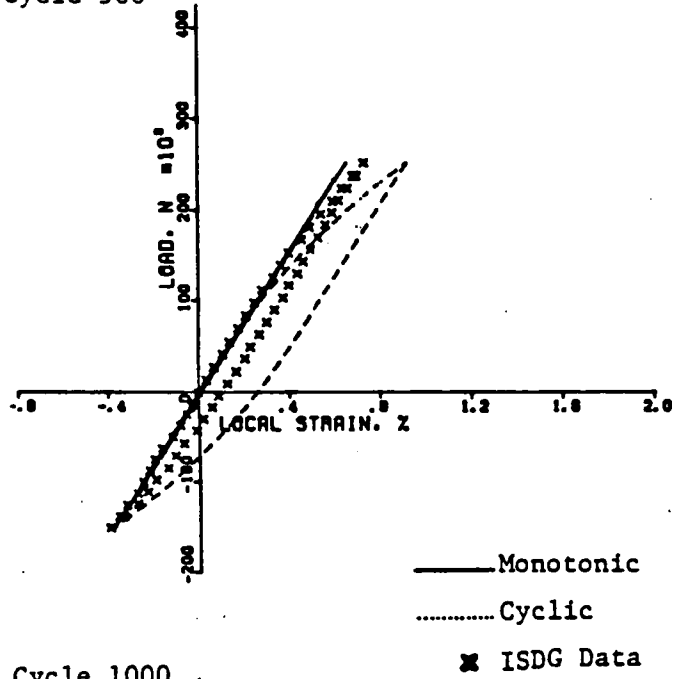
(c) Local Stress Versus Local Strain



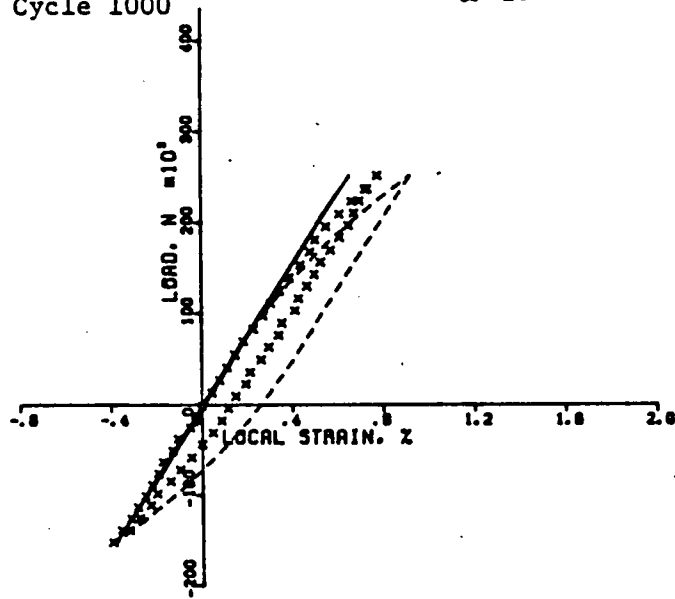
(d) Local Strain Versus Time

Comparison of Neuber Predicted and ISDG Measured First Cycle Stress-Strain Behavior for Inconel 718 Notched Bar at 649° C for Load Pattern I (Continuous Cycle), Test 6.

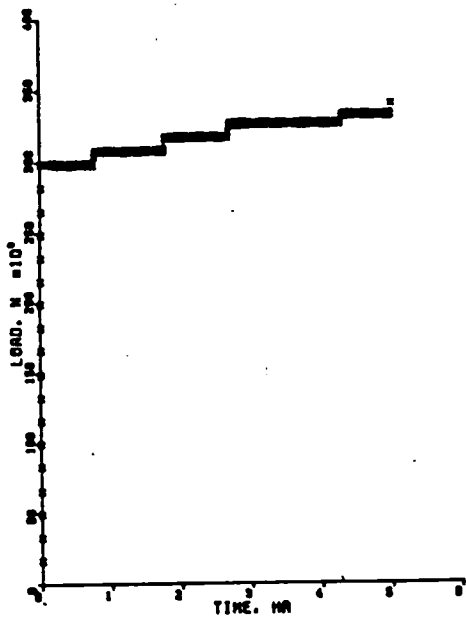
Cycle 500



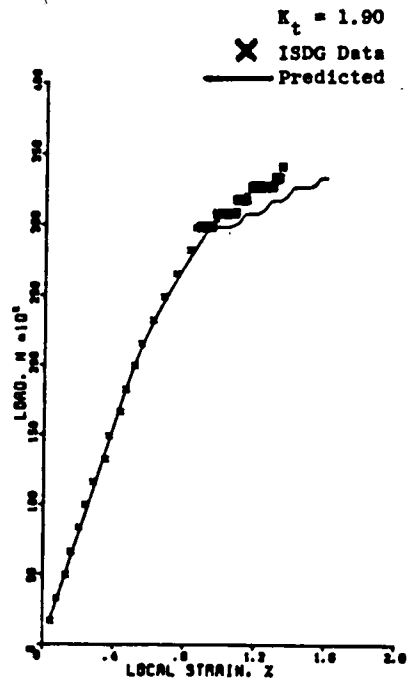
Cycle 1000



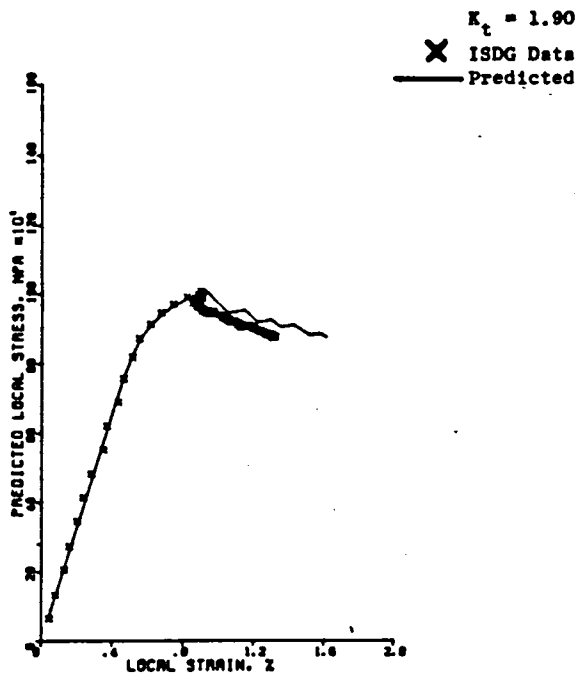
Comparison of Neuber Predictions and ISDG Measured Intermediate Cycle Notch Root Strain Behavior of Test 6 (Continuous Cycle).



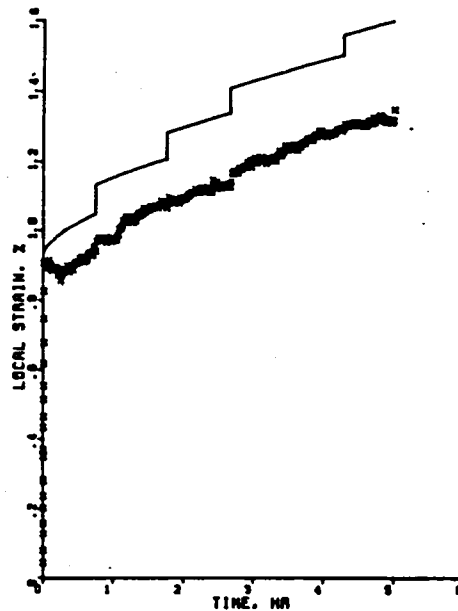
(a) Load Pattern II (Load Versus Time)



(b) Load Versus Local Strain

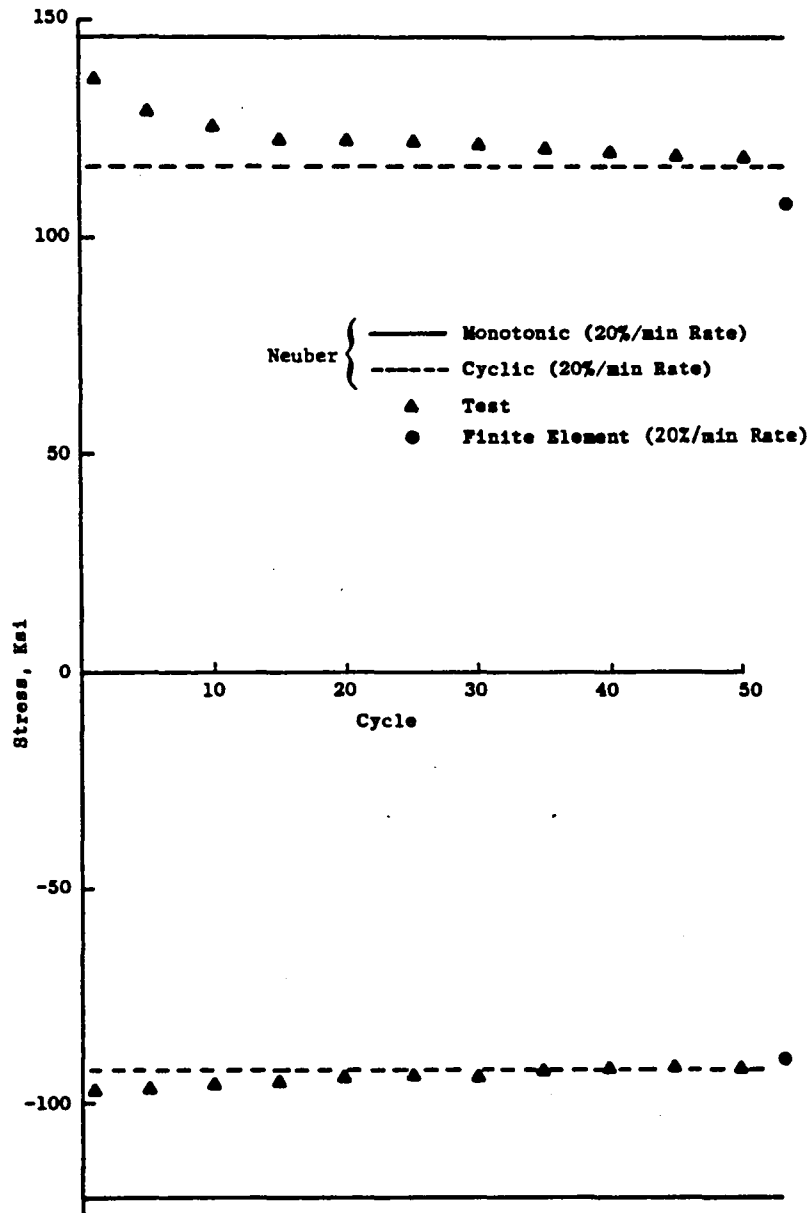


(c) Local Stress Versus Local Strain

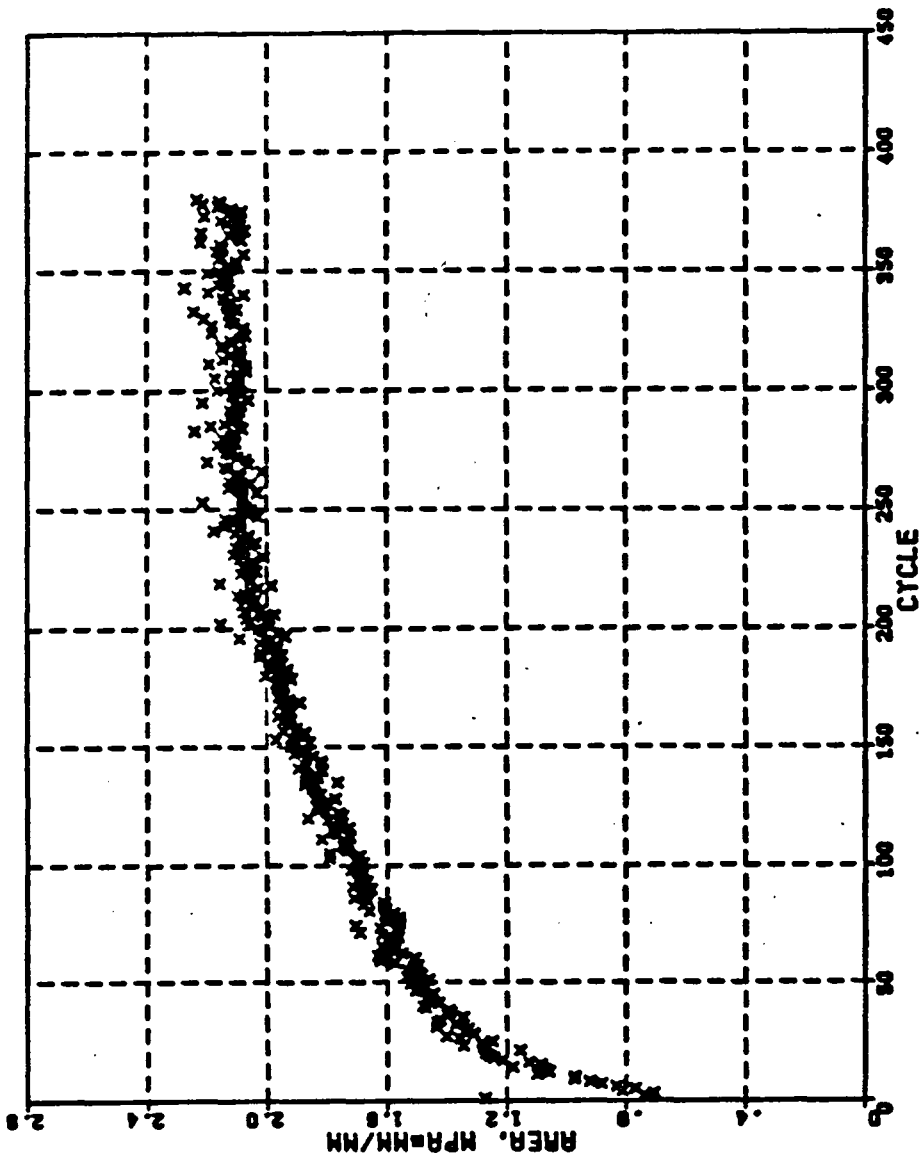


(d) Local Strain Versus Time

Comparison of Neuber Predicted and ISDG Measured First Cycle Stress-Strain Behavior for Inconel 718 Notched Bar at 649° C for Load Pattern II (Creep Test), Test 12.



Comparison of Analytical Predictions of Notch Root Stress and the Result of a Strain-Controlled Smooth Bar Test Using the Strain Pattern Recorded in Test No. 6.



Variation of Hysteresis Loop Area in the Cycle History of Test 10.

Conclusions

- **ISDG System Accuracy of $\pm 3\%$ or ± 150 Microstrain.**
- **When Local Notch Strain Rate Influences were Included, Neuber Provided Excellent Correlation for the Monotonic (Load Up) Case.**
- **Stabilized Cyclic Behavior was not Well Predicted by Neuber. Finite Element Results Improved Correlation but also in Error.**
- **Maximum Monotonic and Cyclic Stress and Minimum Cyclic Stress Levels were Well Predicted. Initial Off Loading (Monotonic) Minimum Stress Was Not.**
- **Computerized Cycle by Cycle Hysteresis Loop Parameter Assessment Useful Tool. The Elastic Slope (Modulus) was Indicator of Notch Root Cracking.**

NONLINEAR STRUCTURAL AND LIFE ANALYSES
OF A TURBINE BLADE

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Abstract

The most critical structural requirements that aircraft gas turbine engines must meet result from the diversity of extreme environmental conditions in the turbine section components. Accurate life assessment of the components under these conditions requires sound analytical tools and techniques, an understanding of the component operating environment, and comprehensive data on component materials. Inadequate understanding of any or all of these areas may result in either a conservative life prediction or component failure.

Much activity has occurred in recent years both through Industry and Government programs to provide the designer with the tools for more accurate design analysis and component life prediction. These efforts encompass advances in analytical stress and life prediction techniques, instrumentation capabilities, and cost-effective, accelerated verification testing.

Advanced structural analysis techniques are available to permit more reliable life prediction in the life-limiting turbine components. However, verification of these methods through application to well-documented failure case histories is lacking.

Although nonlinear stress analysis computer programs are available, they have not been used routinely in hot section component design because of the extensive computation time required for such applications. Furthermore, poorly defined material constitutive equations have hampered more general use of such computer programs.

In addition, several high-temperature, low-cycle fatigue, life prediction approaches have been proposed in recent years. These approaches have not yet been applied extensively to hot section components primarily because critical evaluation through application to well-documented failure case histories is needed.

The objective of this program was to evaluate the utility of advanced structural-analysis techniques and advanced life-prediction techniques in the life assessment of hot-section components. A particular goal was to assess the extent to which a three-dimensional cyclic isoparametric finite-element analysis of a hot-section component would improve the accuracy of component life predictions. At the same time, new high-temperature life-prediction theories such as Strainrange Partitioning and the Frequency Modified approaches were to be applied and their efficiency judged.

A commercial air-cooled turbine blade with a well-documented history of cracking in the squealer tip region was selected as the vehicle for accomplishing the above objective. To perform the stress analysis for this turbine blade, a detailed three-dimensional model of the blade tip region was constructed which consisted of eight-noded isoparametric finite-elements (580 elements and 1119 nodes).

To perform the cyclic nonlinear analysis, a commercially available program, ANSYS, was chosen. For this analysis, previously determined temperature-dependent cyclic stress-strain curves and creep data were used. The kinematic hardening option was selected for the plasticity analysis, and the creep analysis was performed with the time-hardening rule. Seven complete cycles were run, at which time shakedown was determined to have occurred. The computed strain-temperature history at the critical location was used to program a thermomechanical test of an axially loaded specimen.

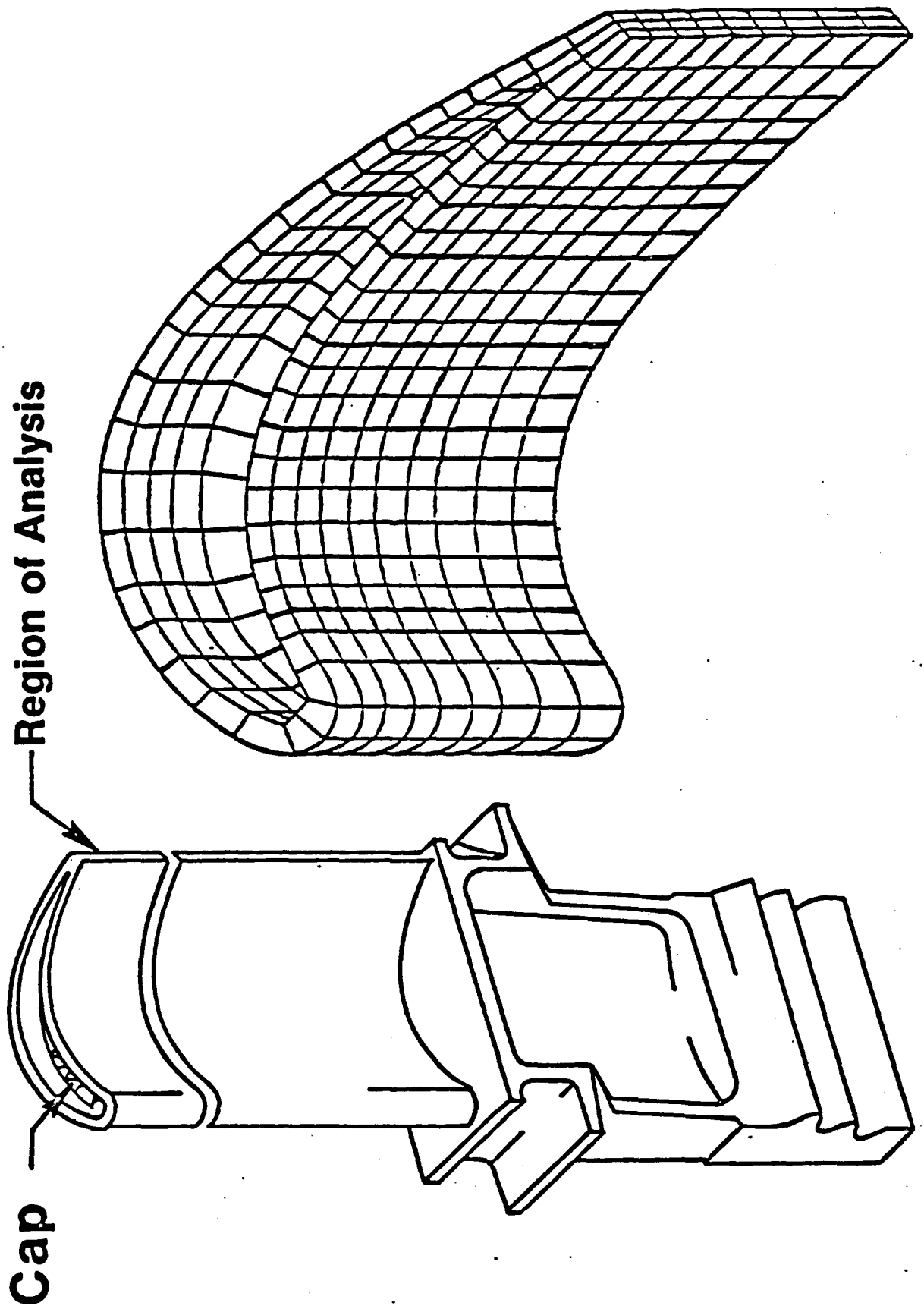
The total strain range at the critical cracking site was calculated using both elastic and inelastic ANSYS analyses. The total strain range values were within less than three percent of each other, thus indicating the potential value of the simpler, much less expensive, elastic analysis. The three dimensional structural analyses produced results in qualitative agreement with the limited experimental evidence. The maximum strain ranges were predicted for the blade tip region where actual cracking occurred.

Tests of a uniaxial strain controlled specimen following the same strain-temperature history as computed at the blade tip crack initiation location showed that the stress-strain response stabilized by the fourth cycle. Analytical simulation of this experiment demonstrated later stabilization of the stress-strain response, higher peak stresses and a smaller amount of stress relaxation than the test results indicated. These discrepancies between analysis and experiment suggest that the creep model and/or data did not accurately represent the material cyclic time dependent behavior.

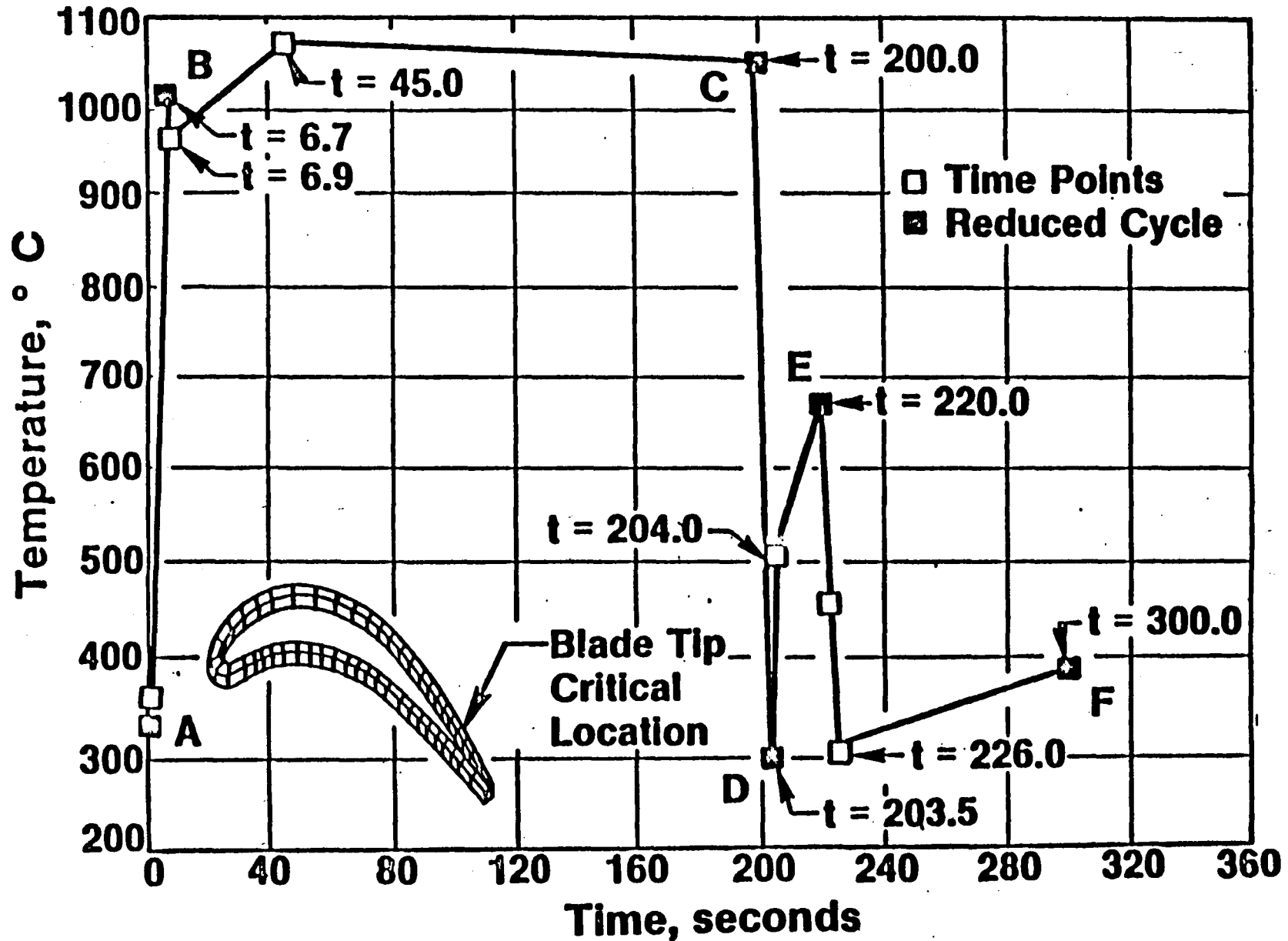
The results of these analyses and the thermomechanical tests were used to make life predictions by several crack initiation life methods, including the Strainrange Partitioning and Frequency Modified Methods.

The evaluation of the life prediction methods indicated that none of those studied were satisfactory. A wide scatter of lives could be predicted with all of the methods because of oversensitivity to input information which was either difficult to calculate accurately or depended on engineering judgements.

Stage 1 High Pressure Turbine Blade and Finite Element Model



Blade Metal Temperature Versus Time at Critical Location

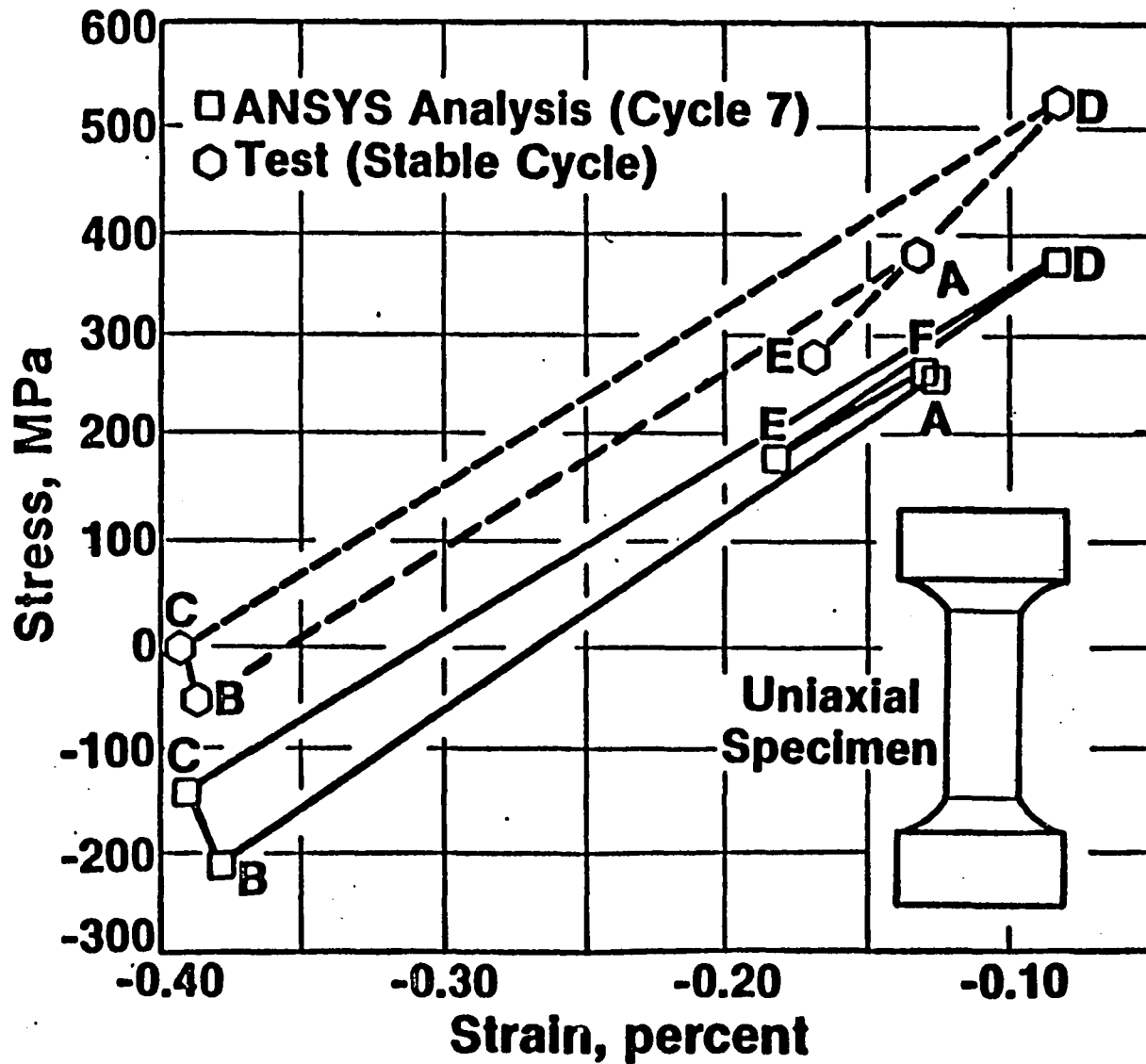


Results of Turbine Blade Tip Structural Analyses

(All Results are for Principal Direction Normal to Radial Crack at Critical Location)

	Elastic	Inelastic		
		Cycle 1	Cycle 2	Cycle 7
Max. Total Strain, %	0.025	-0.05	-0.062	-0.0829
Min. Total Strain, %	-0.2925	-0.3582	-0.371	-0.3918
Total Strain Range, %	0.3175	0.3082	0.3090	0.3089
Mean Stress, MPa	-164.9	-12.9	33.7	80.5

Comparison of Uniaxial Thermomechanical Test and Inelastic Analysis Results



NONLINEAR STRUCTURAL AND LIFE ANALYSES
OF A COMBUSTOR LINER

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Abstract

In this study, three dimensional nonlinear structural analyses were performed for a simulated aircraft combustor liner specimen in order to assess the capability of nonlinear analyses using classical inelastic material models to represent the thermoplastic-creep response of the component. In addition, the computed stress-strain history at the critical location was input into state-of-the-art life prediction methods in order to evaluate the ability of these procedures to predict crack initiation life.

The overall operating cost of the modern gas turbine engine is significantly affected by the durability and efficiency of the major hot section components. These are the combustor and turbine structures in the engine. During each flight cycle, these components undergo large thermally induced stress and strain cycles which include significant amounts of creep and relaxation. Primary responsibilities of the combustor, in the engine cycle, are gas temperature level and pattern control, required for efficient turbine operation, and exhaust emission control at the various flight operating conditions. These goals are accomplished by the precise metering of air throughout the combustor structure. The high pressure and high combustion gas temperature characteristic of this environment require that the combustor liner be cooled for durability. These requirements for control of exit gas temperature, emissions, and metal temperature generate an intense competition for utilization of combustor airflow. The more aggressive performance, efficiency, and emission goals set for current and future engines emphasize the need for development of durable combustor structures which can operate with reduced levels of cooling air. This requires detailed knowledge of the operating environment and the ability to accurately predict structural response for these loadings.

Over the past decade, nonlinear finite-element programs have become available for the structural analysis of multiaxial components subject to cyclic thermo-mechanical loading. These programs involve sophisticated computational algorithms and advanced finite element formulations, yet rely on material models whose applicability to the hot section component environment is questionable. Of primary concern is the response of materials to cyclic loading involving simultaneous creep and plastic behavior. A major need is the development of appropriate hot section component structural response data, sufficient to evaluate the advanced structural analysis capabilities with emphasis on the effectiveness of the material models.

Creep-fatigue life prediction methods used for aircraft engine hot section components are currently calibrated to simplified models for predicting the local cyclic hysteresis response to thermal-mechanical loading. Further, these models generally lack calibration to well-controlled hot section component fatigue test data. Thus, a second major need is for the evaluation of life prediction models for creep-fatigue response of hot section components using the results of nonlinear stress-strain analysis and well-controlled component response data.

This program addresses a critical issue in the development of advanced life prediction technologies-the need to establish the limitations of current nonlinear structural modeling and creep/fatigue life prediction schemes for a major hot section component. In order to make a critical evaluation of these tools, a well controlled component simulation test served as the calibration data source for the program. The component test used a prototypical combustor liner specimen constructed in an identical configuration with current combustor liners in engine service.

A three dimensional non-linear finite element analysis of the liner was conducted with the MARC computer code. The analysis used existing time independent classical plasticity theory with a Von Mises yield surface and the combined (isotropic-kinematic) hardening rule. A constant rate creep model was used to account for instantaneous time dependent plasticity effects. Both the plasticity and creep models were calibrated to isothermal Hastelloy X material response data. The computed strain-temperature history at the critical location of the combustor liner was imposed on a uniaxial specimen in a strain-temperature controlled test. The uniaxial test results were compared to the analytical stress-strain results. The computed strain-temperature history was also input into two life prediction methods (Strain Range Partitioning and the PWA Combustor Life Prediction Method) in order to compare predicted combustor crack initiation life against experimental observations.

The nonlinear structural analysis indicated that the time dependent plasticity model and the creep model did not accurately predict the cyclic thermomechanical response at the louver failure location. Tests of a uniaxial strain controlled specimen run with the same mechanical strain-temperature history as computed at the failure location showed that the stress-strain response stabilized within the first few cycles. Analytical simulation of the experiment with the Hastelloy X creep-plasticity models exhibited continued cyclic hardening (increasing peak tensile stress and reduced inelastic strain range) after many cycles. Potential modification to the plasticity model, including a multi-yield-surface concept, non-linear hardening or use of one of the rate dependent (unified) theories currently under development, may be required to improve the prediction for the varying temperature loading condition. Determination of correct thermomechanical response is critical for the life prediction of the component.

The two high temperature, creep-fatigue life prediction methods considered were the Strain Range Partitioning Pratt and Whitney Aircraft-Commercial Products Division Combustor Life Prediction, and Continuous Damage methods. Both assume that time independent plastic and time dependent creep damage mechanisms are present at elevated temperature. Isothermal fatigue and creep rupture tests are used to define the material life relationships.

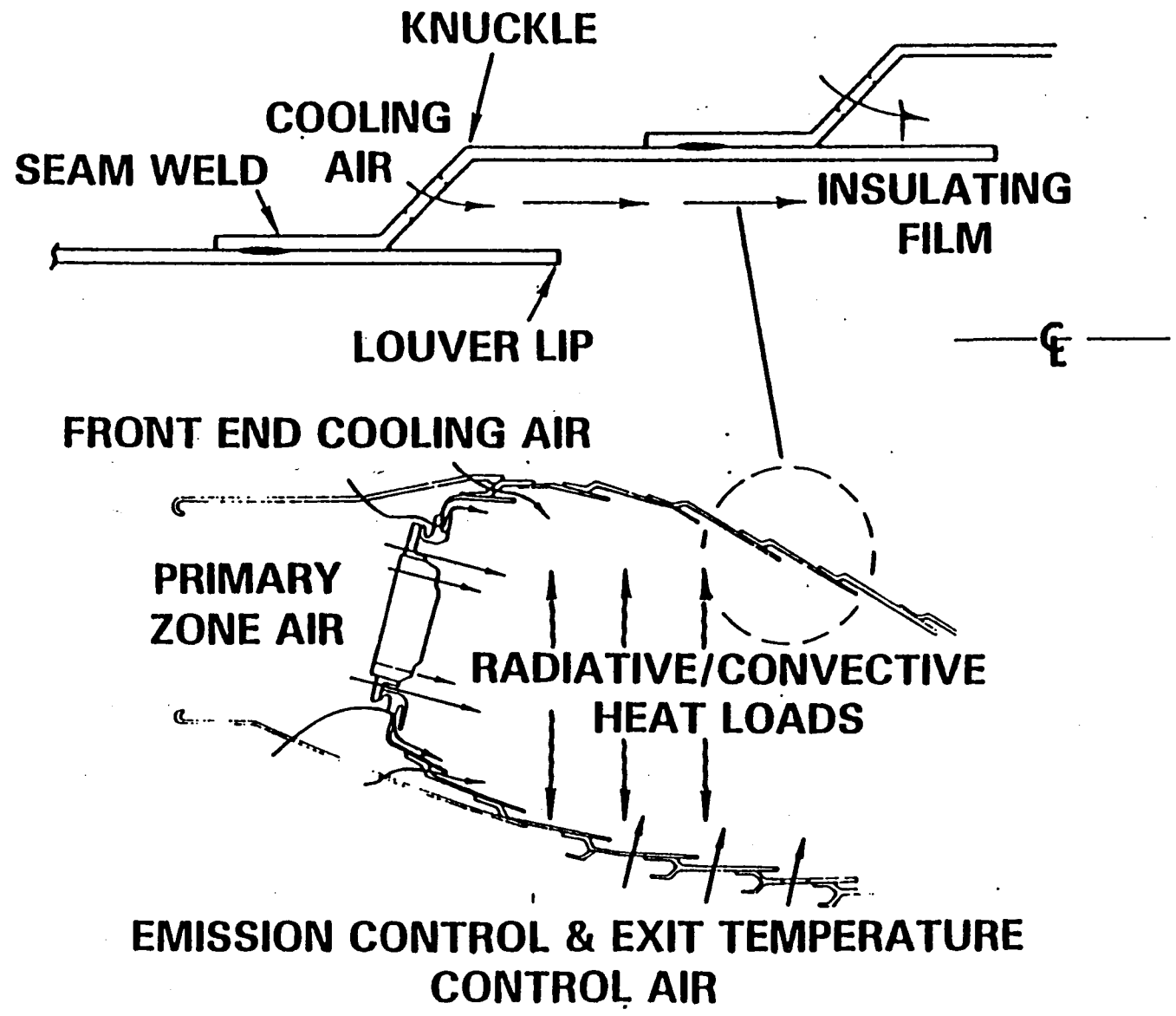
The Strain Range Partitioning and PWA-CPD methods are based on the existence of generic types of fully reversed damage cycles composed of combinations of the plastic and creep mechanisms. For this analysis, the combustor louver lip response contained only the pp(tensile plasticity reversed by compressive plasticity) and pc(tensile plasticity reversed by compressive creep) damage cycles.

The Strain Range Partitioning method overpredicted the louver cracking life (8500 cycles vs. 1000 cycles). Part of this discrepancy may be associated with uncertainty in the definition of the generic pp and pc fatigue life curves. Better definition of the curves may reduce the predicted life (and improve the correlation). However, it appears that the SRP method will overpredict the cracking life by at least a factor of 2.

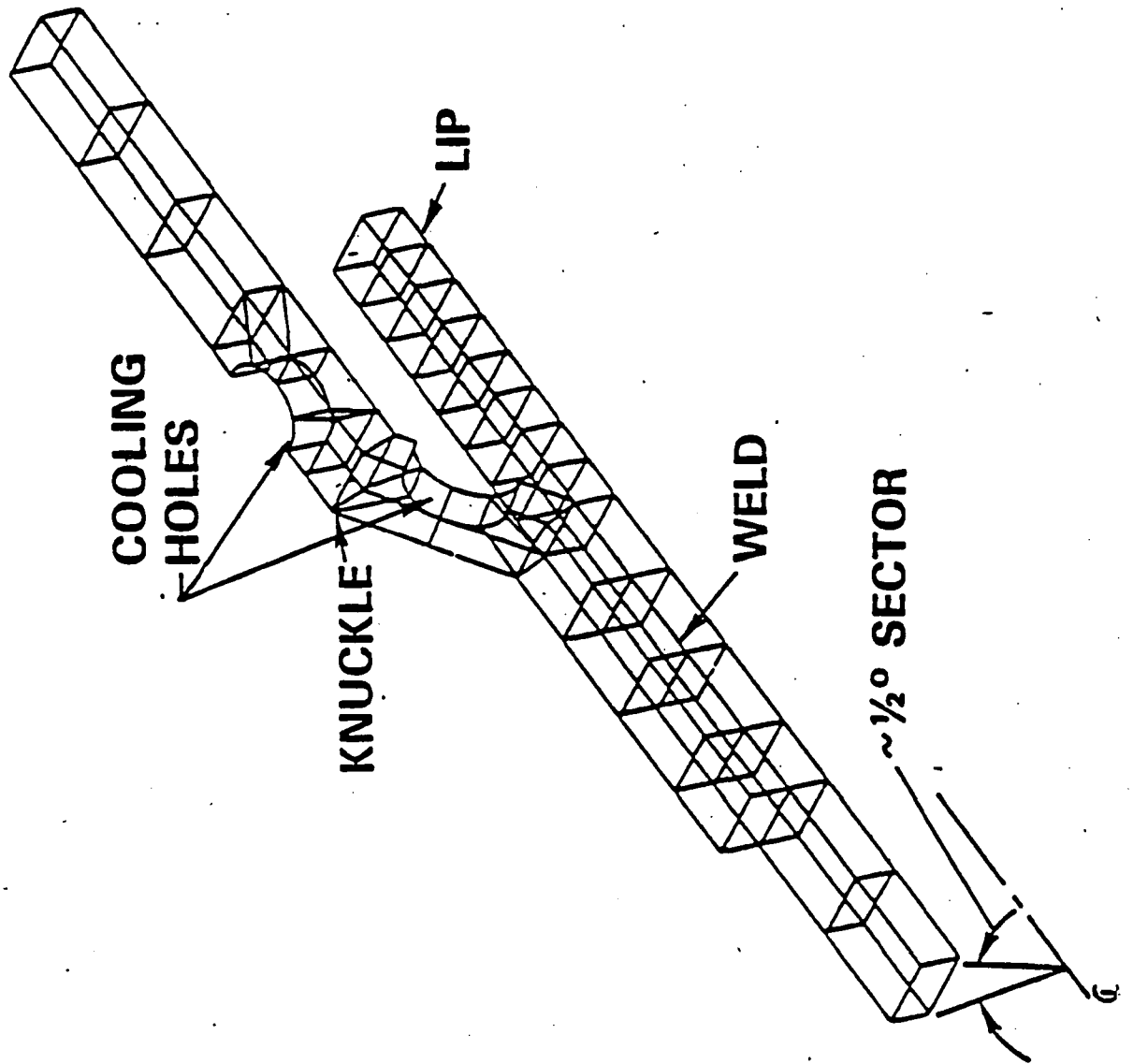
The Pratt and Whitney Aircraft-Commercial Products Division method also overpredicted the louver cracking life (1700 cycles vs. 1000 cycles). This improved correlation (relative to SRP) is due, in part, to the fact that the inelastic strain range predicted by this method is larger than the observed inelastic strain in the louver. Equating the inelastic strain value results in a predicted life of 8000 cycles, which is similar to the SRP calculation. In actual design practice, this method is used with experimental and field service data to assess the overall service life of the component.

The overpredictions in the combustor liner life based on the analyses in conjunction with isothermal, strain controlled fatigue test data suggest that a thermomechanical fatigue cycle may produce damage at a faster rate than a comparable isothermal cycle.

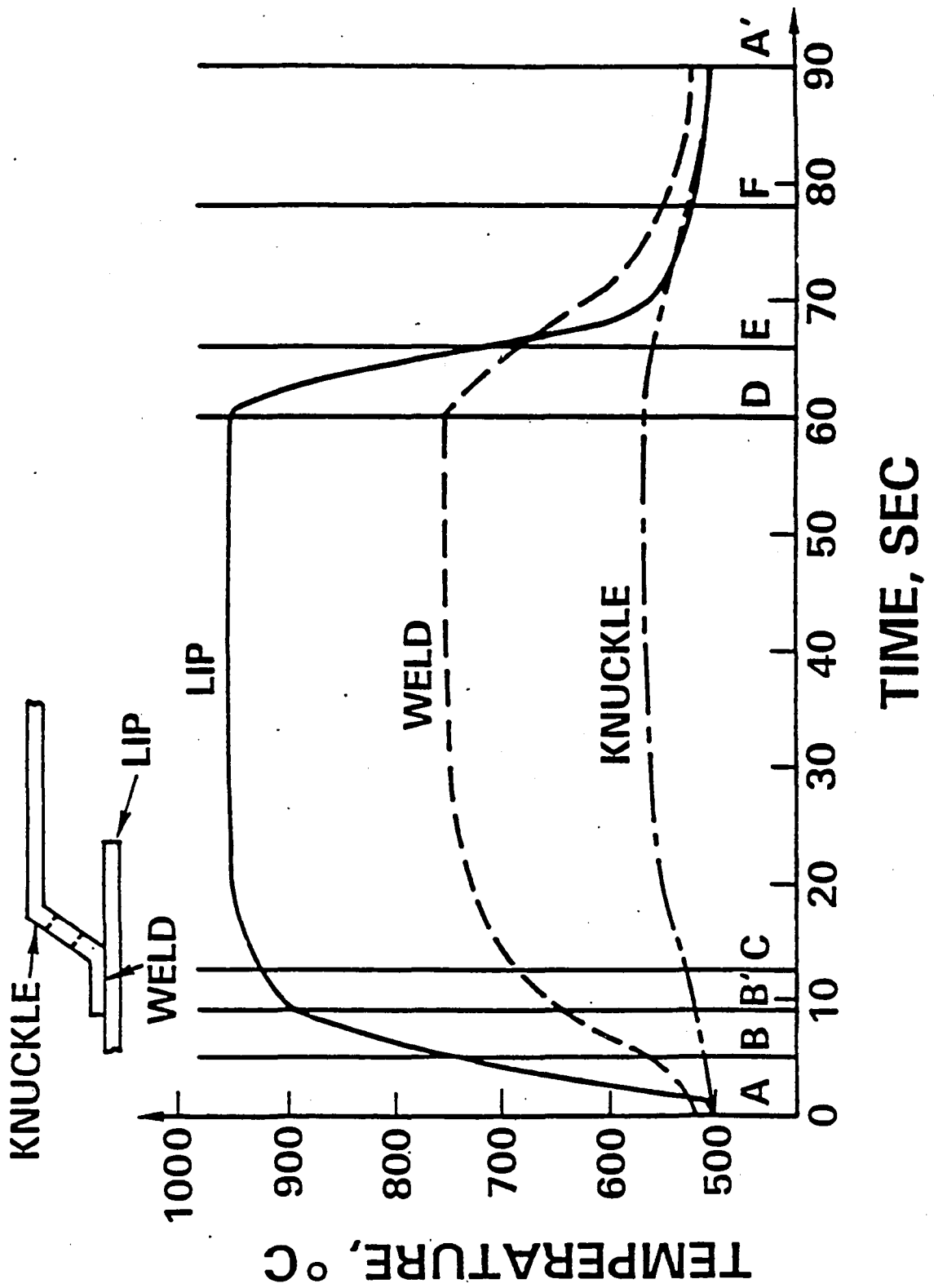
TYPICAL LOUVER COMBUSTOR LINER CONSTRUCTION AND AIRFLOW DISTRIBUTION



FINITE ELEMENT MODEL

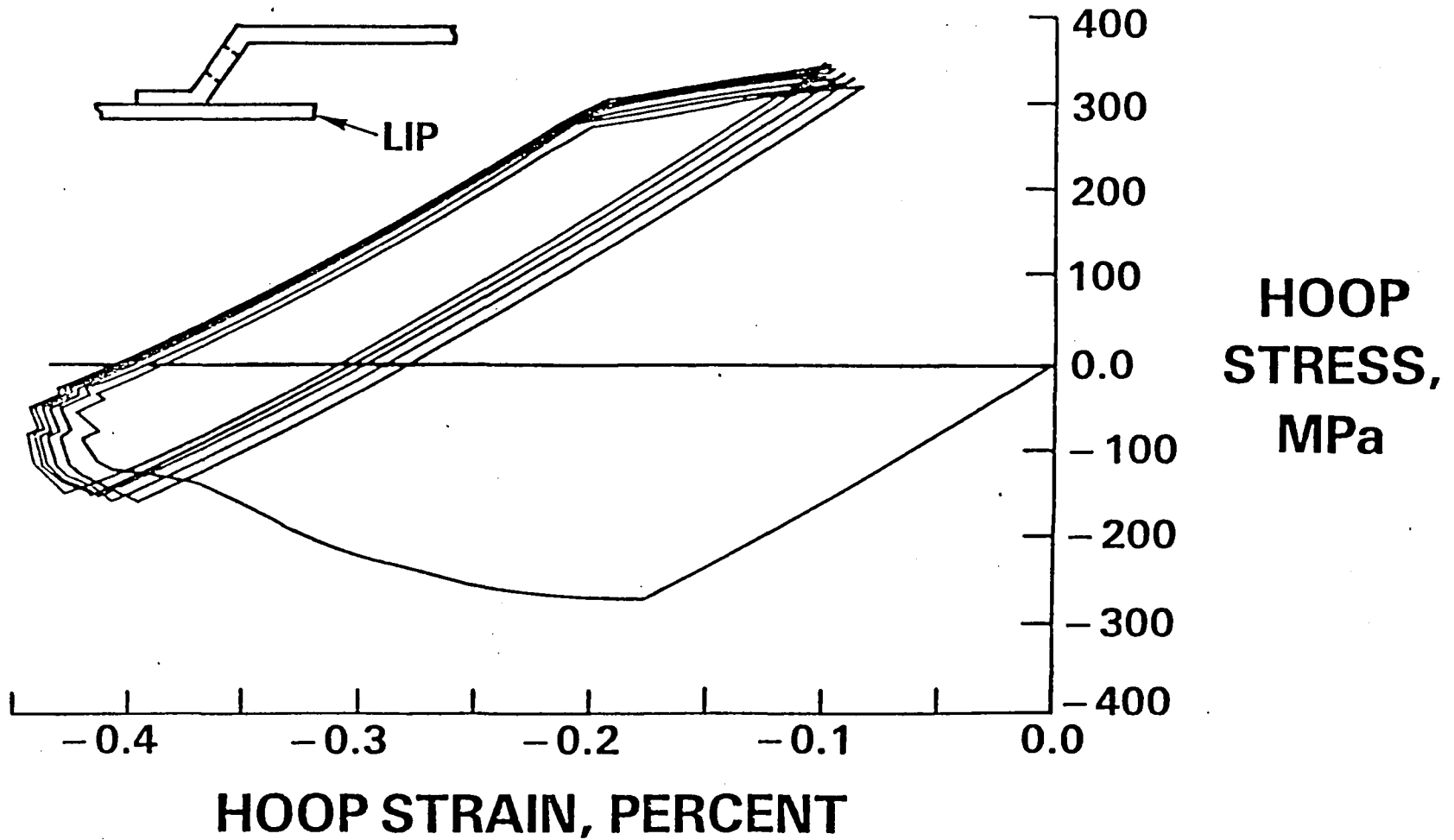


LOUVER TEMPERATURE RESPONSE

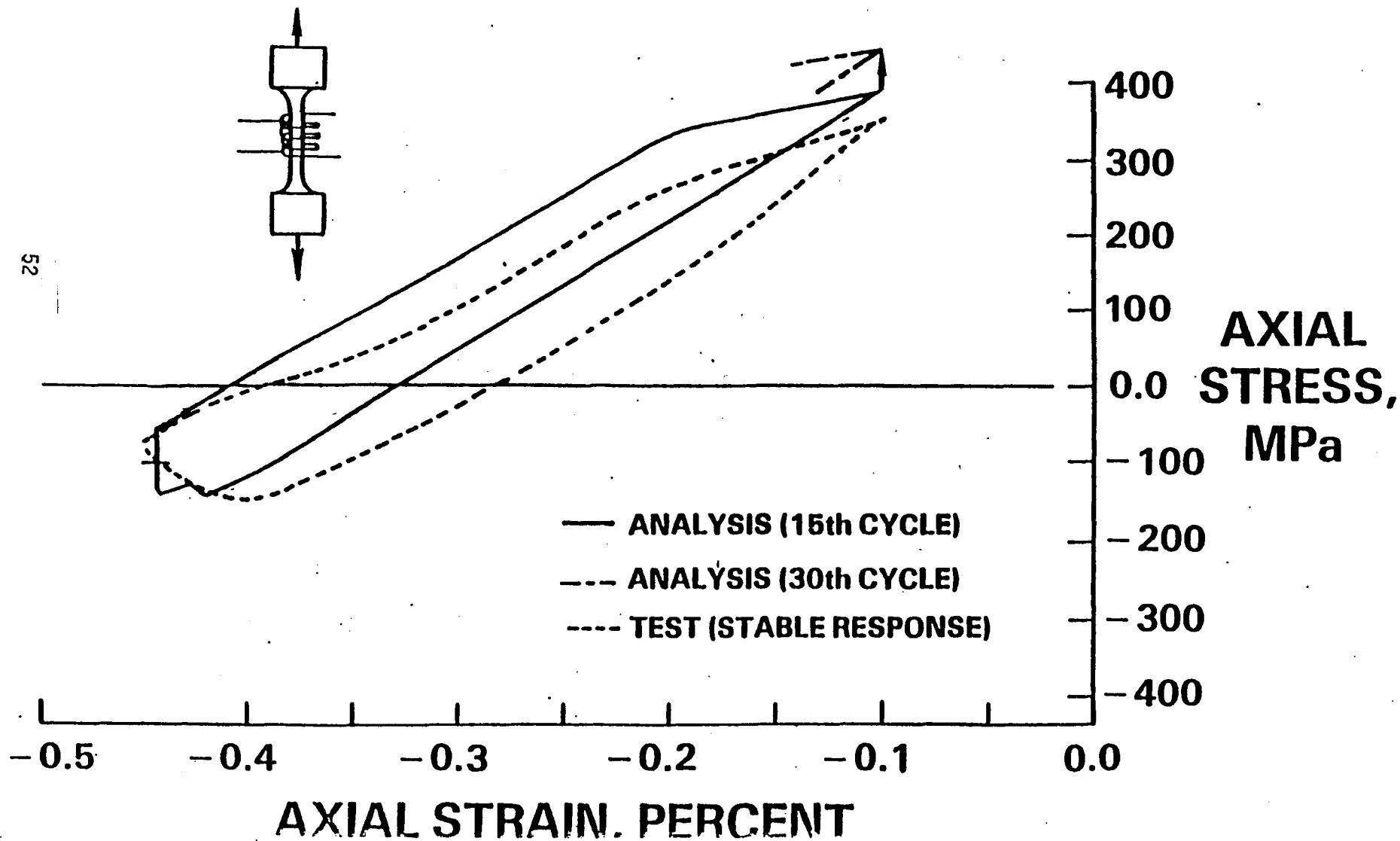


PREDICTED LOUVER LIP RESPONSE FOR SIX (6) LOADING CYCLES

51



COMPARISON OF UNIAXIAL THERMO MECHANICAL TEST AND ANALYTICAL RESULTS



CONCLUSIONS

- ELASTIC ANALYSIS ADEQUATE FOR OBTAINING STRAIN RANGE AND CRITICAL LOCATION
- INELASTIC ANALYSES DID NOT ACCURATELY REPRESENT CYCLIC BEHAVIOR OF MATERIAL
- NONE OF CRACK INITIATION LIFE PREDICTION METHODS WERE SATISFACTORY

PRE-HOST HIGH TEMPERATURE CRACK PROPAGATION
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The Pre-HOST activities in High Temperature Crack Propagation are described in detail in NASA CR-167896, "Fracture Mechanics Criteria for Turbine Engine Hot Section Components." What follows is a brief recap of the highlights of that contract.

This was a 14-month contract awarded in late 1980 to Pratt & Whitney Aircraft. The principal investigator was G. J. Meyers. The program consisted of five technical Tasks.

In Task I, the Contractor was to "establish the locations, characteristic geometry, temperature levels, and stress levels in hot section components of typical advanced turbine engines which present crack initiation and crack propagation conditions that may significantly impact engine operational safety or engine maintenance costs." "The suitability and the limitations of the currently available methods for correlation and generalization of crack propagation data such as linear elastic fracture mechanics parameters" were to be evaluated. The key results of this Task are shown in the first two figures.

In Task II, the Contractor was to identify "the empirical crack growth predictive methods and data necessary for effective design involving the potential cracking conditions identified in the preceding Task I" and also "the test specimen designs and the facility requirements which can provide the required data under suitably controlled thermomechanical crack propagation conditions." A suitable testing program was to be developed.

In Task III, The Contractor was to "conduct an analysis to identify and define the nature and magnitude of the crack initiation and propagation mechanisms at the sites identified in Task I." "An analysis of the corresponding test specimen geometry and loading" was also to be made. The component which received the greatest amount of analysis is the combustor liner shown in Figure 3-1. A finite-element analysis had already been performed under another contract, and the results are shown in Figure 5.2-3. The louver lip (location 1) is the area of interest and contains strong thermal gradients. A more detailed model, Figure 5.3-8, was constructed for J-integral analysis using the MARC program (note the three contours). The calculated J-integral was found to be very sensitive to material property variations within the contour of integration (Figures 5.3-9 and 5.3-10). The results of this analysis indicate that the J-integral calculation resident in the MARC program is not satisfactory for this kind of problem. The specimens that were to be tested in Task IV were analyzed using a modified Shih-Hutchinson approach.

Task IV consisted of the actual testing. The specimens used were tubular specimens with short circumferential through-thickness cracks as shown in Figure 6.2-1, with the initial EDM slot being about 0.040" long. The external-ridge specimens were used for the isothermal tests, the internal-ridge for TMF tests. The TMF cycles that were used are shown in Figure 6.2-2. Cycle I and Cycle II are linear, the "Faithful Cycle" is an approximation to the calculated louver-lip cycle shown earlier. The test matrix is shown in Tables 6-I and 6-II.

The data are correlated and generalized in Task V. As shown in Figures 7.4-2 and 7.4-3, The J-integral was not entirely successful in

correlating the data. One should note, however, that all calculations were effectively based on the assumption that the crack began to open at maximum compressive load. If all cracks opened at the same percentage of max. load, then all calculations are in error by the same constant factor. Based on data spread, the crack tip opening displacement (which was calculated from the J-integral) gave a somewhat better correlation.

There are two significant points to be noted. First, the J-integral calculation resident in the MARC computer program is not satisfactory for problems involving thermal gradients. Second, the specimens tested here were sparsely instrumented, and therefore we can only guess at how they actually responded to the thermal and mechanical cyclings.

DOCUMENTATION OF DAMAGE RESULTS IN JT9D COMBUSTOR LINERS
(Outer and Inner)

Figure Number	Time (% of Calculated 8-50)	Cycles (% of Calculated 8-50)	Calculated Temperature (°F)	Calculated Strain Range (%)	Nature of Damage
COMBUSTOR OUTER LINERS					
3-2	38	20	1780	0.45	<ul style="list-style-type: none"> o Lip Collapse o Coating Spallation o Burning o Extensive Cracking
3-3	84	85	1810	0.45	<ul style="list-style-type: none"> o Cracking and Burning (Similar to Figure 3-2)
3-4	51	88	1780	0.45	<ul style="list-style-type: none"> o Extensive Cracking o Localized Distress
3-5	81	51	1780	0.45	<ul style="list-style-type: none"> o Extensive Cracking (One Severe Crack)
COMBUSTOR INNER LINERS					
3-6	27	10	1730	0.25	<ul style="list-style-type: none"> o Erosion and Burning o Axial and Circumferential Cracking o Dilution Air Hole Cracking
3-7	53	34	1730	0.37	<ul style="list-style-type: none"> o Mild Dilution Air Hole Cracking o Cracking in Aft End

NOTES:

Cooling Type: Film Cooled Material: Hastelloy-X Coating: Metallic-Ceramic Thermal Barrier

Crack Initiation Location: Outer Liner; End of lower lip
Inner Liner; End of lower lip and circumferential seam weld

Liners must be weld-repaired or eventually replaced.

FAILURE CONSEQUENCES:

Outer Liner: Axial cracks link together, resulting in liner deformation. This deformation may affect combustor exit temperature distribution with an ultimate effect on turbine performance and durability.

Inner Liner: Intersection of large axial and circumferential cracks can result in liberation of pieces of the liner, causing secondary damage to turbine blades and vanes.

TABLE 3-II

IMPORTANT DAMAGE MECHANISMS FOR
JT9D HIGH-PRESSURE TURBINE AIRFOILS

<u>Airfoil</u>	<u>Damage Mechanisms</u>
First-Stage Turbine Vane	<ul style="list-style-type: none"> o Cracking (oxidation-assisted) of leading edge and pressure-side wall. o Burning around leading edge cooling holes.
Second-Stage Turbine Vane	<ul style="list-style-type: none"> o Leading edge cracking (early models). o Coating oxidation and impact damage (later models).
First-Stage Turbine Blade	<ul style="list-style-type: none"> o Radial cracking of pressure- and suction-side walls. o Blade tip oxidation. o Stress rupture. o Impact damage.
Second-Stage Turbine Blade	<ul style="list-style-type: none"> o Impact damage. o Stress rupture (early models).

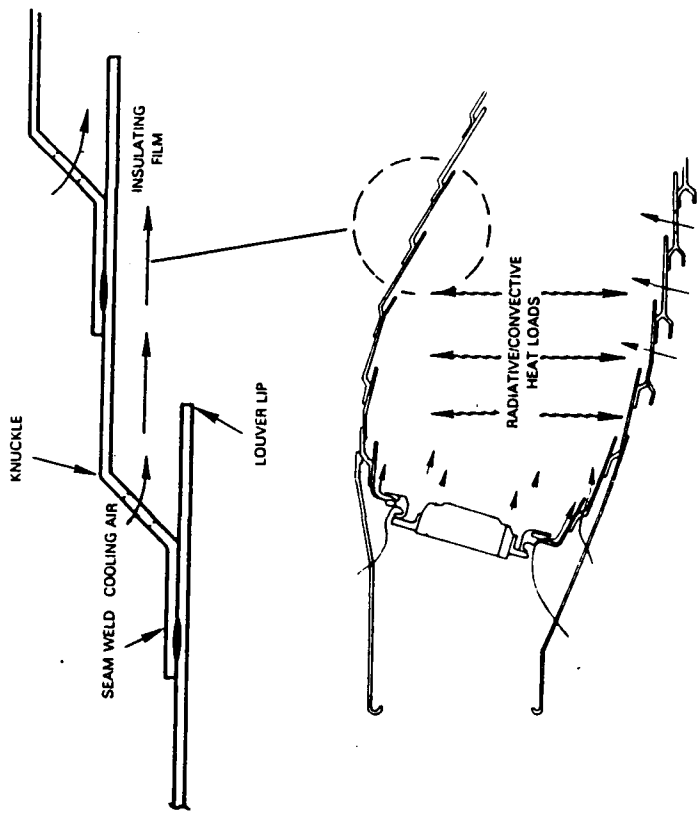


Figure 3-1 Typical Combustor Liner Louvered Construction.

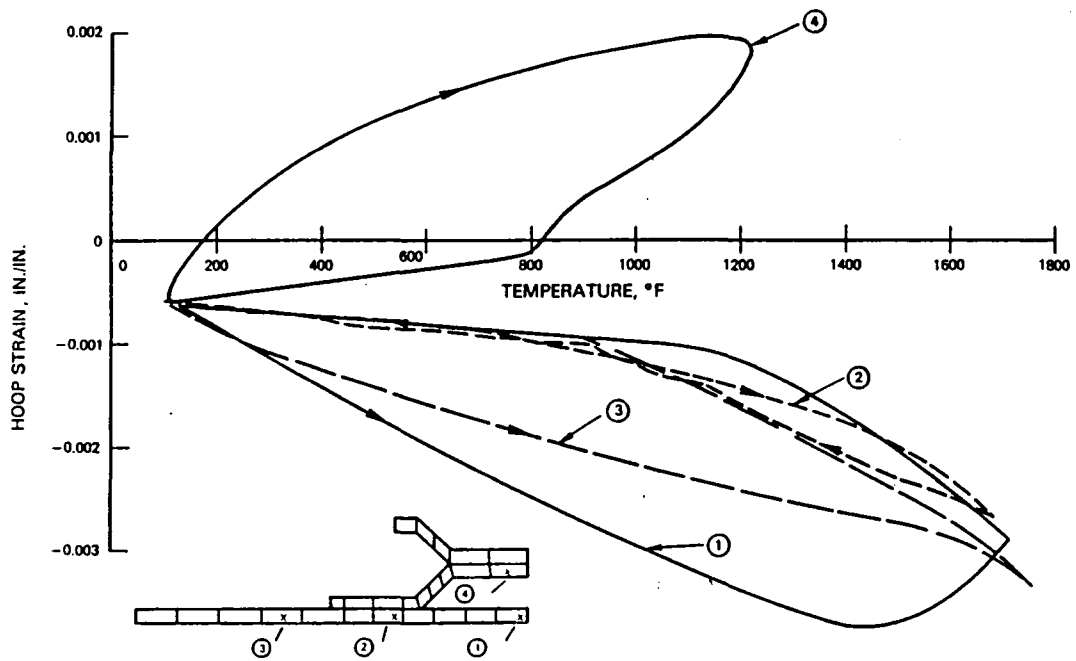


Figure 5.2-3 Strain-Temperature Response at Several Locations Along Combustor Liner Louver.

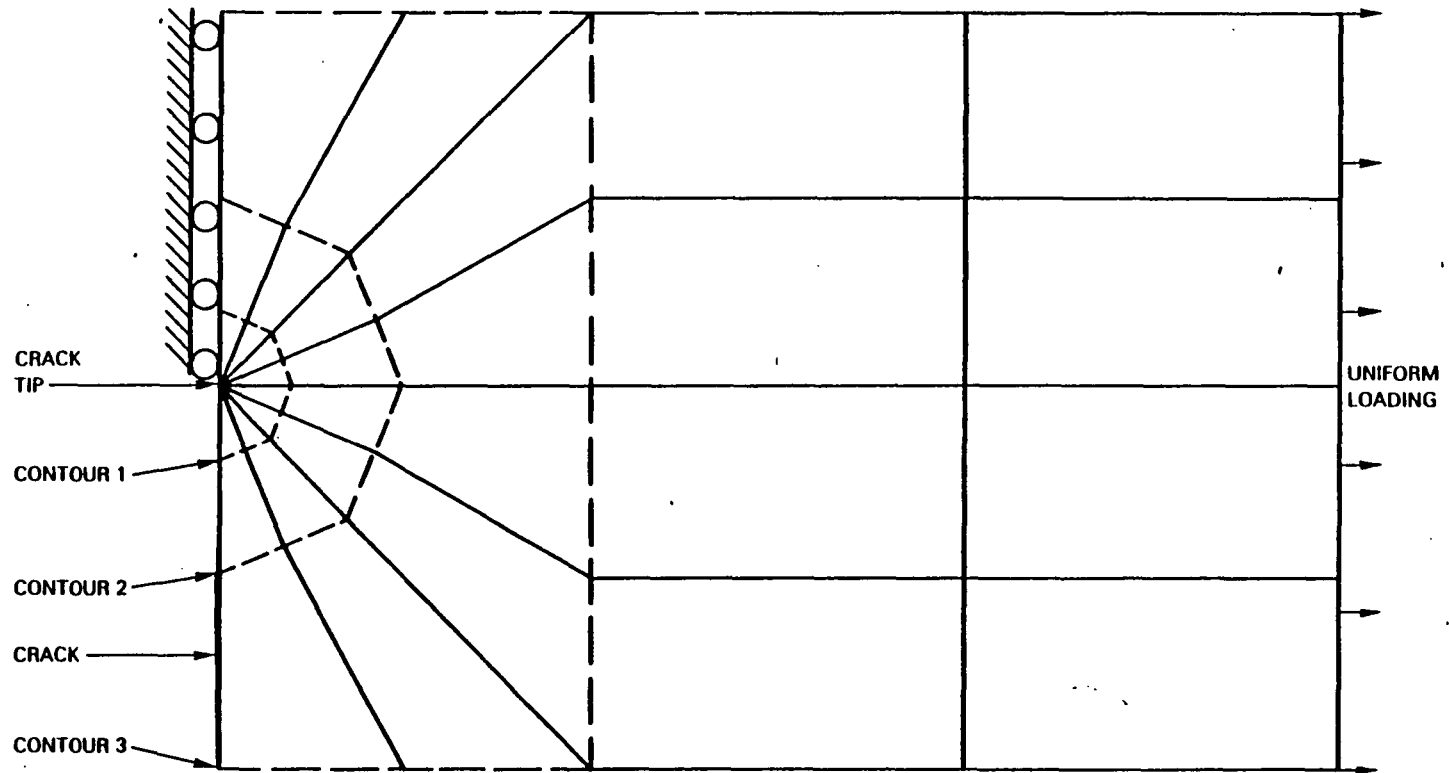


Figure 5.3-8 Coarse Grid Finite Element Mesh for J-Integral Test Cases.

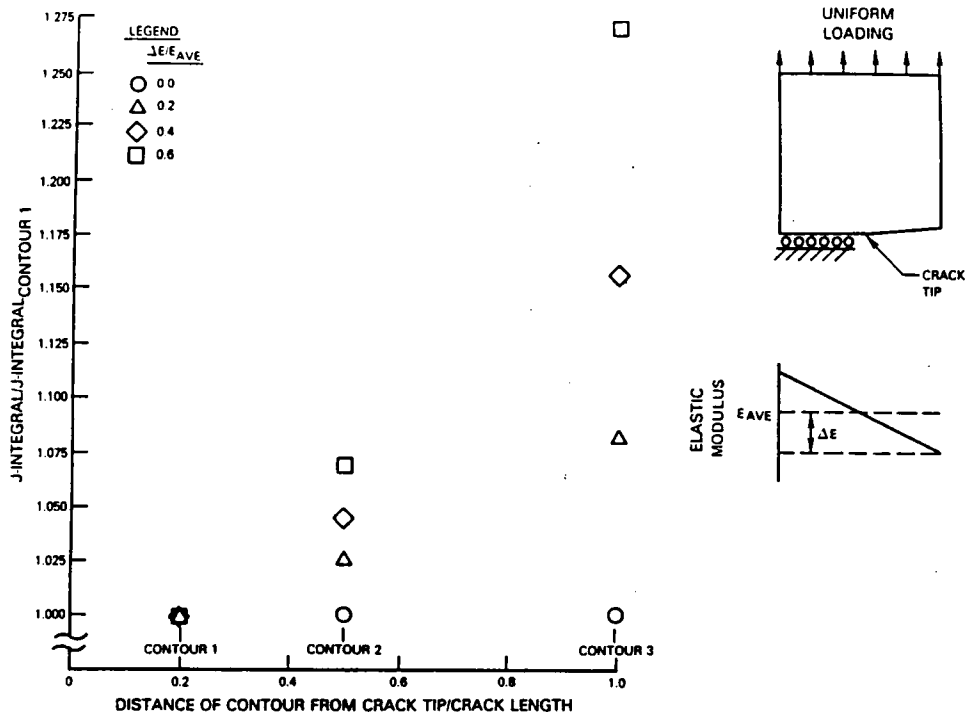


Figure 5.3-9 Effect of Elastic Modulus Variation on J-Integral Calculation using Coarse Grid.

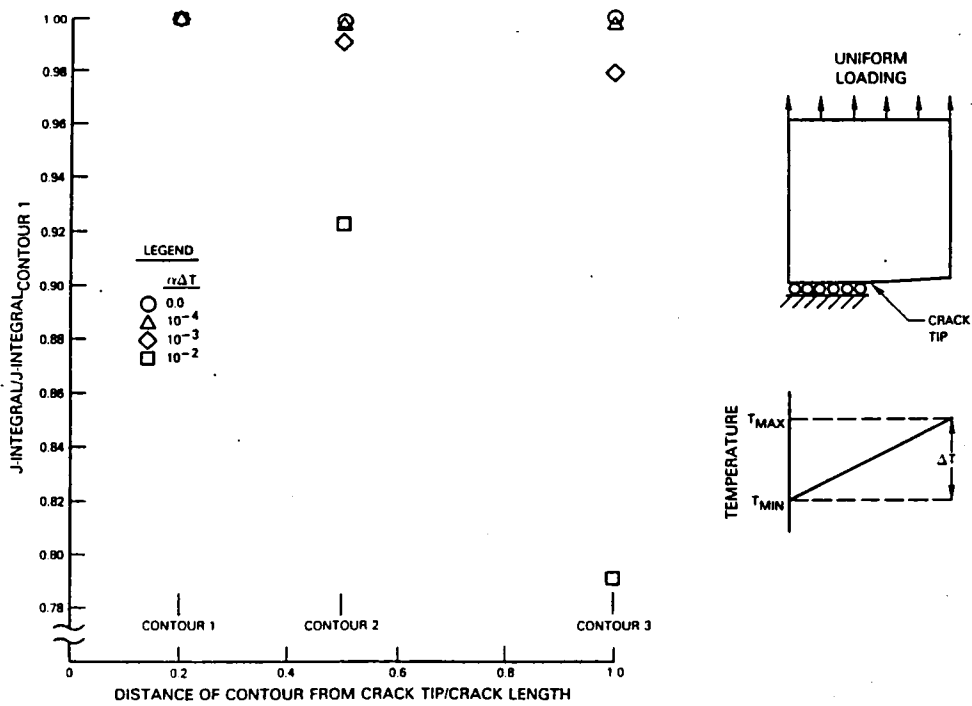
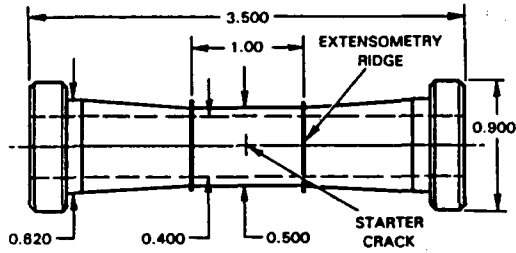
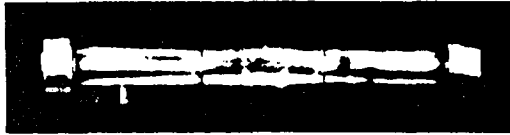
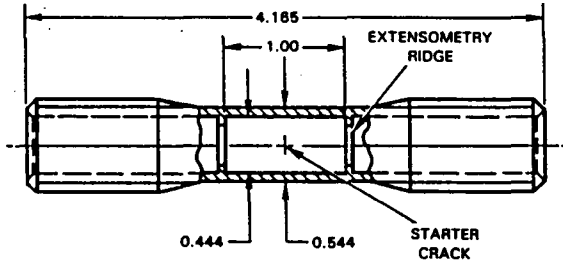
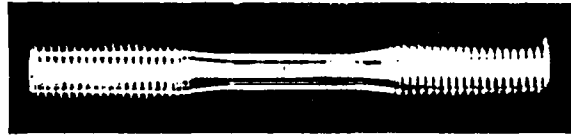


Figure 5.3-10 Effect of Linear Temperature Gradient on J-Integral Calculation Using Coarse Grid.

EXTERNAL RIDGE SPECIMEN



INTERNAL RIDGE SPECIMEN



DIMENSIONS ARE NOMINAL VALUES, GIVEN IN INCHES

Figure 6.2-1 Tubular Strain-Controlled Crack Propagation Specimens.

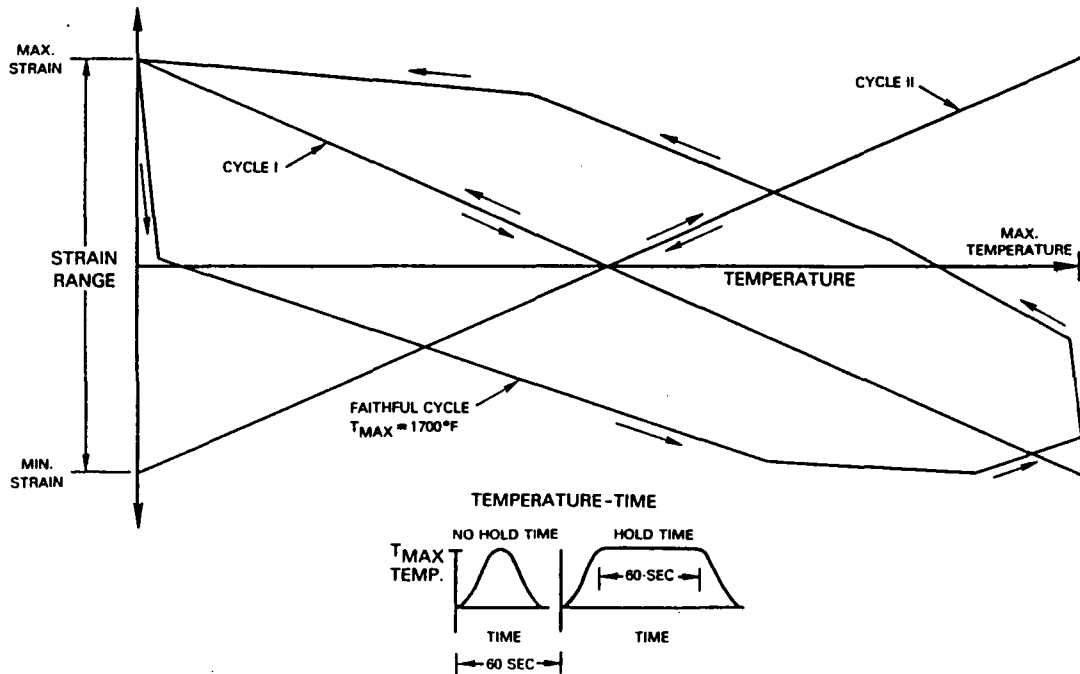


Figure 6.2-2 Strain-Temperature Cycles used in Thermomechanical Fatigue Testing.

TABLE 6-I

CONDITIONS FOR ISOTHERMAL TESTING*

Test No.	Temperature (°C)	Temperature (°F)	Strain Range (%)	Minimum Strain (%)	Maximum Strain (%)	Cyclic Rate (cpm)	Average Strain Rate (in/in)/min	Comments
I-1	427	800	0.15	-0.075	0.075	60	0.18	
I-2	427	800	0.40	-0.20	0.20	10	0.08	
I-3	427	800	0.40	-0.45	-0.05	10	0.08	Mean Strain=-0.25%
I-4	427	800	0.25	-0.125	0.125	10	0.05	
I-6	649	1200	0.15	-0.075	0.075	2.0	0.006	
I-7	649	1200	0.40	-0.20	0.20	1.0	0.008	
I-8	649	1200	0.40	0.05	0.45	1.0	0.008	Mean Strain=+0.25%
I-9	760	1400	0.15	-0.075	0.075	1.0	0.003	
I-10	760	1400	0.25	-0.125	0.125	0.5	0.005	
I-11	760	1400	0.40	-0.20	0.20	1.0	0.004	
I-13	760	1400	0.25	-0.125	0.125	0.5	0.005	1 minute Hold Time
I-14	871	1600	0.15	-0.075	0.075	1.0	0.003	
I-15	871	1600	0.175	-0.0875	0.0875	1.0	0.0035	
I-16	871	1600	0.40	-0.02	0.02	0.5	0.004	
I-18	927	1700	0.15	-0.075	0.075	1.0	0.003	
I-19	927	1700	0.25	-0.125	0.125	1.0	0.005	
I-20	927	1700	0.40	-0.20	0.20	0.5	0.004	
I-21	927	1700	0.25	-0.125	0.125	1.0	0.005	Mean Strain=-0.25%
I-22	927	1700	0.25	-0.125	0.125	0.5	0.005	1 minute Hold Time
I-23	982	1800	0.15	-0.075	0.075	1.0	0.003	
I-23a	982	1800	1.50	-0.75	0.75	1.0	0.030	Large Strain Range
I-24	982	1800	0.25	-0.125	0.125	1.0	0.005	
I-25	982	1800	0.40	-0.20	0.20	0.5	0.004	
I-26	982	1800	0.40	-0.20	0.20	0.5	0.004	Triangular Wave Shape

* All tests had a sinusoidal wave shape, zero mean strain, and no hold time, except where indicated.

TABLE 6-II

CONDITIONS FOR THERMOMECHANICAL FATIGUE TESTING*

Test No.	Temperature (°C)	Temperature (°F)	Strain Range (%)	Minimum Strain (%)	Maximum Strain (%)	Cyclic Rate (cpm)	Average Strain Rate (in/in)/min	Comments
T-1	927	1700	0.15	-0.075	0.075	0.83	0.0025	
T-2	927	1700	0.25	-0.125	0.125	0.83	0.0042	
T-3	927	1700	0.40	-0.20	0.20	0.44	0.0035	
T-4	927	1700	0.25	-0.125	0.125	0.83	0.0042	Cycle II
T-5	927	1700	0.40	-0.20	0.20	0.44	0.0035	Faithful Cycle
T-6	927	1700	0.40	-0.20	0.20	0.30	0.0035	Faithful Cycle; 1.125-minute Hold Time
T-7	982	1800	0.25	-0.125	0.125	0.83	0.0042	
T-8	871	1600	0.25	-0.125	0.125	0.83	0.0042	
T-9	760	1400	0.25	-0.125	0.125	0.83	0.0042	
T-10	649	1200	0.25	-0.125	0.125	0.83	0.0042	
T-11	927	1700	0.40	-0.20	0.20	0.30	0.0035	1.125-minute Hold Time
T-12	871	1600	0.40	-0.20	0.20	0.44	0.0035	

* All tests were Cycle I with no hold time except where indicated.

* All tests had a minimum temperature of 427°C (800°F) and zero mean strain.

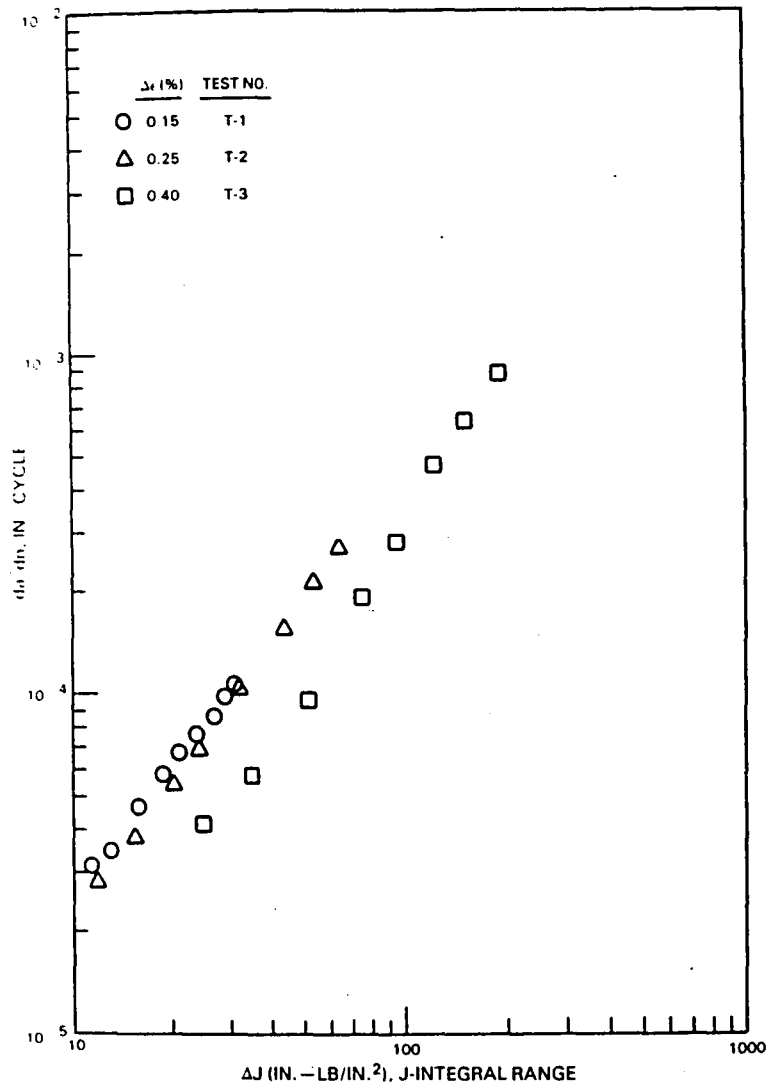
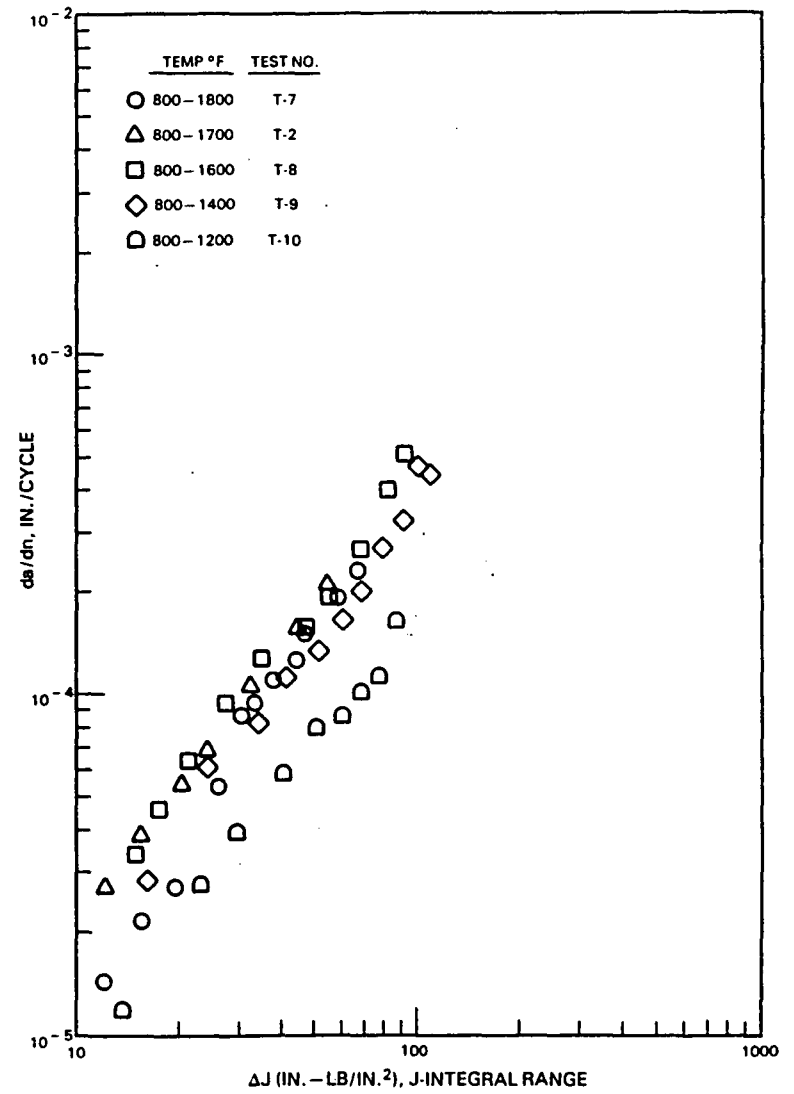


Figure 7.4-2 800 to 1700°F Cycle I Crack Growth Rates Based on J-Integral Range.



re 7.4-3 Cycle I, 0.25 Percent Strain Range Crack Growth Rates Based on J-Integral Range.

HOST INSTRUMENTATION R&D PROGRAM

OVERVIEW

by

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The HOST Instrumentation R&D program is focused on two main classes of instrumentation. The first class is for characterizing the environment around the turbine engine components. These instruments include those for measuring gas flows, gas temperatures, and heat fluxes. The second class of instruments is for characterizing the effect of the environment on the turbine engine components. The second class includes strain measurements and an optical system for viewing various other structural responses such as cracking, buckling, spalling, carbon buildup, etc.

The HOST Instrumentation R&D program was formulated to concentrate on the critical measurements that could not be made with commercially available instruments or with instruments that were already under development via NASA- or DOD-funded efforts or in IR&D programs.

The HOST Instrumentation R&D Program Schedule showing the current active efforts is included in the accompanying figures. The program schedule shows all HOST-funded efforts plus selected non-HOST-funded efforts initiated during the year prior to the start of the HOST program that directly relate to the HOST goals. Each line represents a separate contract, grant, or LeRC in-house effort.

The heaviest resources are concentrated on the measurements of strain and gas flow since these measurements are extremely critical to the success of the HOST program and the HOST requirements differ from the current state of the art by a considerable margin. Followup and complementary efforts not shown in the schedule are being planned for the strain measurement area.

TURBINE ENGINE HOT SECTION TECHNOLOGY

INSTRUMENTATION RESEARCH AND DEVELOPMENT

CS-82-2681

HOST INSTRUMENTATION R&D PROGRAM

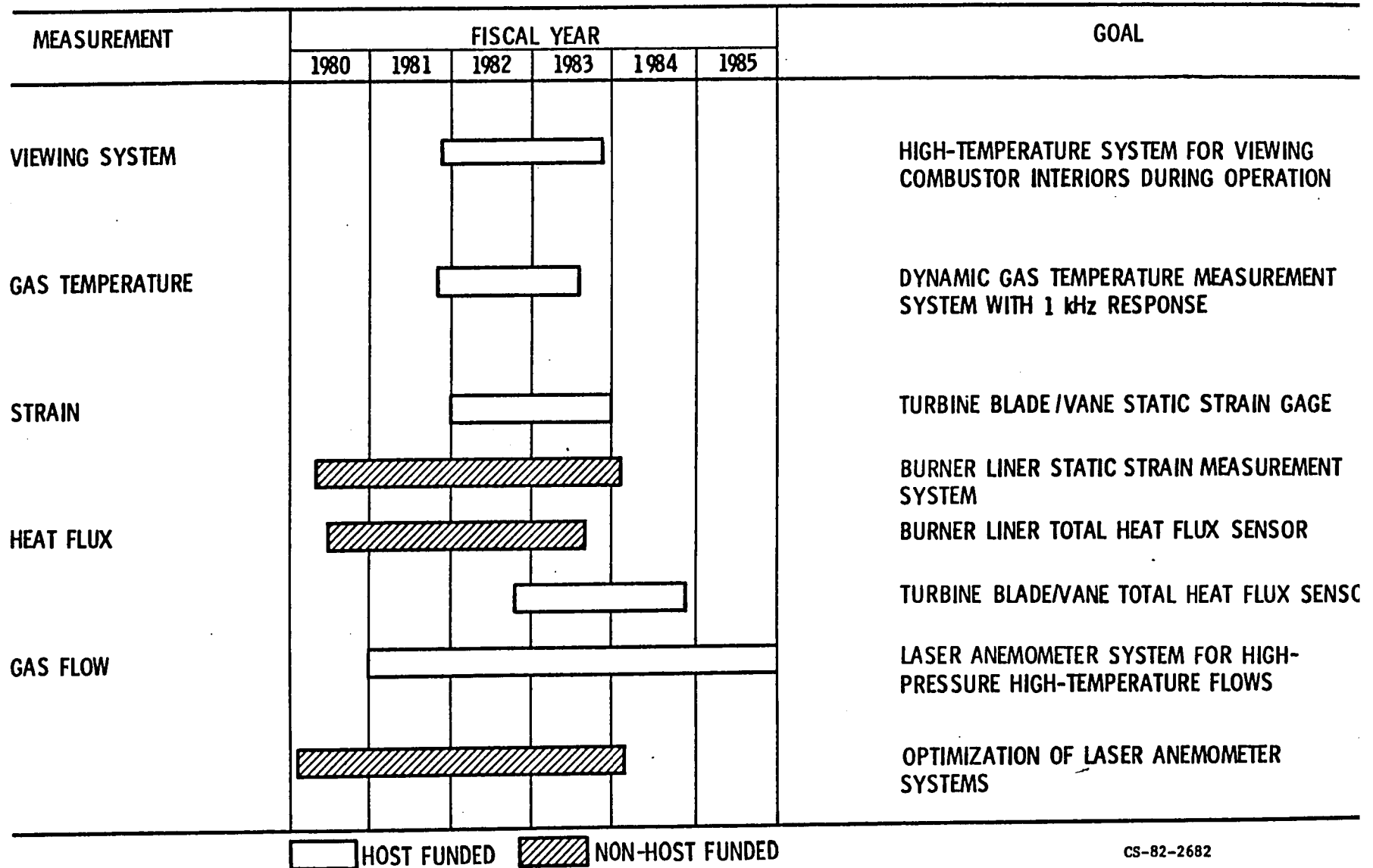
GENERAL GOALS:

- **DEVELOP INSTRUMENTATION FOR CHARACTERIZING THE ENVIRONMENT AROUND TURBINE ENGINE COMPONENTS**
- **DEVELOP INSTRUMENTATION FOR CHARACTERIZING THE EFFECT OF THE ENVIRONMENT ON THE TURBINE ENGINE COMPONENTS**

CS-82-2680

HOST INSTRUMENTATION R&D PROGRAM

(SHOWING ACTIVE EFFORTS AS OF 10/82)



TURBINE ENGINE HOT SECTION TECHNOLOGY

INSTRUMENTATION SESSION AGENDA

OVERVIEW

N. WENGER, LeRC

COMBUSTOR VIEWING SYSTEM

W. MOREY, UTRC

DYNAMIC GAS TEMPERATURE PROBE

D. ELMORE, P&W GPD

TURBINE BLADE/VANE STATIC STRAIN GAGE

F. LEMKEY, UTRC

LASER SPECKLE TECHNIQUE FOR BURNER
LINER STRAIN MEASUREMENTS

K. STETSON, UTRC

HEAT FLUX SENSORS FOR BURNER LINERS AND
TURBINE BLADES AND VANES

G. ALWANG, P&W CE

HOT SECTION LASER ANEMOMETRY

W. NIEBERDING, LeRC
R. EDWARDS, CWRU

CS-82-2679

COMBUSTOR VIEWING SYSTEM

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United Technologies Research Center
East Hartford, Connecticut 06108

The overall objectives of the hot section viewing program (Slide 1) are to develop an optical system for viewing the interior of a combustor during high temperature, high pressure operation, and to produce a visual record of some causes of premature hot section failures. The program designed to accomplish these objectives (Slide 2) includes: identifying and analyzing system designs that will provide clearest images and be able to survive in the hostile environment inside the combustion chamber; a preliminary test program to investigate the performance of primary system components and identify problem areas; and examination of computer techniques for image enhancement and analysis of combustor images. The final phase of the program will conclude with the design, fabrication, demonstration, and delivery of a prototype system.

Viewing the inside of a combustor (Slide 3) will be useful to determine the location and shape of the flame, and if and under what operating conditions the flame is impinging on the combustor wall. Flame instability and turbulence could be observed or measured and possibly related to combustor acoustics. The condition of the combustor liner, the fuel nozzles, and the turbine vanes could be examined during combustor operation, and the prognosis of liner damage or hot spots may be followed. It may also be possible to observe the accumulation of coking deposits, observe the fuel spray pattern, and measure relative or absolute flame temperatures. It follows, then, that the immediate applications (Slide 4) for a combustor viewing probe would be as a diagnostic tool for the study of combustor failure modes and as an aid in the design and analysis of future combustors. The probe may be especially helpful in more critical combustor designs that operate at higher temperatures or burn alternate fuels. Future applications with a combustor viewing or optical detection probe may include routine diagnostic inspection on an operating engine for preventive maintenance, an in-flight engine monitor, and an engine control input.

The hot section viewing program (Slide 5) is a twenty-four month program. During the first year of the program, which has recently been completed, the analysis of system designs, a preliminary test program design, and a preliminary test program to look at problem areas was carried out. In the second year of the program, which has just been initiated, we will design and fabricate a prototype viewing system. The prototype viewing probe will be tested in a high pressure combustor rig and the program concluded with a post test analysis.

One of the problem areas investigated during the preliminary test program was viewing a combustor liner surface through an oil burning flame. An oil flame is highly luminous due to the generation of very small soot particles, most of which are eventually consumed. The flame creates an intense background light making it difficult to view or record images of the combustor surfaces. A pulsed laser illumination source and computer image enhancement were two techniques tested to reduce the effect of the flame background.

An atmospheric pressure oil burner rig was constructed in the laboratory (Slide 6) for the viewing tests. Six viewing ports were placed along each side of the combustion chamber so that views could be taken of objects through the flame.

A pulsed dye laser was used as an illumination source (Slide 7). The laser beam was focused into a single large core fiber that transmitted the beam to the combustor viewing port alongside the viewing probe. The viewing system consisted of a lens, a fused fiber bundle consisting of 70,000 fibers or picture elements, and a camera to record the transmitted image. A view of a combustor liner section, taken with the viewing system, can be seen in the photograph (Slide 8) with the combustor off and on. With a standard incandescent illuminator the liner piece could not be seen or recorded with a camera due to the presence of the bright flame background. The pulsed laser illumination is considerably brighter than the flame, however, and can discriminate against the flame radiation by allowing an exposure that is too short in time for the flame radiation. We can obtain additional discrimination against the flame by wavelength filtering to pass only the dye laser region of the spectrum.

The use of computer image analysis techniques to expand the contrast and enhance one's ability to see through the flame background is demonstrated in the next photographs (Slides 9 and 10). In this latter case, the flame was not nearly as dense as in the previous case where the dye laser illuminator was used.

Optical distortion is another problem that will occur when viewing inside combustors. In the highly turbulent combustion zone thermal gradients may exist that range from combustor inlet temperatures to maximum combustion temperatures over distances of a few centimeters. As a consequence, the light signal from a point source of light or a high resolution view of a combustor surface will be distorted to some extent. The extent of the distortion in the laboratory rig, about 1 mrad, is shown in the next slide (Slide 11). Here a point source of light is magnified at the output of the image bundle. The individual fiber elements (10 μm dia.) can be seen in the photo. A computer analysis of the images with and without the combustor in operation can give the amount of distortion spread of the image and also give the fundamental point spread function of the viewing system so that distortion corrections could be applied to the image (Slide 12). The distortion one could expect in a high pressure combustor may be five to six times greater than we observed in the laboratory rig.

A preliminary test probe was designed and fabricated (Slide 13) to test the ability to cool a probe placed in the primary combustion zone and to examine different configurations for purging the exposed viewing surface of soot deposits. Copper was the material chosen for the probe due to its superior heat transfer capability. From results taken in the laboratory rig, we estimate that the probe could easily survive in the high heat transfer environment of a gas turbine combustor with high pressure water cooling for the probe. A best gas flow design for keeping a viewing window at the end of the probe free of soot in the primary combustion was also determined from a series of tests.

HOT SECTION VIEWING SYSTEM

- **Objectives**
 - **Develop an optical system for viewing the interior of a combustor during high temperature, high pressure operation**
 - **Produce a visual record of some causes of premature hot section failures**

Slide 1

APPROACH

- **Identify and analyze system designs that will provide clearest images and survive in the hostile combustor environment**
- **Conduct a preliminary test program to investigate the performance of primary system components and identify problem areas**
- **Examine computer techniques for image enhancement and analysis of combustor images**
- **Design, fabricate, demonstrate, and deliver a prototype system**

Slide 2

USE OF COMBUSTOR IMAGE

- **Location and shape of flame**
- **Flame instability and turbulence**
- **Relative and absolute flame temperature**
- **Liner damage**
- **Condition of fuel nozzles**
- **Coking deposits**
- **Fuel spray pattern**
- **Condition of turbine vanes**

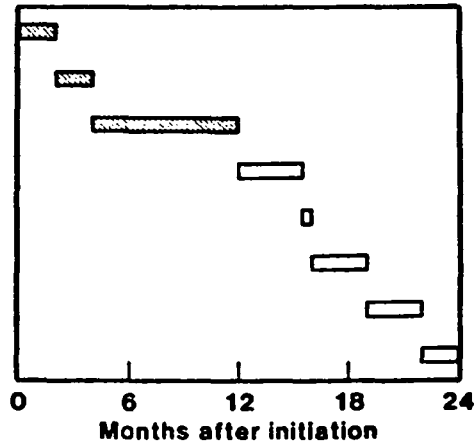
Slide 3

- **Immediate applications**
 - **Diagnostic tool for study of combustor failure modes**
 - **Aid in combustor analysis and design**
- **Future applications**
 - **Routine diagnostic inspection for engine maintenance**
 - **Inflight engine monitor**
 - **Engine control input**

Slide 4

HOT SECTION VIEWING SYSTEM PROGRAM

- Analysis
- Program design
- Preliminary test
- Design prototype
- Design test apparatus
- Fabrication
- Testing
- Post test analysis



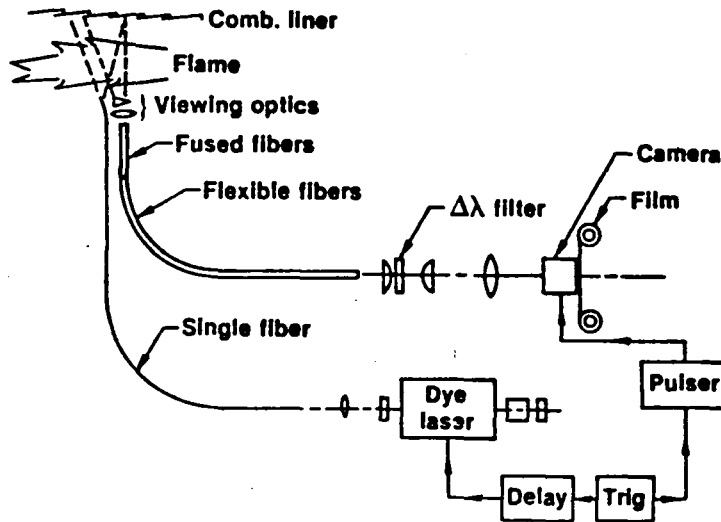
Slide 5

COMBUSTOR FOR VIEWING TESTS



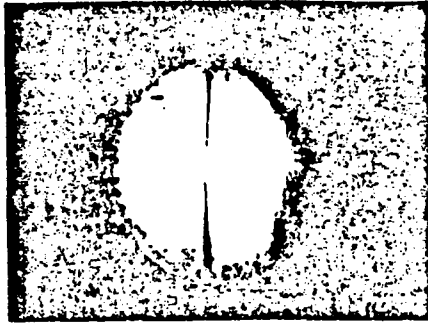
Slide 6

COMBUSTOR VIEWING SYSTEM

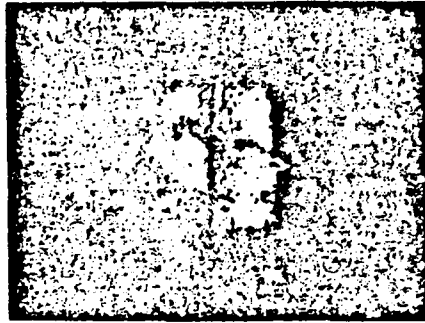


Slide 7

VIEW OF COMBUSTOR LINER WITH LASER ILLUMINATION



Laser illumination, combustor off



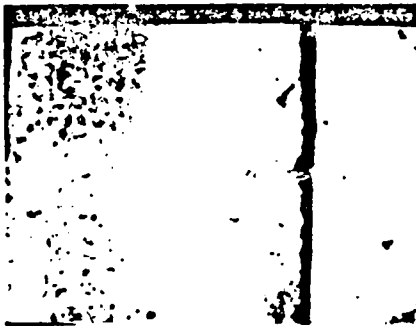
Combustor on

Slide 8

LINER CRACK IMAGE

Flame Off
Digitized

Original gray scale



Histogram equalization

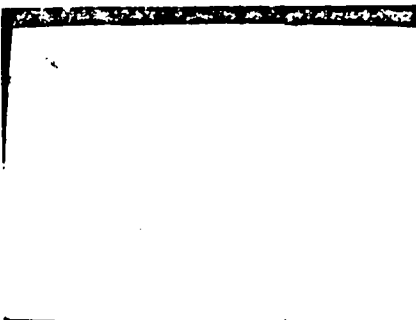


Slide 9

LINER CRACK IMAGE

Flame On
Digitized

Original gray scale

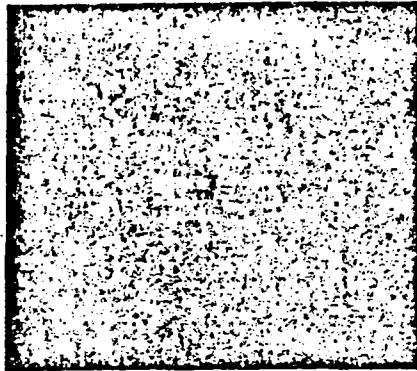


Histogram equalization

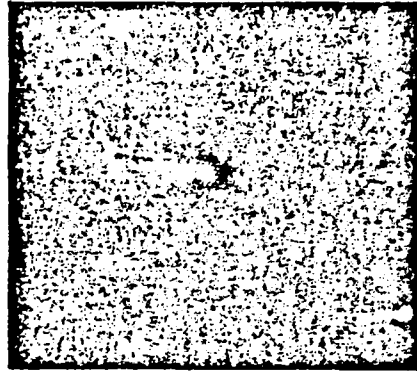


Slide 10

OPTICAL DISTORTION IN COMBUSTOR



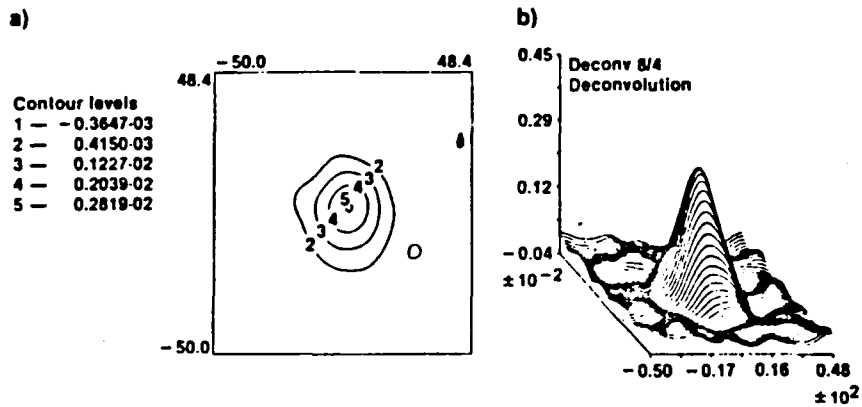
Combustor off



Combustor on

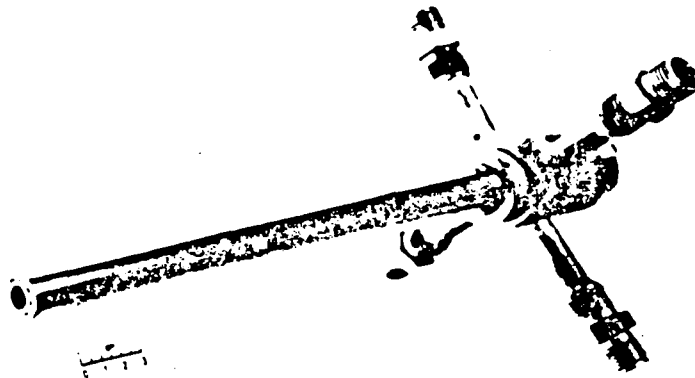
Slide 11

POINT SPREAD FUNCTION Derived from Distortion Measurements



Slide 12

PRELIMINARY TEST PROBE



Slide 13

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Dennis Elmore

Pratt & Whitney Aircraft
Government Products Division

INTRODUCTION

The objective of this effort is to develop an advanced measuring system which is capable of measuring the rapidly varying gas temperature at the exit of an aircraft jet engine combustor during ground based testing of hot section components. The following presentation gives a brief review of the contract objectives/sensor guidelines, the technical approach/program schedule, and the accomplishments to date.

The Sensor must be designed for installation in an annular jet engine combustor operating with jet A fuel and air, and must have a compensated frequency response up to 1 K Hz. The environment of present-generation combustors is detailed in Table I. The program goals for measurement uncertainty are 5% or less for frequencies up to 200 Hz and 10% or less from 200 Hz to 1000 Hz.

The Program is organized into eight tasks with the technical effort concentrated into the following six tasks:

- Task 1 - Study and Selection of Methods
- Task 2 - Analysis of Methods and Selection of the Most Promising Method
- Task 3 - Review
- Task 4 - System Design and Test Plan Formulation
- Task 5 - Fabrication and Tests
- Task 6 - Analysis of Results

The contract was initiated on 10 August 1981 and will run for an 18 month period. Tasks 1 through 4 have been completed and Task 5 is currently in process. The results of these efforts are briefly discussed in the following paragraphs.

PROGRESS

Task 1 consisted of identifying and evaluating candidate measurement concepts. The effort included performing a literature survey using the Lockheed Dialog, the NASA Recon and the Defense Technical Information Center information retrieval systems. A list of candidate measurement methods is shown in Table II. These candidates were evaluated against contract technical criteria and rated either acceptable or unacceptable. Results of the evaluation revealed that only two concepts meet all contract requirements: a) The dual-wire passive thermocouple and b) single-wire pulse heated thermocouple.

The dual-wire passive thermocouple concept uses two different diameter, beadless-junction thermocouples to measure heat transfer coefficient in-situ and thereby compensate thermocouple response. The single-wire pulse heated thermocouple concept uses an electrically-pulsed over-temperature condition to determine the time constant from the pulse decay rate. Construction techniques are very similar for both designs. Both approaches use type B wire elements with beadless welded junctions to provide a uniform cylinder in cross flow geometry which is easily analyzed. Both are mounted similarly in high temperature ceramic insulators. The length to diameter ratio for thermocouple elements of both designs is less than 20 to 1.

The two concepts were further analyzed in Task 2. Each concept was analyzed for structural, thermal, and data acquisition/reduction requirements. Both concepts were rated acceptable structurally and thermally. The single pulse-heated concept would have to be pulsed near the melting temperature of the thermocouple material. In addition, the pulse-heating amplitude was not large enough to obtain the contract accuracy requirements using a reasonable number of decay pulses and sampling times. Based on these results, the dual passive thermocouple concept was selected for use in the remaining tasks involving detailed design, fabrication and testing.

The analyses also revealed that both concepts had end conduction losses which could not be ignored and would have to be corrected during the data analyses.

Task 4 efforts consisted of: 1) design of a temperature measuring system based on the dual-wire passive thermocouple approach, including sensor detailed design and design of the data acquisition/reduction system; and 2) the definition of the test plans.

The sensor design effort was based on the thermal and structural analyses of Task 2. The final probe configuration was designed for installation in an F100 engine borescope plug location at turbine inlet. The probe was designed structurally to P&WA design criteria and the final thermocouple wire element lengths were reduced from preliminary designs due to higher aerodynamic loading in the full scale engine.

The final data acquisition/reduction system design was also based on results of Task 2. The computer software was modified from a simple first order system to a program using a finite element model which accounts for sensor wire end conduction losses. The data acquisition system records each thermocouple signal on FM tape through AC coupling and low noise differential amplifiers (Preston DX-A3). Data reduction system consists of an FM tape reproduce and a Hewlett Packard Model 5451 C Fourier Analyzer.

The data reduction process consists of computing the theoretical response (gain and phase) of each thermocouple over a range of discrete frequency points (generally between 2 Hz to 40 Hz) as a function of heat transfer coefficient using the finite element analysis. The range of heat transfer coefficient is selected to cover the anticipated value for the engine test conditions. From these theoretical data, the transfer function (ratio of gains) of the two thermocouples for the same discrete frequency points as a function of heat transfer coefficient is derived.

Engine test data for both T/C's is digitized (4 KHz sampling rate) into the Fourier system and converted to temperature using an NBS curve fit. From these data the transfer function of the two thermocouples is computed vs. frequency using FFT techniques. This measured transfer function data is used to determine the in-situ value of heat transfer coefficient from the theoretically derived relationships of the transfer function of the two thermocouples as a function of heat transfer coefficient.

The in-situ value of heat transfer coefficient is then used to compute the theoretical response of the 76 μm T/C to the gas stream temperature fluctuations using the finite element analysis. Gain and phase response is computed at discrete frequencies from 2 Hz to 2KHz. This response curve is then used to compensate the 76 μm thermocouple engine data to yield a compensated frequency spectrum. The compensated frequency spectrum can be Fourier transformed to yield the compensated temperature time wave form.

The test plans consist of three test series: 1) System shakedown laboratory tests; 2) Laboratory burner tests; and 3) An engine test.

A summary of environmental conditions for each of these tests is shown in Table III.

The system shakedown lab tests will verify system function. The testing will be done in a laboratory using an electrically heated air blower. The accuracy of the compensation will be verified using an electrical analog of the thermocouple finite element array using first order low pass RC filters which correspond to the individual time constants of each node. These filters will then be substituted for the thermocouple and a random noise signal will be input to simulate the thermocouple signal levels.

The second test series will be conducted using a laboratory burner test rig. These tests will allow evaluation of the sensor and measurement system in an elevated temperature combustor environment.

The third test series will be performed by installing and evaluating on a non-interference basis in an experimental F100 engine test. The sensor will be installed in the turbine-inlet borescope plug. These tests will provide the final evaluation of the sensor at the required environmental conditions.

The program status as of the first of October 1982 was that Tasks 1 through 4 has been completed and Task 5 was in process. The sensor fabrication and software program were completed.

FUTURE WORK

Future work will complete the three test series described above and process the data taken. The test program is a stepwise, iterative approach which allows for necessary problem-solving and technical flexibility. At the present rate, data acquisition should be complete by the end of 1982 and final data reduction/analysis should be well underway.

SUMMARY

An approach to dynamic temperature measurement has been identified. The method uses two beadless junction Type B thermocouples to measure heat transfer coefficient in-situ. Heat conduction effects are accounted for using a finite element model of the thermocouple. Fabrication of thermocouples is complete. A test program which verifies measurement system function will be accomplished in the next few months.



DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

*Host Annual Contractor Workshop
NASA-LEWIS Research Center*

Prepared Under Contract NAS 3-23154 for
NASA-LEWIS Research Center

Presented by
Denny Elmore
Assistant Project Engineer

19 October 1982

GOVERNMENT PRODUCTS DIVISION
P.O. Box 2691
West Palm Beach, Florida 33402

GP 82-683

AV260062 822408 0029

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Objectives

To develop an advanced measuring system capable of measuring the rapidly varying gas temperature at the exit of an aircraft jet engine combustor during ground-based testing of hot section components

AV260064 822408 0029

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Overview

- Contract objectives and sensor guidelines
- Technical approach/program schedule
- Program status

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Table I sensor design and environment guidelines

Geometry: Annular combustor, $2 \text{ cm} < H < 8 \text{ cm}$

Temperature: $\bar{T} \sim 1400^\circ\text{K}$ (2060°F); $T \sim 500^\circ\text{K}$ (900°F)

Frequency response: 1 kHz

Pressure: $10 < P < 20 \text{ ATM}$

Flow: $\bar{v} \sim 150 \text{ m/s}$; $v \sim 50 \text{ m/s}$

Gas composition: Fuel (nominal jet A) and air

Sensor life: 5 hr minimum

Accuracy: Temperature uncertainty $\leq 5\%$ for $f \leq 200 \text{ Hz}$
Temperature uncertainty 10% for $200 \text{ Hz} < f < 1 \text{ kHz}$

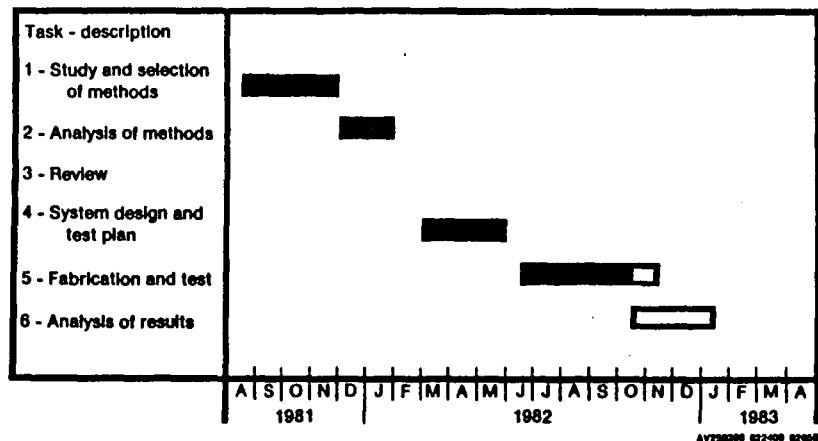
Spatial resolution: $D \leq 0.5 \text{ cm}$

Vibration 10g

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DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Technical approach/program schedule



79

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Table II concepts evaluated

Single wire thermocouple

- Passive
- Pulse-heated

Dual wire thermocouple

- Passive
- Pulse-heated

Resistance thermometer

- Wire
- Film

Ultrasonic thermometer

Fluidic resonator

Gas sampling

Johnson noise thermometer

Piezoelectric resonator

Vibrating wire

Radiation pyrometry

High speed photography

Coherent anti-stokes raman spectroscopy

AV200000 022-000 02000

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Task 1 - Study and selection of methods

Literature survey

- Concepts identified

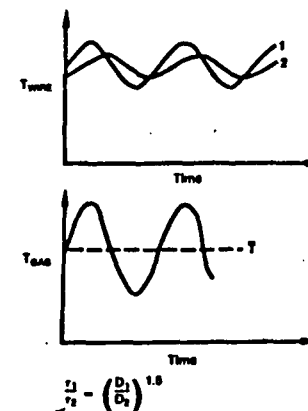
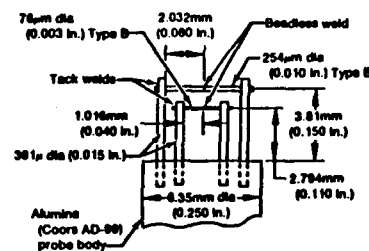
Evaluation/screening

- Selected - dual passive thermocouple
- single pulse-heated thermocouple

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DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Dual passive thermocouple concept

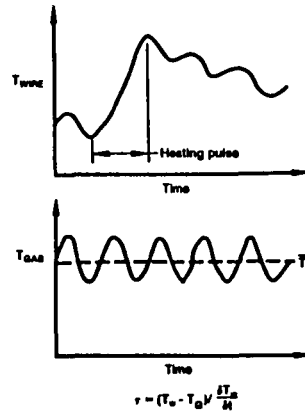
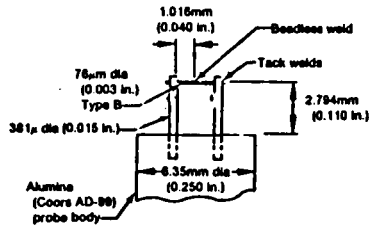


$$\frac{T_1}{T_2} = \left(\frac{D_1}{D_2} \right)^{1.0}$$

AV200000 023-000 02271

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Single pulse-heated thermocouple concept



AV250398 822-408 872288

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Task 2 - Analysis of methods and selection of the most promising method

- Structural analysis
- Thermal analysis
- Data acquisition/reduction analysis

AV250398 822-408 82428

80

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Task 2 analysis results

	Single-wire pulse-heated	Dual-wire passive
Structural	Acceptable: $d > 0.008\text{CM}(0.003\text{ in})$ $L/D < 20$	Acceptable: $d > 0.008\text{CM}(0.003\text{ in})$ $L/D < 20$
Thermal	Acceptable: end-conduction correction necessary (pulse approaches wire melting point)	Acceptable: end-conduction correction necessary
Data acquisition and reduction	Unacceptable: pulse-heating amplitude required is too large for required accuracy and conditions	Acceptable:

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DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Task 4 - System design and test plan formulation

Sensor design

- Mechanical
- Structural
- Thermal

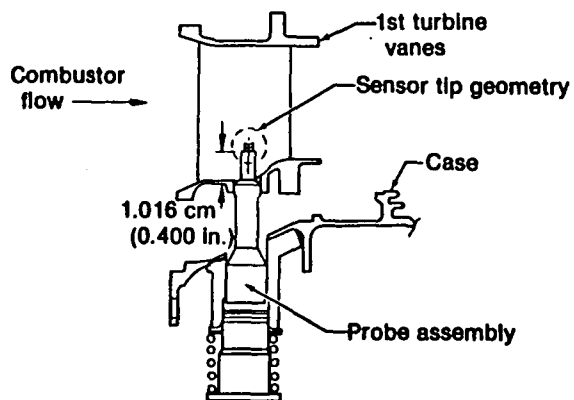
Data acquisition and data reduction system

Test plan

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DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

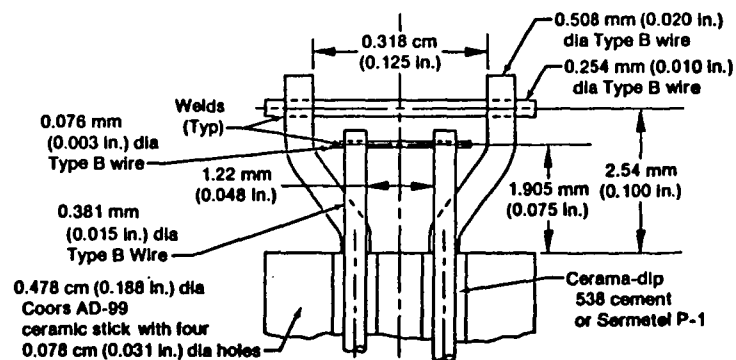
Probe designed for F100 turbine installation



AV790204 022-000 00000

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Sensor tip geometry



AV790205 022-000 00000

81

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

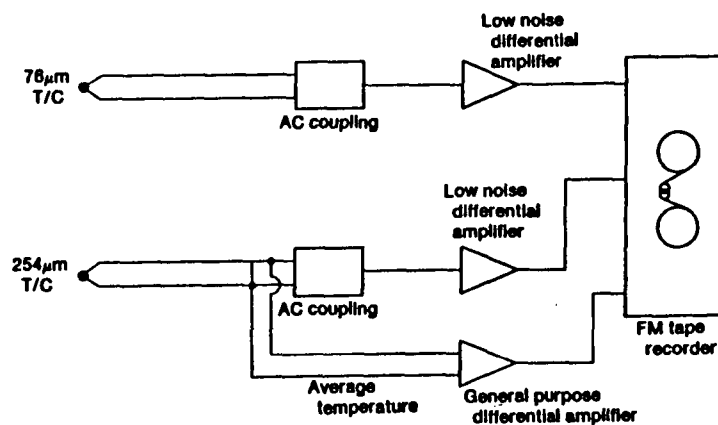
Completed sensor



AV790206 022-000 00000

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

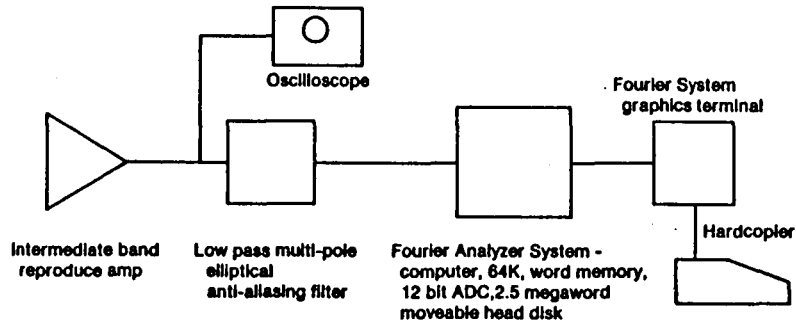
Data acquisition system



AV790207 022-000 00000

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Data playback system



AV250300 822-000 82500

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Table III test plan summary

Criteria	System shakedown and compensation verification	Lab burner rig	Full-scale engine tests
Temperature (1900°K peak; ± 500°K fluctuations)			X
Frequency response (1000Hz compensated)	X	X	X
Pressure (10 to 20 atm)			X
Flow (150m/s with ± 50m/s fluctuations)			X
Gas composition (products of combustion and air)		X	X
Sensor lifetime (5 hr minimum)		X	X
Accuracy	X		
Vibration (10g loading up to 500Hz)			X

AV250400 822-000 82500

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Compensation method

- Compute theoretical response (76 μ m and 254 μ m T/C's) vs heat transfer coefficient (finite element conduction effects included) over frequency range
- Measure (data) response of 76 μ m and 254 μ m T/C's over frequency range (using FFT techniques)
- Determine actual heat transfer coefficient from computed and measured response
- Generate theoretical response of 76 μ m T/C for actual heat transfer coefficient for frequency range
- Compensate 76 μ m T/C data in frequency domain
- Inverse fourier transform to time domain

AV250300 822-000 82500

DYNAMIC GAS TEMPERATURE MEASUREMENT SYSTEM

Program status

Completed Tasks 1 through 4

Task 5 in process

- Hardware fabrication
- Software programming
- Testing

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High Temperature Static Strain
Sensor Development Program

NAS3-23169

United Technologies Research Center
Charles Hulse, Frank Lemkey, Richard Bailey, Howard Grant

The purpose of this program is to develop resistance strain gages useful for static strain measurements on nickel or cobalt superalloy parts inside a gas turbine engine on a test stand. Measurements of this type are of great importance in meeting the goals of the Host Program because, without reliable knowledge of the stresses and strains which exist in specific components, it will be difficult to fully appreciate where improvements in design and materials can be implemented. The first year of effort has consisted of a strain gage alloy development program which is to be followed by an optional second year of work to investigate complete strain gage systems which will use the best of the alloys developed together with other system improvements.

The specific goal for the complete system is to make measurements to $2,000\mu\epsilon$ with error of only $\pm 10\%$ over a 50 hour period. In addition to simple survival and stability, attaining a low thermal coefficient to resistivity, of order 100 ppm/K or less, is also a major goal. This need results from the presently unavoidable uncertainty in measurements of the exact temperatures in the turbine. The size and thickness requirements to avoid aerodynamic effects suggests the use of the sputtering technique as the best system fabrication approach. The first task of the program was to select candidate alloys or alloy systems using a search of the literature and the available metallurgical theory. Alloy candidates were evaluated and compared using a grading system consisting of the product of the following factors with their total weight potential given in parentheses: Repeatability (20), Oxidation (18), Resistivity (16), Thermal coefficient of resistivity (14), Elastic range (12), Differential thermal expansion (10) and Miscellaneous judgments (10). After discussion and review with NASA, the following alloys, indicated in weight percent, were deemed the best candidates: 45Pt-45Pd-10Mo, 60Pd-30Ag-10Mo, FeCrAl type, 84.4Ni-14.2Cr-1.45Si, Pt-10W and Pd-30 Cr.

In addition, fabrication and test plans were also developed and given NASA approval. The initial intention was to fabricate alloy samples by sputtering on thin alumina substrate plates. Because of the possibilities of delays and that it would probably be difficult to screen many alloys within the time constraints of the program, the alternative approach of drop casting directly into wire form was adopted. In this process, small buttons of the alloy were repeatedly melted on a cold copper or tungsten hearth using tungsten electrodes and then cast inside

tubes of SiO₂ or Al₂O₃ by suddenly changing the pressure of the argon cover gas. Partly because of the simplicity of this approach, we were able to make evaluations on a total of 37 different samples of 29 different compositions which can be compared with a maximum of 12 alloy samples required by the work statement.

A specially constructed thermal cycling apparatus was developed to make resistivity measurements by the use of a split metal tube heater which could be cycled or held at a constant temperature under program control. The test sample was positioned axially in the center of this tube with Platinum leads for voltage measurements and five thermocouple wires attached in the constant temperature section by spot welding. Part of the way through the test program, a H.P. 9826A computer and a voltmeter capable of measurement down 100 nanovolts were added to the apparatus.

The first part of the experimental program consisted of homogenizing the cast samples via heat treatments followed by metallographic and/or SEM investigations of the microstructures produced to verify that they were essentially phase pure. Measurements of electrical resistivity, the continuous measurement of the changes in resistivity for samples cycled from R.T. to 1250K at 10,50 and 250K/min. and drift of resistance with time at 1250K were then made for each alloy. This series of tests was iterated for at least a total of six different alloy compositions of each alloy type unless the data obtained suggested that experiments with that type be discontinued. Significant problems of immiscibility in Pd-Ag-Mo resulted in that system being dropped. Work on Nicrosil (84.4Ni-14.2Cr-1.4Si) was also discontinued because of a high thermal rate sensitivity and apparent metallurgical instability at about 900K. The alloy compositions judged to be the best using the evaluation process described above were: 48Pt-40Pd-12Mo, 77Pt-6.5W-6.5Re, 77.5Fe-11.9Al-10.6Cr and 83Pd-13Cr-4Co (in weight percent).

The second part of the test plan consisted of evaluations of the oxidation resistances, chemical compositions, thermal expansions, melting points, stress-strain and creep behavior of the best alloys of each type discovered in part one. The oxidation testing consisted of periodic measurements of weight change during a 50 hour exposure to air at 1250K. As was predicted, the Pd-13Cr-4Co and FeCrAl modification #3 gained in weight while the Pt-W and Pt-Pd-Mo alloys lost weight. The total weight gain of Mod #3, approximately 0.3 mg/cm², was slightly greater than that for Kanthal A-1 which is an alloy known for its good resistance to oxidation. All the other FeCrAl alloys lost weight because the coatings were nonprotective and spalled off.

The final part of the test program consisted of an attempt to demonstrate that sputtered films, after being fully stabilized and annealed, had essentially the same electrical properties as the cast material of identical composition. Sputtered samples, approx. 1.3 microns thick, were prepared of FeCrAl Mod #3 and Kanthal A-1 for comparison. Attempts to test the Kanthal A-1 sample resulted in erratic results and finally failure of the gage to conduct. The problem was traced to oxidation caused by small amounts of oxygen in the argon purge gas. The oxide films were not completely continuous. Those results suggested that much thicker sputtered films are required.

In summary, equipment and techniques were developed suitable for iterative studies of a variety of compositions. Many compositions were examined and some significantly improved alloys were identified. Additional iterative alloy development work is desirable, especially with regard to the electrical and environmental stability of (coated) sensor gages.

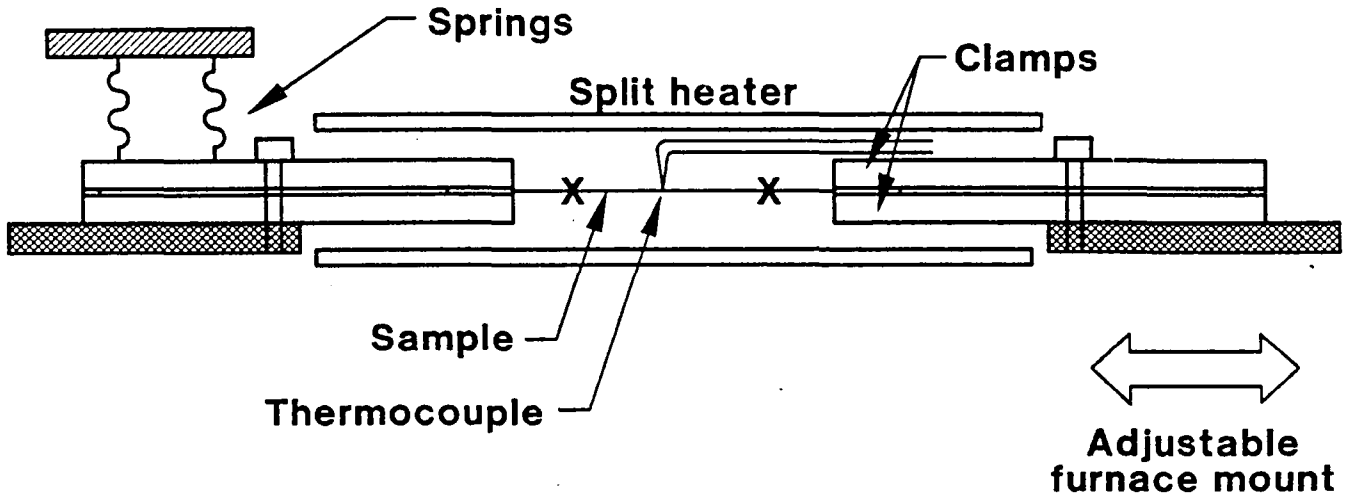
OBJECTIVE: DEVELOP IMPROVED STATIC STRAIN GAGE ALLOYS

- Useful to 1250K
- 50 hrs. life
- 2,000 $\mu\epsilon$, \pm 10%
- Jet engine environment

TEST PLAN

- Part I - Measure ρ , α & drift at 1250K iterate
- Part II - Determine additional properties of selected alloys
- Part III - Demonstrate transfer of properties to sputtered thin films

HIGH-SPEED THERMAL CYCLE/ RESISTIVITY MEASUREMENT APPARATUS



RANKING METHODOLOGY

<u>Factors</u>	<u>Scale</u>
Repeatability, R	1 - 20
Oxidation, O	1 - 18
ρ	1 - 16
α	1 - 14
ϵ_{el}	1 - 12
ΔCTE	1 - 10
Judgment, J	1 - 10

Σ upper scale factors = 100

$$\text{Ranking} = (R) \cdot (O) \cdot (\rho) \cdot (\alpha) \cdot (\epsilon_{el}) \cdot (\Delta CTE) \cdot (J)$$

RANKING OF ALLOY COMPOSITIONS

<u>Alloy type</u>	<u>Rank</u>	
	<u>3/10/82</u>	<u>8/25/82</u>
FeCrAl, A-1 Fe-22Cr-5.5Al-0.5Co	9.2	
FeCrAl, Mod. #3 Fe-11.9Al-10.6Cr		83.8 *
Nicrosil Ni-14.2Cr-1.4Si	110.9	16.1
45Pt-45Pd-10Mo	148.3	

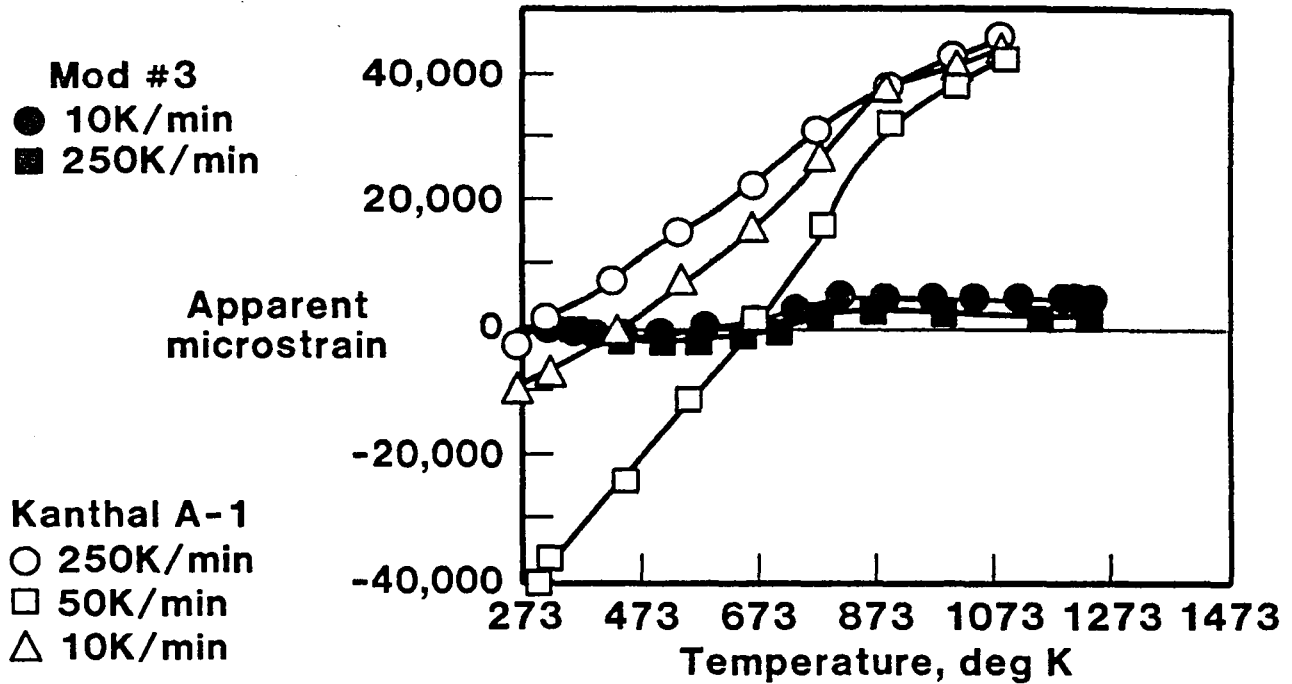
* = recommended

RANKING OF ALLOY COMPOSITIONS (Cont.)

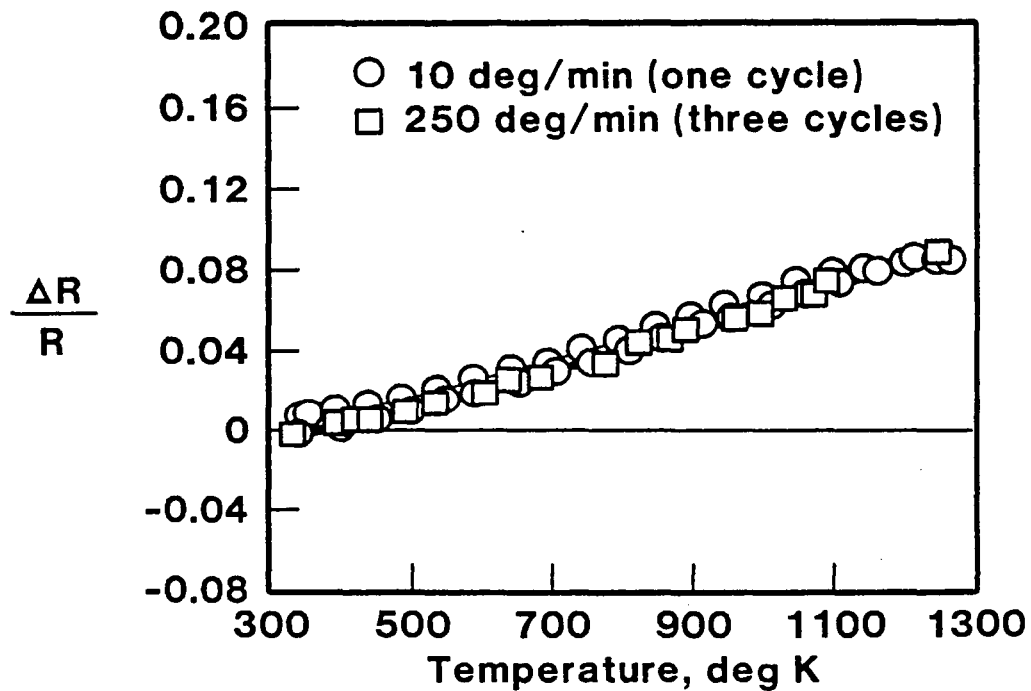
<u>Alloy type</u>	<u>Rank</u>	
	<u>3/10/82</u>	<u>8/25/82</u>
48Pt-40Pd-12Mo Mod. #4		42.6
Pd-20Cr	94.1	
Pd-13Cr-4Co		44.2
Pt-10W	57.6	
Pt-6.5W-6.5Re		145.2 *
Pd-Ag-Mo Immissibility - Alloy dropped		

* = recommended

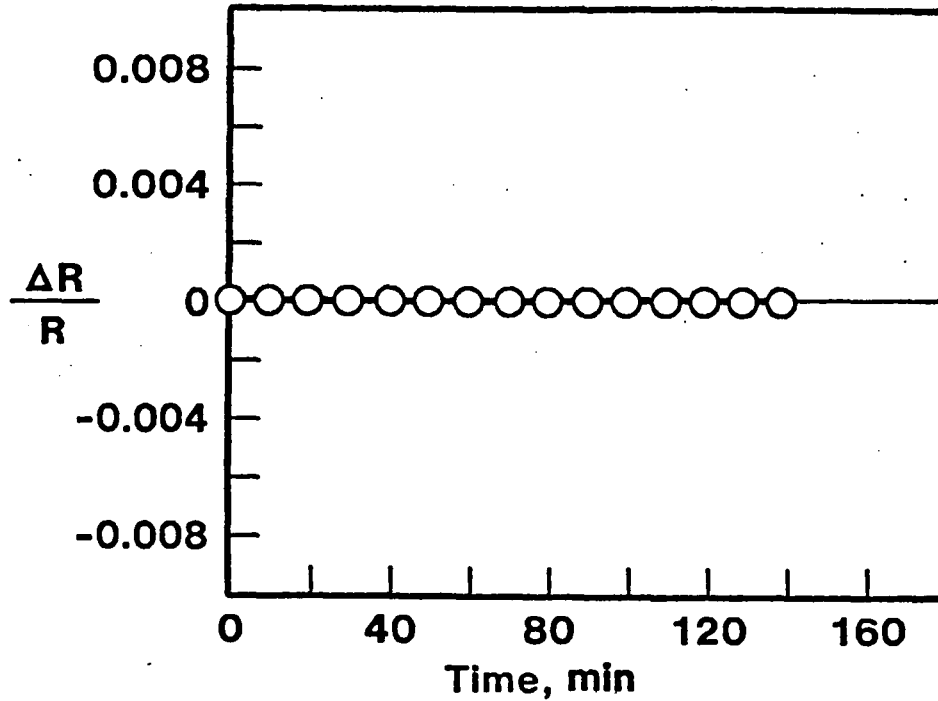
KANTHAL A-1 AND FeCrAl MOD #3 AFTER 2 HRS. AT 1153°K



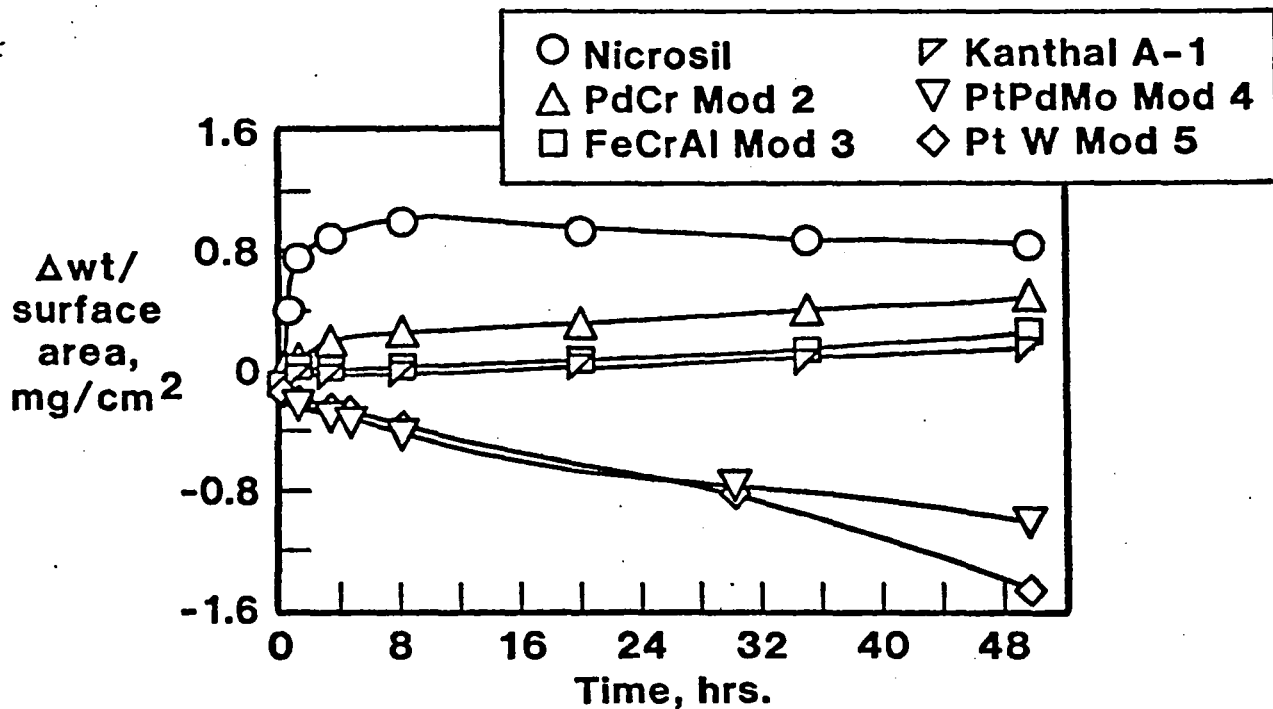
ELECTRICAL RESISTANCE FOR Pt-6.5W - 6.5 Re vs TEMPERATURE



RESISTANCE DRIFT OF FeCrAl MOD #3 AT 1250°K



WEIGHT CHANGE OF STRAIN GAGE ALLOYS EXPOSED TO AIR AT 1250°K



RA420TX.016

MAJOR RESULTS

- Testing facilities developed
- Simple fabrication technique established
- Numerous compositions examined
- Pt-W-Re and FeCrAl alloy identified
- Oxidation & strain range limitations observed
- Further optimization advisable

STATUS

- Improved compositions discovered
- Optimization desirable

LASER SPECKLE TECHNIQUE FOR BURNER LINER STRAIN MEASUREMENTS

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East Hartford, Connecticut 06108

ABSTRACT

Thermal and mechanical strains have been measured on samples of a common material used in jet engine burner liners, which were heated from room temperature to 870°C and cooled back to 220°C, in a laboratory furnace. The physical geometry of the sample surface was recorded at selected temperatures by means of a set of twelve single-exposure specklegrams. Sequential pairs of specklegrams were compared in a heterodyne interferometer which allowed high-precision measurement of differential displacements. Good speckle correlation was observed between the first and last specklegrams also, which showed the durability of the surface microstructure, and permitted a check on accumulated errors. Agreement with calculated thermal expansion was to within a few hundred microstrain over a range of fourteen thousand.

BACKGROUND

A number of problems confront the use of double-exposure speckle photography for strain measurement. First and foremost, there are limits to the minimum and maximum speckle displacements that can be determined from a specklegram. These lie at approximately one and fifty speckle diameters, where a speckle diameter may be approximated by the Airy disk associated with the imaging system. These limits impose severe restrictions in the bulk displacements allowed for an object under test. This difficulty has been dealt with by using separate photographic plates for each of the deformation states of the object. When separate specklegrams are compared, bulk displacements may be eliminated by translating and rotating one with respect to the other.

A second problem lies in the accuracy required when optical strain measurement is forced to compete with electrical strain gaging. For example, if strain is to be measured on gage length of 1 mm to within 10 microstrain, the end-point displacements must be measured to within 10 nanometers. This level of performance is routine for electrical gages, but it is quite taxing for optical systems. Assume, for example, that relative speckle displacements can be measured to 1.0 percent of a speckle diameter. The f/number of the imaging system must be reduced to about 1.3 before the speckles are sufficiently small to permit a 10 nanometer displacement measurement. Many troubles attend the use of such low f/numbers

in speckle photography; for example, lens aberrations become severe and lead to different displacements for different speckle sizes and shapes, and depth of focus becomes very short.

The photogrammetric comparison of specklegrams by heterodyne interferometry was introduced recently¹ to deal with these problems. Separate specklegrams were placed in the two paths of an interferometer in such a way that their far-field diffraction patterns (halos) could be combined and made to interfere. Simultaneous translation of the two specklegrams (via a common translation stage) caused the number of fringes across the output plane to increase or decrease at a rate proportional to any strain present between the two specklegrams. The introduction of a doppler shift between the two beams of the interferometer caused the fringe pattern to sweep across the output plane at a constant velocity. A pair of detectors in the output plane were used to generate electrical signals whose phase difference was proportional to the number of fringes spanned by the detectors. Connecting these signals to a phase meter allowed measurement of changes in phase to 0.1° . This corresponded to a measurement of relative speckle displacements to one thirty-six-hundredth of a speckle diameter. This increased precision in the measurement of speckle displacements made practical use of the optical systems with reasonable f/numbers (e.g., f/10) for high resolution strain measurements.

Two problems were identified with the initial implementation of heterodyne speckle photogrammetry: a random pattern of apparent strain that was highly correlated to the stage position, and an apparent strain associated with thermal drift of the equipment. Both problems have been dealt with by subsequent work.² Data averaging was used to eliminate the effects of thermal drift and other sources of random error. The stage dependent error was eliminated by reconfiguration of the interferometer, so that the plates could lie in the same plane, and by refined adjustment procedures.

With the major problems of heterodyne speckle photogrammetry under control, thought was given to its application to high temperature strain measurement. It was reasoned that speckle photographs could be made of a test sample at various times during a test program. Once developed and fixed, these photographs would serve as a permanent record of the deformation states of the test sample, and they could be processed later to yield strain distributions. Unlike strain gages, which provide continuous strain data at discrete geometrical locations, the speckle photographs would provide contiguous strain data at discrete time intervals. There were considered to be sufficient cases of thermomechanically generated strains, where the geometrical strain pattern was of more importance than its detailed temporal history, to warrant exploration of the technique. This decision was also encouraged by the fact that the technique should be practicable at very high temperatures.

This presentation covers the results to date of a program carried out to apply heterodyne speckle photogrammetry to high temperature strain measurement. It will begin with a consideration of the optical, mechanical, and electronic problems concerned with such a test. This will be followed by a description of the test procedures and the results obtained.

It will be shown that heterodyne speckle photogrammetry has significant potential application to strain measurement of objects at high temperatures. Because it is necessary to subtract thermal expansions, however, it is necessary to measure accurately the temperature of the sample surface under consideration. Because it is capable of generating strain distributions, attention must also be paid to temperature distributions on the sample. If varying temperature gradients exist in gas through which speckle photographs are recorded, then their effect must be removed by independent measurement and computation. Temporal resolution of changing strain patterns will be compromised by the need to record separate photographs of the sample surface at discrete time intervals. This may be compensated, however, by the extraordinary amount of spatial strain information available. In 1983, it is planned to apply this technique to the measurement of strain on the surface of a JT8D burner liner in a test stand.

REFERENCES

1. Smith, G. B., and K. A. Stetson: "Heterodyne Readout of Specklegram Halo Interference Fringes." Appl. Opt., Vol. 19, pp. 3031-3033, 1980.
2. Stetson, K. A.: "Strain Measurement on Rough Surfaces by Optical Heterodyning." Proc. 1981 Spring Meeting, SESA, June 1-4, 1982, Dearborn, Mich., pp. 236-241.

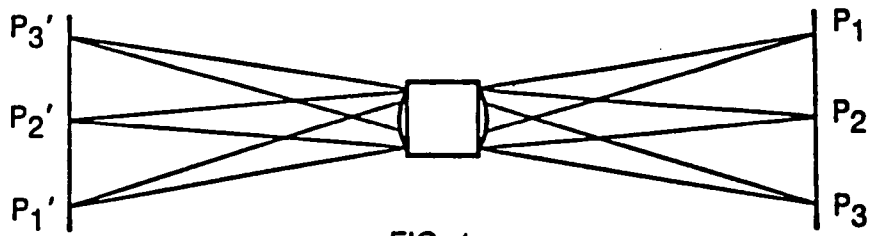


FIG. 1-a

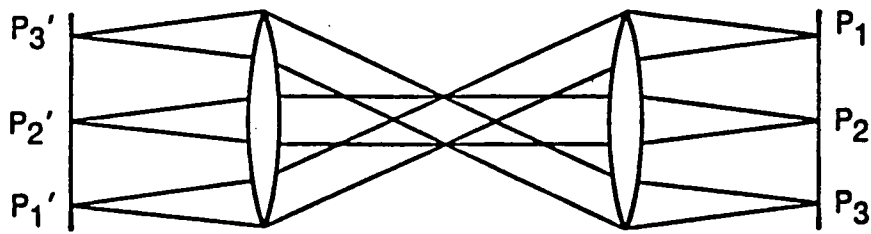


FIG. 1-b

CONVENTIONAL VERSUS TELECENTRIC LENS SYSTEM

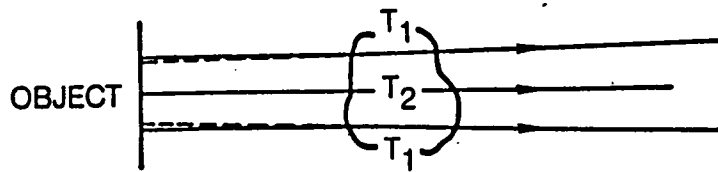


FIG. 2-a

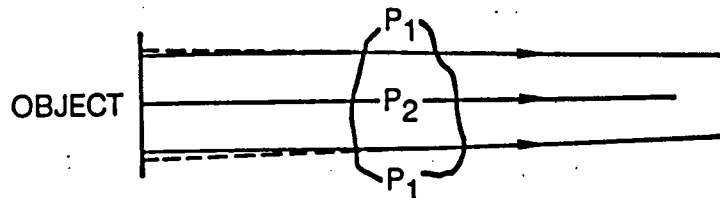


FIG. 2-b

LENSLETS CREATED BY TEMPERATURE AND PRESSURE INHOMOGENEITIES

- a) Thermal Lenslet $T_2 > T_1$.
- b) Pressure Lenslet $P_2 > P_1$.

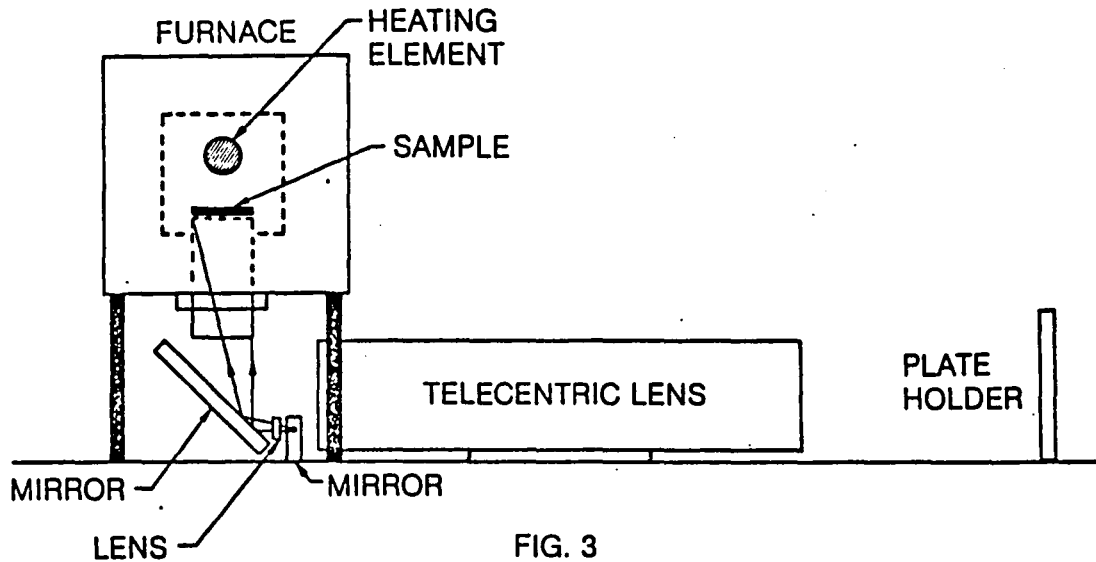


FIG. 3

LABORATORY FURNACE AND SPECKLEGRAM RECORDING SYSTEM

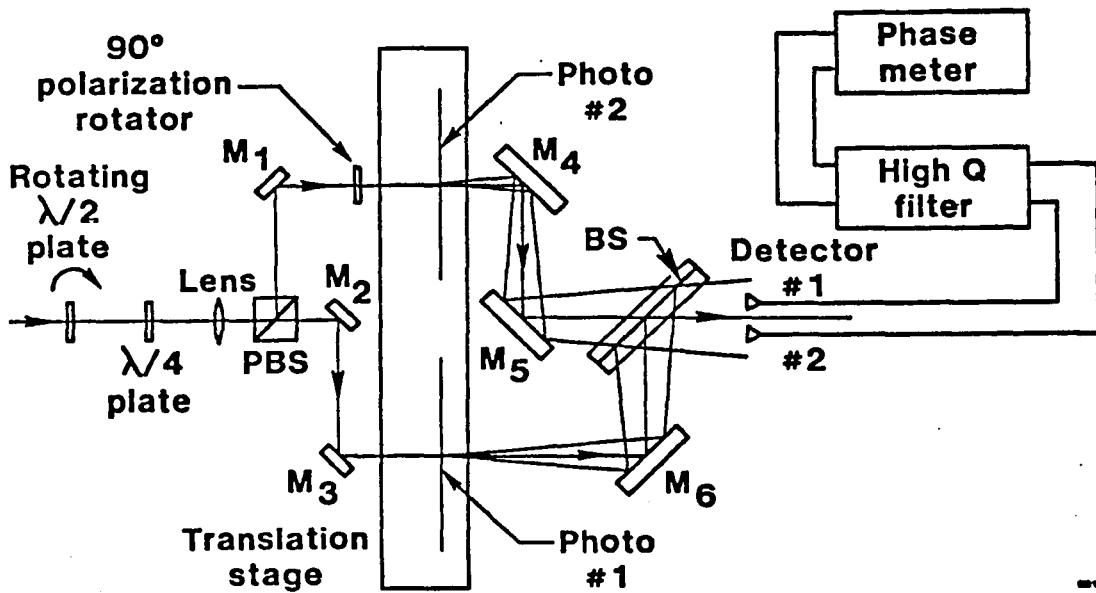


FIG. 4

INTERFEROMETRIC COMPARATOR FOR HETERODYNE READOUT OF SPECKLEGRAM HALOS

8-7-64-3

8-107L-000

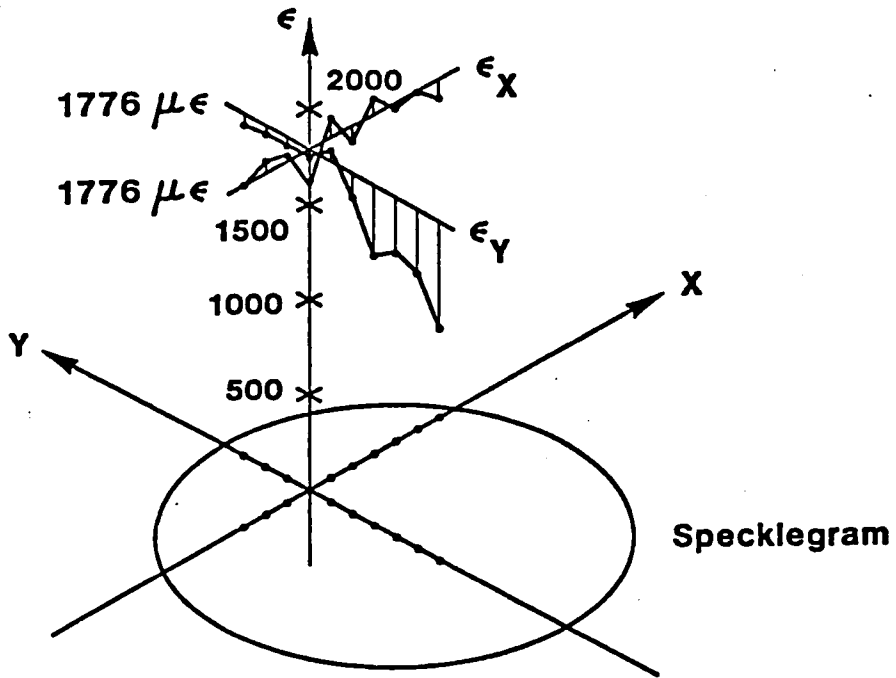


FIG. 5

ISOMETRIC PLOT OF A THERMAL STRAIN DISTRIBUTION

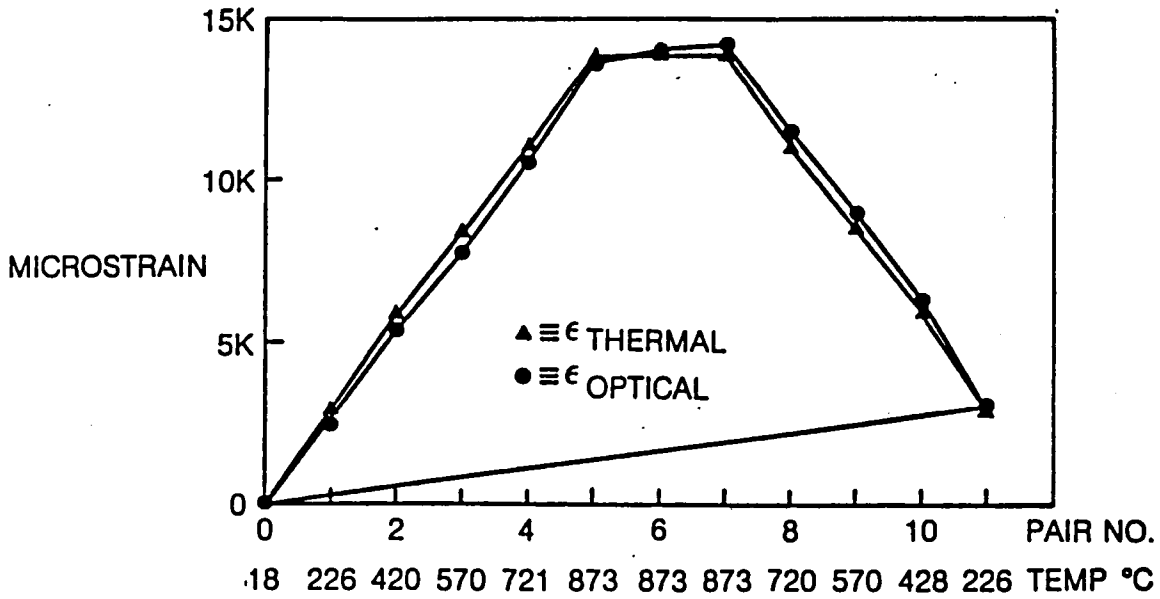


FIG. 6

STRAIN HISTORY OF AN UNCONSTRAINED SAMPLE OF BURNER LINER MATERIAL

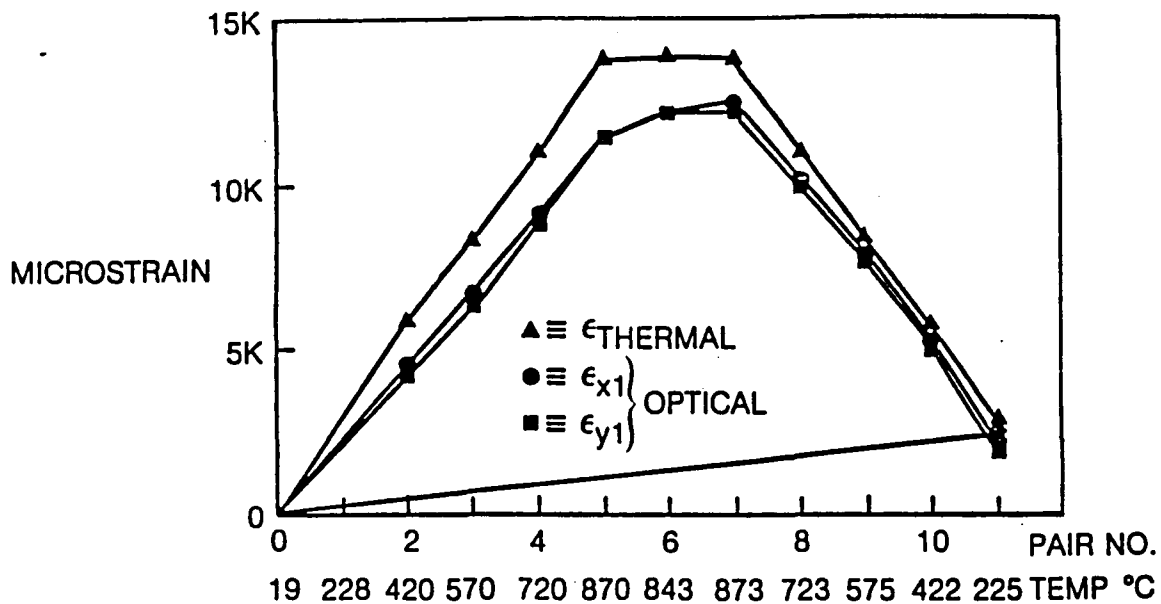


Fig. 7

STRAIN HISTORY OF A CONSTRAINED SAMPLE OF A BURNER LINER MATERIAL

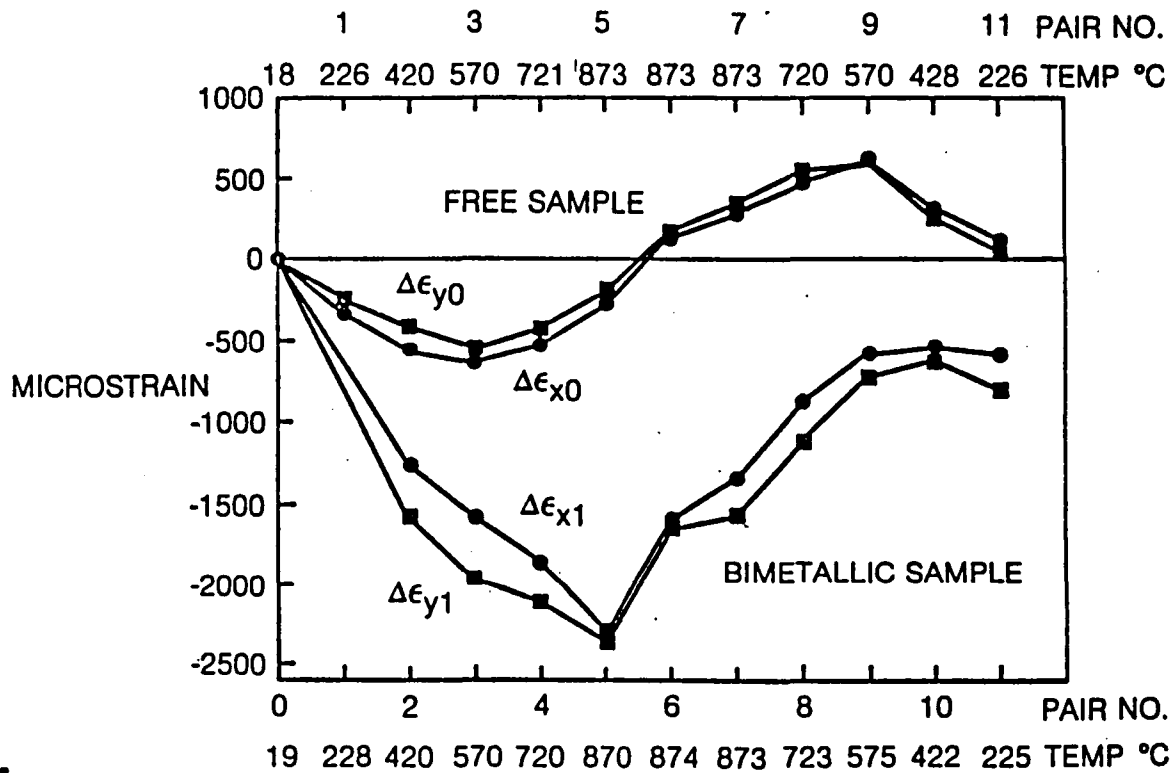


FIG. 8

MECHANICAL STRAIN OF THE CONSTRAINED AND UNCONSTRAINED SAMPLES

"Heat Flux Sensors for Burner Liners and Turbine Blades and Vanes"

W. G. Alwang
Pratt & Whitney Aircraft Group
Commercial Engineering

The major challenge to designers of advanced gas turbine engines continues to be: how does one make significant improvements in fuel efficiency without compromising the durability of hot section hardware? Although great strides are being made in efficiency improvement via better gas path clearance control and less lossy aerodynamic design, the principal approach to efficiency improvement is determined by the fundamental thermodynamics of the gas turbine cycle, that is, increased cycle temperature and pressure. Thus, hot section hardware durability must be maintained in an increasingly hostile environment.

The NASA HOST program is devoted to the development of design methodology which will allow these problems to be overcome. Early in the evolution of the program it was recognized that several formidable experimental difficulties existed with respect to obtaining suitable data in support of the desired analytical techniques for life prediction. These were principally difficulties in the measurement of critical hot section parameters such as gas temperature distribution, metal temperature, high temperature static and dynamic strain, and heat flux. Consequently, the program addressed development of measurements technology very early.

The particular programs which I will discuss cover the measurement of heat flux. This work is being carried out under two contracts as follows;

- 1) "Advanced High Temperature Heat Flux Sensor Development," NAS3-22133.
- 2) "Turbine Blade and Vane Heat Flux Sensor Development and Experiment," NAS3-23529.

The first, which was a precursor to the HOST program, covers the development of total heat flux sensors for burner liners and also the demonstration of total and radiant heat flux sensors in a combustor test. The second covers total heat flux sensors for turbine blades and vanes.

At the present time very little data is available on heat flux in realistic gas turbine environments. At lower combustor temperatures convective heat transfer dominates but with increases in pressure and temperature as well as use of more luminous broad spec fuels radiant heat transfer must also be accounted for.

A thorough review of potential approaches was conducted including both transient and steady state measurements. Measurement of total heat flux was emphasized rather than measurement of hot side or cold side heat transfer coefficients. Consequently, configurations were sought which produce minimum disturbance to the heat flux which would be present without the sensor in place. Three basic types of devices were selected for further study; the embedded thermocouple sensor, the laminated sensor and the Gardon gauge. These sensors were analytically modelled and detailed heat transfer calculations carried out to provide data for the design of non-perturbing sensor installations. This work also included an investigation of hot side boundary layer effects using the program STAN5. The experimental development program consisted of the fabrication of sensors which were then subjected to a series of thermal soak tests, thermal cycle tests and pre and post test calibrations. Calibration

facilities were constructed using electrically heated filaments and quartz lamp banks. A major requirement of the program was the achievement of repeatable calibration with 5% uncertainty or less at realistic gas turbine temperature conditions. The results of this work and the sensor designs developed are contained in the contract final report which will be published shortly.

A follow on to this program was recently begun which will be completed in mid '83. In this work the total heat flux sensors developed in part 1 of the program will be run in a high pressure combustor environment as part of the NASA Broad Spec Fuels Combustor Technology Program (NAS3-23269). In addition, two types of radiant heat flux sensors will be run, a porous plug radiometer and an optical radiometer. The two radiometers are significantly different in principle and in their spectral and spatial sensitivity. On completion of this contract, HOST participants will have sensor design information and performance data available to provide the basis for total heat flux measurements on combustor liners and also performance data on two types of radiant heat flux sensors.

The turbine blade and vane heat flux sensor program, NAS3-23529, was recently begun and is planned to be completed in two years. It has two basic objectives. First, building on experience in the above programs, it will develop total heat flux sensor concepts which apply to the considerably more difficult geometry of cooled vanes and blades. In addition to wire thermocouple based approaches to the sensor design, sputtered thin film thermocouples, fiber optic and other alternative sensor designs will be considered. The most promising sensor designs will be run in a demonstration test employing a well known geometry in which the output of the sensors under development will be compared with alternative measurements of heat flux. This program will result in sensor design and performance data suitable for use by other HOST participants in blade and vane heat flux measurement.

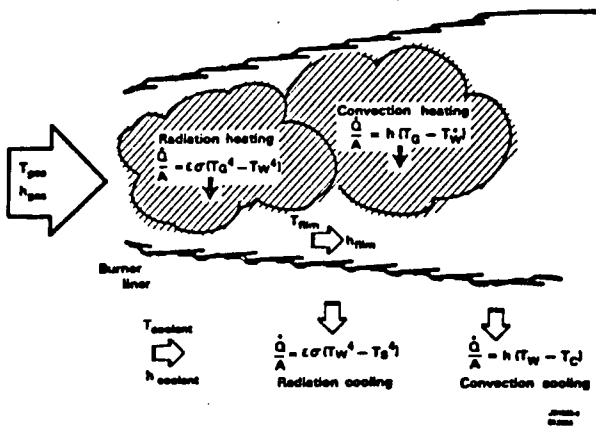
"HEAT FLUX SENSORS FOR BURNER LINERS AND TURBINE BLADES AND VANES"

W.G. Alwang
P&WA
Commercial Engineering

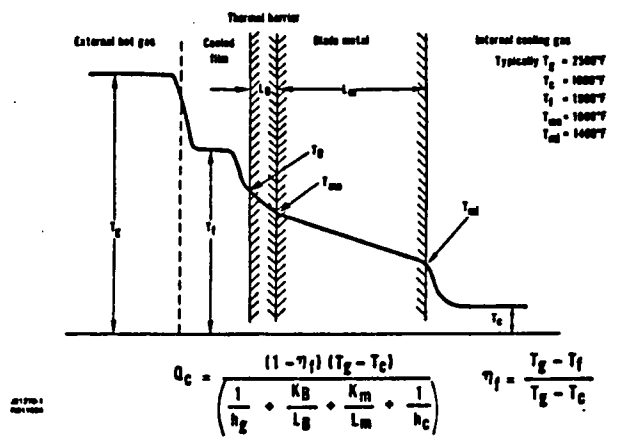
CRITICAL PARAMETERS IN TURBINE COOLING

- Metal temperature
- Heat flux:
convective and radiative
- Gas temperature
- Strain

BURNER LINER HEAT TRANSFER



HEAT FLUX IN A COOLED AIRFOIL



HOST HEAT FLUX SENSOR PROGRAMS

- "Advanced high temperature heat flux sensor development" (NAS3-22133)

Phase 1 — Develop sensor for burner liner good to 1 megawatt/m²
(Final report October 1982)

Phase 2 — Demonstrate total and radiative heat flux measurement
in NASA broad spec fuel program (Test schedule May 1983)

- "Turbine blade and vane heat flux sensor development and experiment" (NAS3-23529)

Phase 1 — develop sensor for cooled blades and vanes
(complete August 1983)

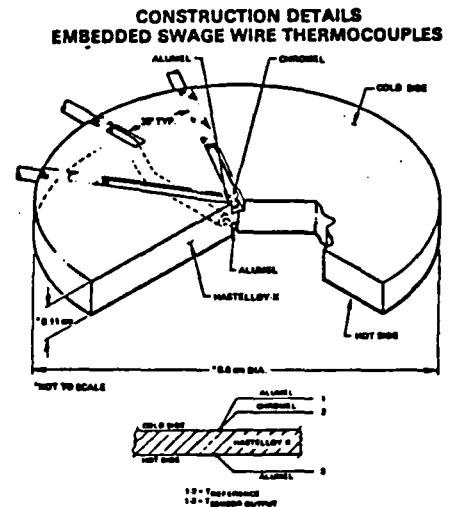
Phase 2 — demonstrate in experiment at turbine conditions
(complete August 1984)

GENERAL METHODS OF MEASURING TOTAL HEAT LOADS

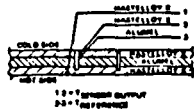
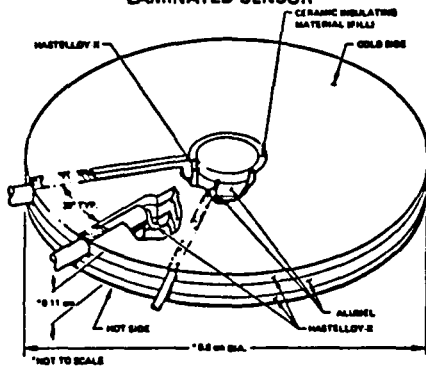
- Measurement of differential temperature across a thermal barrier caused by the heat flow
- Measurement of the temperature history of a known mass of material
- Measurement of input power required to maintain a constant surface temperature

THREE APPROACHES SELECTED FOR BURNER LINERS (ALL STEADY STATE)

- Embedded thermocouple sensor
- Laminated sensor
- Gardon gauge

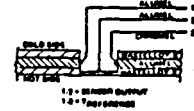
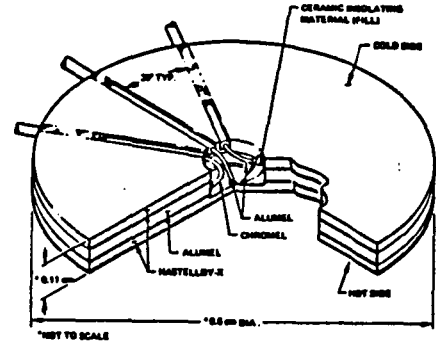


**CONSTRUCTION DETAILS
LAMINATED SENSOR**



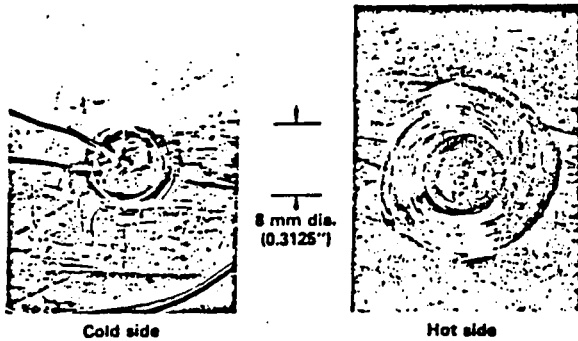
2725-10
2825-10

**CONSTRUCTION DETAILS
GARDON GAUGE**



2725-10
2825-10

**HIGH TEMPERATURE STEADY STATE
HEAT FLUX SENSOR FOR
COMBUSTOR LINER APPLICATIONS**

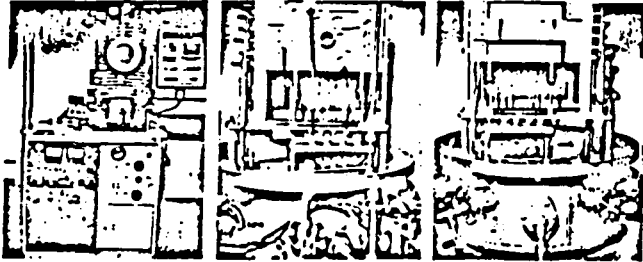


**DEVELOPMENT TESTING
OF SENSORS**

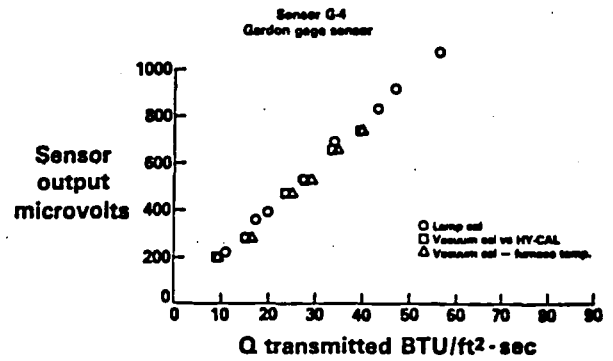
- Thermal soak tests
- Thermal cycle tests
- Calibrations
 - Absolute
 - Comparative

2725-10
2825-10

THREE FILAMENT RADIATIVE CALIBRATION FACILITY



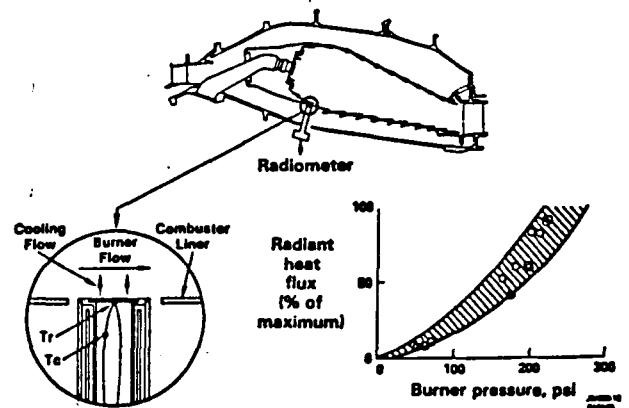
COMPARISON OF ABSOLUTE AND COMPARATIVE CALIBRATIONS

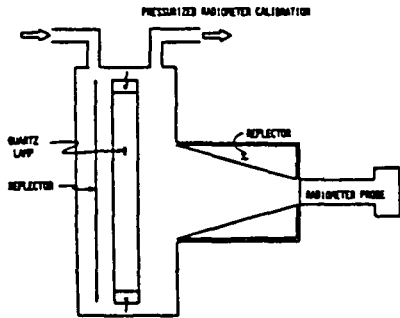
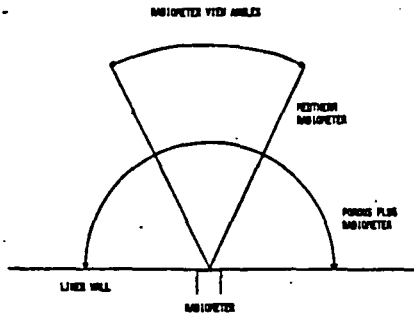


GENERAL METHODS OF MEASURING RADIATIVE HEAT TRANSFER

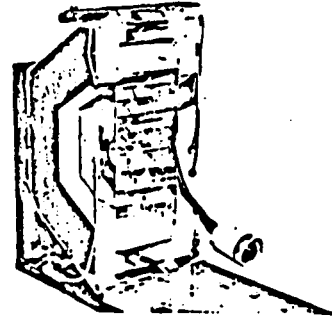
- Radiometers with windows
- Radiometers with gas purge
- Porous plug radiometers

POROUS PLUG RADIOMETER

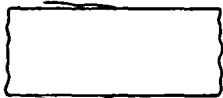




QUARTZ LAMP RADIATIVE CALIBRATION FACILITY



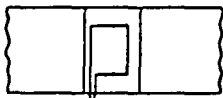
ONE DIMENSIONAL STEADY STATE SENSORS



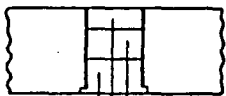
Surface thermocouples



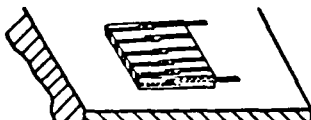
One dimensional sensor



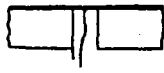
Differential one dimensional sensor



Laminated sensor

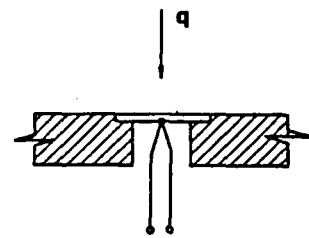


Thin thermopile on hot side surface



Gardon gauge

THIN FOIL TRANSIENT HEAT FLUX SENSOR



Thermocouple

HOT SECTION LASER ANEMOMETRY

by

William C. Nieberding
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

The Lewis Research Center is sponsoring an in-house and grant program to develop a laser anemometry system for hot section applications. The goal of this program is to be able to map the flow profiles through the vanes, between the vanes and blades, and between the rotating blades of a turbine. We are specifically aiming at developing a system for the Lewis High Pressure Turbine (HPT).

In-house work on laser anemometry at Lewis has been under way since about 1973. All the work has been in cold axial flow cascades and in single-stage axial flow compressor facilities. Under these conditions we have found that the fringe-type, single axis system has worked well for axial and circumferential flow component measurements. The radial component has been quite a bit more difficult to measure since we never have more than a minimum size window to access the rig. In addition, we never have the option of putting the optical axis more than a few degrees from the radial direction of the machine to measure the radial component because the view is always blocked by the blades or vanes. This led us to the development of a laser anemometer system which measures the Doppler shift directly along the optical axis. This is particularly difficult because the radial component is not only a very small fraction of the perpendicular components but is also a very small absolute velocity which causes very little Doppler shift. A paper has just been published on this topic.¹

All the problems we have had so far are expected to be much worse when we face the hot, high pressure flows in the HOST program. The velocity profiles which will be measured will include all three components--axial, circumferential, and radial. Our experience with the radial component in cascades warns us that this will be the most difficult component to obtain under the limited optical access conditions under which we are forced to work. We plan to measure the mean flows and angles, the turbulence parameters, the Reynolds stresses, and the power spectra.

Because running the HPT is very expensive, we are going to develop the system as much as possible in lower cost rigs and move up in phases until we finally get to HPT. We just can't afford to be doing much hardware or software debugging while HPT is running.

The first step, which is currently getting under way, is to solve as many problems as possible in a small bench top combustor facility. This is an open jet burner operating at one atmosphere. Here we will investigate various seeding materials and techniques. Windows with various thicknesses, geometries, temperature gradients, and stresses will be tried to determine whether

we can make the laser beams behave properly under these conditions of nonuniform refractive index. Here also we will try various signal processing techniques to extract the signal from the background noise radiation and the reflections from nearby surfaces. A computer is being installed here which is identical to the one we would put in HPT so that all of the interfacing, control, and data gathering, handling, and processing software will be as complete as possible before moving to the more expensive facilities. We will even simulate the rotation of the stage here so that the synchronization software and hardware that we developed for compressor stages can be incorporated into the system.

The next step up will be to put an almost identical system into the Lewis Warm Turbine Facility. Here we can get all of the conditions of HPT except the pressure and temperature. All the desired flow mapping can be done here. This alone would be very valuable data even if we never made it to HPT. In the meantime, the system that was installed on the open jet bench top rig will remain operational so that continued development can be done there. Problems encountered on the Warm Turbine also can be attacked on this little rig.

As can be seen from the schedule, the timetable on this program calls for fixing the design for the Warm Turbine Facility by fall of 1983 so that we can be running in that facility throughout 1985 and then in HPT in 1986. So far we are doing pretty well on this schedule and have high hopes of staying on it.

A continuing problem which plagues laser anemometry systems used in high operating cost facilities is getting a sufficient data rate so that the flow maps can be obtained without a phenomenal amount of running. The way to get the data rate up is to carefully optimize seeding, beam geometries, signal processing, and data reduction. These problems are being addressed in a grant that we are funding at Case-Western Reserve University. The next speaker, Professor Robert V. Edwards, will present some of the results of his study.

¹Seasholtz, Richard G. and Goldman, Louis J., "Laser Anemometer Using a Fabry-Perot Interferometer for Measuring Mean Velocity and Turbulence Intensity Along the Optical Axis in Turbomachinery," NASA TM 82841

HOST LASER ANEMOMETRY

GOAL:

- CHARACTERIZE VELOCITY FIELD IN HIGH PRESSURE TURBINE
 - o THREE SPATIAL COMPONENTS
 - o INTER AND INTRA BLADE
 - o MEAN AND TURBULENT PARAMETERS

III

SEQUENCE:

1. START WITH SYSTEMS DEVELOPED FOR LOW TEMPERATURE CASCADES AND ROTATING BLADE ROWS
2. APPLY THESE SYSTEMS (MODIFIED) TO BENCH TOP COMBUSTOR FACILITY
3. COMPLETE DESIGN FOR HIGH TEMPERATURE FACILITIES
4. INSTALL SYSTEM IN WARM TURBINE FACILITY
5. INSTALL SYSTEM IN HIGH PRESSURE/TEMPERATURE TURBINE FACILITY

HOST LASER ANEMOMETRY SCHEDULE

73 74 81 82 83 84 85 86

LOW TEMPERATURE WORK

CASCADES
 ROTATING COMPRESSOR STAGES

BENCH TOP COMBUSTOR

WINDOWS/SEEDING
 SIGNAL PROCESSING
 SOFTWARE
 RADIAL COMPONENT

WARM TURBINE

FACILITY MODIFICATIONS
 LA SYSTEM DESIGN CHOSEN
 INSTALLATION & OPERATION

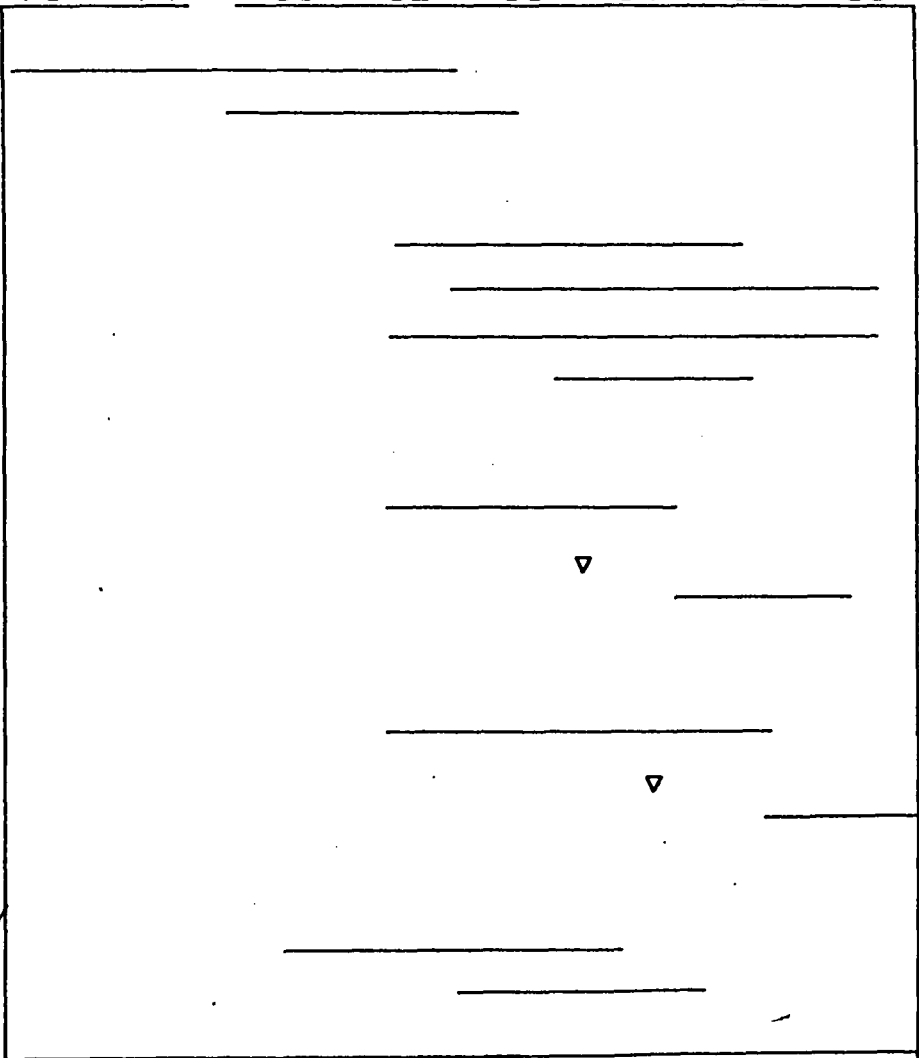
HIGH PRESSURE TURBINE

FACILITY MODIFICATIONS
 LA SYSTEM DESIGN CHOSEN
 INSTALLATION & OPERATION

SIGNAL PROCESSING OPTIMIZATION

ANALYTICAL
 EXPERIMENTAL

112



LASER ANEMOMETER OPTIMIZATION

P.I. - Robert V. Edwards
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Case Western Reserve University
Cleveland, Ohio 44106

The purpose of this section of the project is to design, construct, and test laser anemometer configurations for Hot Section velocity measurements. Optimizing the laser anemometer system necessarily included the data processing algorithms used. It is felt that the requirements here are too demanding for standard laser anemometer systems.

Relevant Hot Section Properties

- 1) High temperature with possibility of a large background radiation
- 2) Difficult optical access
- 3) Large flow velocity variation - especially in the rotating sections
- 4) Presence of solid surfaces that generate spurious reflections
- 5) Low seed particle density

The laser anemometer works by detecting light scattered by small particles entrained in a flow as they pass through a well-defined region of space. This region is illuminated by a pattern of light (see Figures 1 and 2). The measurement problem is to detect the "signature" of a particle as it appears and to extract the velocity from that signature. Any detected light that is not coming from a particle within the desired region, interferes with the measurement. The desire to make measurements near walls makes it essential to design a system that is particularly effective at rejecting light coming from outside the wanted region (see Figure 3). The optical system must be designed so that all expected flow angle fluctuations generate a recognizable, measurable signature. The rest of the detector system must be robust enough to deal with the wide range of velocities encountered in rotating systems.

In the past few years, the laser scattering group at Risø, Denmark, under the direction of Lars Lading, and the laser scattering group at Case Western Reserve University under Robert V. Edwards, have worked together to develop procedures for the optimal design of laser anemometry systems. The principles derived are being used to design the system for Hot Section measurements.

The system decided on is a so-called time-of-flight anemometer with elliptical spots (see Figures 4 and 5). In terms of laser light utilization, this optical pattern gives the biggest "bang for the buck". The version of the time-of-flight designed for this project contains two new features:
1) Elliptical spots - This gives the wide flow angle acceptance characteristics of a "fringe" anemometer combined with the superior spatial resolution or a time-of-flight anemometer. 2) Part of the normal time-of-flight signal

processing is performed optically - The velocity information in a time-of-flight is obtained by timing the interval between the pulses from the two spots. In a light-scattering experiment the received pulses always contain noise. Therefore, the position of the peak of the pulse cannot be obtained by differentiation of the signal. The derivative of the noise will overwhelm the signal. It can be shown that the optimal method of detecting the pulse position in the presence of noise involves transforming the received pulse into a "sideways S" pulse as shown in Figure 6.

The prototype for the Hot Section measurements uses a unique optical coding to transform the pulse into the optimal form for pulse position sensing (See Figures 7 and 8). Heretofore, this required rather complex and inflexible electronic circuitry. This optical processor is intrinsically free from some of the errors to which the electronic circuits were prone.

The optical prototype has been constructed and is in the initial phase of testing. In the next year, the full system will be built and tested for accuracy, robustness and spatial discrimination. The theory from which the system was derived is being written up for publication and should appear in the next year.

Recent Publications

1. Edwards, R. V.: A New Look at Particle Statistics in Laser Anemometer Measurements, J. Fld. Mech. Vol. 105, pp. 317-325, 1981.
2. Edwards, R. V., Lading, L. and Jensen, A. S.: Optimizing and Comparing Laser Anemometers, International Symposium on Applications of Laser-Doppler Anemometry to Fluid Mechanics, Lisbon, Portugal, July 1982.
3. Edwards, R. V. and Jensen, A. S.: Output Statistics of Laser Anemometers in Sparsely Seeded Flows, International Symposium on Applications of Laser-Doppler Anemometry to Fluid Mechanics, Lisbon, Portugal, July 1982.

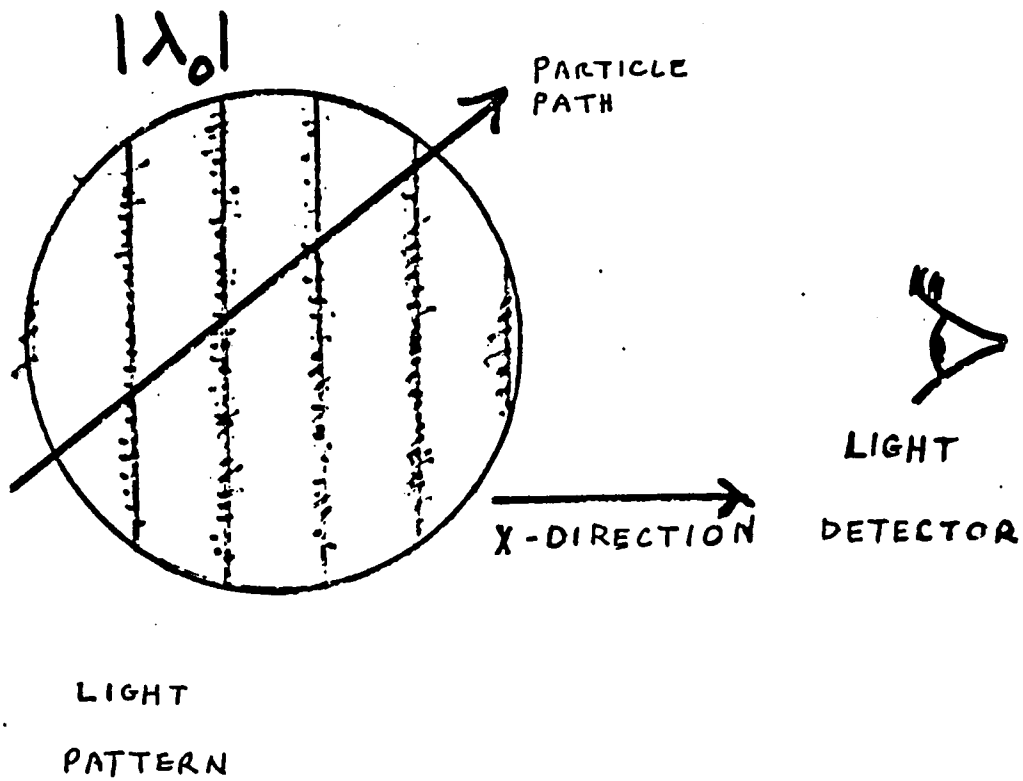
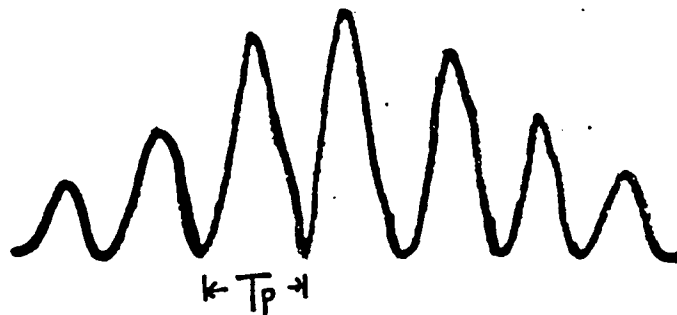


Figure 1

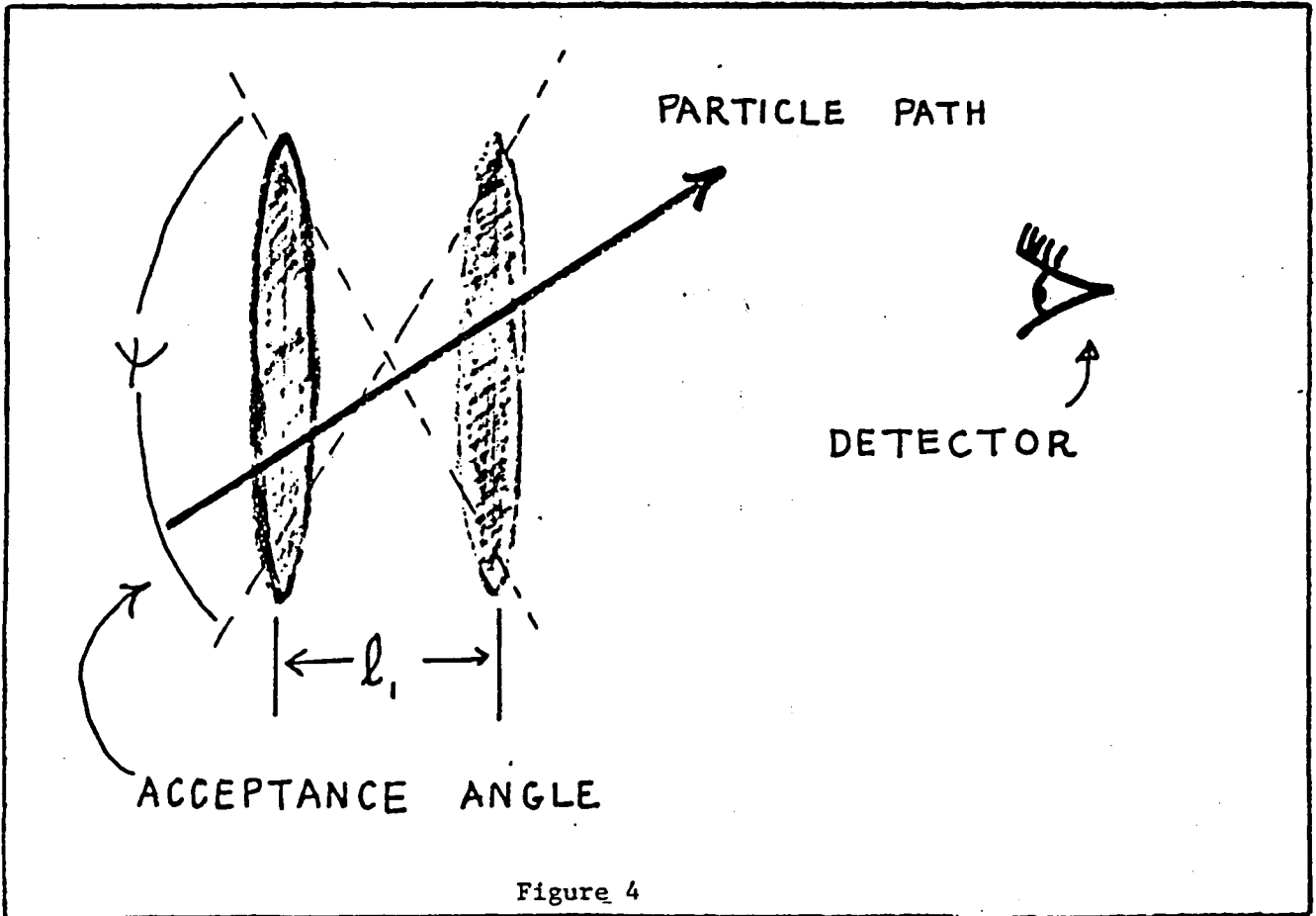
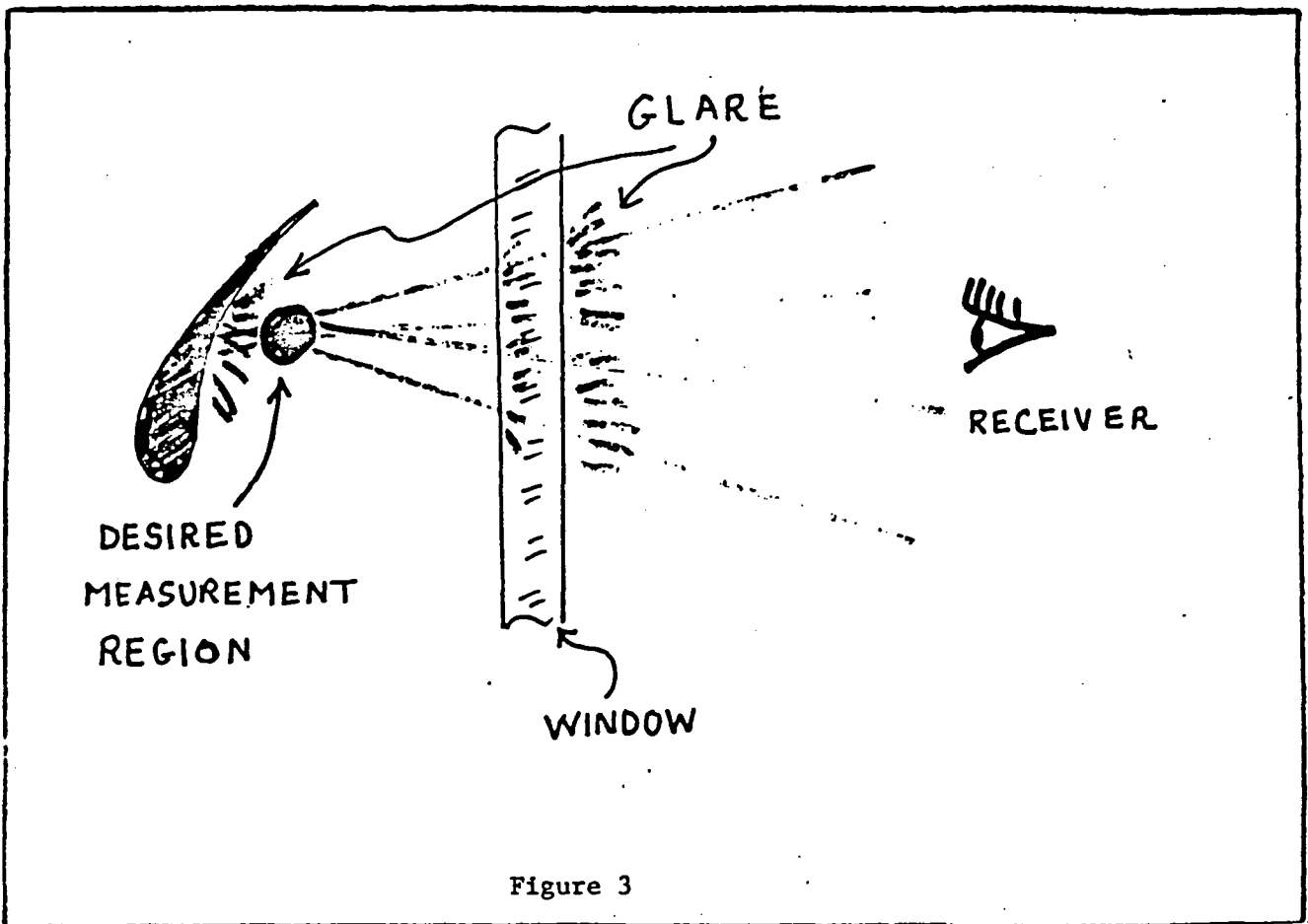
PARTICLE "SIGNATURE"



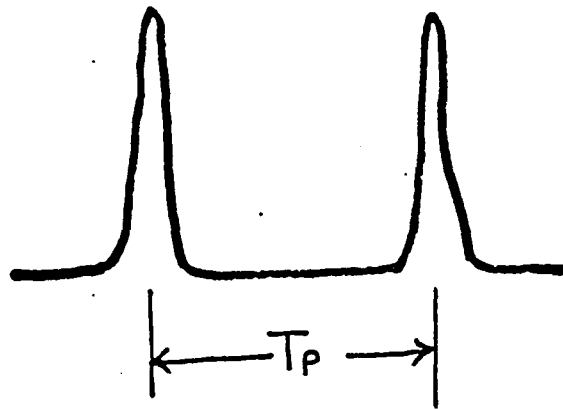
IN X DIRECTION , $V_{Px} T_p = \lambda_0$.

λ_0 IS KNOWN , SO MEASURING T_p
GIVES X - COMPONENT OF PARTICLE VELOCITY.

Figure 2

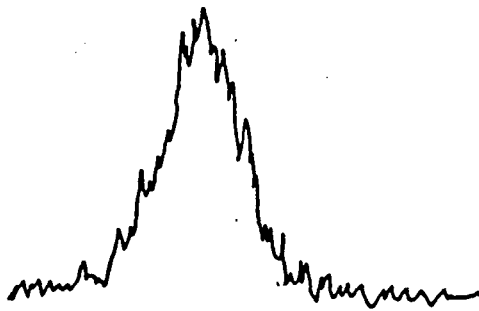


"SIGNATURE"

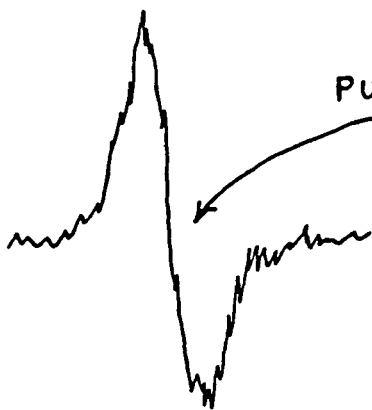


$$V_{Px} = \frac{l_1}{T_P}$$

Figure 5



NOISY PULSE



PULSE POSITION

TRANSFORMED
PULSE

Figure 6

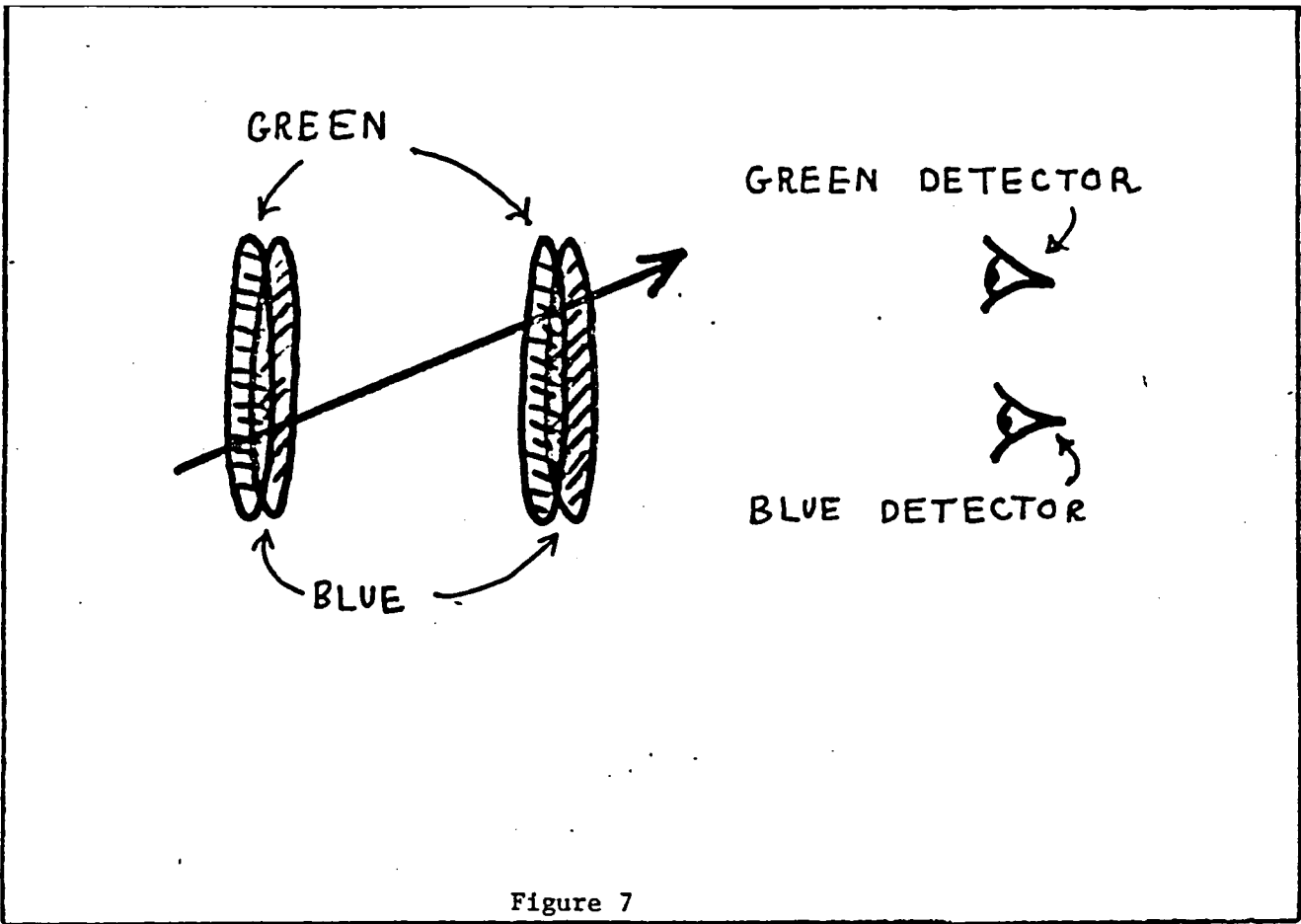


Figure 7

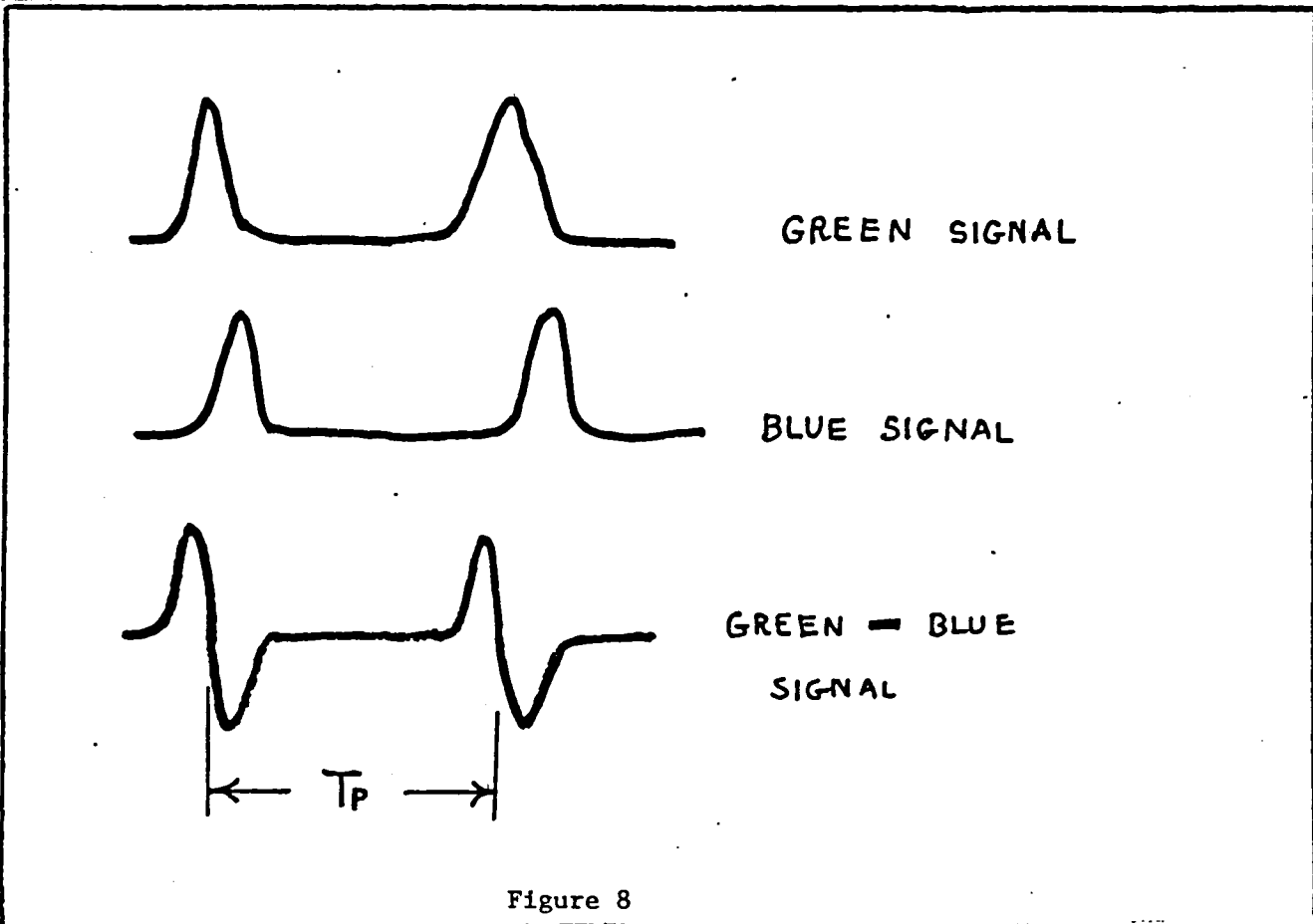


Figure 8

TURBINE HEAT TRANSFER

John E. Rohde
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

Improved turbine durability and performance and reduced development cost will all result from improved methods of predicting turbine metal temperatures. As you know, better metal temperature prediction methods require improvements in the method of determining the hot gas flow through the turbine passage and the cooling air flow inside the airfoil and in the methods of predicting the heat transfer rates on the hot gas-side and on the coolant-side of the airfoil. The overall turbine heat transfer effort is directed at improving all four of these areas of concern.

Achievement of these improvements requires a rigorous and systematic research effort from both the experimental and analytical sides. The experimental approach being pursued starts with fundamental experiments with simple shapes and flat plates; progresses on to more realistic cold, warm, or hot cascades; continues to progress on to more realistic warm turbine, large low-speed turbine, or transient turbine tests; and finally combines all the interactive effects in real-engine environment turbine tests. Analytical approaches being pursued also start with relatively simple mathematical models and progress to more realistic cases that include more interactive effects, and finally combines all the interactive effects of the turbine operating in the real engine environment.

Currently, contract and grant activities are being (or will be) conducted to obtain fundamental experimental data and to develop and/or compare analytical methods in all four areas of concern. These contract and grant activities will be discussed in detail later in this meeting by the respective principal investigators.

Major NASA Lewis in-house turbine research efforts are being pursued to obtain more realistic and real-engine type turbine experiments. The NASA

Lewis Research Center is in the process of activating our High Pressure-High Temperature Facility (HPF) with initial 20 atmosphere and 2500^o F experimentation scheduled for the last quarter of 1982. HPF will provide the country with a known real-engine environment in which to conduct controlled aerothermodynamic and structural research studies. We envision a multiple role for HPF in providing engineering-quality research data for modeling and code verification, in defining a real-engine environment, and in evaluating advanced turbine cooling technology in a real-engine environment.

The major turbine research parameters of interest that will be measured or determined to provide a better understanding of the thermal, aerodynamic, and mechanical performance of air-cooled turbine airfoils are the following:

- 1) local hot gas recovery temperatures along the airfoil surfaces,
- 2) local airfoil wall temperature,
- 3) local hot gas-side heat transfer coefficients on the airfoil surfaces,
- 4) local coolant-side heat transfer coefficients inside the airfoils,
- 5) local hot gas flow velocities and secondary flows at real-engine conditions, and
- 6) local delta strain range of the airfoil walls.

Currently, little of this type experimental research information exists with controlled warm or real-engine conditions and known boundary conditions.

These in-house turbine research efforts will be conducted using the best available analyses to help define the test configurations, the types of research measurements, and/or the test conditions and for the comparison with the measured research results. Analytical efforts will initially use the best available flow and heat transfer codes such as a two- or three-dimensional inviscid flow code and a two- or three-dimensional boundary layer heat

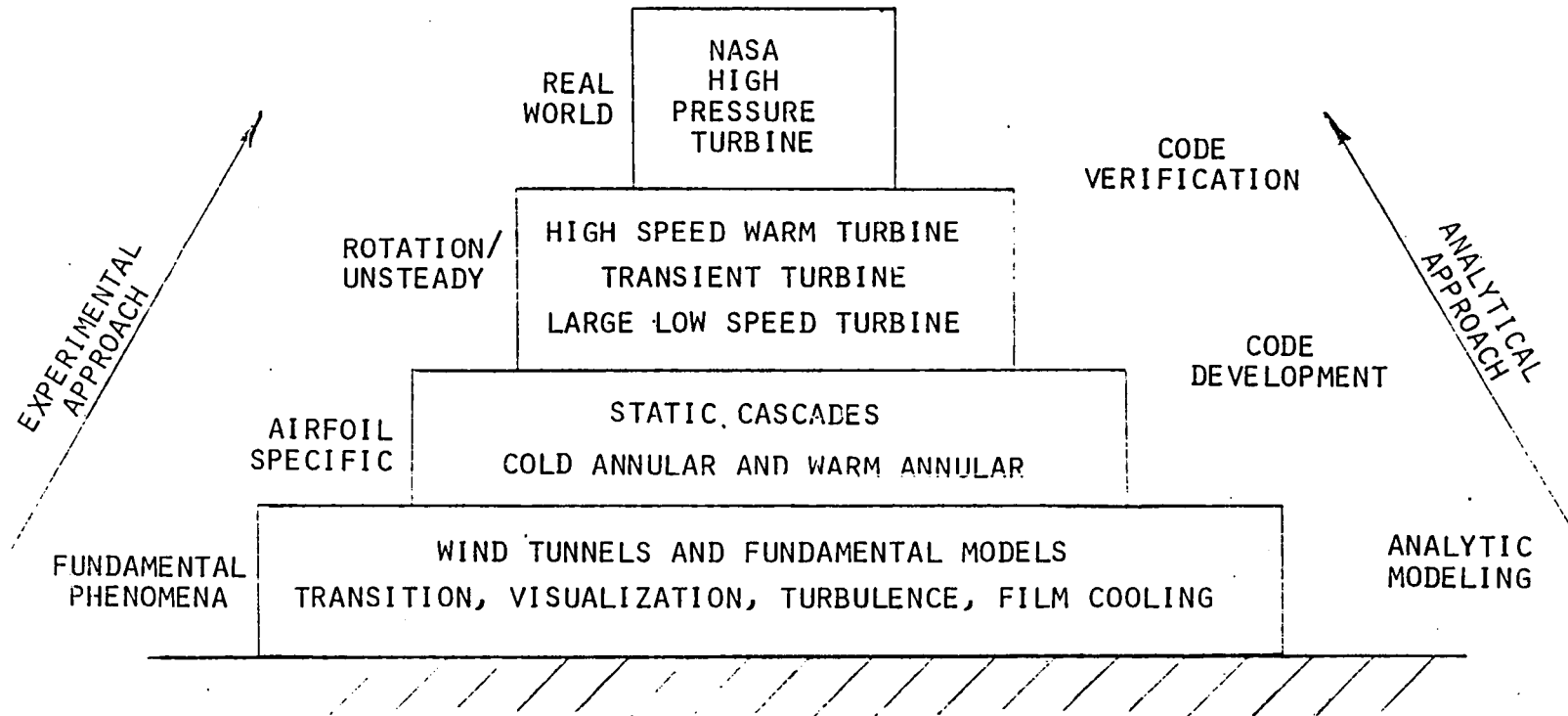
transfer code. These analyses will be applied at the mid-span section and possibly at the hub and tip sections or other local zones of the passage. More sophisticated three-dimensional viscous codes and three-dimensional viscous codes with boundary layer resolution will be used as they become available. These analytical efforts will be conducted using the best available source or sources in-house and on contract with industry and universities.

TURBINE HEAT TRANSFER

OBJECTIVES: IMPROVE ACCURACY OF PREDICTING LOCAL BLADE METAL TEMPERATURES USING COMPUTER CODES THAT ARE COMPATIBLE WITH STRUCTURAL ANALYSIS CODES

- APPROACH:
- INVESTIGATE HOT GAS STREAM AND COOLANT PASSAGE FLOW MECHANICS AND HEAT TRANSFER
 - OBTAIN BENCHMARK-QUALITY AND ENGINEERING-QUALITY DATA FOR EVALUATION AND IMPROVEMENT OF MODELS PRESENTLY USED IN PREDICTION CODES
 - UTILIZE IMPROVED MODELS TO IMPROVE ACCURACY OF PREDICTING LOCAL GAS-SIDE AND COOLANT-SIDE HEAT TRANSFER COEFFICIENTS
 - UTILIZE FLOW MODELS TO IMPROVE ACCURACY OF PREDICTING LOCAL HOT GAS STREAM ENVIRONMENT THROUGH TURBINE ROWS AND COOLANT FLOW CONDITIONS INSIDE THE AIRFOIL
 - INTEGRATE IMPROVEMENTS IN PREDICTION OF HOT GAS STREAM ENVIRONMENT AND COOLANT FLOW AND LOCAL HOT GAS-SIDE AND COOLANT-SIDE HEAT TRANSFER COEFFICIENTS INTO IMPROVED METAL TEMPERATURE PREDICTION CODES
 - PROVIDE ENGINEERING-QUALITY TEST CASES FOR EVALUATION OF ACCURACY OF PREDICTION AND INPUT TO STRUCTURAL ANALYSIS CODES

BUILDING BLOCK AEROTHERMAL TURBINE RESEARCH APPROACH



TURBINE HEAT TRANSFER

PLANNING SCHEDULE

ELEMENT	FY	81	82	83	84	85	86	87	EXPECTED RESULTS
GAS-SIDE HEAT TRANSFER, NON-ROTATING, 2-D		[Solid bar]							--DETERMINE INFLUENCE OF VARIABLES ON FLOW TRANSITION AND DURATION AND IMPROVED MODELS
GAS-SIDE HEAT TRANSFER, NON-ROTATING, FILM			[Solid bar]						--SAME AS ABOVE WITH FILM COOLING
GAS FLOW ENVIRONMENT AND HEAT TRANSFER, NON-ROTATING			[Solid bar]			[Dashed bar]			--OBTAIN BENCHMARK QUALITY AEROTHERMODYNAMIC DATA AND IMPROVED THREE-DIMENSIONAL VISCOUS FLOW CODES, NO ROTATION
GAS FLOW ENVIRONMENT AND HEAT TRANSFER, ROTATING, 3-D					[Dashed bar]				--SAME AS ABOVE WITH ROTATION
MULTIPLE JET ARRAY IMPINGEMENT		[Solid bar]							--IMPROVED HEAT TRANSFER CORRELATION AND MODEL FOR IMPINGEMENT COOLING
COOLANT SIDE HEAT TRANSFER WITH ROTATION AND ENTRANCE GEOMETRY				[Solid bar]					--HEAT TRANSFER CORRELATIONS, INCLUDING EFFECTS OF ROTATIONS AND ENTRANCE GEOMETRY
METAL TEMPERATURE PREDICTION CODES				[Dashed bar]					--METAL TEMPERATURE PREDICTION CODES WITH IMPROVED HEAT TRANSFER MODELS/CORRELATIONS
IN-HOUSE RESEARCH AND VERIFICATIONS			[Solid bar]						--VERIFICATIONS OF FLOW, HEAT TRANSFER METAL TEMPERATURES, AND STRAIN PREDICTIONS, AT NEAR AND REAL-ENGINE TYPE CONDITIONS

TURBINE HEAT TRANSFER

POSSIBLY CONTRACT AND/OR
IN-HOUSE:

METAL TEMPERATURE PREDICTION CODES

SCOPE:

REVIEW AND MODIFY EXISTING AIRFOIL METAL TEMPERATURE
CODES FOR EFFICIENT INCORPORATION OF DEVELOPED
MODELS AND FOR INTERFACING WITH STRUCTURAL ANALYSIS
CODES

DURATION:

THREE YEARS

APPROACH:

- REVIEW EXISTING CODES
- INCORPORATE IMPROVED MODELS DEVELOPED UNDER HOST
- ASSURE EFFICIENT INTERFACING WITH STRUCTURAL
CODES

TURBINE HEAT TRANSFER

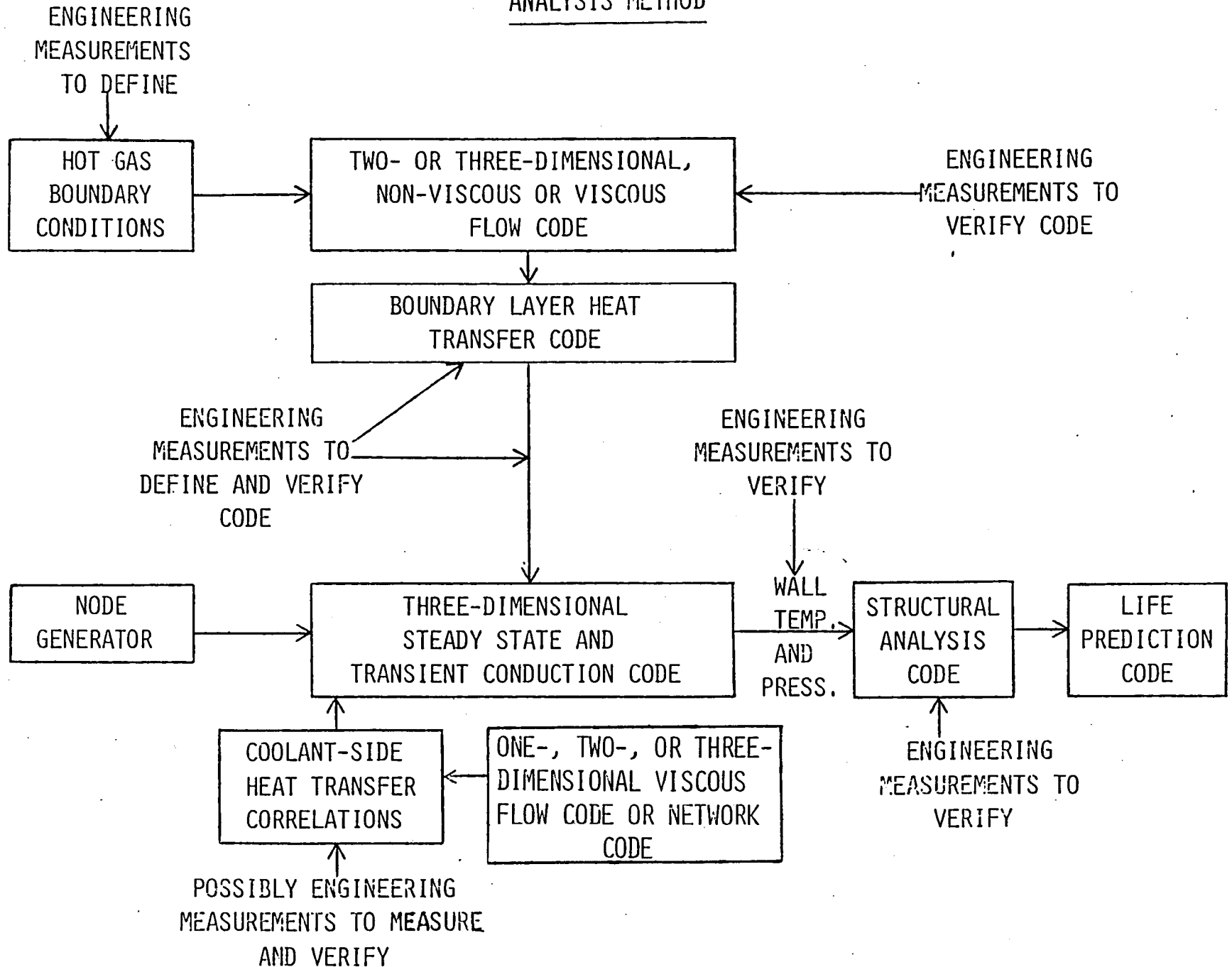
IN-HOUSE RESEARCH AND VERIFICATIONS

SCOPE: EXPERIMENTS AND ANALYSIS TO SUPPLEMENT CONTRACTUAL AND GRANT EFFORTS ON IMPROVING ACCURACY OF FLOW ENVIRONMENT AND HEAT TRANSFER PREDICTIONS AND THE VERIFICATION OF DEVELOPED/IMPROVED PREDICTION METHODS

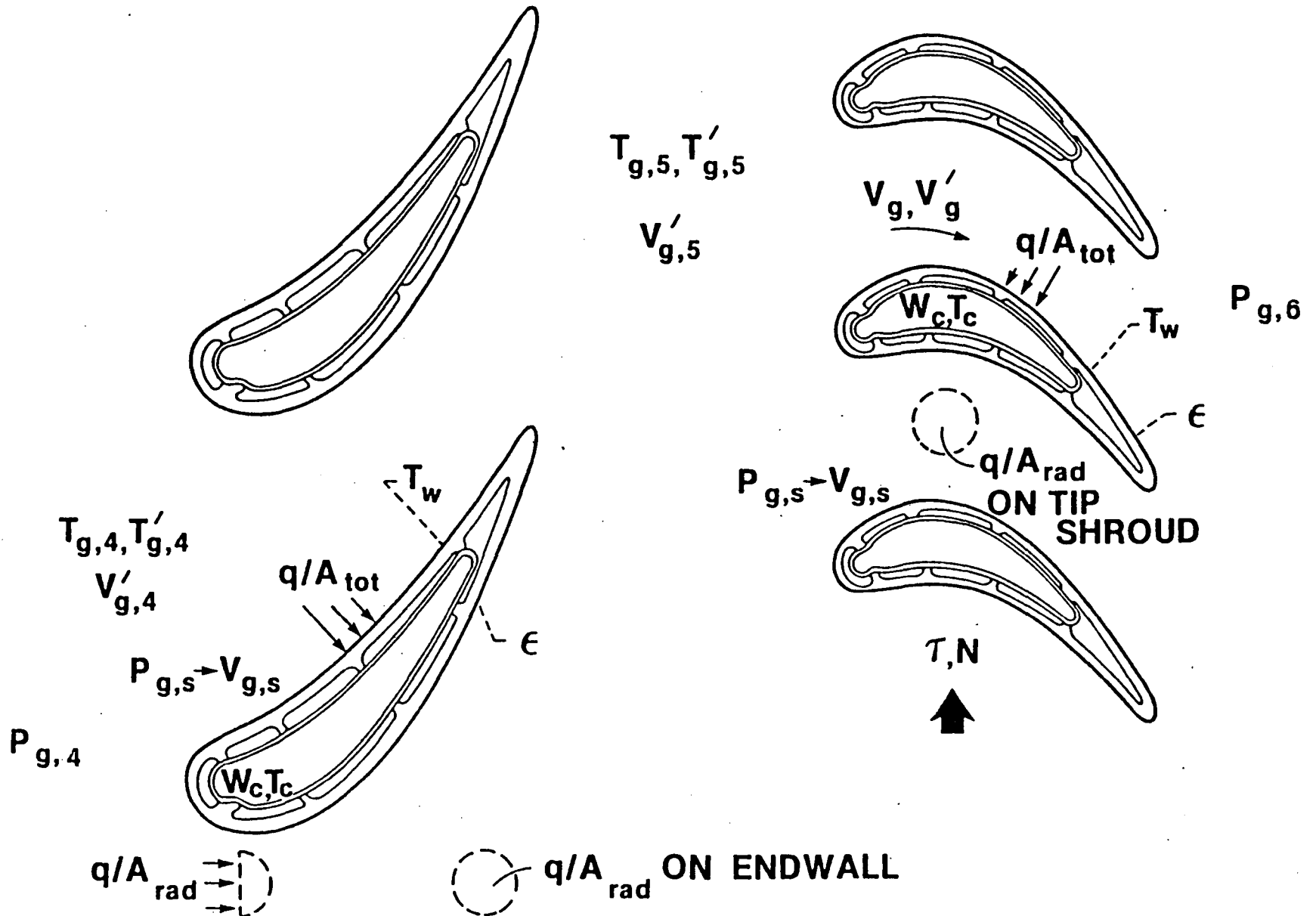
DURATION: SIX YEARS

- APPROACH:
- MEASURE LOCAL HEAT TRANSFER COEFFICIENTS OVER A STATOR VANE AT NEAR-REAL ENGINE CONDITIONS AND COMPARE WITH PREDICTION
 - PROCURE LASER ANEMOMETER SYSTEM AND INSTALL AND CHECK-OUT IN NASA WARM TURBINE
 - OBTAIN MEASUREMENTS IN NASA HIGH PRESSURE TURBINE (HPT) AT NEAR REAL ENGINE CONDITIONS TO EVALUATE PREDICTION ACCURACIES OF CODES FOR HOT GAS FLOW, HOT GAS ENVIRONMENT, HOT GAS-SIDE AND COOLANT-SIDE HEAT TRANSFER COEFFICIENTS, METAL TEMPERATURES, AND STRAIN

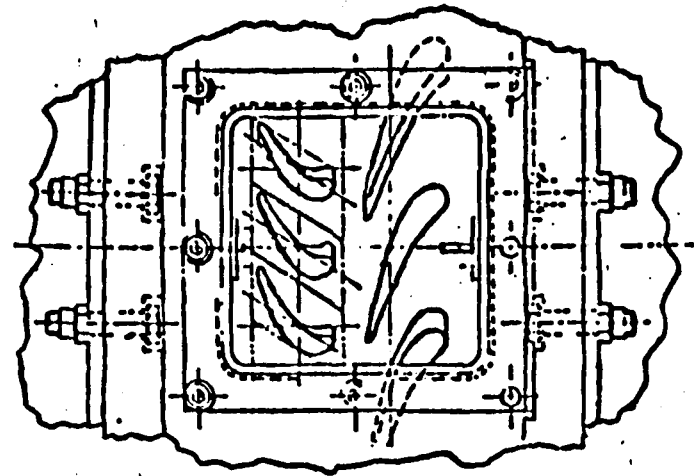
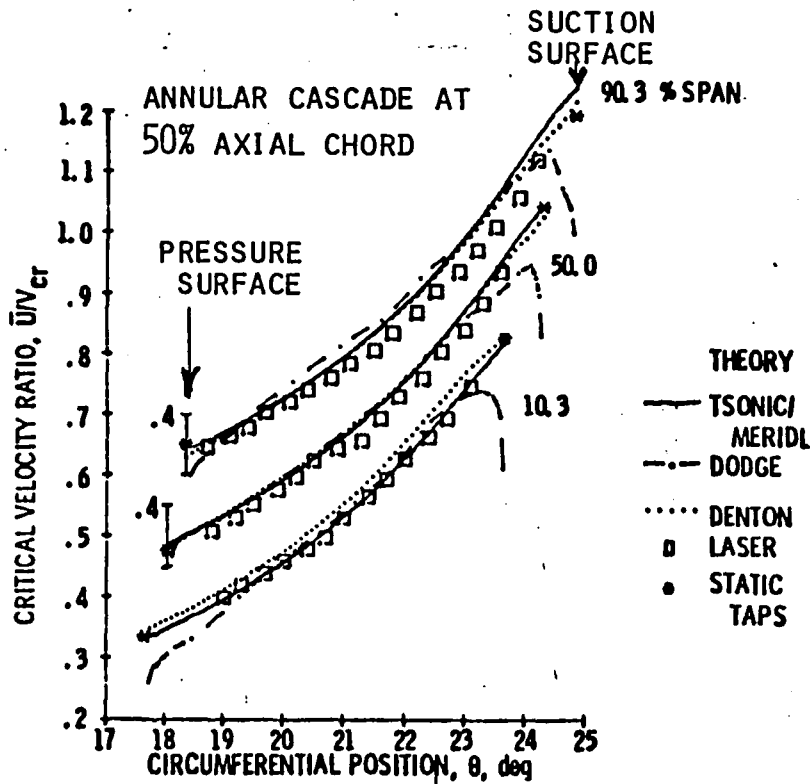
TYPICAL AIRFOIL AEROTHERMODYNAMIC
ANALYSIS METHOD



ENGINEERING-TYPE MEASUREMENTS DESIRED



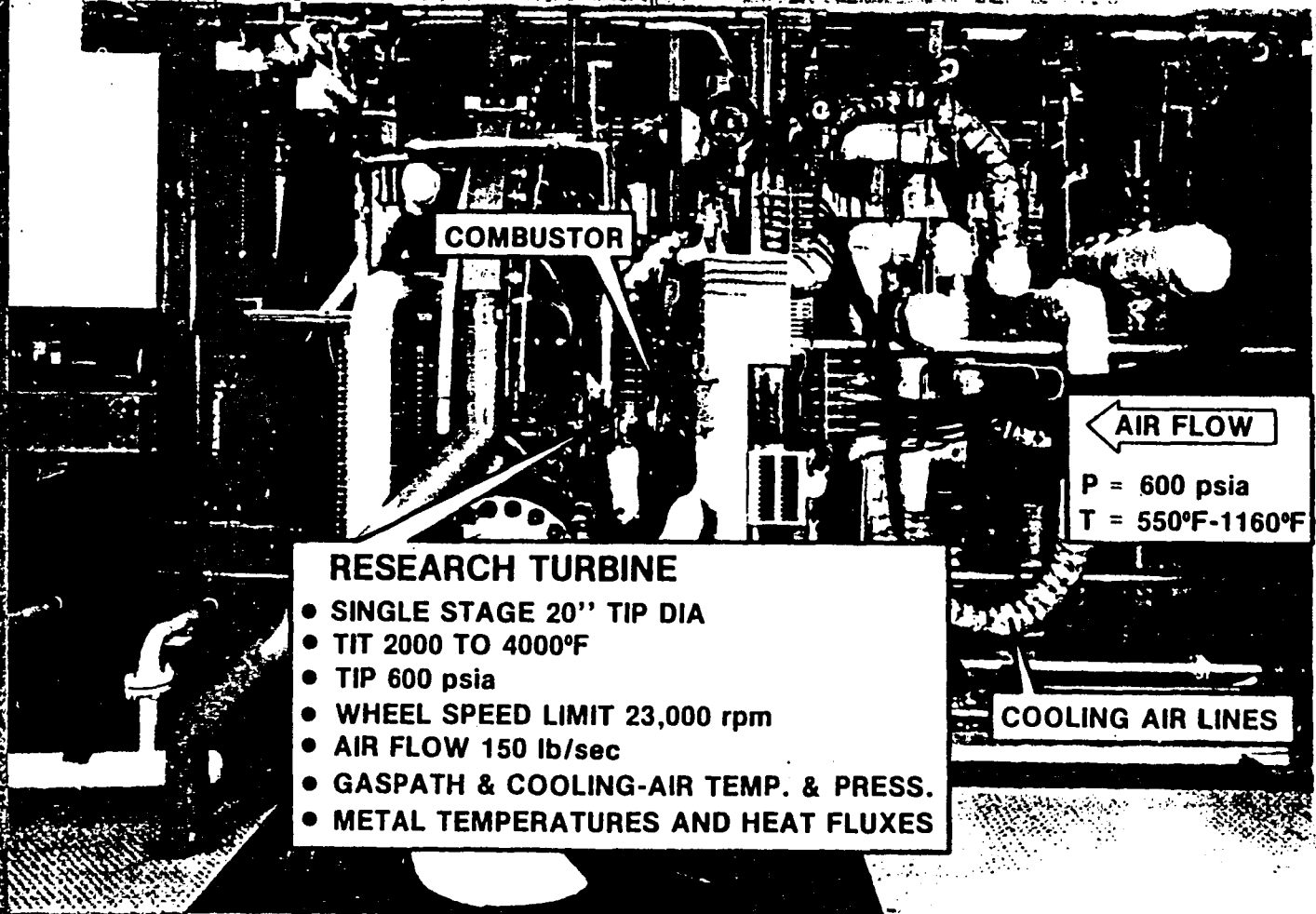
**NONINTRUSIVE VELOCITY MEASUREMENTS
THROUGH COMPLETE TURBINE STAGE WITH
UNIFORM AND NONUNIFORM INLET
TEMPERATURE PROFILES**



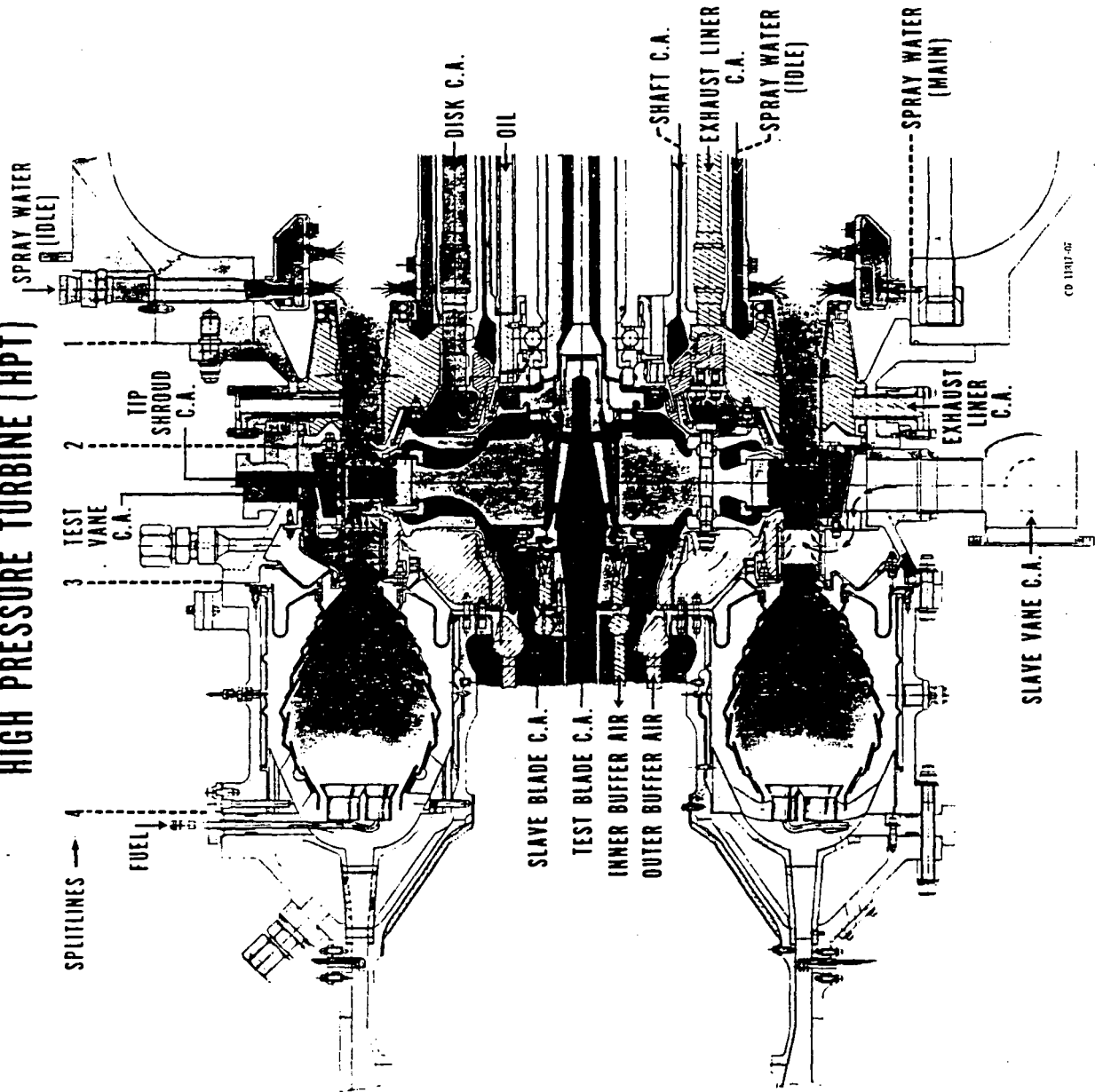
WARM TURBINE
LASER SURVEY
WINDOW

**BENCHMARK DATA FOR EVALUATION OF
ADVANCED 3-D FLOW CODES FOR AERO-
DYNAMICS AND HEAT TRANSFER**

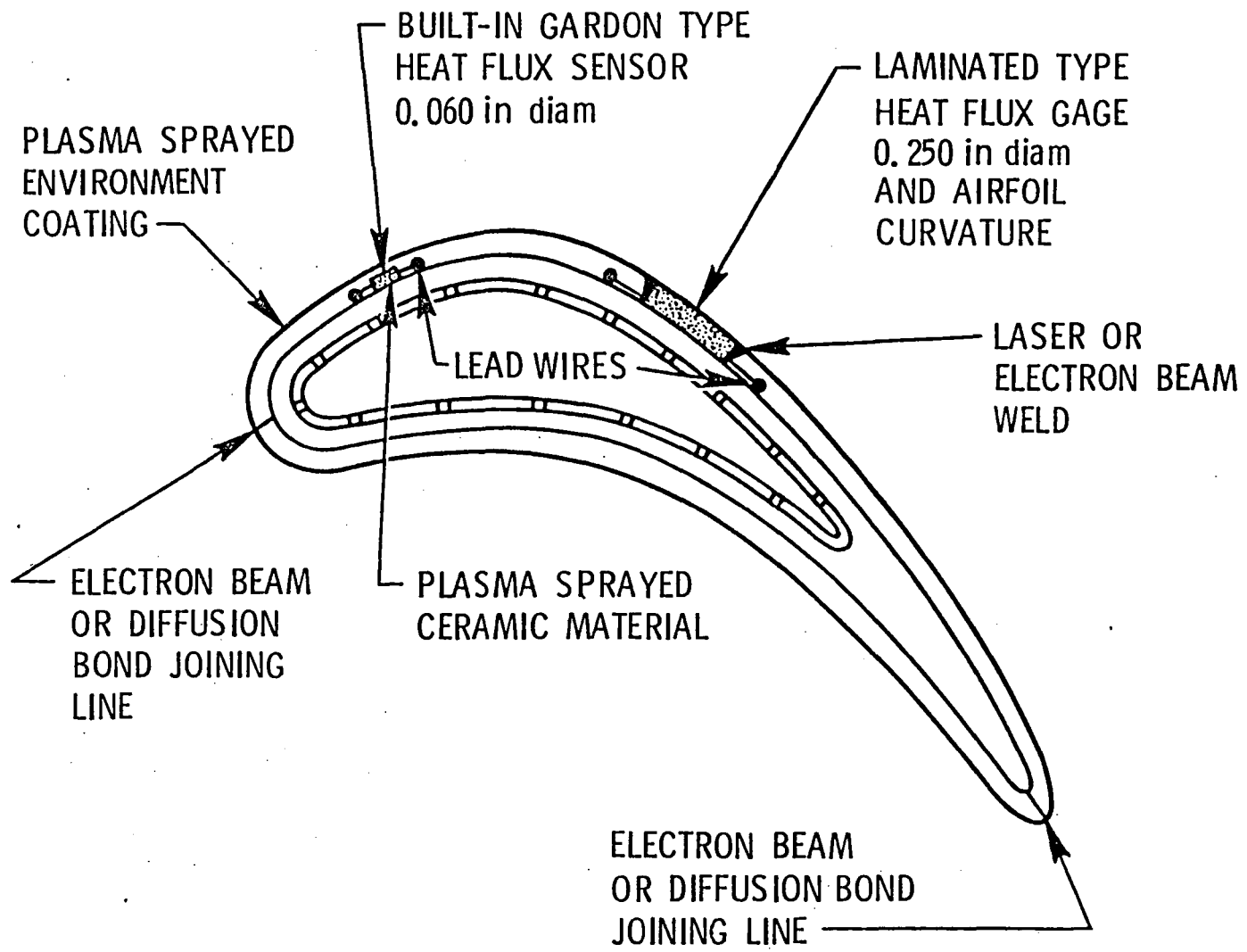
HIGH PRESSURE AND TEMPERATURE TURBINE RESEARCH FACILITY REAL ENGINE ENVIRONMENT



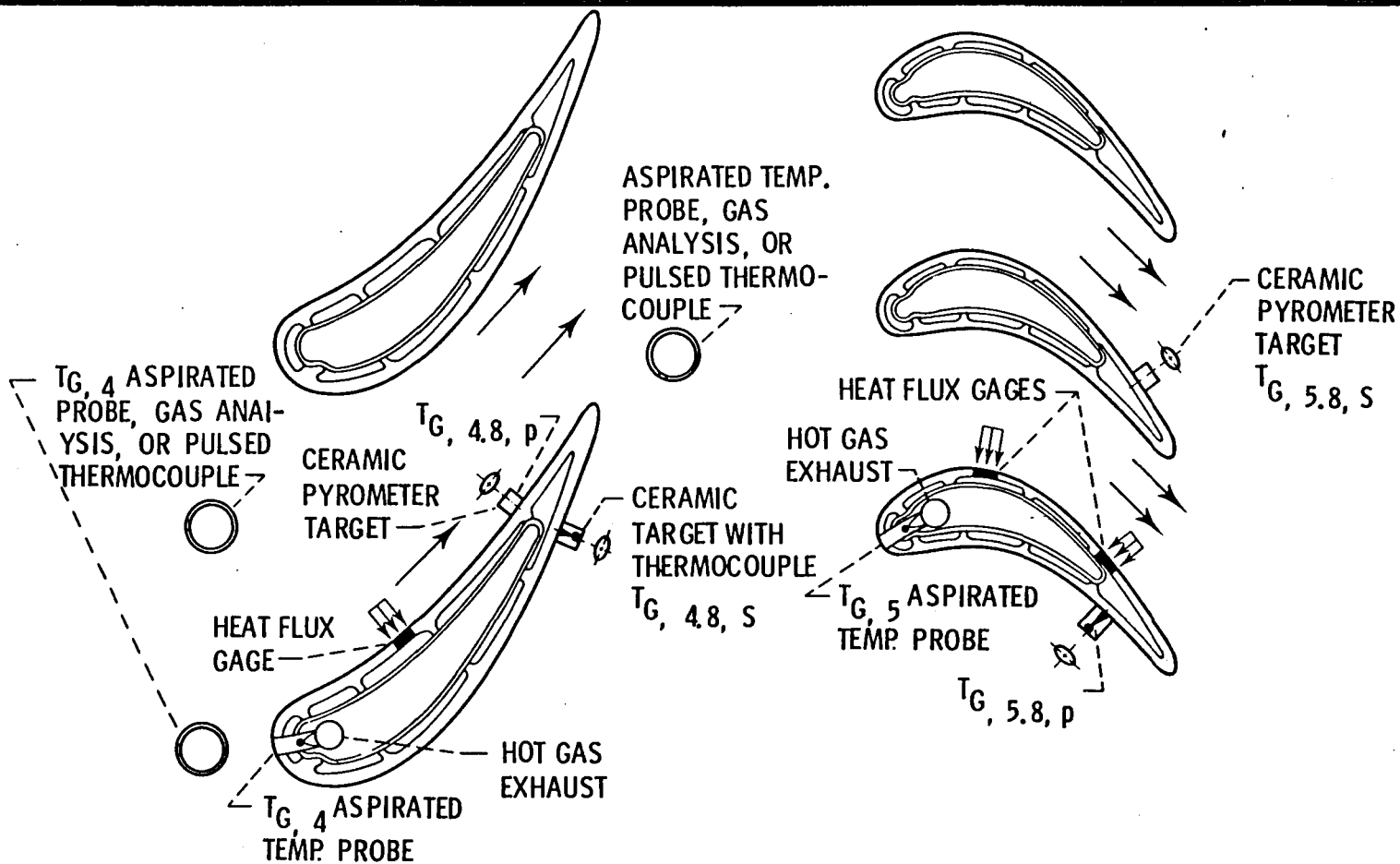
HIGH PRESSURE TURBINE (HPT)



CO 1197-66

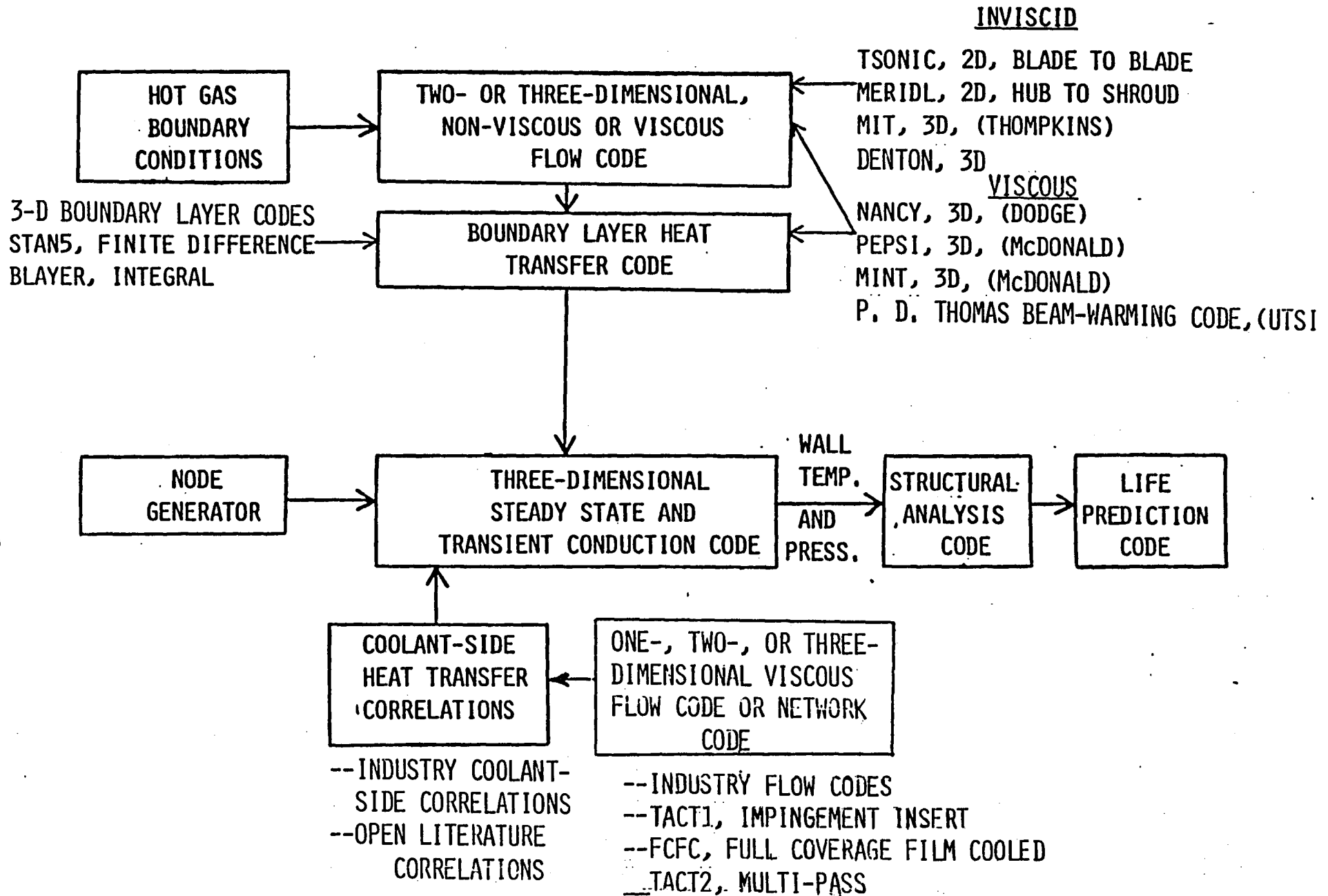


IN-HOUSE METHODS FOR THE FABRICATION
OF AIRFOILS WITH HEAT FLUX SENSORS



LOCAL HOT GAS TEMPERATURE AND HEAT FLUX INSTRUMENTATION

TYPICAL AIRFOIL AEROTHERMODYNAMIC ANALYSIS METHOD



HOST IN-HOUSE HIGH TEMPERATURE TURBINE TESTING

ELEMENT	FY	82	83	84	85	86	87	EXPECTED RESULTS
CONDUCT HEAT TRANSFER STUDIES OVER THE RANGE OF REAL-ENGINE CONDITIONS WITH NON-FILM COOLED AIRFOILS				☐ T _G	☐ H _G & H _C			--REAL ENGINE VALUES OF LOCAL HOT GAS TEMPERATURES, LOCAL HOT GAS-SIDE HEAT TRANSFER COEFFICIENTS, AND LOCAL COOLANT-SIDE HEAT TRANSFER COEFFICIENTS FOR COMPARISON WITH ANALYTICAL MODELS AND CODES
VERIFICATION TEST OF THE NASA BASELINE RESEARCH TURBINE WITH AIRFOIL METAL TEMPERATURE PREDICTION CODES						☐		--ASSESSMENT OF AIRFOIL METAL TEMPERATURE PREDICTION CODES WITH THE NASA BASELINE RESEARCH TURBINE
MAP THE THREE-DIMENSIONAL FLOW FIELD IN A ROTATING PASSAGE WITH NO FILM COOLING				☐ WARM TURBINE	☐ HPT			--VERIFY THE INVISCID FLOW REGION OF THE PASSAGE, WHICH WILL HAVE BEEN ESTABLISHED IN WARM TURBINE TESTS AND ESTABLISH THE ACTUAL VELOCITIES IN THE SECONDARY FLOW REGIONS OF THE PASSAGE FOR COMPARISON WITH FLOW CODES
MEASURE BI-AXIAL STRAIN DELTAS AT CRITICAL LOCATIONS ON STATIC AND ROTATING AIRFOILS AND THE DISK						☐		--ASSESSMENT OF STRUCTURAL CODES WITH KNOWN THERMAL AND MECHANICAL BOUNDARY CONDITIONS
VERIFICATION TEST OF ADVANCED TECHNOLOGY COOLED TURBINE WITH AIRFOIL METAL TEMPERATURE PREDICTION CODES							☐	--ASSESSMENT OF AIRFOIL METAL TEMPERATURE PREDICTION CODES WITH AN ADVANCED TECHNOLOGY COOLED TURBINE

GAS SIDE HEAT TRANSFER - 2-D FLOW

Ronald York
Detroit Diesel Allison
Indianapolis, Indiana

The work reported herein is being performed under NASA contract NAS 3-22761 entitled, "Development of Analytical Techniques for Improved Prediction of Local Gas-to-Blade Heat Transfer". The objectives of the program were to assess the capability of currently available modeling techniques to predict airfoil surface heat transfer distributions in a 2-D flow field, acquire experimental data as required for model verification, and to make and verify improvements in the analytical models. Chart No. 1 summarizes the program task structure.

Two data sets, Turner⁽¹⁾ and Lander⁽²⁾, were selected from the literature for use in evaluating models in Task I. Two additional airfoils were chosen for cascade testing under this contract. These airfoils, the Mark II and C3X, are representative of highly loaded, low solidity airfoils currently being designed. Cross sections of the four airfoils and the grid used to make inviscid flow predictions for each airfoil are shown in Chart No. 2. Note the significant variation in airfoil geometry. This variation is intended to provide a significant test of the analytical models. Predicted surface pressure distributions for the four airfoils are shown in Chart No. 3.

The two heat transfer cascades that were operated under Task II were run in the Detroit Diesel Allison Aerothermodynamic Cascade Facility (ACF). The facility, described in Chart No. 4, provides the capability of obtaining both heat transfer and aerodynamic measurements at simulated engine conditions. The method employed in the facility to obtain airfoil surface heat transfer measurements is shown schematically in Chart No. 5. Basically, the exterior of the airfoil is instrumented with grooved surface thermocouples with this data serving as the exterior boundary condition input to a finite element analysis. The internal boundary conditions are calculated heat transfer coefficients in the internal cooling holes. These values are calculated from measurements of the coolant temperature, pressure and flowrate. A photo of the three-vane, 2-D C3X cascade is shown in Chart No. 6. The vane surface thermocouples appear as lines on the surface. This center airfoil also contains static pressure taps, thus permitting simultaneous measurement of the surface pressure and heat transfer distributions.

The test matrix over which both cascades were operated is shown in Chart No. 7. Data was obtained at two exit Mach numbers, 0.9 and 1.05, and over a range of exit Reynolds numbers from 1.5×10^6 to 2.5×10^6 . The inlet turbulence intensity and wall-to-gas temperature ratio were also varied. Chart No. 8 shows typical data from the Mark II cascade. The effect of variation in Reynolds number is clearly evident. The heat transfer measurement technique is capable of detecting the rapid increase in heat transfer caused by separation and reattachment on the suction surface. This separation is a result of the large adverse pressure gradient on the Mark II airfoil. Similar data for the C3X cascade is shown in Chart No. 9. Here the suction surface demonstrates transition and the effect of Reynolds number on the location of the start of transition can be clearly seen. Chart No. 10 illustrates the Mach number effects on heat transfer in the Mark II cascade. As would be expected the Mach number effects are seen only on the suction surface in the region where the Mach number affects the pressure distribution. A summary of the results found with the Mark II and C3X cascades is shown in Chart No. 11.

Charts 12-22 present some preliminary results of the analytical work being done using 2-D boundary layer theory for establishing an acceptable approach for predicting gas turbine airfoil heat transfer over a wide range of operating conditions and geometries. Starting with the STAN5 mixing length (ML) theory turbulence modeling evaluated in Task I, Task III efforts have been directed towards supplying systematic boundary and initial conditions and structuring a realistic gas turbine airfoil environment ML turbulence model which reflects free-stream turbulence, transition, curvature, pressure gradient, etc.

The importance of specifying realistic velocity (pressure) boundary conditions over the entire airfoil surface and generating appropriate initial velocity and thermal profiles is often understated because of more noticeable inadequacies in turbulence modeling. Specifying the correct velocity boundary conditions is very important since the Reynolds number and pressure gradient play important roles physically and in the development of turbulence and/or transition models. Specification of correct initial thermal profiles near the stagnation point are essential to accurately obtain leading edge heat loads. Boundary conditions for all boundary layer calculations are currently being obtained from the inviscid blade-to-blade Euler solver developed by Delaney at DDA. The ability of this method to accurately predict the inviscid pressure distribution is illustrated by Chart 12 for the Mark II airfoil. In addition, because the Delaney method uses a body-centered coordinate system (example shown on Chart 2), it provides excellent resolution of the velocity distribution from the stagnation point to the trailing edge. This ability to resolve the velocity field in the near-stagnation-point region has provided much insight and has guided modeling efforts aimed at generating initial velocity and thermal profiles. Initial profiles are generated from boundary layer similarity solutions assuming stagnation flow, i.e. Euler number equal to unity. Based on the work of Miyazaki and Sparrow⁽³⁾, who extended the laminar similarity solution concept to include the effects of free-stream turbulence for flow normal to a cylinder, a more general system of equations and turbulence model was developed at DDA to reflect the differences between stagnation flow on an airfoil from that on a circular cylinder. These changes are summarized on Chart 13. Changes include recasting the equations in compressible form and directly using the near-stagnation-point pressure-gradient calculated from the inviscid analysis rather than assuming an isolated cylinder value. Also, the Miyazaki and Sparrow turbulence model was generalized to reflect variable stagnation point pressure gradient. The results of this approach for predicting airfoil stagnation point heat transfer are shown in Chart 14. The open symbols are predictions using the Miyazaki and Sparrow turbulence model and the solid symbols represent predictions using DDA's generalized form. As can be seen, the present scheme is capable, in most instances, of predicting stagnation point heat transfer within $\pm 10\%$ of the experimental mean. These results are encouraging because the data span a wide range of turbulence levels (0.45 - 18%) and geometries. These results are a direct outcome of relaxing the cylinder-in-crossflow assumption, and properly modeling the near-stagnation-point velocity field using the body-centered coordinate system Euler solver.

The remaining charts 15-22, highlight progress towards developing a realistic ML turbulence model for a 2-D finite difference boundary layer code. Chart 15 presents the analytical form of the effective viscosity (μ_{eff}) being studied. Note the presence of two turbulent terms, μ_t and μ_{TU} . The principle difference in the two terms is the velocity scale assumed. The additional term is included to account for the significant effects of free-stream turbulence on heat transfer for nominally laminar boundary layers observed on turbine airfoils. The intermittency functions γ_t and γ_{TU} incorporate information regarding transition behavior. Besides evaluating promising published μ_{eff} models, a significant inhouse effort is in progress aimed at taking advantage of the large amount of data collected within this program. At the present time, the majority of the modeling effort has gone into structuring a suitable model for the turbulence viscosity term μ_{TU} . As contrasted on Chart 15, a major difference between Task I prediction methodology and the Task III concept is the inclusion of μ_{TU} . The general functional form for μ_{TU} , currently being explored, is shown in Chart 15. Although the exact form is not finalized, one key feature is that the model is compatible with the similarity solution stagnation flow analysis in that the effects of free-stream turbulence incorporated in the function f is carried forward into the surface boundary layer computation.

The last charts (15-22) present heat transfer predictions compared to experimental data from four different airfoil heat transfer experiments. Predictions are shown using the original STAN5 mixing length turbulence model with the transition Reynolds number set to 250, and the current Task III concept turbulence model for μ_{TU} . For the later predictions, the term γ_t was set to zero to allow only the influence of μ_{TU} to be seen in the μ_{eff} definition. Attention should be directed to the pressure surface predictions where the effects of free-stream turbulence are most dominant and therefore, the modeling of μ_{TU} more important.

Predictions for the Mark II airfoil, for one Reynolds number level, using the Task I and present Task III approaches are shown in Chart 16. Of particular interest is the better pressure surface prediction obtained using the additional "turbulence" viscosity term over a standard laminar/transitional/turbulent approach. Suction surface predictions were only possible up to the location of the shock, where the boundary layer code reached a separation condition. Chart 17 shows Task III concept predictions for the Mark II airfoil at three different exit Reynolds number levels and indicate, at least on the pressure surface, that both trends and levels are reasonably well predicted.

Chart 18 shows predictions for the C3X airfoil at a single exit Reynolds number and again there is a significant improvement in pressure surface prediction. The suction surface prediction is in poor agreement beyond the transition point, but as mentioned earlier transition modeling has not yet been addressed in the Task III predictions since $\gamma_t = 0$.

Charts 19 and 20 show predictions for the Turner airfoil and serve to illustrate the influence of free-stream turbulence. Since the original STAN5 ML turbulence model does not model free-stream turbulence effects directly, only one prediction is possible for the three different experiments as shown on Chart 19. In contrast, Chart 20 shows the effects of free-stream turbulence are adequately modeled on the pressure surface using the current μ_{TU} concept, but the suction surface predictions only indicate proper trends not levels.

Finally, predictions for the Lander airfoil are presented in Charts 21 and 22 respectively. It is clear that substantial predictive improvement can be attributed to the use of the turbulence viscosity although still, predictions beyond the indicated transition zone are not well modeled with this approach.

In conclusion, to date significant progress has been made in advancing the idea of establishing a unified approach for predicting airfoil heat transfer for a wide range of operating conditions and geometries. Preliminary results are encouraging and further ML turbulence modeling ideas will be explored in the remaining phase of the program, primarily concentrating on transition behavior through γ_t and γ_{TU} modeling.

REFERENCES

- (1) A. B. Turner, "Local Heat Transfer Measurements on a Gas Turbine Blade", Jour. of Mechanical Engng. Sciences, Vol. 13, 1971, pp. 1-12.
- (2) R. D. Lander, "Effect of Free-Stream Turbulence on the Heat Transfer to Turbine Airfoils", Tech. Rept. AFAPL-TR-69-70, Air Force Systems Command, Sept. 1969.
- (3) H. Miyazaki and E. M. Sparrow, "Analysis of Effects of Free-Stream Turbulence on Heat Transfer and Skin Friction", Jour. of Heat Transfer, Trans. ASME, Vol. 99, 1977, pp. 614-419.

LIST OF SYMBOLS

C	Chapman-Rubens parameter, $\rho\mu/\rho_e \nu_e$
F	Similarity function related to velocity
f	Empirical function in turbulence viscosity definition
G	Similarity function related to temperature or enthalpy
g	Empirical function in turbulence viscosity definition
H	Shape factor
l	Length scale in mixing length hypothesis
M	Mach number
p	Pressure
Pr	Prandtl number
Re	Reynolds number
T	Temperature
u	Velocity
X	Coordinate in streamwise direction
y	Coordinate normal to streamwise direction

GREEK

β	Euler number
γ	Intermittency function, or specific heat ratio when unsubscripted
δ	Boundary layer thickness
ϵ	Eddy viscosity
Λ	Pressure gradient parameter
μ	Molecular viscosity
ν	Kinematic viscosity
ρ	Density

SUBSCRIPTS

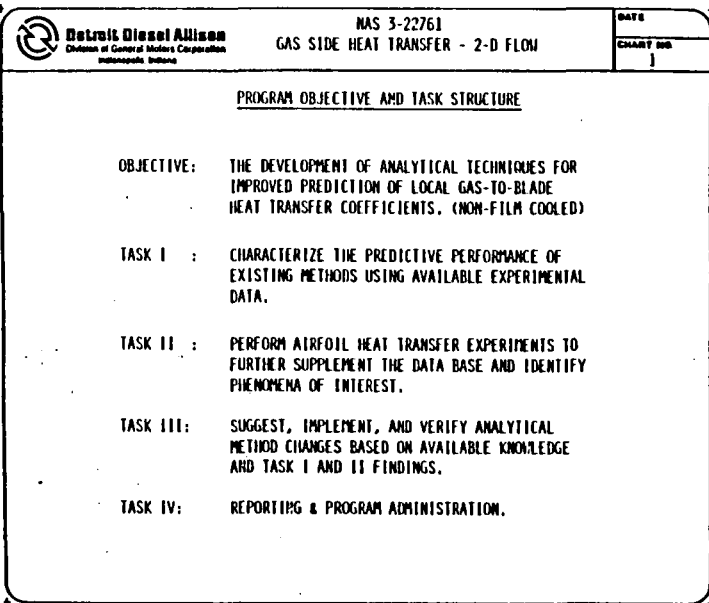
D	Refers to leading edge diameter or twice radius of curvature
e	Refers to outer edge of boundary layer
eff	Effective quantity
m	Refers to momentum
t	Refers to turbulent
tr	Refers to transition
TU	Refers to turbulence intensity
θ	Momentum thickness
*	Upstream quantity

SUPERSCRIPTS

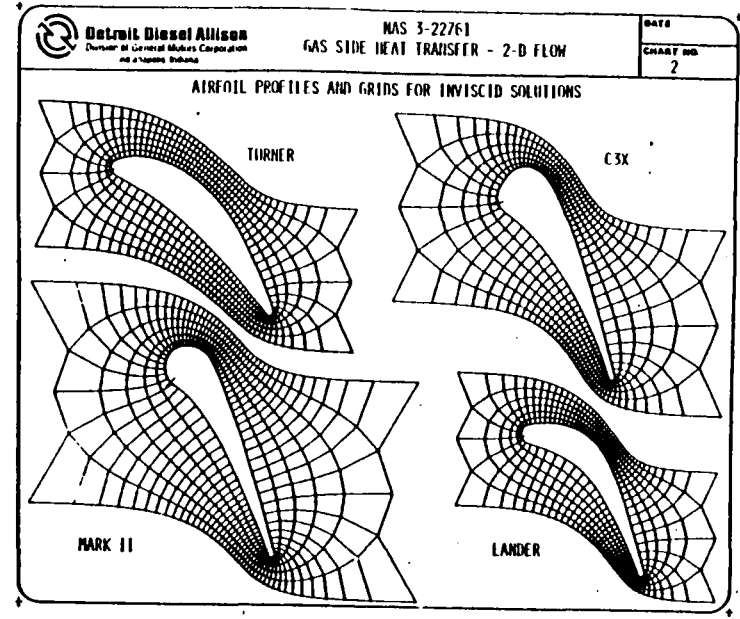
^	Refers to fluctuating quantity
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OTHER

< >	Refers to root-mean-square (RMS) value
	Absolute value

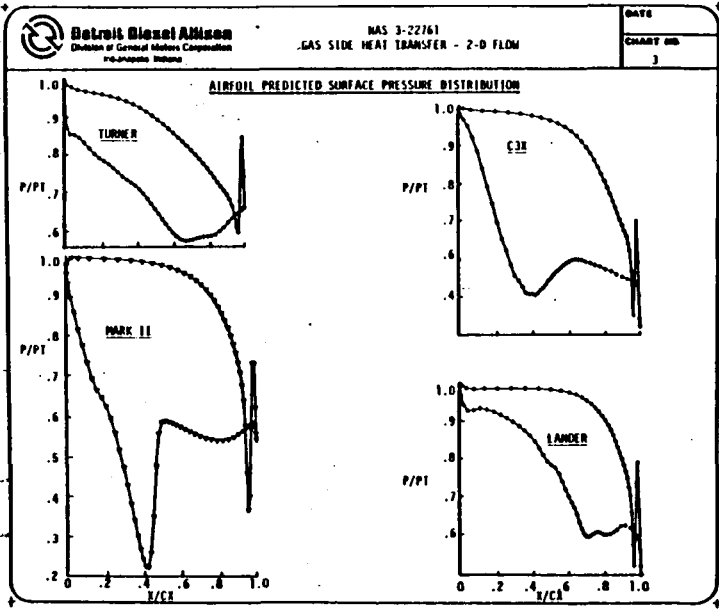


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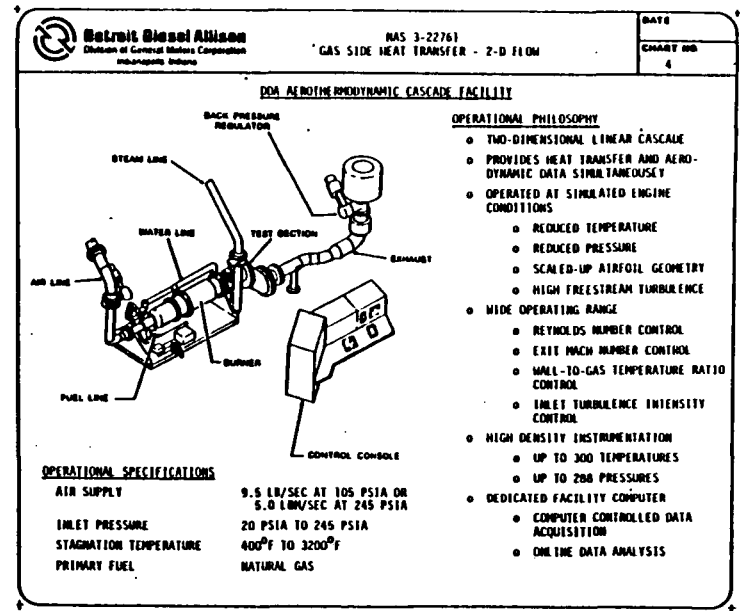


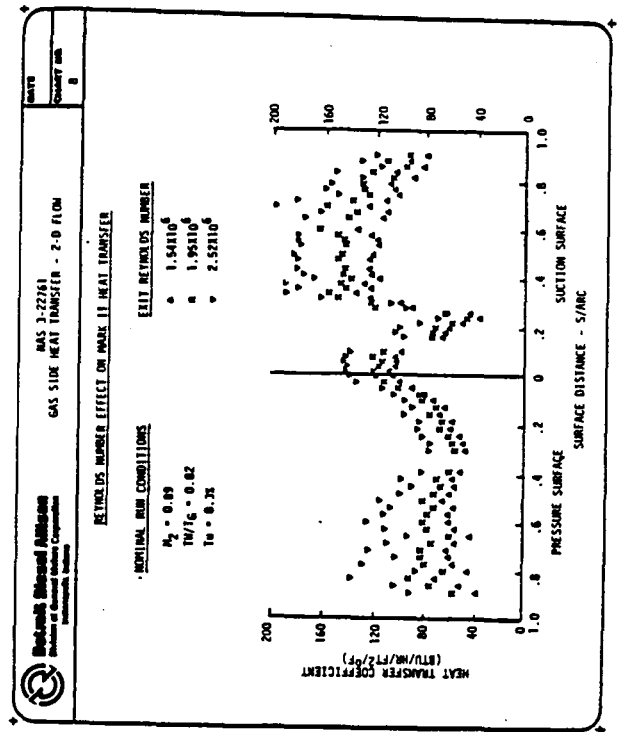
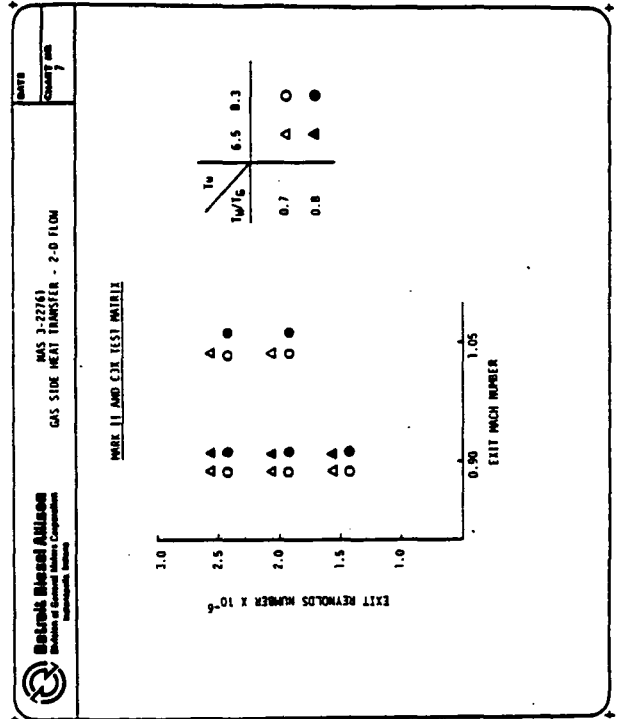
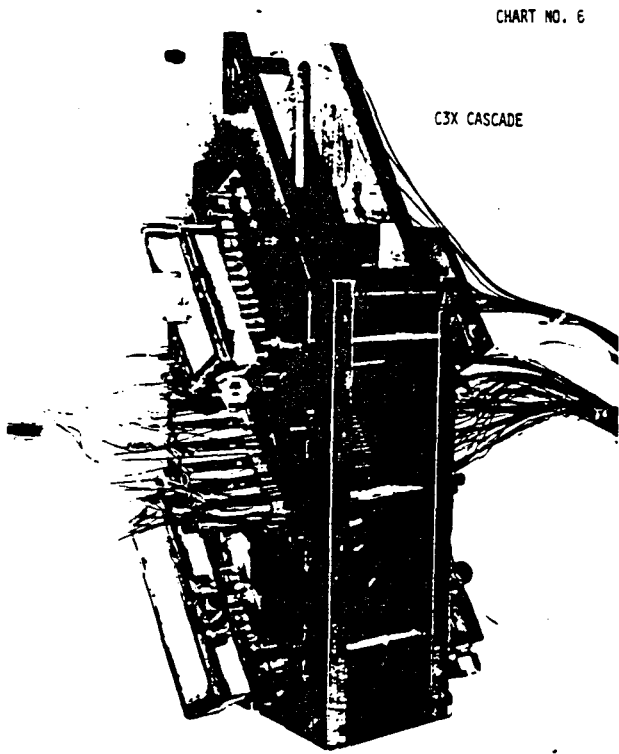
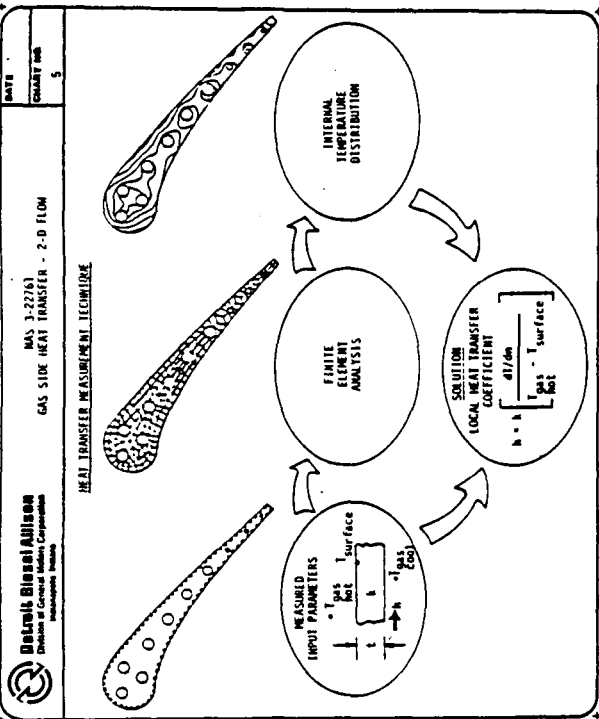
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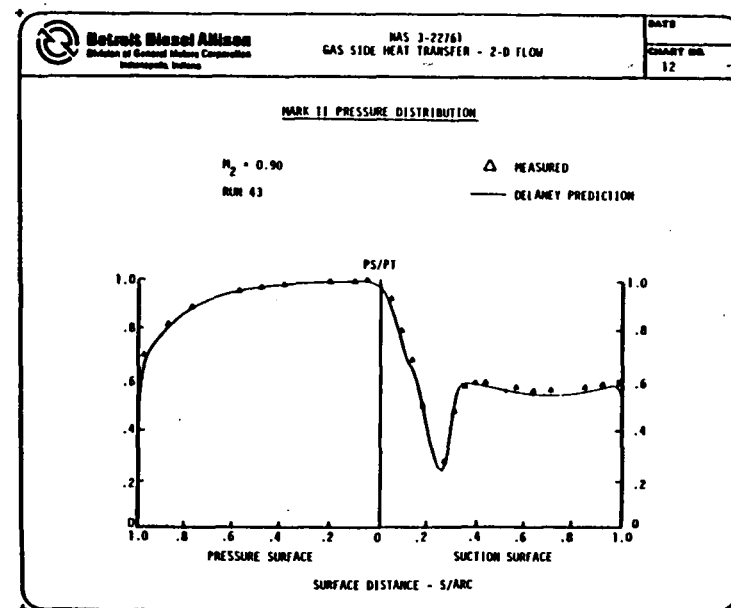
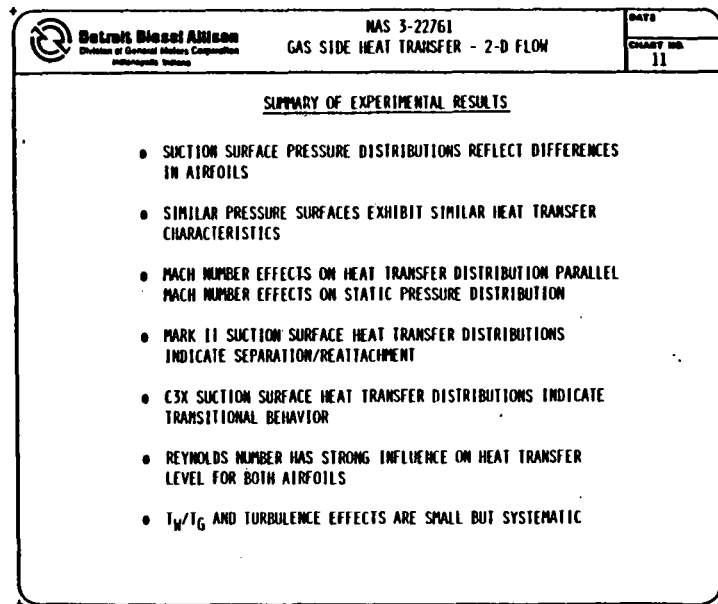
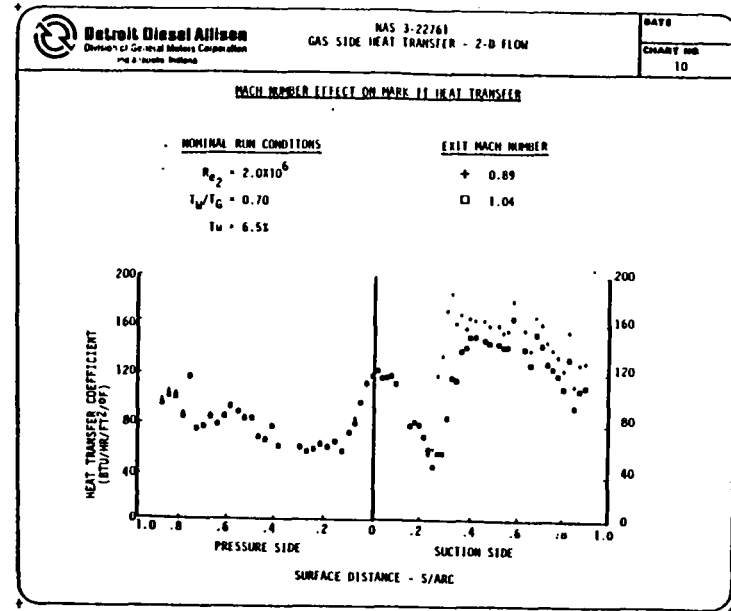
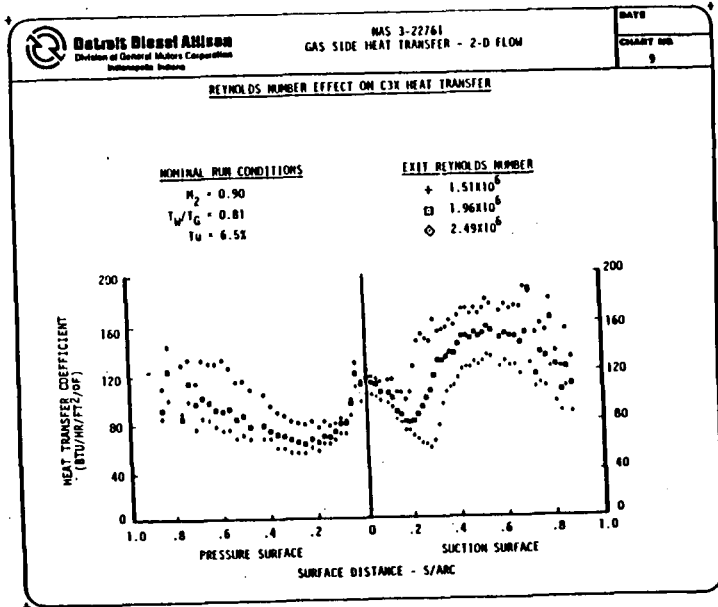
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FROM DDAD(REV. 5-75)







Decatur Diesel Allison
Division of General Motors Corporation
Warren, Michigan, Detroit

MAS 3-22761
GAS SIDE HEAT TRANSFER - 2-D FLOW

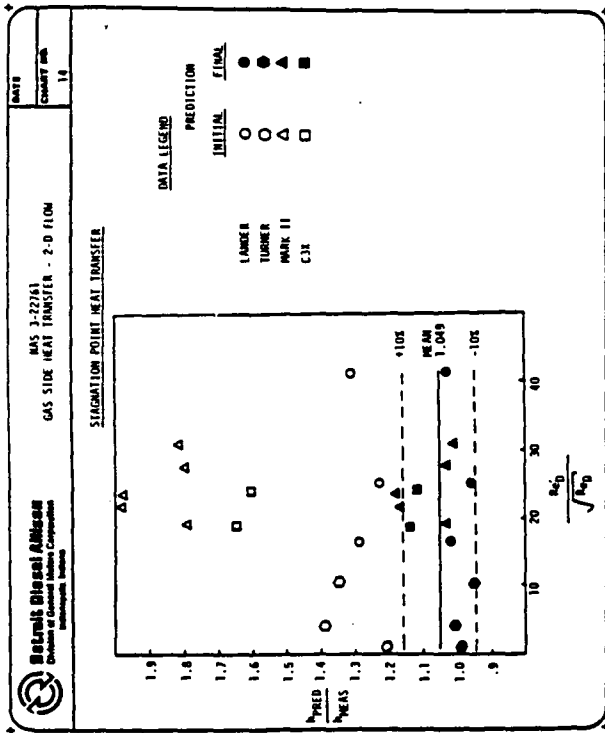
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EXTENDED SIMILARITY SOLUTION

$$\left[c \left(1 + \frac{1}{\sigma} \right) F'' \right]' + FF'' + \beta \left[G - (F')^2 \right] = 0 \quad \text{-- MOMENTUM}$$

$$\left[c \left(\frac{1}{\sigma} + \frac{1}{Pr} \frac{1}{\sigma} \right) G' \right]' + FG' - c(\sigma - 1)F'^2(F')^2 = 0 \quad \text{-- ENERGY}$$

- TERMS ① AND ② COMPRISE TURBULENT EXTENSION OF COMPRESSIBLE BOUNDARY LAYER SIMILARITY SOLUTION. (MIZAZAKI AND SPARROW 1977)
- c_m IS MODELED USING MIXING LENGTH THEORY TO ACCOUNT FOR FREE-STREAM TURBULENCE AND PRESSURE GRADIENT EFFECTS.
 - MIZAZAKI AND SPARROW
 - $c_m = 2.2 \epsilon^{-0.1} \frac{\mu}{\rho u_\infty^2}$
 - $c_m = 1 \left(\frac{Re_\infty}{Re_D} \right)^{0.1} \left(\frac{Re_\infty}{Re_D} \right)^{-0.5} \frac{\mu}{\rho u_\infty^2}$ -- ODA GENERALIZED FORM
- INITIAL PROFILES FOR STARTING BOUNDARY LAYER ANALYSIS ARE OBTAINED BY NOTING THAT:
 - $f(0) = 0$
 - $g(0) = 1$



Decatur Diesel Allison
Division of General Motors Corporation
Warren, Michigan, Detroit

MAS 3-22761
GAS SIDE HEAT TRANSFER - 2-D FLOW

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EFFECTIVE VISCOSITY CORRELATION

$$\mu_{eff} = \mu \left(1 + \gamma_L \gamma_{10}^{-1} \right) \gamma_{10}^{-1}$$

γ_L = MOLECULAR VISCOSITY
 $\gamma_L = \mu \left(1 + \frac{2}{Pr} \right) \left(\frac{\mu}{\rho u_\infty^2} \right)$ (TURBULENT VISCOSITY)
 $\gamma_{10} = \mu \left(1 + \frac{2}{Pr} \right) \left(\frac{\mu}{\rho u_\infty^2} \right)$ ("TURBULENCE" VISCOSITY)

γ_L, γ_{10} = INTERMITTENCY FUNCTIONS REFLECTING TRANSITIONAL BEHAVIOR

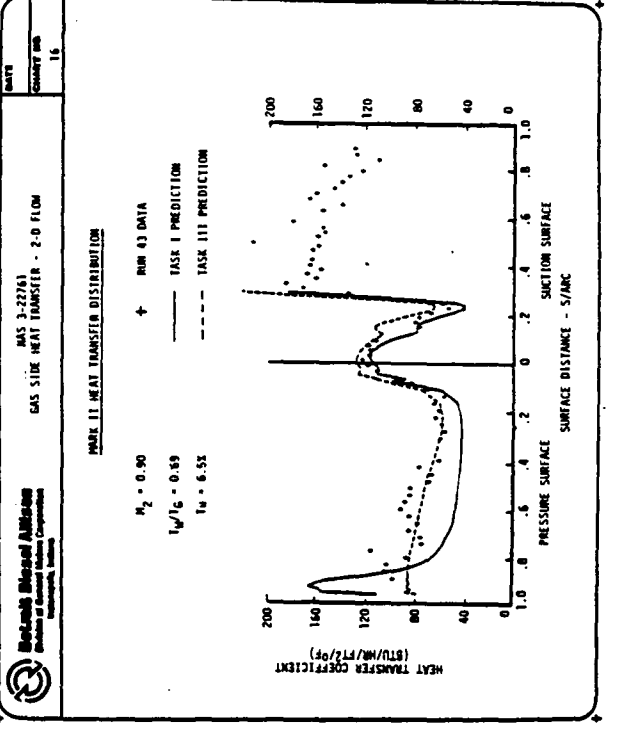
$$\left[0 \leq \gamma_L, \gamma_{10} \leq 1 \right]$$

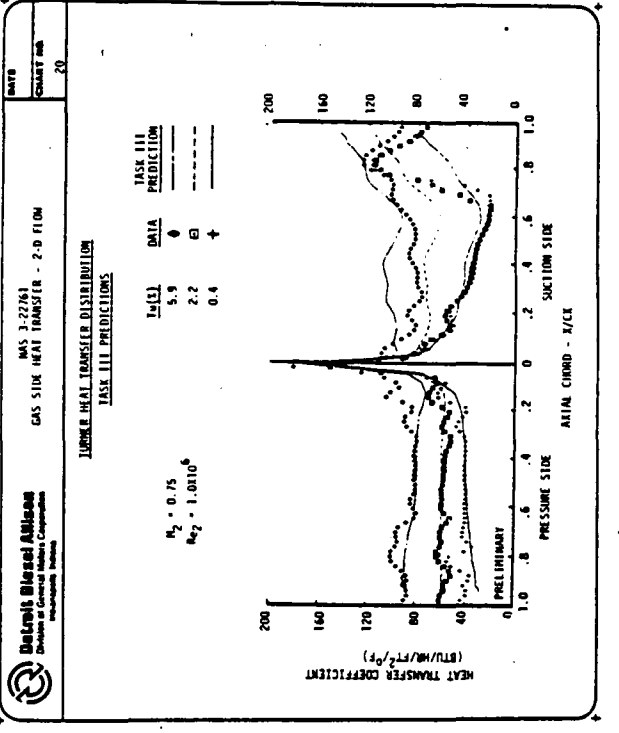
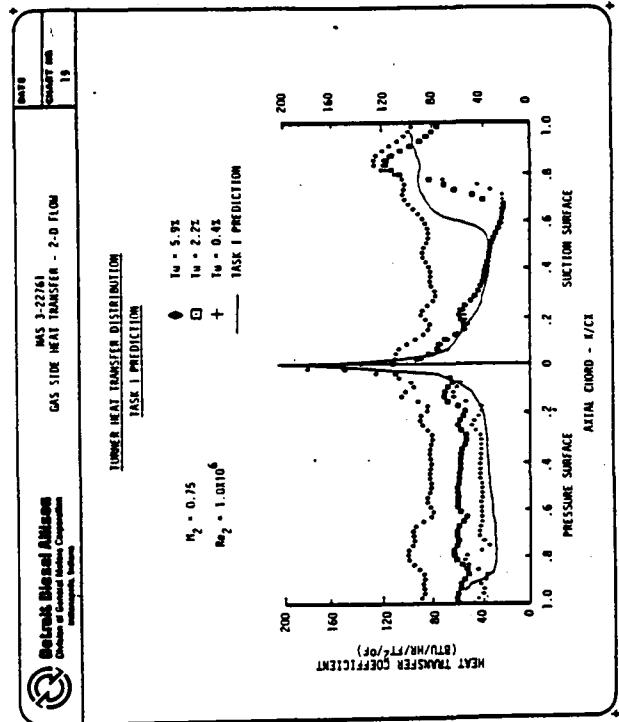
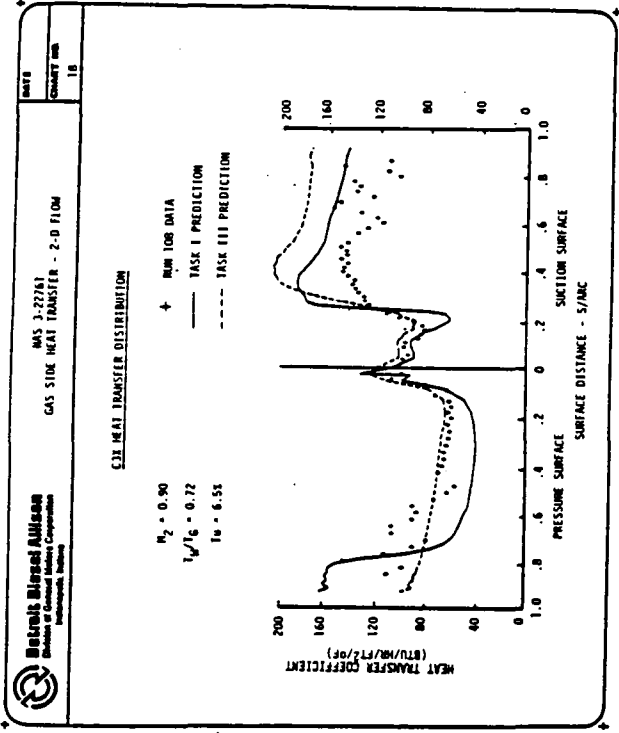
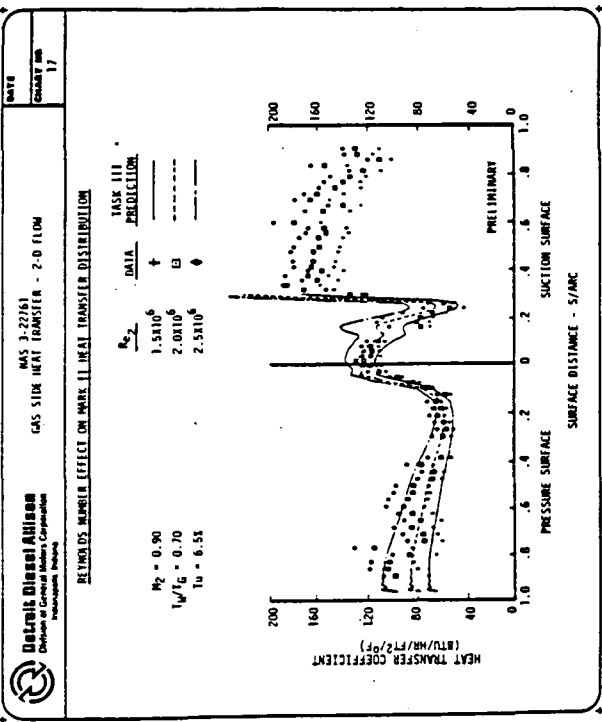
TASK I COMPUTATIONS
STANS - CHAMFORD AND GATS (1972)
MIXING LENGTH THEORY, 1, a.:

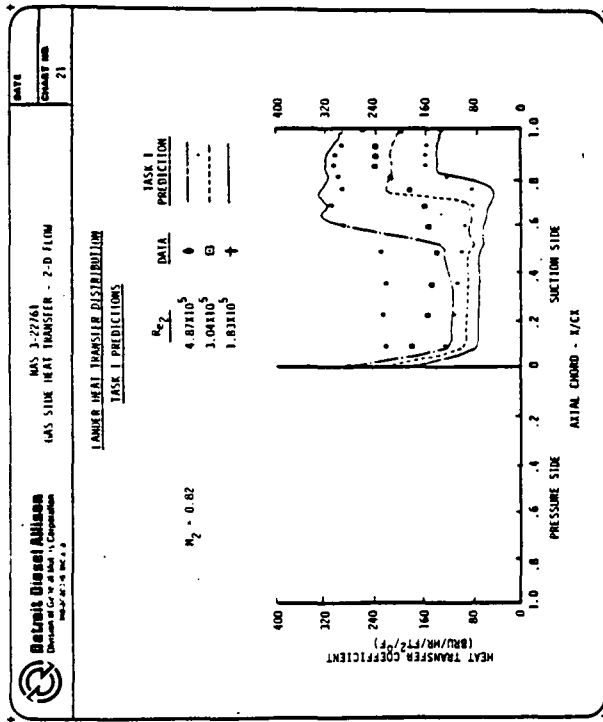
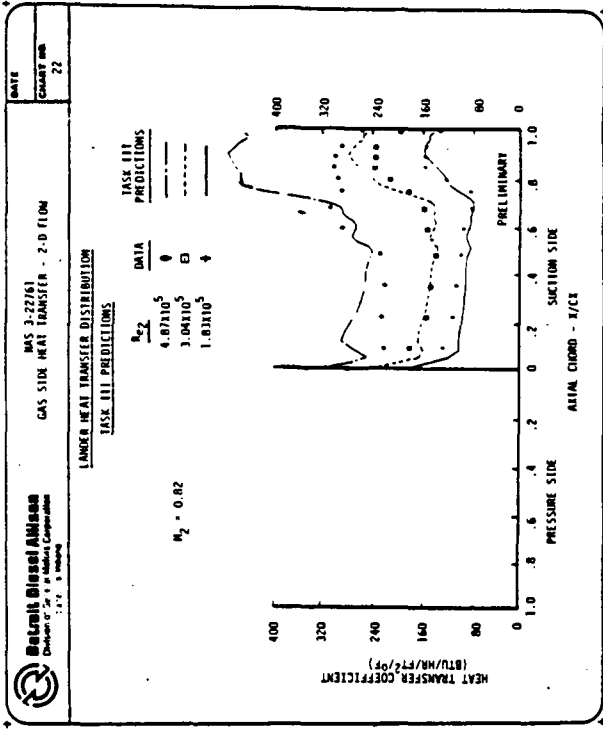
- $\gamma_{10} = 0$
- $\gamma_L = \left(\frac{0.1 Re_\infty Re_{D,cr}}{1 + Re_\infty Re_{D,cr}} \right)^{0.5}$
- $\gamma_L = \mu \left(1 + \frac{2}{Pr} \right) \left(\frac{\mu}{\rho u_\infty^2} \right)$
- $\gamma_{10} = \left(\frac{0.1 Re_\infty \sqrt{Re_{D,cr}}}{1 + Re_\infty \sqrt{Re_{D,cr}}} \right)^{0.5} \left(\frac{\mu}{\rho u_\infty^2} \right)$ (NOT MODELED)

TASK III MODELLING CONCEPTS

- $\gamma_{10} = \left(\frac{0.1 Re_\infty Re_{D,cr}}{1 + Re_\infty Re_{D,cr}} \right)^{0.5}$
- $\gamma_L = \left(\frac{0.1 Re_\infty Re_{D,cr}}{1 + Re_\infty Re_{D,cr}} \right)^{0.5} \left(\frac{\mu}{\rho u_\infty^2} \right)$
- $\gamma_L = \left(\frac{0.1 Re_\infty \sqrt{Re_{D,cr}}}{1 + Re_\infty \sqrt{Re_{D,cr}}} \right)^{0.5} \left(\frac{\mu}{\rho u_\infty^2} \right)$ (NOT MODELED)
- $\gamma_{10} = \left(\frac{0.1 Re_\infty \sqrt{Re_{D,cr}}}{1 + Re_\infty \sqrt{Re_{D,cr}}} \right)^{0.5} \left(\frac{\mu}{\rho u_\infty^2} \right)$ (NOT MODELED)







GAS FLOW ENVIRONMENT AND HEAT TRANSFER

NONROTATING 3D PROGRAM

Roy J. Schulz
The University of Tennessee Space Institute
Tullahoma, Tennessee

OBJECTIVES

The experimental contract objective is to provide a complete set of "benchmark" quality data for the flow and heat transfer within a large rectangular turning duct. These data are to be used to evaluate, and verify, three-dimensional internal viscous flow models and computational codes. The analytical contract objective is to select such a computational code and define the capabilities of this code to predict the experimental results obtained experimentally. Details of the proper code operation will be defined and improvements to the code modeling capabilities will be formulated.

The experimental and analytical efforts are being conducted under a coordinated multiphase contract. Phase one, the current work, is the study of internal flow in a large rectangular cross-sectioned, 90° bend turning duct, and is planned as a 28 month study which started in April, 1982. Phase one is divided into five tasks, numbered I through V. Future work to be performed at NASA's option includes the investigation of flow over an airfoil cascade, with and without film cooling, inside the turning radius of the duct. This future work is designated as phases two and three of the contract. Phase two, consisting of Tasks VI through XI, will consider a large scale cascade where "benchmark" data will be obtained to document the viscous flow field, pressure distribution and heat transfer phenomenon. Phase three of the contract, consisting of Tasks XII through XVI, includes mass flow injection from cascaded airfoils to document the flow field and heat transfer with film cooling of the blades.

APPROACH

Separate experimental and analytical approaches have been undertaken to attain the contract objectives. The experimental approach for Phase 1, the current work, initiated with design, fabrication, and instrumentation of a large rectangular turning duct with a 90° bend. Air flow will be drawn through the duct using an induced draft fan with variable blade pitch and variable rotation speed, to provide both a range of flow conditions in the duct and controllability for maintaining constant conditions during testing. The duct has been designed to be assembled in modules, allowing simple modifications of the duct for varying the inlet length or the wall boundary layer conditions. The duct construction is designed to allow detailed measurements to be made for the following three duct wall conditions: 1) an isothermal wall with isothermal flow, 2) an adiabatic wall with convective heat transfer by mixing between unheated surrounding flow, and 3) an isothermal wall with heat transfer from a uniformly hot inlet flow. Measurements for all three conditions will be made at two bulk Reynolds numbers and different inlet lengths to provide both laminar and fully turbulent boundary layer flows approaching the duct turn. The flow velocities for both Reynolds numbers will be low enough to remain well within the incompressible Mach number range so that only thermally induced density gradients will be encountered.

The primary instrumentation being assembled for the flow measurements is a three-dimensional, vector laser velocimeter (LV). The LV will use two colors and Bragg diffraction beam splitting/frequency shifting to separate the three simultaneous, orthogonal, vector velocity components. The LV signal processors determine digital values of velocity for the seeded flows from particle crossing the laser beam generated probe volume. To simplify and speed up digital data acquisition, the LV processors are built around an S-100 bus Z-80 microprocessor, which provides the additional advantage of on-line, near-real time data reduction. This on-line data reduction capability will be used to assess the adequacy and precision of the data as it is acquired and recorded for more complex, off-line detailed analysis. To help qualify the measurements as "benchmark" data, the LV measurements will be compared with both pitot probe and hot wire anemometer measurements for flow conditions which permit these comparisons.

The analytical approach initiated with a search for candidate state-of-the-art numerical solution procedures for internal, three-dimensional viscous flows. The two candidate codes selected were the P. D. Thomas Beam-Warming code and the Briley-McDonald "MINT" code. With both codes and their available documentation obtained, a comparison of them will be made in terms of user orientation, documentation, numerics, physical modeling, accuracy, grid sensitivity, boundary conditions, formulation, CPU time and ease of extension to future problems. Initial results will be obtained for calibration purposes and compared to published results, and with flows visualized and measured with a 2-D LV system in a 1/3 scale duct. Then numerical convergence properties and grid refinement techniques will be studied, both by uniform refinement and by local clustering of the mesh points. After the effects of discretization error have been estimated the code predictions will be compared with full scale experimental data as it is obtained, for laminar flow and turbulent flow to evaluate available turbulence models. Finally, heat transfer modeling will be evaluated, and the best of the codes selected for a detailed comparison with the experimental data. This detailed comparison will determine the accuracy of the code's calculated flow field and generate recommendations and formulations for code improvement.

CURRENT RESULTS AND PLANS

Both the experimental and analytical phases are currently underway. The experimental phase has involved test facility and instrumentation design and fabrication. The turning duct design has been approved by NASA program monitors and is essentially an induced draft wind tunnel with a 90° bend test section. Nominal bulk flow velocities of 6.096 m/sec (20 ft/sec) and 60.96 m/sec (200 ft/sec) and different straight inlet lengths were selected for the low and high Reynolds number cases to provide moderately thick laminar and turbulent boundary layers entering the bend. The contour of the inlet transition, or bellmouth, was designed by numerical calculations to provide smooth acceleration to the inlet section. The flow rates corresponding to the selected velocity were used to specify an induced draft fan for the duct exit, which in turn, established the diffuser configuration. The 90° curve section of the duct will be assembled from a series of flanged interchangeable modules. Additionally, interchangeable instrument access modules provide access for probing the flow with the LV and with the pitot and hot wire probes. The duct will

be insulated for the adiabatic wall test cases, and water jackets will be added to establish the isothermal wall test cases. The LV window module allows optical access from all four walls of the duct, with an unobstructed field of view so that the LV probe volume can be traversed from wall to wall. The window segments are thin, optical flats which are only 1.27 cm (1/2 in) wide along the inside wall. The windows were designed to minimize deviations from wall curvature on the convex and concave duct sidewalls. This test facility is presently in the materials acquisition and mechanical fabrication stage.

The LV optical assembly will be mounted on a box beam structure which fits around the duct. The LV scans the duct cross-plane and cross-stream coordinate directions by activating an LV microprocessor-controlled mill bed, to which the box beam is mounted. The optical arrangement allows the three pairs of laser beams to cross coincident within ± 0.5 mm, as well as forward scatter light collection for greatest signal to noise ratio. The signal to noise ratio and processor accuracy are also improved by mixing the collected signals with accurately known oscillators that downbeat the signals to lower frequencies. The reduced signal frequencies eliminate the effects of small deviations centered on very high RF carrier frequencies from affecting the measurements. Hardware is being acquired and software is being developed for improved on-line graphics for data reduction and presentation which allows assessment of the validity and precision of the LV measurements as they are being acquired. This significantly improves the established levels of statistical confidence and also helps assure that all critical physical flow features are being measured. LV system development is moving into the shakedown stage, which is being initiated by evaluating the flow in a 1/3 scale model duct. An evaluation of LV compatibility with the windowed optical access module is also planned before LV and intrusive probe comparison testing is initiated.

The analytical effort is also well underway. One of the selected computational codes, the P. D. Thomas code, was received for evaluation, and requests have been made for a recent version of the Briley-McDonald "MINT" code. The P. D. Thomas code has extensive documentation and has been adapted to the experimental geometry of the 90° bend duct. Numerically converged results have been obtained for a published, laminar flow case. Also, the initial results were compared to published solutions of the "MINT" code for the same flow, which includes duct geometry and Reynolds number, although the flow inlet Mach number used for the Thomas code was greater. With either computer code discrepancies between computed solutions and experimental measurements for both laminar and turbulent flows have been reported. The source of the discrepancy seems to be discretization error, and derive a set of approximate solution errors. Also, solutions of both codes will be compared with flow visualization in a 1/3 scale duct, to help identify problem areas with both code results.



**GAS FLOW ENVIRONMENT AND HEAT TRANSFER -
NONROTATING 3D PROGRAM**

NASA CONTRACT NAS3-23278

Roy J. Schulz, Ph.D., P.E.

OBJECTIVES

- PROVIDE "BENCHMARK" DATA DELINEATING 3-D VISCOUS FLOW WITH HEAT TRANSFER ON A LARGE, RECTANGULAR, 90° BEND TURNING DUCT

PHASE

1. FLOW DEVELOPMENT IN THE BEND WITH DIFFERENT INLET BOUNDARY LAYERS AND WALL HEAT FLUX DISTRIBUTIONS (TASK I-V)
 2. FLOW DEVELOPMENT IN THE BEND WITH AN IMBEDDED CASCADE BLADE SYSTEM FOR DIFFERENT INLET BOUNDARY LAYERS AND WALL HEAT FLUX DISTRIBUTION (TASKS VI-XI)
 3. FLOW DEVELOPMENT IN THE BEND WITH AN IMBEDDED CASCADE BLADE SYSTEM USING AIR INJECTED THROUGH THE BLADES TO SIMULATE FILM COOLING FOR DIFFERENT INLET BOUNDARY LAYERS AND WALL HEAT FLUX DISTRIBUTIONS (TASK XII - XVI)
- SELECT, EVALUATE, MODIFY AND/OR DEVELOP A STATE-OF-THE-ART 3-D VISCOUS FLOW COMPUTER CODE BY CONFRONTATION WITH EXPERIMENTAL DATA (SAME 3 PHASES)
 - VALIDATE CODES' CAPABILITY/ADAPTABILITY
 - ADD, IF NECESSARY, ENERGY TRANSFER/CONSERVATION EQUATION
 - EVALUATE TURBULENT TRANSPORT MODELS FOR THIS CLASS OF FLOWS
 - DEFINE MESH ESTABLISHMENT AND RESOLUTION REQUIRED FOR ACCURATE COMPUTATION OF THIS CLASS OF FLOWS.

EXPERIMENTAL APPROACH

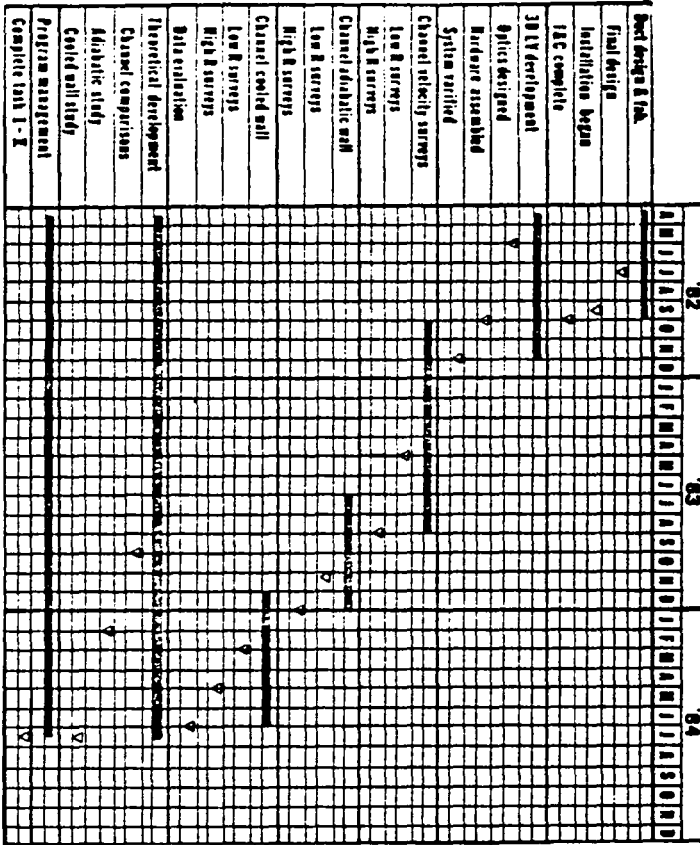
- BUILD FACILITY AND INSTRUMENTATION
 - MODIFIABLE CURVED DUCT
 - VARIABLE AIR FLOW
 - SEGMENTED CONSTRUCTION
 - LASER VELOCIMETER INSTRUMENTATION
 - SIMULTANEOUS 3 COMPONENT DETERMINATION
 - BRAGG SYSTEM (VELOCITY VECTOR MEASUREMENTS)
 - MICROPROCESSOR BASED SYSTEM - OUTLINE STATISTICAL DATA DETERMINATION
 - VALIDATED AGAINST PITOT AND HOT WIRE SYSTEM
- DUCT EXPERIMENTS - TO FACILITATE ANALYTICAL COMPARISONS
 - UNHEATED FLOW - 2 ENTRANCE CONDITIONS
 - MIXING OF HOT AND COLD STREAMS WITH ADIABATIC WALL
 - HOT FLOW WITH ISOTHERMAL WALL
- CASCADE EXPERIMENT - BLADES INSTALLED IN DUCT BEND
 - SAME SERIES AS DUCT EXPERIMENTS
 - REPEAT SERIES WITH SIMULATED FILM COOLING

ANALYTICAL APPROACH

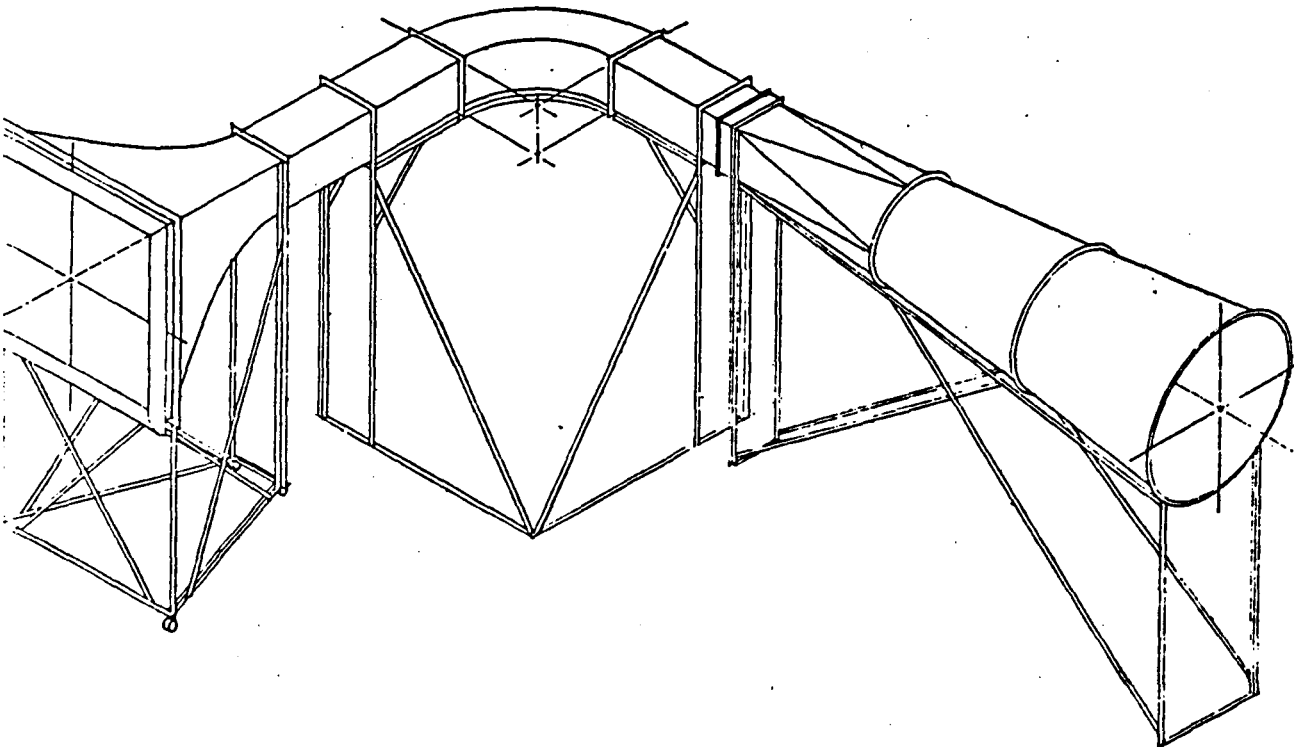
- COMPARE APPLICABLE 3-D CODES WITH ADEQUATE DOCUMENTATION
 - LAMINAR FLOW CALCULATIONS WITHOUT HEAT TRANSFER
 - ADAPT CODE TO PROBLEM - BOUNDARY AND INITIAL CONDITIONS
 - COMPARE WITH FLOW VISUALIZATION RESULTS
 - COMPARE WITH PUBLISHED RESULTS FOR SIMILAR CONFIGURATIONS
- PRELIMINARY COMPARISON WITH DATA (INCLUDING TURBULENT FLOW AND HEAT TRANSFER), AS IT IS OBTAINED
 - COMPARE CODES FOR RESOLUTION WITH EQUIVALENT GRIDS, COMPUTATIONAL TIME, STORAGE, EASE OF IMPLEMENTATION AND EASE OF MODIFICATION
 - SELECT CODE FOR DETAILED COMPARISON WITH DATA
- DETAILED COMPARISONS WITH DATA
 - FORMULATE IMPROVEMENTS
 - EVALUATE SENSITIVITY TO GRID SPACING AND TIME STEP SELECTION

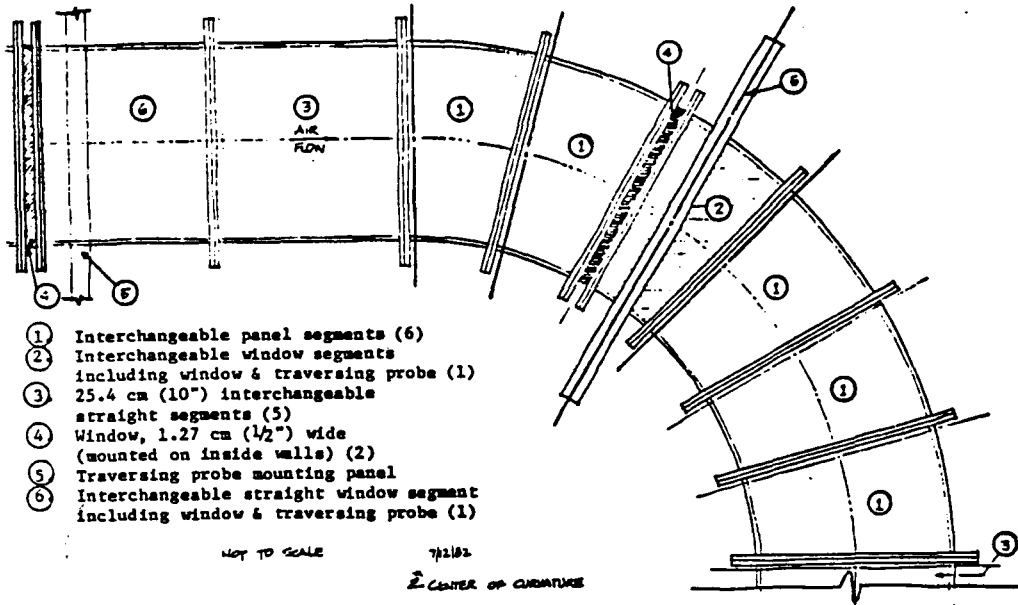
SCHEDULE

Phase 1



90° BEND TURNING DUCT

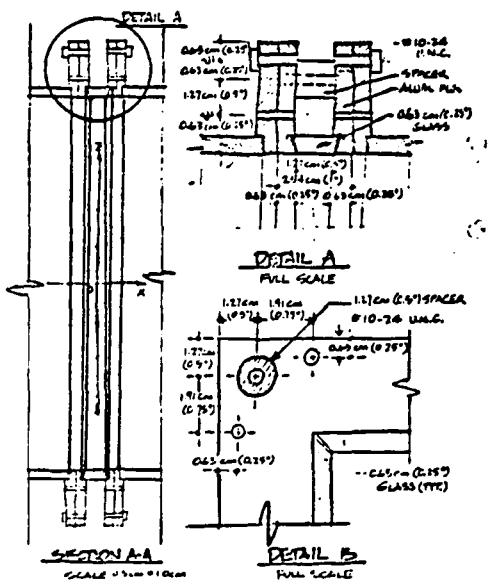
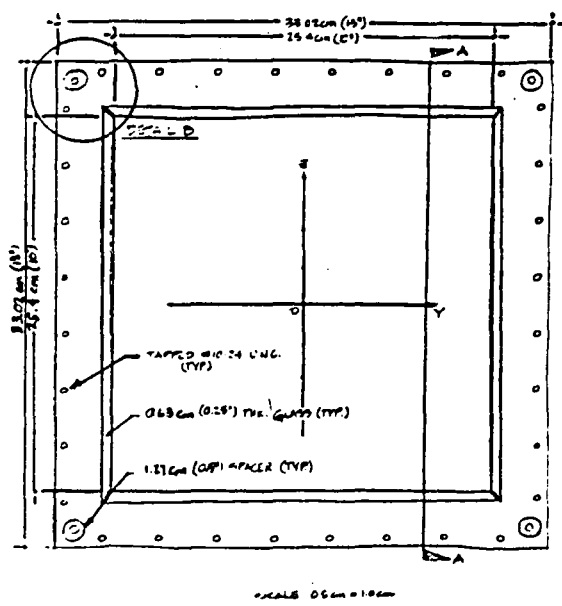




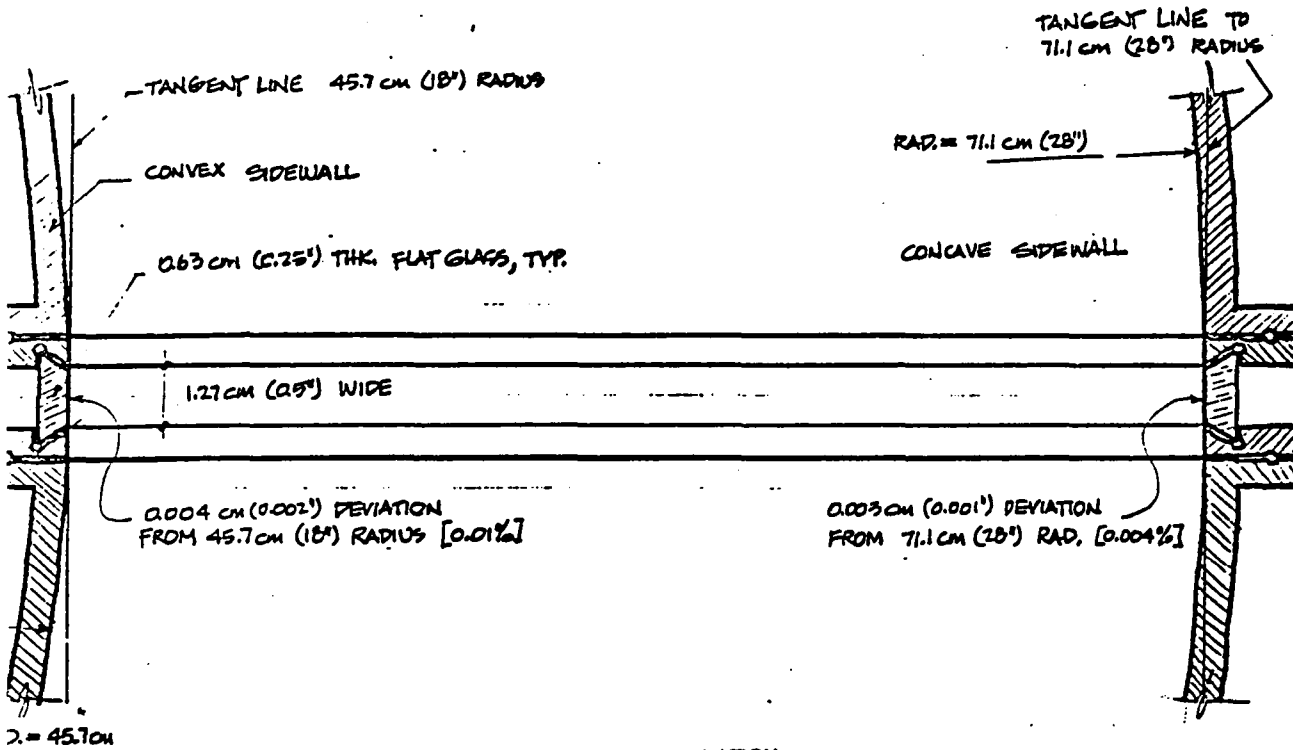
- ① Interchangeable panel segments (6)
- ② Interchangeable window segments including window & traversing probe (1)
- ③ 25.4 cm (10") interchangeable straight segments (5)
- ④ Window, 1.27 cm (1/2") wide (mounted on inside walls) (2)
- ⑤ Traversing probe mounting panel
- ⑥ Interchangeable straight window segment including window & traversing probe (1)

NOT TO SCALE
 7/2/82
 CENTER OF CURVATURE

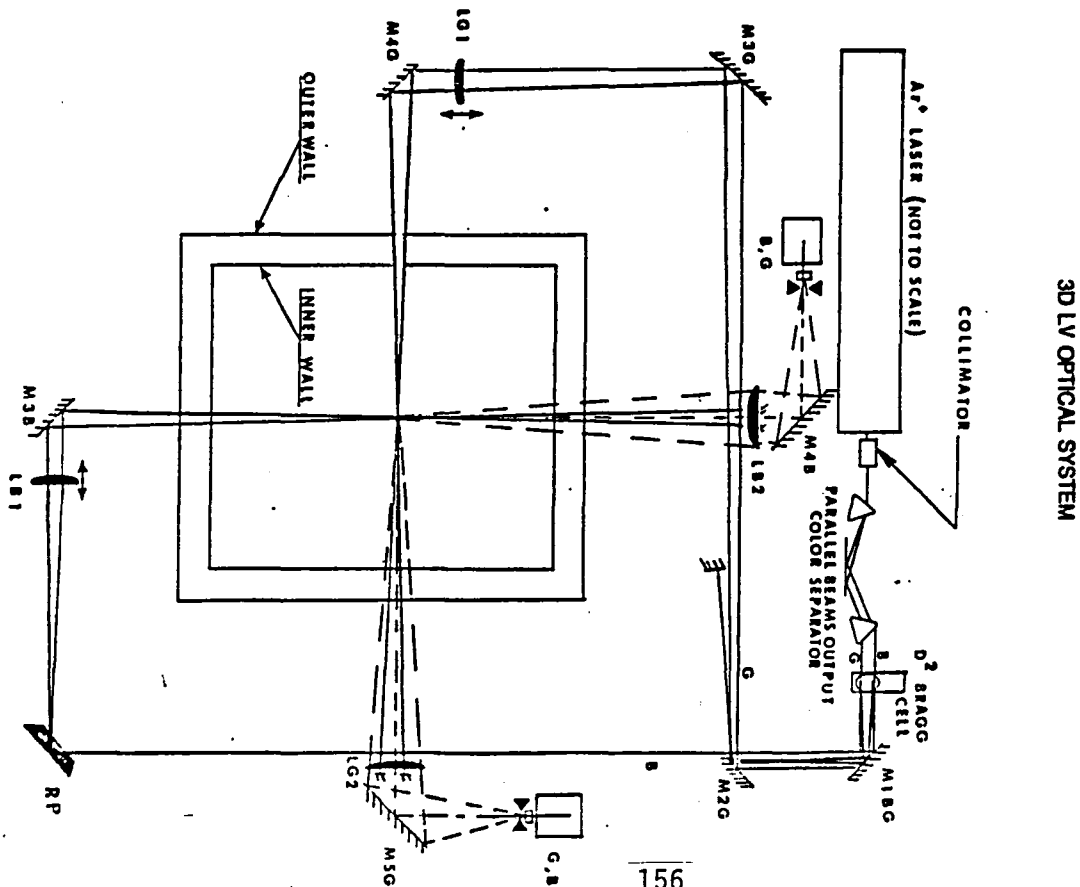
SEGMENTATION OF BEND



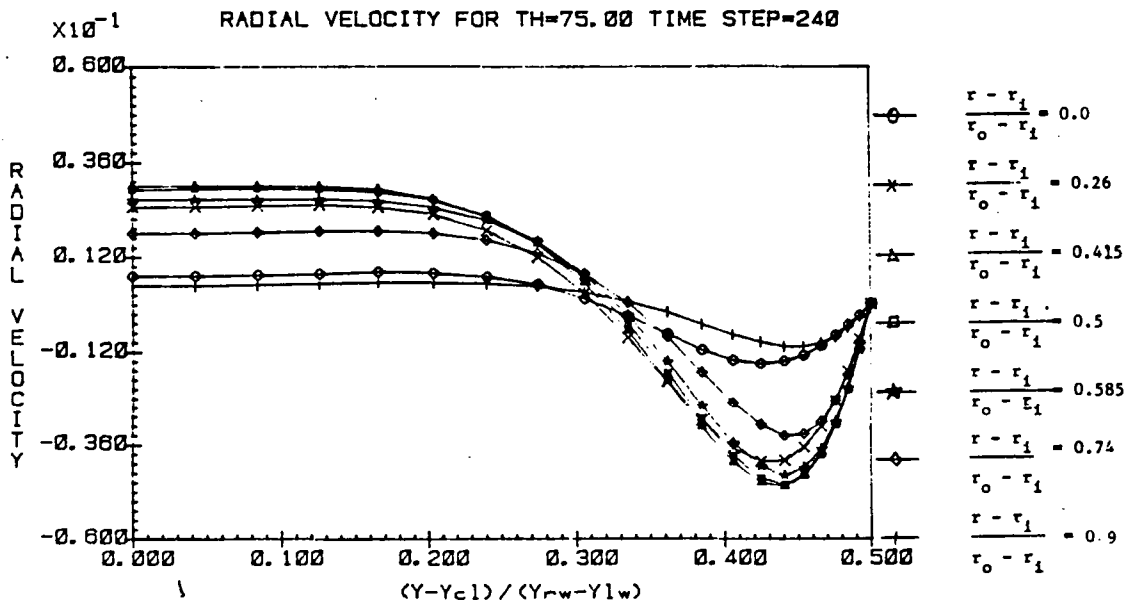
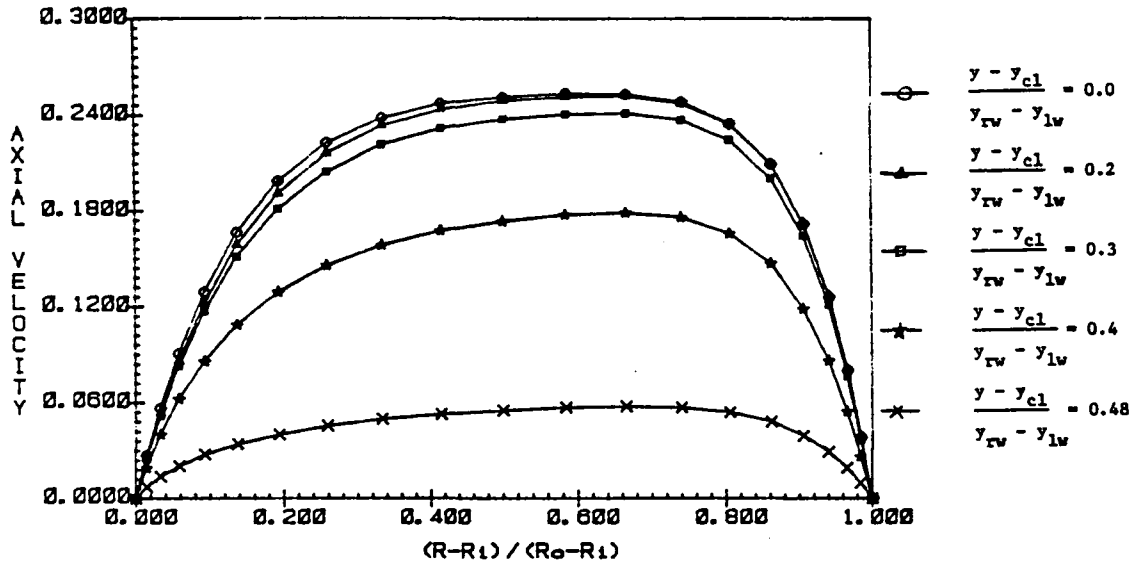
WINDOW MODULE



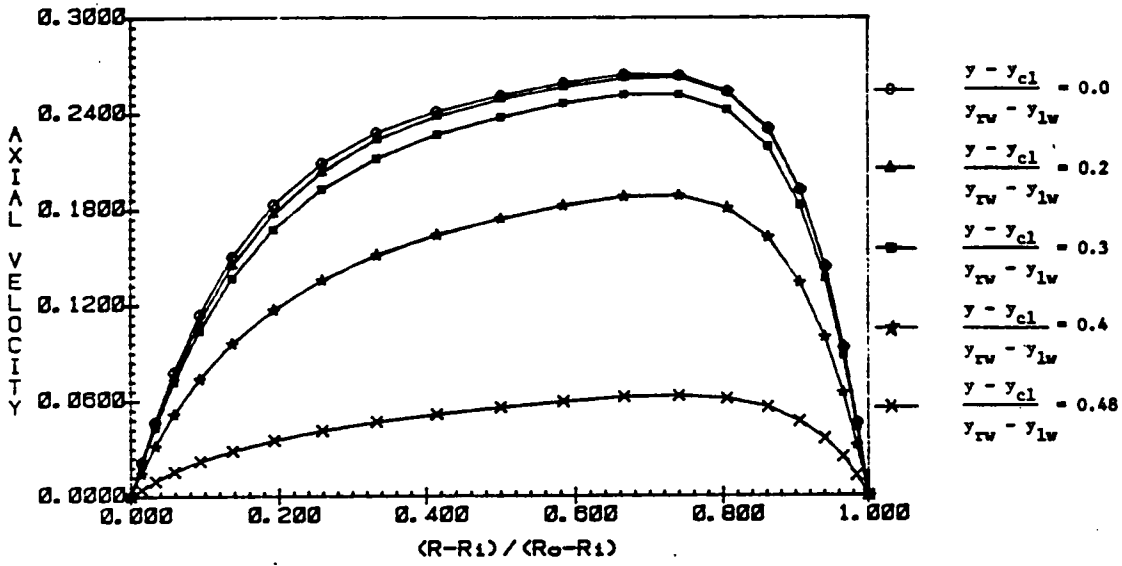
FLAT WINDOW INSTALLATION



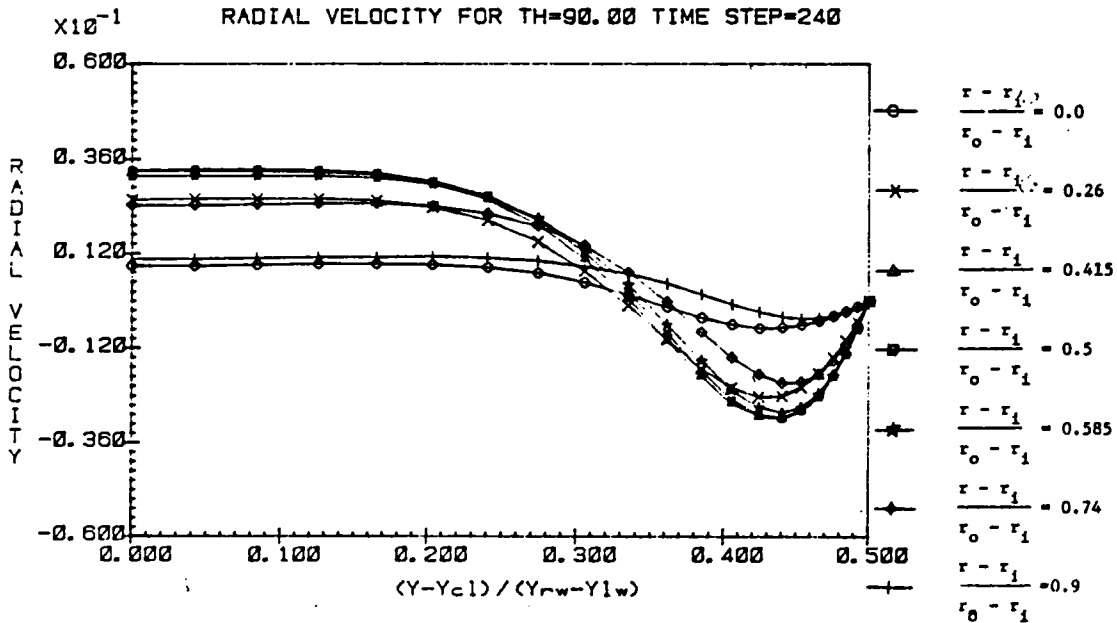
VELOCITY PLOT FOR TH=75.00 TIME STEP=240



VELOCITY PLOT FOR TH=90.00 TIME STEP=240



RADIAL VELOCITY FOR TH=90.00 TIME STEP=240



STATUS SUMMARY

EXPERIMENTAL

- DUCT DESIGN APPROVED - DETAILED DRAWINGS AND FABRICATION STAGES
- LV OPTICAL DESIGN BREADBOARDED - PREPARING FOR SHAKEDOWN TESTS (1/3 SCALE DUCT)
- LV PROCESSOR MODIFICATIONS UNDERWAY
 - 500 MHz CLOCK
 - 3D SIMULTANEITY
 - COMPUTER CONTROLLED POSITIONING
 - ON LINE DATA GRAPHICS

ANALYTICAL

- CODES SELECTED FOR EVALUATION
 - P. D. THOMAS
 - "MINT"
- P. D. THOMAS CODE SET UP/OPERATIONAL FOR FLOW IN 90° BEND DUCT
 - RESULTS AGREE WITH PUBLISHED LITERATURE
- NOT YET ABLE TO ACQUIRE "MINT"

FUTURE PLANS

Phase 1

EXPERIMENTAL

- COMPLETE DUCT FABRICATION, INSTALLATION AND SHAKEDOWN
- INITIATE 2-D LV MEASUREMENTS ON 1/3 SCALE DUCT TO DEVELOP ON-LINE DATA REDUCTION AND TRAVERSING SYSTEM SOFTWARE
- COMPLETE 3-D LV DEVELOPMENT AND SHAKEDOWN, ASCERTAINING MEASUREMENT UNCERTAINTIES AND COMPARABLE ACCURACY TO PROBES
- INITIATE ISOTHERMAL FLOW TESTS

ANALYTICAL

- OBTAIN "MINT" AND INPUT BOUNDARY AND INITIAL CONDITIONS FOR DUCT WITH LAMINAR FLOW TO COMPARE WITH PUBLISHED RESULTS.
- INITIATE P. D. THOMAS AND "MINT" CODE COMPARISON WITH LAMINAR FLOW VISUALIZATION IN 1/3 SCALE DUCT
- EVALUATE/MODIFY TURBULENCE MODEL AND PREDICT TURBULENT FLOW WITH BOTH CODES.

JET ARRAY IMPINGEMENT FLOW DISTRIBUTIONS AND
HEAT TRANSFER CHARACTERISTICS - EFFECTS OF
INITIAL CROSSFLOW AND NONUNIFORM ARRAY GEOMETRY

L.W. Florschuetz and D.E. Metzger
Department of Mechanical and Aerospace Engineering
Arizona State University
Tempe, Arizona 85287

SUMMARY

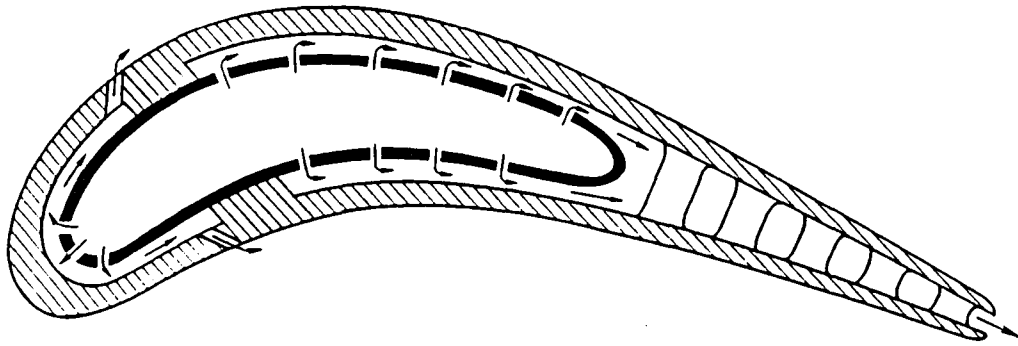
The work reported on is divided into two major parts, each of which represents an extension of work completed in earlier phases of an overall investigation. The overall investigation was directed toward the determination of flow distributions and heat transfer characteristics for two-dimensional arrays of circular air jets impinging on a surface parallel to the jet orifice plate. The configurations considered were intended to model those of interest in current and contemplated gas turbine airfoil midchord cooling applications. The geometry of the airfoil applications considered dictates that all of the jet flow, after impingement, exit in the chordwise (i.e., streamwise) direction toward the trailing edge. The accumulated flow from upstream jet rows in the array acts as a crossflow to downstream rows. In some cooling schemes an initial crossflow arising from air used to cool the leading edge approaches the midchord jet array. The temperature of this initial crossflow air can be several hundred degrees higher than the cooling air introduced to the jet array.

The early work in the study dealt with arrays of uniform geometries not subject to an initial crossflow. These arrays had streamwise hole spacings of 5, 10, and 15 hole diameters, spanwise hole spacings of 4, 6, and 8 diameters, and jet exit plane-to-impingement surface spacings (channel heights) of 1, 2, and 3 hole diameters, with 10 spanwise rows of holes. Spanwise averaged heat transfer coefficients, resolved in the streamwise direction, were measured and correlated in terms of individual spanwise row jet and crossflow velocities, and in terms of the geometric parameters. These results were reported in detail in two previously published NASA reports.

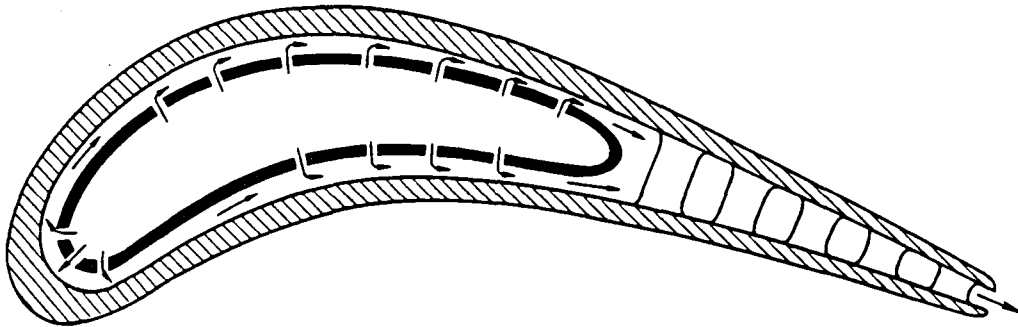
Part I of the present study deals with experimental results for the effects of an initial crossflow on both flow distributions and heat transfer characteristics for a number of the prior uniform array geometries. Heat

transfer coefficients and adiabatic wall temperatures resolved to one streamwise hole spacing were determined for ratios of the initial crossflow-to-total jet flow rate ranging from zero to unity. The adiabatic wall temperatures depend on the relative flow rates and relative characteristic temperatures of both the jet air and the initial crossflow air, as well as on the geometric parameters. Both Nusselt number profiles and dimensionless adiabatic wall temperature ("effectiveness") profiles were determined and considered in relation to the flow and geometric parameters. For some conditions "effectiveness" profiles cover nearly the entire range between zero and unity, and Nusselt numbers at upstream rows are reduced significantly compared with zero initial crossflow values, even for initial crossflow-to-total jet flow ratios as small as 0.2. Special test results which showed a significant reduction of jet orifice discharge coefficients owing to the effect of a confined crossflow were obtained, and a flow distribution model which incorporates those effects was developed.

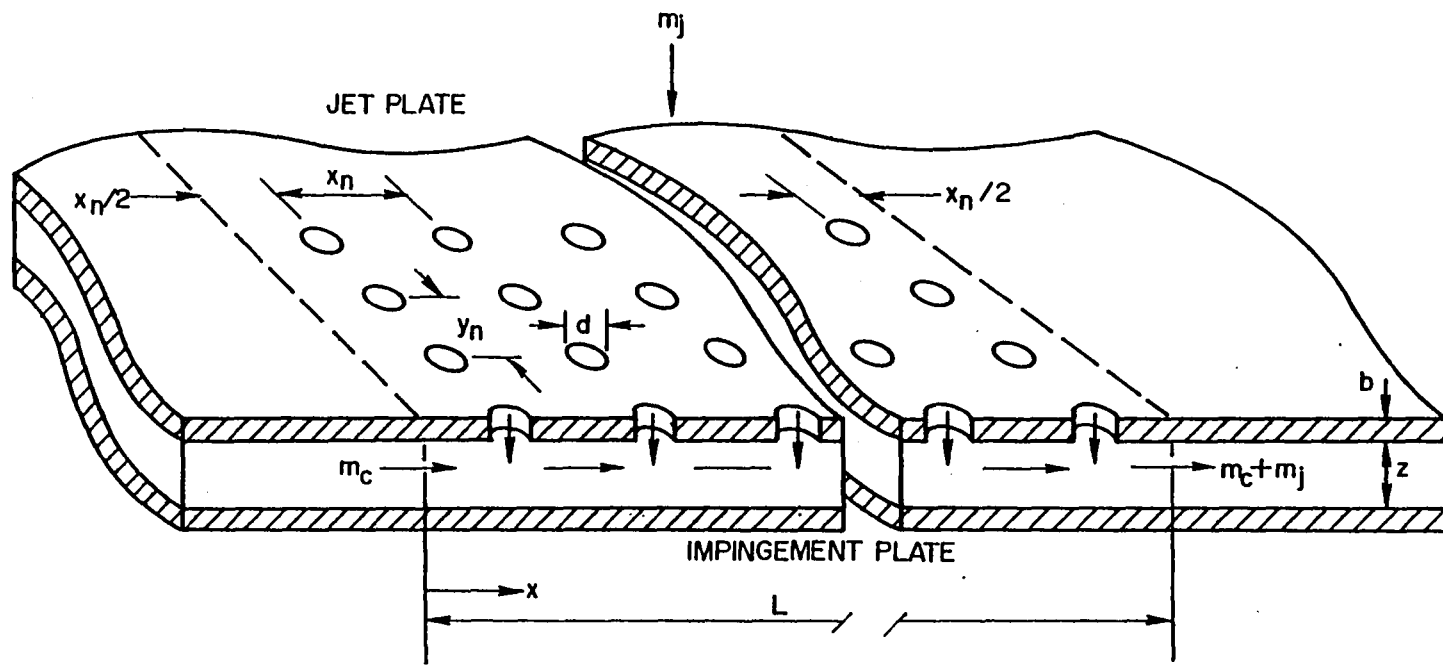
Part II deals with experimental results for the effects of nonuniform array geometries on flow distributions and heat transfer characteristics for noninitial crossflow configurations. The nonuniform arrays are comprised of two different regions each of which has a uniform geometry. Either hole spacing or hole diameter has a different value in the two regions. The previously developed flow distribution model for uniform arrays was extended to nonuniform arrays and validated by comparison with the measured flow distributions. The validated flow distribution model was then employed to compare the nonuniform array streamwise resolved heat transfer coefficient data with the previously reported uniform array data and with the previously developed correlation based on the uniform array data. It was found that the uniform array results can, in general, serve as a satisfactory basis from which to predict heat transfer coefficients at individual rows of nonuniform arrays. However, significant differences were observed in some cases over the first one or two rows downstream of the geometric transition line of the nonuniform array. For practical purposes the "entrance" or "adjustment" length for a downstream region could be considered as requiring from zero to at most two jet rows, depending on the particular case.



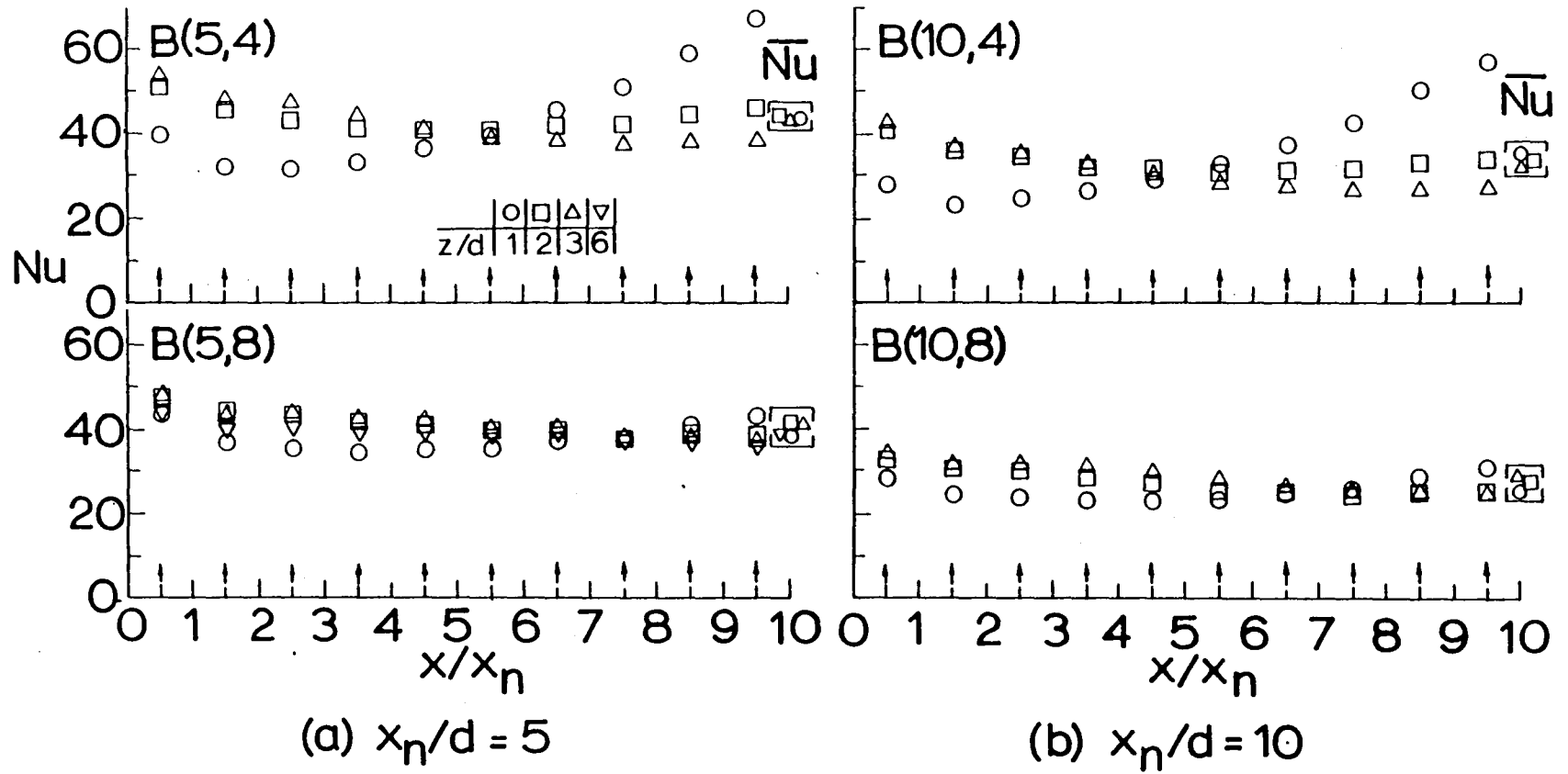
Impingement cooled airfoil - midchord arrays not subject to initial crossflow.



Impingement cooled airfoil - midchord jet arrays subject to initial crossflow.



Initial crossflow basic test model geometry and nomenclature.



Typical chordwise Nusselt number profiles without initial crossflow

**Geometric Parameters and Mean Discharge Coefficients
for Jet Plates Tested with Initial Crossflow.**

Jet Plate B($x_n/d, y_n/d$)I	A_o^*	d and b (cm)	N_s	N_s'	C_D
B(5,4)I(S)	0.0393	0.254	12	18	0.85
B(5,8)I	0.0196		6	9	0.80
B(10,4)I	0.0196	0.127	24	36	0.76
B(10,8)I	0.0098		12	18	0.76

Channel heights, $(z/d) = 1, 2, \text{ and } 3$

Fixed Parameters:

Channel width (span), $w = 18.3 \text{ cm}$.

Heat transfer test plate width, 12.2 cm

Heat transfer test plate length, 39.4 cm

Overall channel length, 43.2 cm

Initial crossflow channel length, 26.0 cm

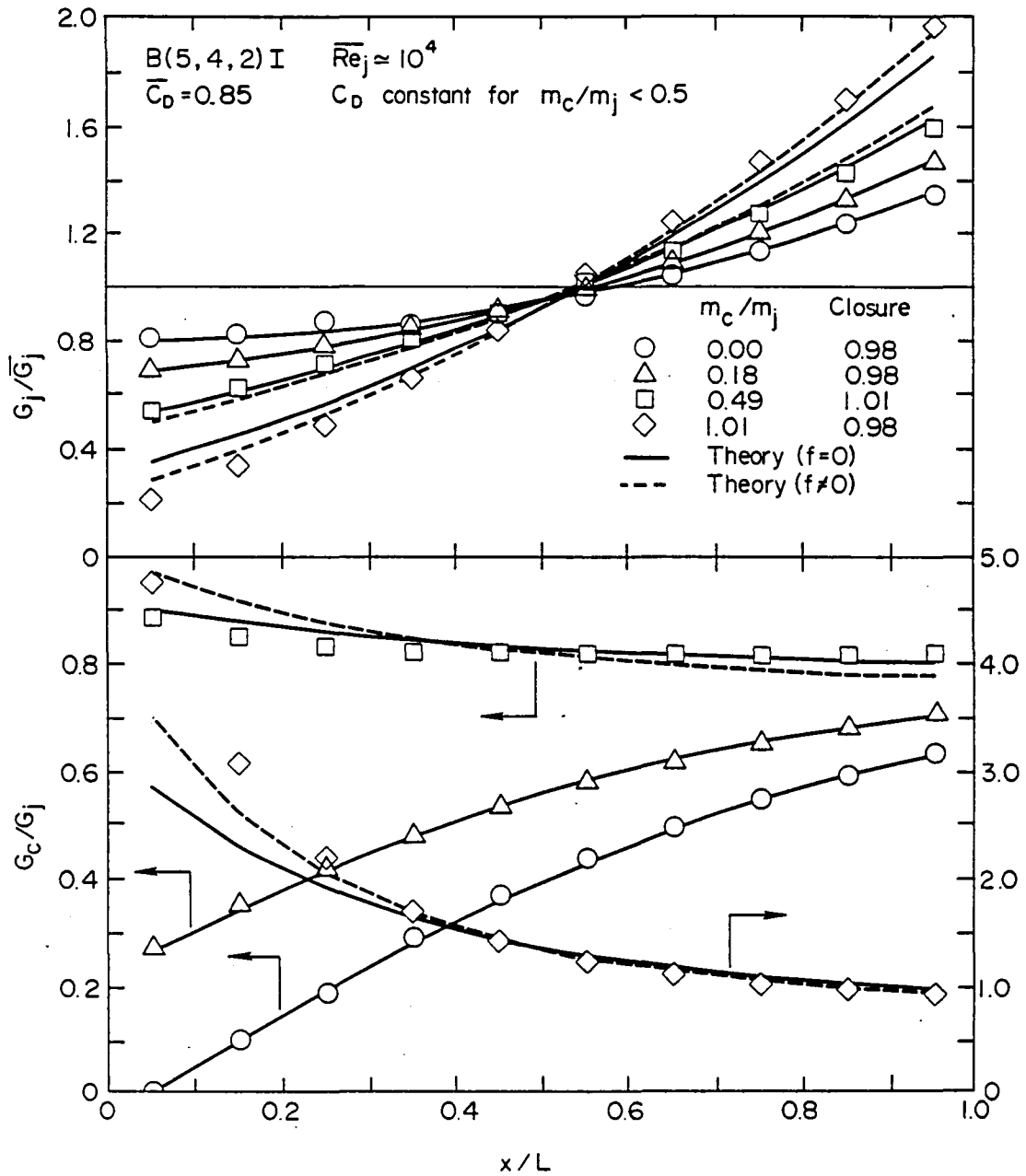
B-size jet array and plenum length, $L = 12.7 \text{ cm}$

Downstream exit length, 4.5 cm

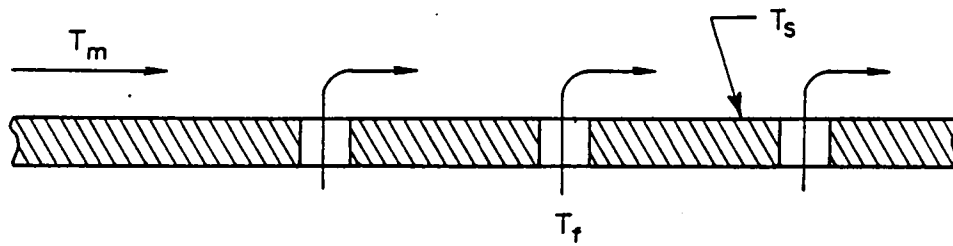
Initial crossflow development length, 24.1 cm

Number of spanwise rows of jet holes, $N_c = 10$

I = Inline, S = staggered hole pattern

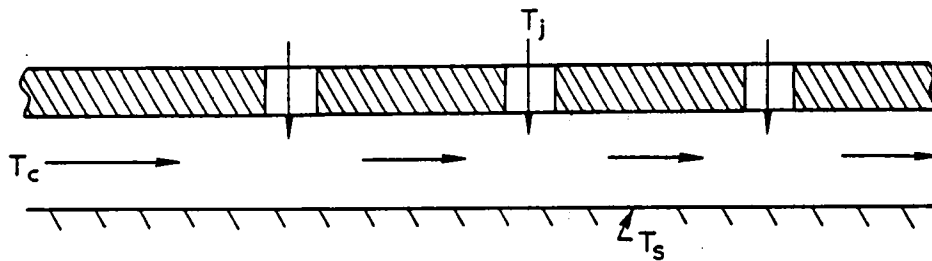


Effect of initial crossflow on jet array flow distribution (G_j/\overline{G}_j) and cross-to-jet mass velocity ratio (G_c/G_j) for $B(5,4,2)I$ geometry - experimental data compared with predictive model.



$$q = h(T_s - T_{aw}) \quad \eta = \frac{T_{aw} - T_m}{T_f - T_m}$$

Film cooling as a three-temperature problem.

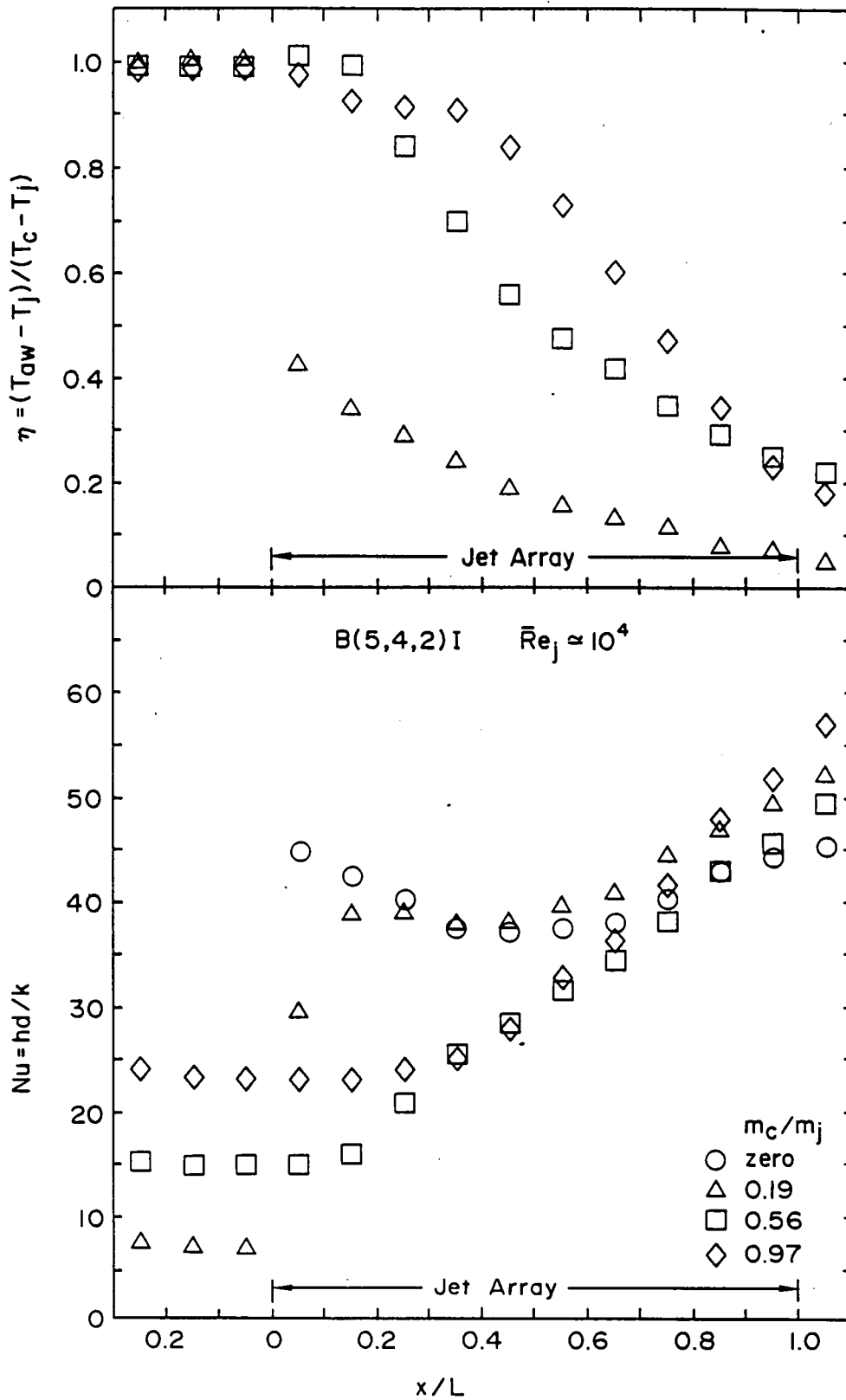


$$q = h(T_s - T_{aw}) \quad \eta = \frac{T_{aw} - T_j}{T_c - T_j}$$

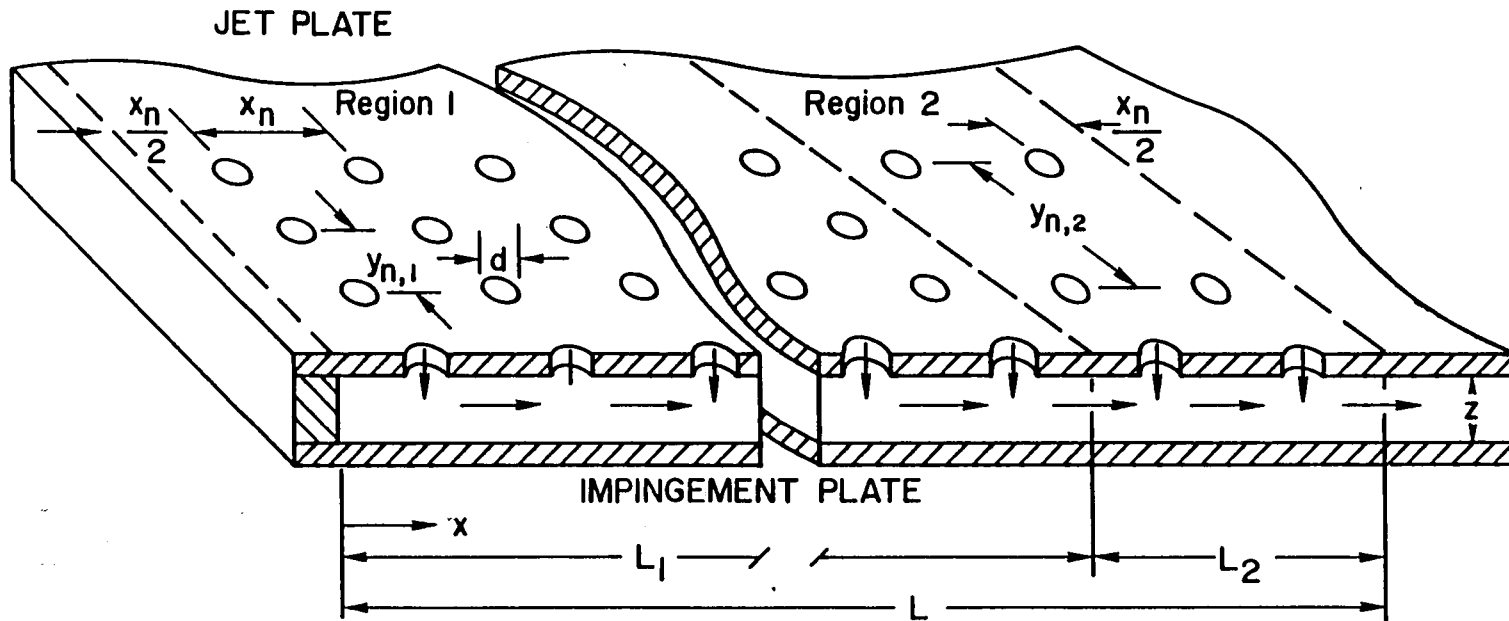
or

$$q = h[(1 - \eta)(T_s - T_j) + \eta(T_s - T_c)]$$

Jet array impingement with initial crossflow as a three-temperature problem.



Effect of initial crossflow rate on η and Nu profiles for B(5,4,2)I geometry.



Example of basic test model geometry and nomenclature for nonuniform array.

Nonuniform Array Geometries Tested

Nonuniform Parameter	Array Geometry†		Number of Rows		d (cm)	Test Series #
	Region 1	Region 2	Region 1	Region 2		
x_n	B(10,8,3)I	B(5,8,3)I	4	2	0.254	1X*
y_n	B(5,8,3)I	B(5,4,3)I	8	2	0.254	2Y*
	B(5,4,3)I	B(5,8,3)I	4	6	0.254	3Y*
	B(5,8,3)I	B(5,4,3)I	2	8	0.254	4Y*
	B(5,4,3)I	B(5,8,3)I	1	9	0.254	5Y*
	B(10,8,3)I	B(10,4,3)I	8	2	0.127	6Y
	B(5,8,2)I	B(5,4,2)I	8	2	0.254	7Y
	B(10,8,2)I	B(10,4,2)I	8	2	0.127	8Y
	B(10,8,2)I	B(10,4,2)I	5	5	0.127	9Y
					<u>d_1 (cm)</u>	
d	B(10,8,2)I	B(5,4,1)I	5	5	0.127	1D*
	B(5,4,1)I	B(10,8,2)I	5	5	0.254	2D*
	D(15,6,3)I	D(10,4,2)I	5	5	0.254	3D
	D(10,4,2)I	D(15,6,3)I	5	5	0.381	4D

† (x_n/d , y_n/d , z/d)

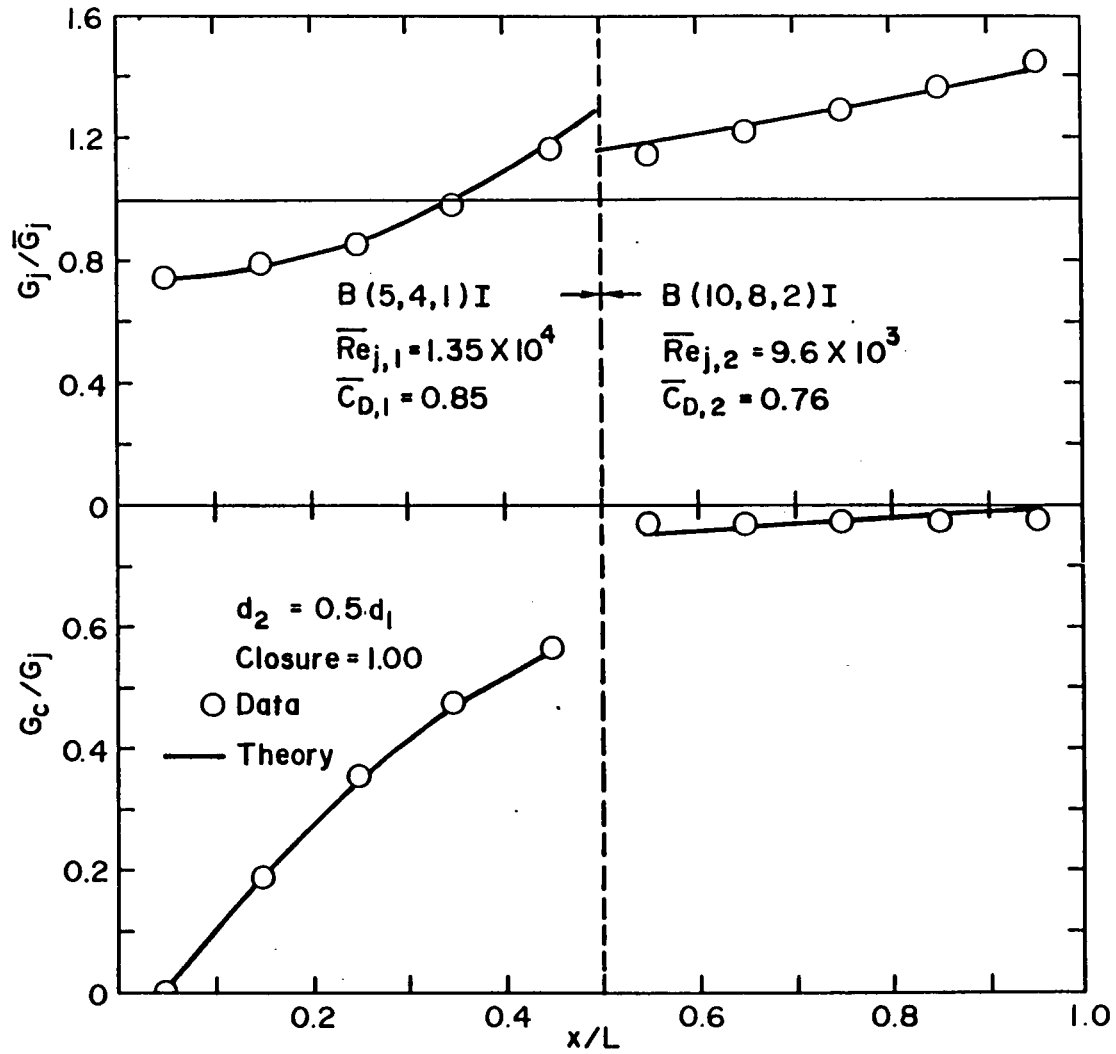
Prefix designates overall array length: B(L = 12.7 cm), D(L = 38.1 cm)

Suffix designates hole pattern: I = Inline

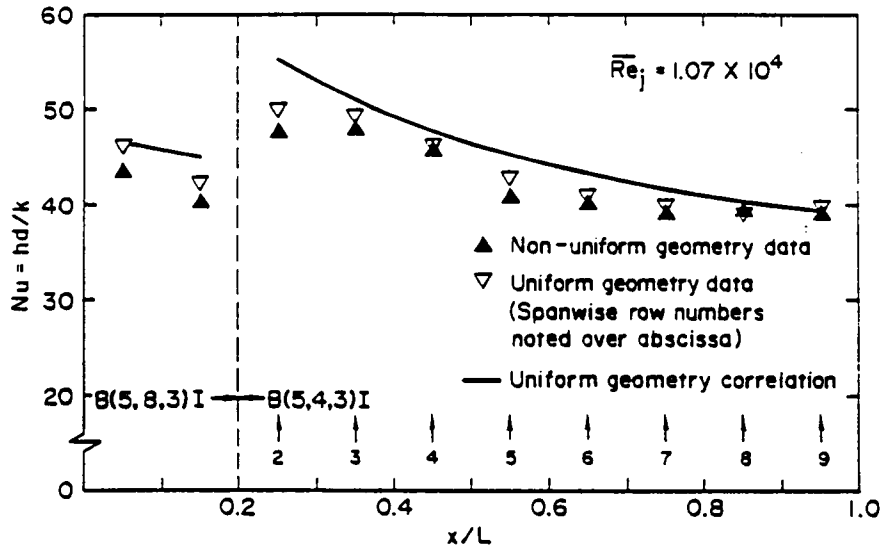
* Flow distribution (row-by-row) measured in addition to heat transfer coefficients

Note: b = d, for Test Series 1X and for 2Y through 9Y

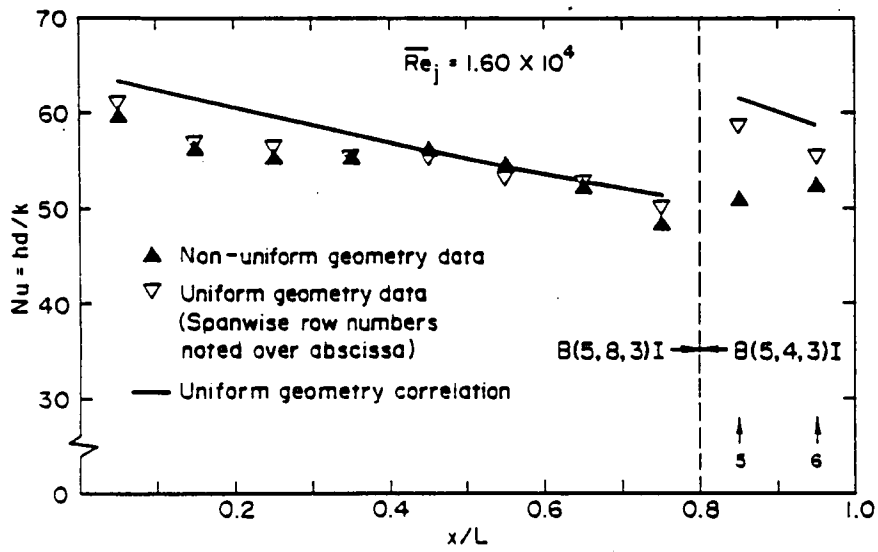
b = largest of d_1 or d_2 , for Test Series 1D through 4D



Flow distribution data for nonuniform diameter array compared with theory. Test #2D.



Nusselt number data for nonuniform y_n array compared with uniform array data and correlation. Test #4Y.



Nusselt number data for nonuniform y_n array compared with uniform array data and correlation. Test #2Y.

COOLANT PASSAGE HEAT TRANSFER WITH ROTATION

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One of the areas where technology is lacking is in predicting local heat transfer and local coolant flow conditions within coolant passages of rotating blades and particularly in blades that have multi-pass type coolant passages. Although the effects of the coriolis and buoyancy forces due to rotation on coolant-side heat transfer are generally not currently included in the design methods for blades, the influence of these forces could be large. Comparisons of non-rotating heat transfer data and extrapolations of available correlation for the average heat transfer coefficients with radial outflow of cooling air, showed that neglecting rotation at gas turbine engine conditions would result in variations in the heat transfer coefficient by as much as 45 percent. This, in effect, results in blade metal temperatures running as much as 100° F different from predicted values. This also may explain why rotating blade metal temperatures in engine tests are often higher than expected from results obtained in non-rotating cascade tests.

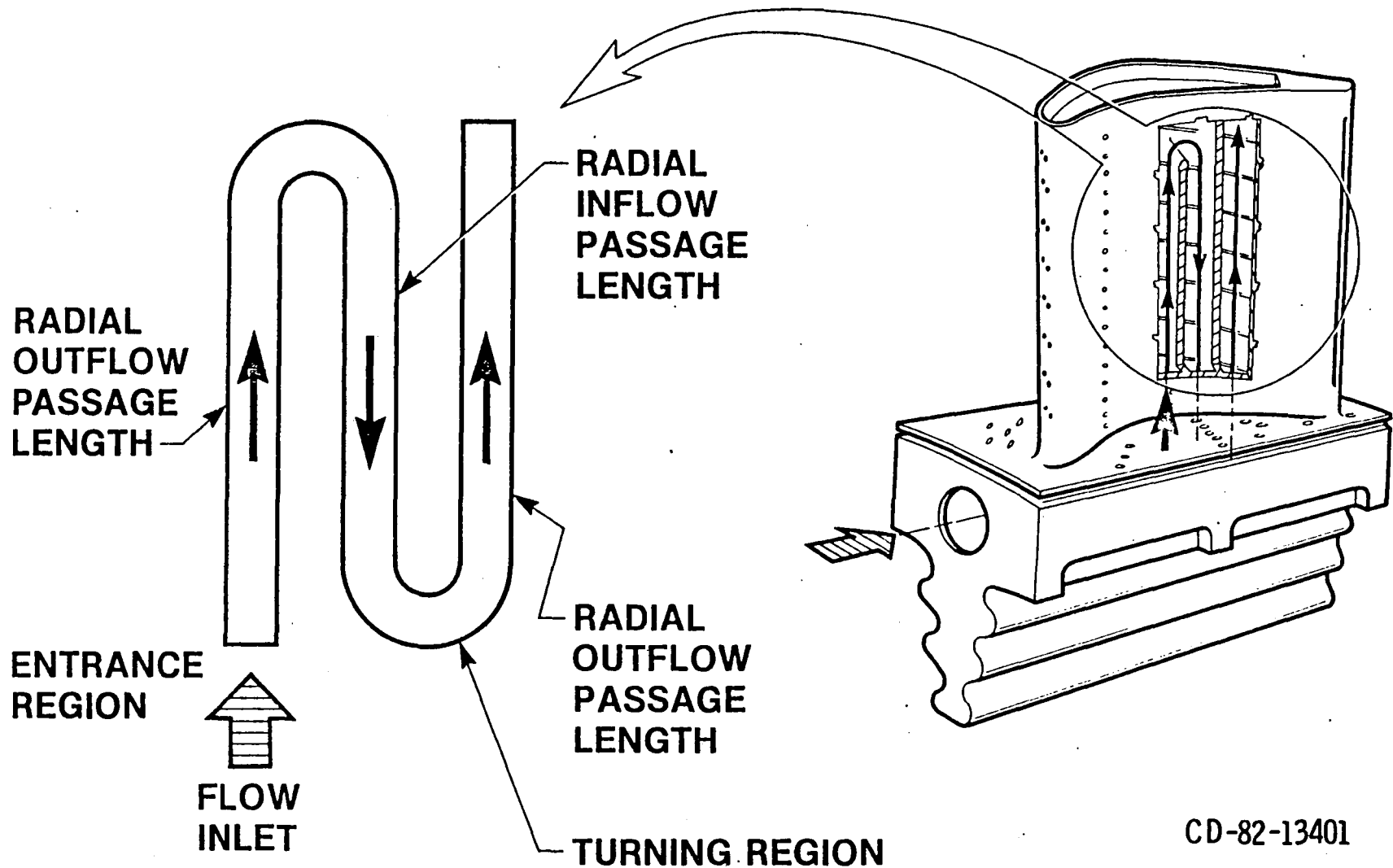
Although analytical computer code predictions have been obtained for such variations as coolant velocity and coolant temperature profiles within rotating coolant passages, and relatively good agreements have been obtained for predicted and measured heat transfer and friction factors, the results were for relatively low centrifugal buoyancy conditions. The high pressure and heat flux and the resultant large fluid density gradients combine with the high rotational speed of current and advanced gas turbine engines to require inclusion of the effects of coriolis and buoyancy forces in the predictive models. Generally beneficial effects on the average heat transfer are predicted from the effect of coriolis forces caused by rotation, but the buoyancy forces are expected to negate and augment the coriolis effect on heat transfer in coolant passages depending on the radial outflow and inflow of the

coolant flow respectively.

Although correlations have been obtained for the average heat transfer in rotating smooth passages with radial outflow of coolant for a large range of test conditions, further research is needed. Rotating experiments are needed to extend to local heat transfer along and around the passages and to the higher Rayleigh and Reynolds numbers that simulate geometry and the large fluid density gradients and buoyancy forces expected in advanced gas turbine engines. Rotating experiments are also needed dealing with the radial inflow of coolant, the interactive effects of entrance and turning regions, and the effect of wall boundary layer trips (turbulators).

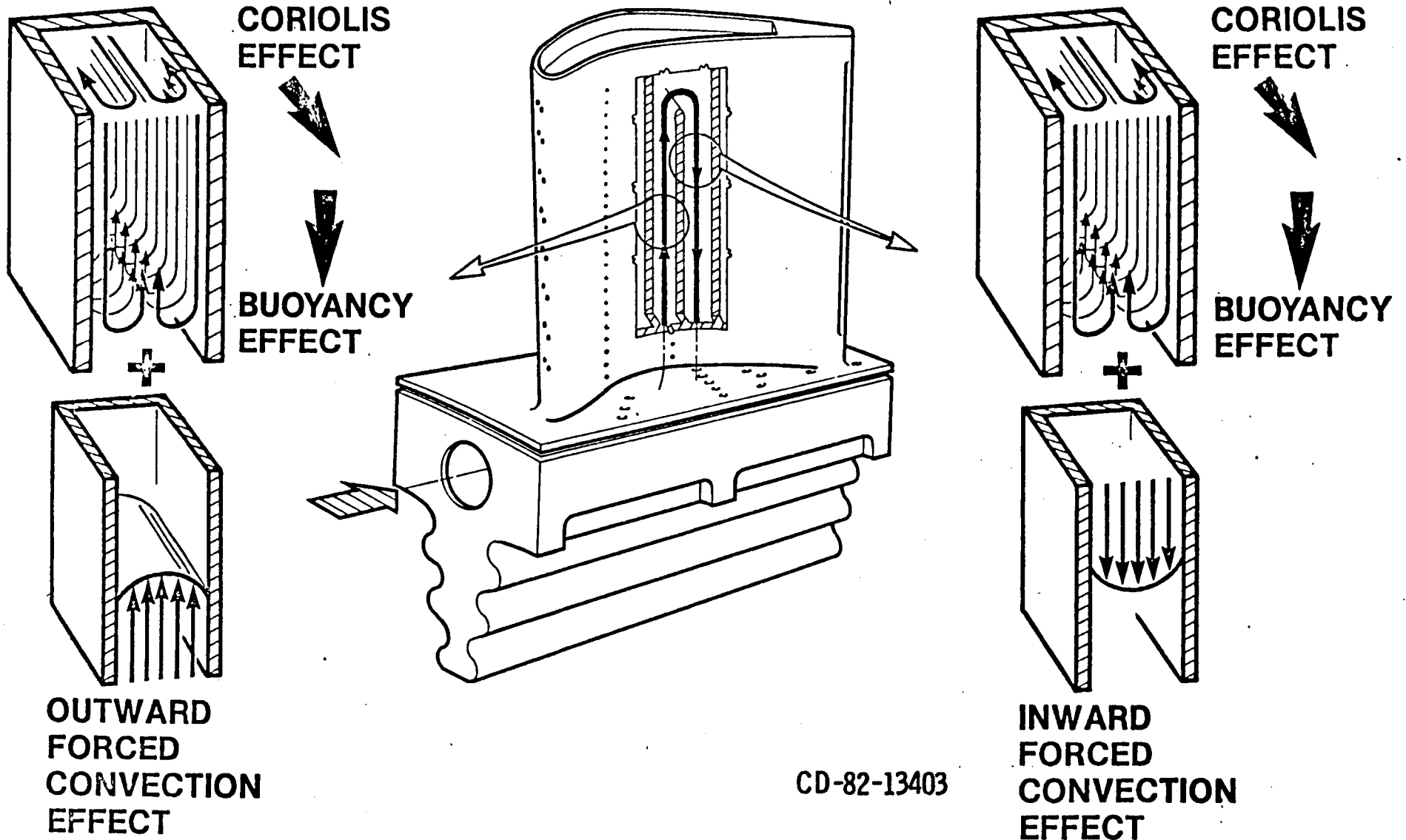
Correlations, analytical models, and computer codes will be developed and improved in order to accurately predict the heat transfer coefficients and local flow conditions within a multi-pass rotating turbine blade. These computer codes and correlations will be developed or improved to predict the local heat transfer coefficient at the entrance to the coolant passage, along the length of the passage, and in any flow recirculation regions of the passage, and the associated local coolant flow conditions. These requirements dictate initially a quasi three-dimensional viscous flow code with full consideration of rotation and buoyancy effects.

SCHEMATIC OF COOLANT PASSAGE TEST UNIT



CD-82-13401

COOLANT SIDE HEAT TRANSFER WITH ROTATION



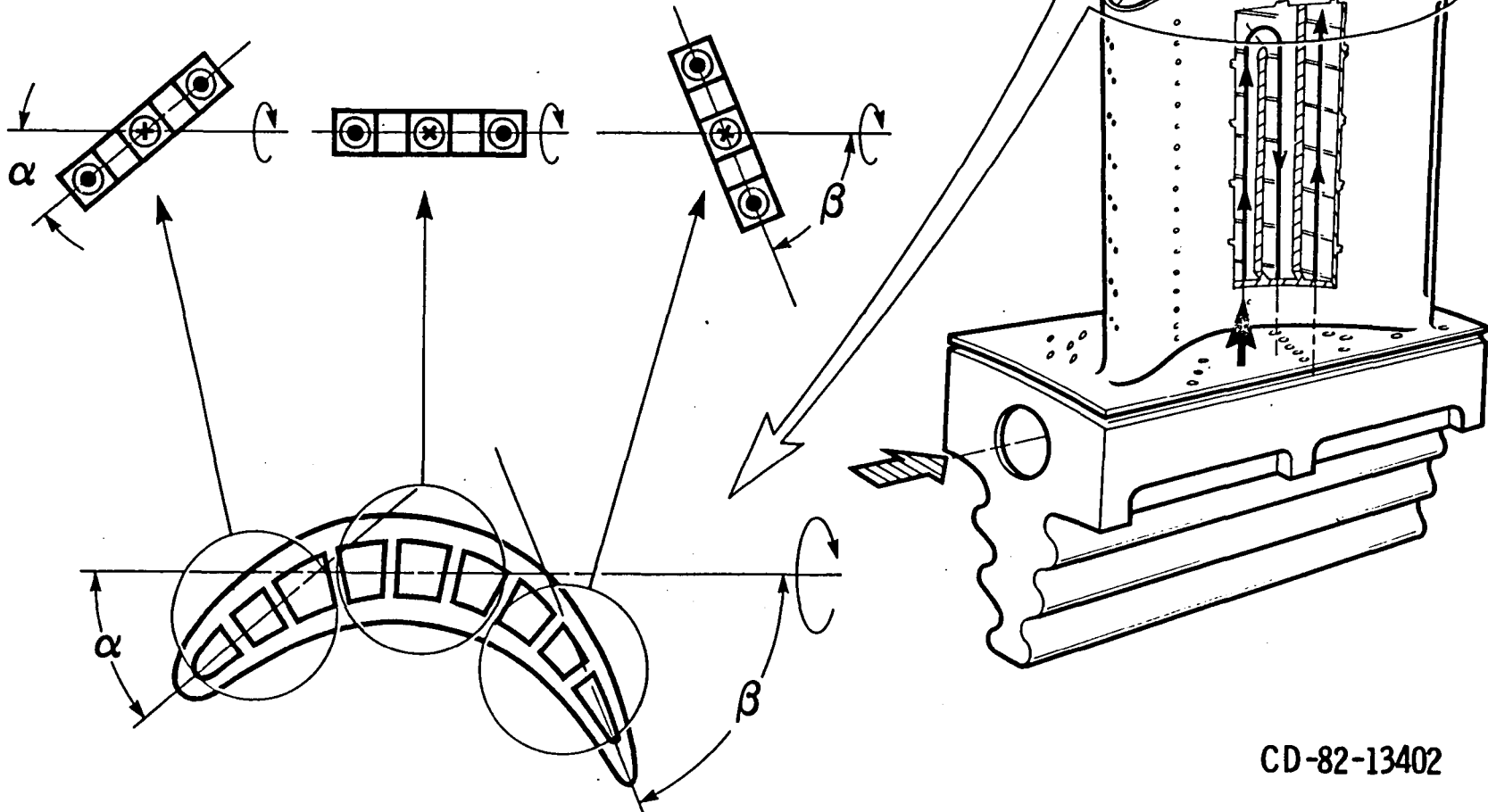
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CD-82-13403

SCHEMATIC OF PASSAGE ROTATIONS

⊗ RADIAL INFLOW

⊙ RADIAL OUTFLOW



STRUCTURAL ANALYSIS

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Hot section components of aircraft gas turbine engines are subjected to severe thermal-structural loading conditions, especially during the start-up and take-off portions of the engine cycle. The most severe and damaging stresses and strains are those induced by the steep thermal gradients induced during the start-up transient. These transient stresses and strains are also the most difficult to predict, in part because the temperature gradients and distributions are not well known or predictable, and also because the cyclic elastic-viscoplastic behavior of the materials at these extremes of temperature and strain are not well known or predictable.

A broad spectrum of structures-related technology programs is either underway or will be in the near future to address the deficiencies previously mentioned. The problems are being addressed at the basic as well as the applied level, including participation by industry and universities as well as in-house at NASA Lewis. In addition to the HOST program, some elements are being supported through our Base R&T program.

One element of the structures program will develop improved time-varying thermal-mechanical load models for the entire engine mission cycle from start-up to shutdown. The thermal model refinements will be consistent with those required by the structural code including considerations of mesh-point density, strain concentrations, and thermal gradients. Models will be developed for the burner liner, turbine vane and turbine blade. One aspect of this part of the program is a thermal data transfer module currently under development which will automate the transfer of temperatures from available heat transfer codes or experimental data sets to the structural analysis code. Another part of the program which will soon be initiated is an automated component-specific geometric modeling capability which will produce 3-D finite element models of the components. Self-adaptive solution strategies will be developed and included to facilitate selection of appropriate elements, mesh sizes, etc.

Another major part of the program is the development of new and improved nonlinear 3-D finite elements and associated structural analysis programs, including the development of temporal elements with time-dependent properties to account for creep effects in the materials and components. Improved constitutive models to facilitate improved prediction of cyclic thermomechanical viscoplastic material behavior are also under development. Experimental facilities to aid in developing and verifying theories and models are currently being established in-house at Lewis.

Further explanation and some details about the various aspects of the structures program mentioned above will be given in the following write-ups.

HOST
STRUCTURAL ANALYSIS

OBJECTIVE:

TO DEVELOP AND VALIDATE INTEGRATED, TIME-VARYING THERMAL/MECHANICAL LOAD MODELS FOR IMPROVED STRESS/STRAIN/DEFORMATION PREDICTIONS IN GAS TURBINE ENGINE BURNER LINERS, TURBINE BLADES AND VANES. ALSO TO DEVELOP COMPONENT-SPECIFIC AUTOMATED GEOMETRIC MODELING AND SOLUTION STRATEGY CAPABILITIES AND ADVANCED INELASTIC ANALYSIS METHODS, INCLUDING PLASTICITY AND CREEP EFFECTS, FOR NONLINEAR, ANISOTROPIC, FINITE ELEMENT STRUCTURAL ANALYSIS AND DESIGN COMPUTER CODES.

STRUCTURAL ANALYSIS

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
THERMAL DATA TRANSFER		▬						COMPUTER MODULE LINKING THERMAL AND STRUCTURAL ANALYSES
COMPONENT SPECIFIC MODELING			▬	▬	▬	▬	▬	COMPONENT-RELATED, TIME VARYING, THERMAL-MECHANICAL LOAD HISTORY & GEOMETRIC MODELS
3-D INELASTIC ANALYSES				▬	▬	▬	▬	ADVANCED 3-D INELASTIC STRUCTURAL/STRESS ANALYSIS METHODS AND SOLUTION STRATEGIES
LINER CYCLIC RIG		(IH)	▬	▬	▬	▬	▬	BURNER STRUCTURAL/LIFE EXPERIMENTS
HIGH-TEMPERATURE STRUCTURES LAB		(IH)	▬	▬	▬	▬	▬	INTEGRATED EXPERIMENTAL /ANALYSIS RESEARCH
MATERIAL BEHAVIOR TECHNOLOGY		(IH)	▬	▬	▬	▬	▬	MATERIAL CONSTITUTIVE MODELS

STRUCTURES

THERMAL/STRUCTURAL DATA MODULE

DR. R. J. MAFFEO, G. E.

COMPONENT-SPECIFIC MODELING

DR. M. HIRSCHBEIN, NASA

INELASTIC ANALYSIS METHODS

DR. C. CHAMIS, NASA

BURNER LINER CYCLIC RIG

DR. R. THOMPSON, NASA

BURNER LINER THERMAL/STRUCTURAL LOAD MODELLING

R.J. Maffeo

AEBG

General Electric Co.

The objective of this program is to develop a thermal data transfer computer program module for the Burner Liner Thermal-Structural Load Modelling Program. This will be accomplished by (1) reviewing existing methodologies for thermal data transfer and selecting three heat transfer codes for application in this program, (2) evaluating the selected codes to establish criteria for developing a computer program module to transfer thermal data from the heat transfer codes to selected stress analysis codes, (3) developing the automated thermal load transfer module, and (4) verifying and documenting the module.

In aircraft turbine engine hot section components, cyclic thermal stresses are the most important damage mechanism. Consequently, accurate and reliable prediction of thermal loads is essential to improving durability. To achieve this goal, a considerable effort over the past 20 years has been devoted to the acquisition of engine temperature test data, as well as the development of accurate, reliable, and efficient computer codes for the prediction of steady-state and transient temperatures and for the calculation of elastic and inelastic cyclic stresses and strains in hot section components. There is a need for continued development of these codes, because the availability of more accurate analysis techniques for complex configurations has enabled engine designers to use more sophisticated designs to achieve higher cycle efficiency and reduce weight.

It has become apparent in recent years that there is a serious problem of interfacing the output temperatures and temperature gradients from either the heat transfer codes or engine tests with the input to the stress analysis codes. With the growth in computer capacity and speed and the development of input preprocessors and output postprocessors, the analysis of components using hundreds and even thousands of nodes in the heat transfer and stress models has become economical and routine. This has exacerbated the problem of manual transfer of output temperatures from heat transfer nodes to stress analysis input to where the engineering effort required is comparable to that required for the remainder of the analysis. Furthermore, a considerable amount of approximation has been introduced in an effort to accelerate the process. This tends to introduce errors into the temperature data which negates the improved accuracy in the temperature distribution achieved through use of a finer mesh. There is, then, a strong need for an automatic thermal interface module.

The overall objectives of this thermal transfer module are that it handle independent mesh configurations, perform the transfer in an accurate and efficient fashion and that the total system be flexible for future improvements.

Based on our study of existing thermal transfer modules, and our previous experience with TITAN (a 2-D thermal transfer module developed by us) we have identified three levels of criteria for the program development associated with this contract.

Level I contains the general criteria which must be satisfied for a usable product.

Level II contains specific criteria which must be satisfied to meet the requirements associated with gas turbine design problems. This list stems mainly from our internal experience.

Level III contains criteria which are desirable but not necessary. In most cases, items in this class can be achieved through a multi-step process. Total automation might be desirable, but we do not believe this effort is warranted at this time.

Level I: General Criteria for A Thermal Transfer Module

IA) Independent Heat Transfer and Stress Geometry Meshes

This criteria lies at the heart of our effort. Useful thermal transfer modules must address this feature. Automatically included in this is the ability to transfer from finite difference heat transfer to finite element stress analysis mesh.

IB) Accurate Transfer of Data

Simplistic approaches such as averaging the closest nodes do not always yield accurate results, and the utility of the transfer program is questionable. This criteria will be met by using all available temperature information to do the interpolation and by using different mappings to correspond with different heat transfer elements.

IC) User Friendly

Programs which do not meet this criteria tend to be used incorrectly or as a last resort. We plan to construct our thermal transfer module to encourage the analyst to use it. Any errors encountered by the module will be reported in a clear diagnostic, and "help" commands will be available for beginning users.

ID) Computationally Efficient

We will code the program to achieve an efficient flow of data. Our past experience with TITAN has led to several improvements over our original efforts, and we expect to produce similar gains in 3D transfer problems. This criteria covers both searches to find the proper heat transfer element for a stress node as well as the single element inverse mapping functions.

IE) Flexible

We plan to construct the thermal transfer module such that future modifications, or even different applications (pressure, or boundary condition transfer, for instance) could be accomplished without a full re-write. This criteria stems from past experience in having to improve or draw upon techniques which could almost, but not quite, perform the required task. Our transfer module will transfer temperatures in a state-of-the-art manner, but it will also provide a vehicle for numerous other 3D interpolation based problems.

Level II: Specific Criteria Required For Gas Turbine Design Problems

IIA) Coordinate Transforms

Coordinate transformation that will allow the heat transfer model to be aligned with the structural model.

IIB) "Out-of-Box" Provision

Provision to account for stress nodes that lie just outside the heat transfer model due to slight differences in the dimensions used in the heat transfer and stress analysis models, as a result of using different tolerances on the actual component dimensions.

IIC) Windowing

Capability to "window in" on a smaller portion of the heat transfer model.

IID) Selected Time Steps

Ability to select temperature distributions at specific time steps from a large transient thermal analysis.

Level III: Desirable But Not Essential Features

IIIA) Automatic Handling of Temperature Discontinuities

In our module these will be treated in a two-step manner. Total automation is possible, but probably not necessary at this point.

IIIB) Scaling of Temperatures Based on Variation In Engine Power Level Settings

Such scaling will not be done inside our transfer module, but could, if desired, be applied by another program to the original results from our transfer module.

IIIC) "Altered" Stress Geometry

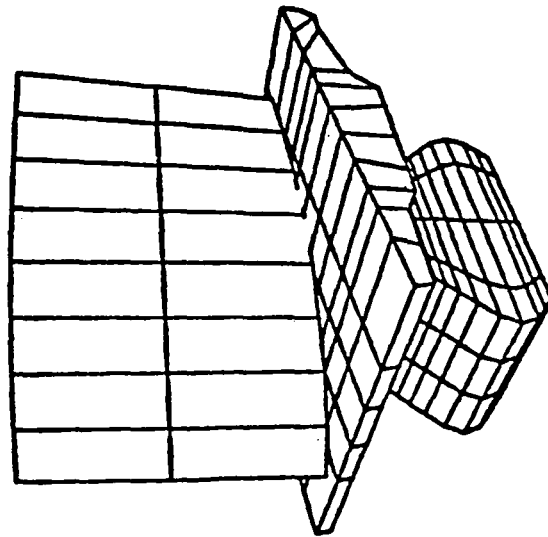
Many times the stress analyst wants to alter the part geometry to reduce his stresses, but the deviations will not, in the judgement of the heat transfer analyst, affect the temperatures. We have in the past used "ad hoc" procedures to transfer temperatures to the new stress geometry. This approach is not optimum, but we do not plan to include any capability for this case in our transfer module.

This module, once it is developed, will transfer thermal data from heat transfer meshes to stress analysis meshes. But it will have the capability to do much more. The basic features of 3-D search and interpolation will make it an outstanding foundation for automatic construction of embedded meshes, local element refinement, and transfer of other mechanical loadings.

OBJECTIVES

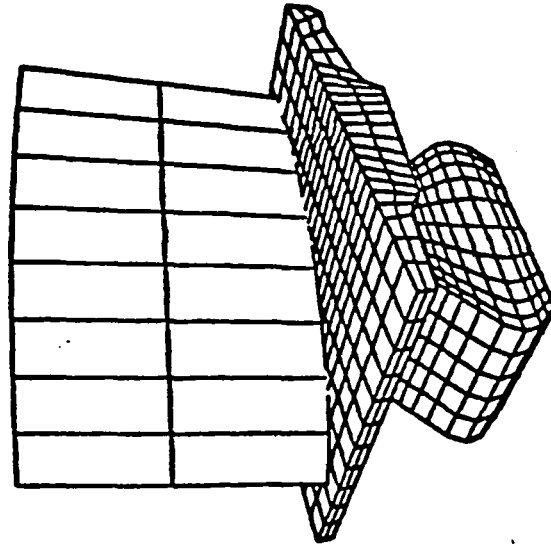
- TRANSFER TEMPERATURES FROM A HEAT TRANSFER STUDY TO A STRESS ANALYSIS
 - INDEPENDENT MESHES
 - ACCURATE/EFFICIENT TRANSFER
 - FLEXIBLE

OBJECTIVE



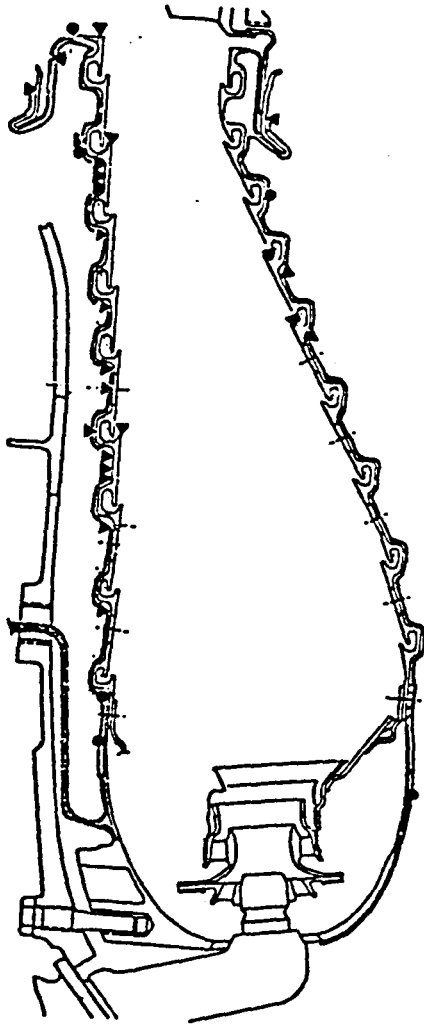
3D HEAT TRANSFER MODEL

TEMPERATURES

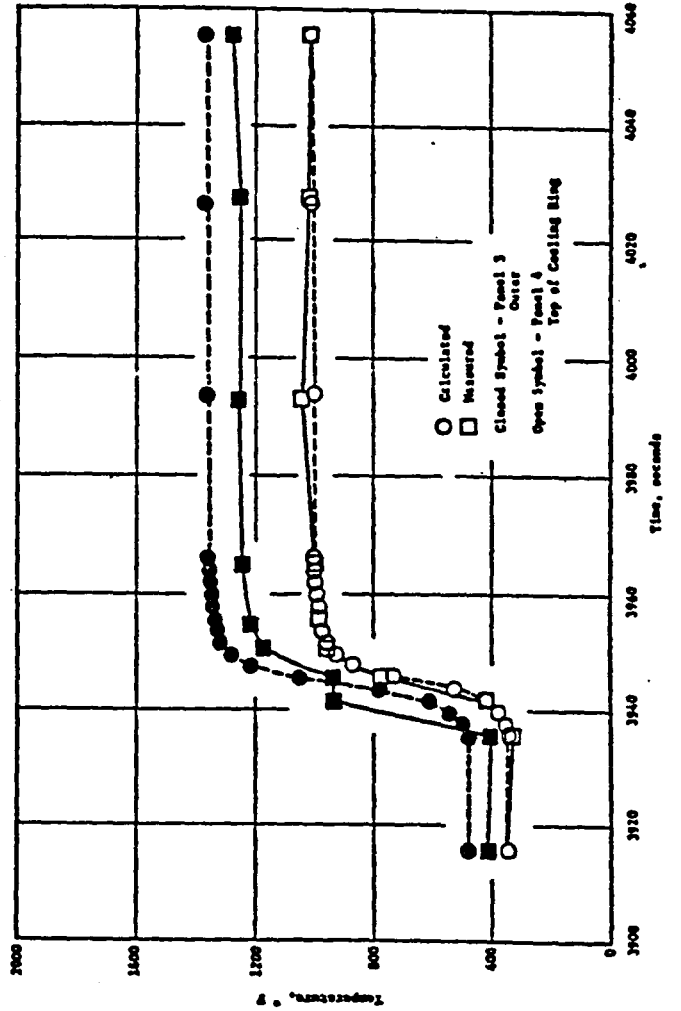


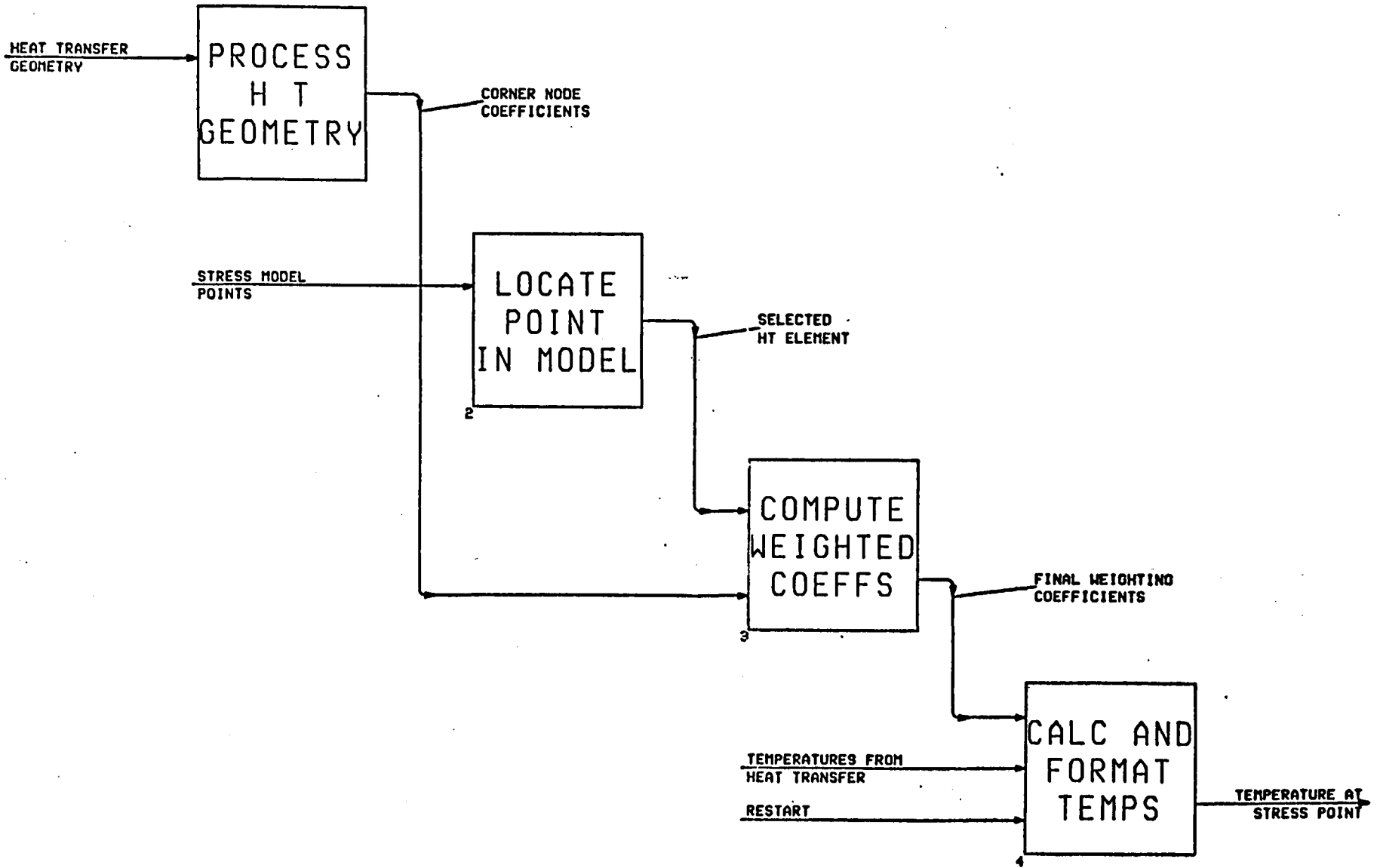
3D STRESS MODEL

EXPERIMENTAL DATA SET



▼ THERMOCOUPLES





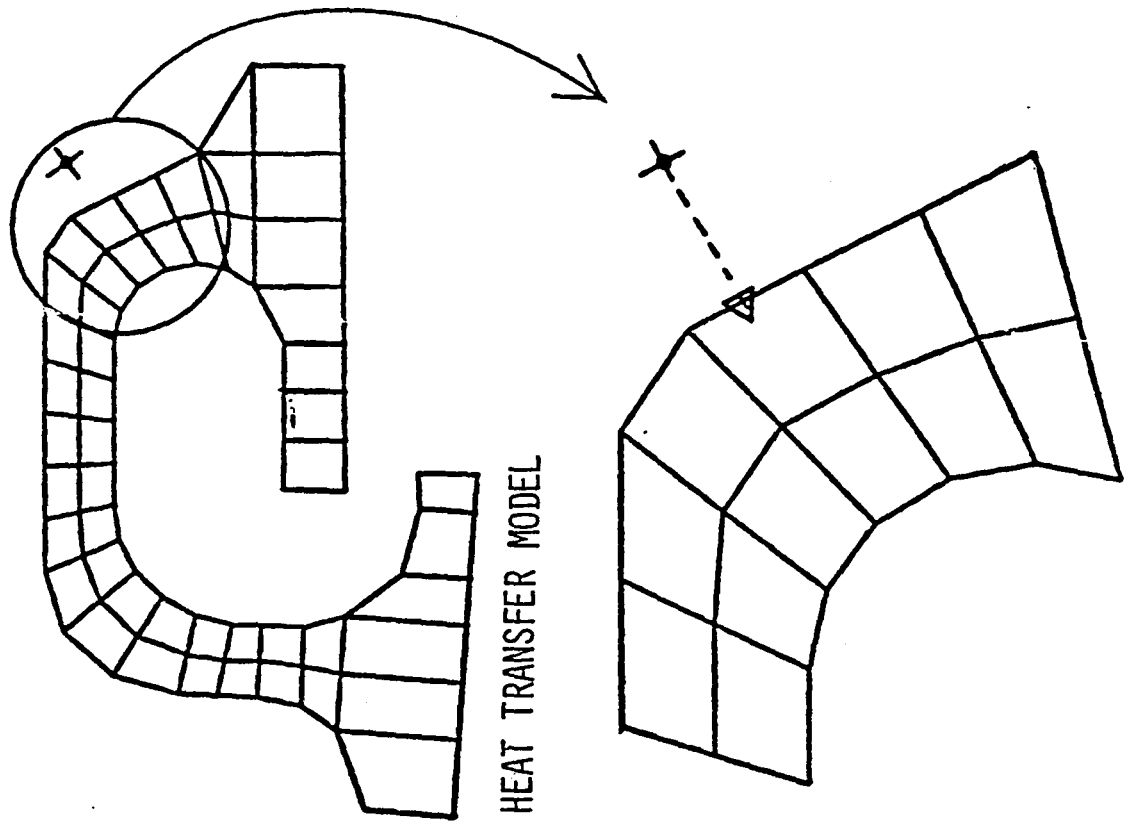
THERMAL TRANSFER MODULE

OVERALL SYSTEM

REVISION- 1.1

09-20-02

EXTERIOR POINT METHOD

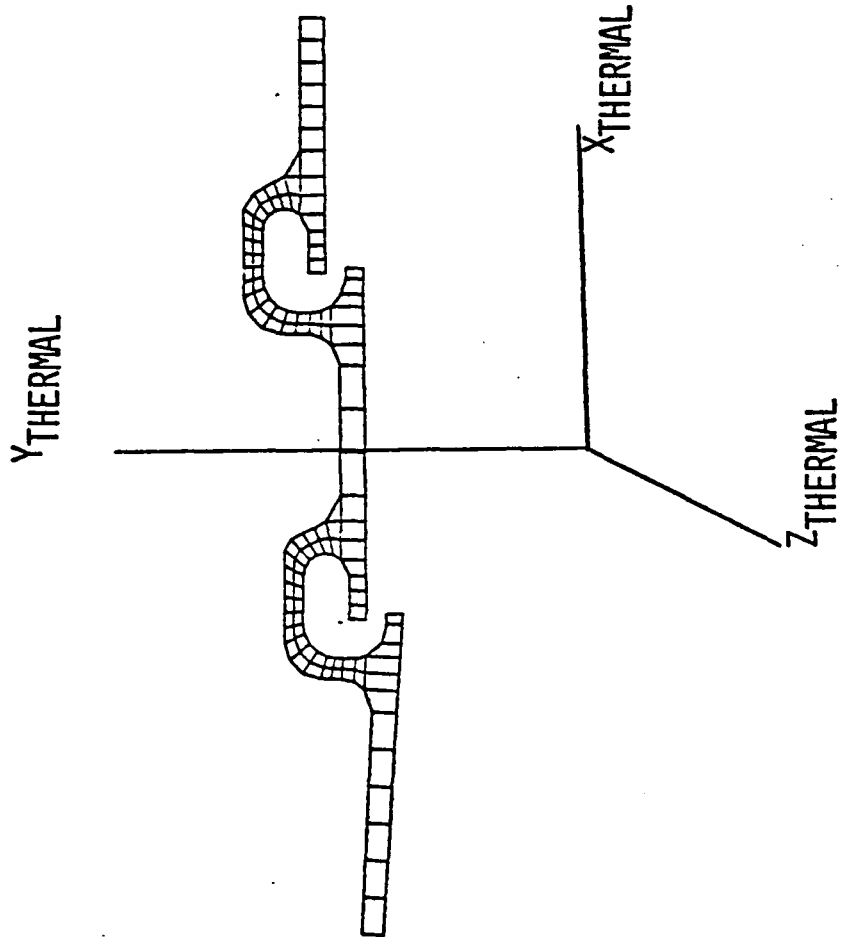
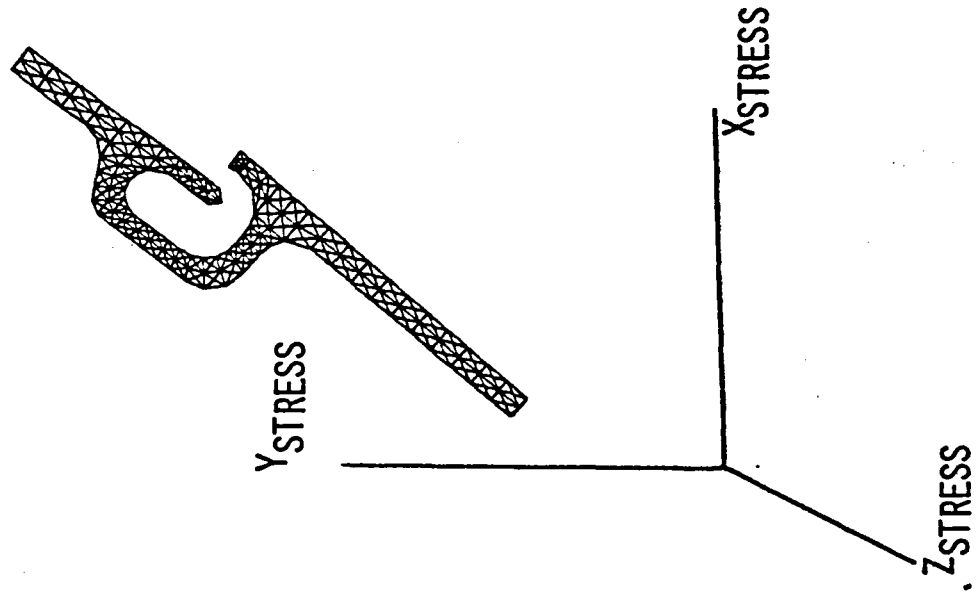


X STRESS POINT

△ SURFACE TEMPERATURE LOCATION

HEAT TRANSFER MODEL

'BUILT IN' TRANSFORMATIONS/WINDOWING



FLEXIBILITY

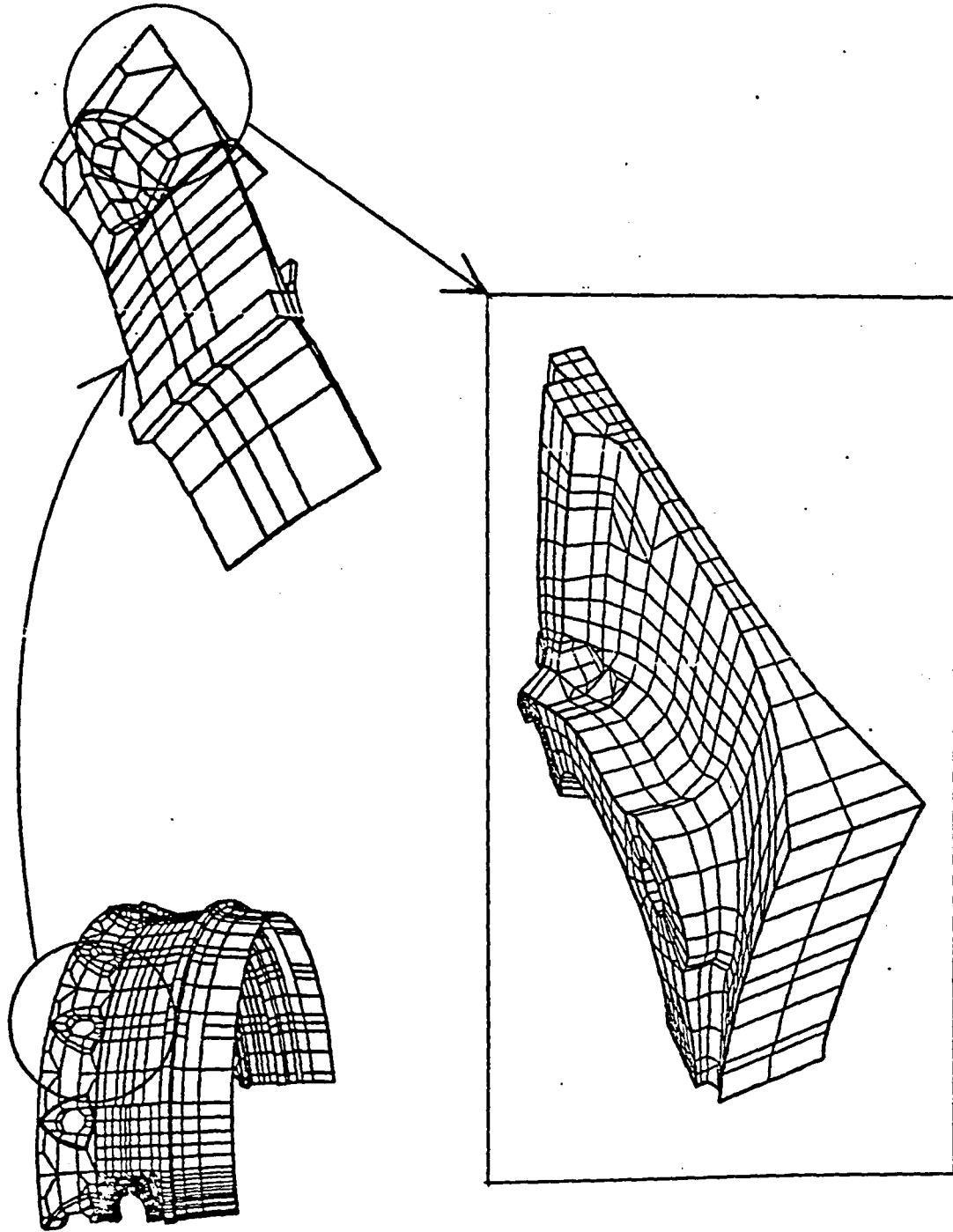
- TRANSFER MODULE NOT KEYED TO ANY SPECIFIC CODES
- INITIAL EFFORT DIRECTED AT TRANSFER OF THERMAL DATA

HOWEVER

BASIC TECHNOLOGY CAN BE APPLIED TO MANY AREAS.

- TRANSFER MECHANICAL LOADS
- TRANSFER OF BOUNDARY CONDITIONS FOR MESH REFINEMENT
- COMPUTATION OF CONSTRAINT EQUATION COEFFICIENTS

FUTURE APPLICATIONS
AUTOMATIC BOUNDARY CONDITION ASSIGNMENT



COMPONENT-SPECIFIC MODELING

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As a result of the recent drastic increases in fuel costs, the aircraft gas turbine engine industry has placed much higher technical priorities on reducing engine weight and increasing engine efficiency. As part of this effort, engine temperatures, internal gas pressures, and rotational speeds are being increased, while the size and weight of the engine components are generally being reduced. The result is that components, in many cases, are operating closer to their structural limits. This places much greater importance on the ability to accurately structurally analyze engine components to assure that they can survive for their designed lifetime in an increasingly harsh environment.

The burner liner, turbine blades and vanes are among the most structurally burdened and analytically complex components in the engine. High mean temperatures with severe transients, local hot spots, and steep gradients characterize the thermal environment of all three components. Additionally, the vanes and turbine blades are subjected to the highest gas pressures in the engine, and the turbine blades are loaded even further by strong centrifugal forces. The geometry of these parts is equally complex. The burner liner can be designed as overlapping stepped louvers with many cooling holes. The turbine blades and vanes may also have regions densely packed with tiny slanted cooling holes as well as complex internal gas paths along the span. In addition, there are small radius fillets near the base of the blades and vanes. As a result of the geometry and thermo-mechanical loading, these components have locally steep stress and strain gradients, regions which undergo varying degrees of cyclic plasticity and creep deformation, and material properties which can vary significantly in time and space.

In order to increase the durability and life of hot section components, new high-temperature materials and fabrication techniques are being applied to these components. These new materials have significantly anisotropic material behavior, such as with single crystal blades and directional solidification. This makes accurate and efficient structural analysis of hot section components even more difficult.

Currently, there are basically two general approaches to structurally analyze complex engine parts. One is to use general purpose analysis codes, such as NASTRAN and MARC. These are extremely powerful tools which can be applied to a wide variety of problems. However, they are not tailored to the needs of any one problem and depend very heavily on the user to adapt them to specific problems. Furthermore, these programs are designed as "one-shot" problem solvers based on the finite element method of analysis. That is, a problem is first modeled by whatever method the program is based on, and then the whole problem is solved at one time. It is up to the user to decide how the problem is modeled and whether the solution obtained is accurate to the desired degree. If the problem is to be re-solved, say with a finer finite

element mesh, or if a non-linear approach is required, it is up to the user to remodel and to develop his analysis strategy. All sub-problems, approximate solutions, or sensitivity studies must be controlled by the user. To some extent this effort can be reduced by writing geometric and discretization pre-processors, and in the case of NASTRAN, new Rigid Formats can be added. Essentially, with general purpose codes, the analyst tends to overpower very complex problems, but at great expense in computational effort and man-hours.

The second approach is similar to the first, except that the analysis codes are streamlined to meet the needs of more focused problems. These codes have built-in pre- and post-processors to reduce the modeling effort, facilitate sequencing of programs, and to make the display of data easier. However, they still rely on a single analysis method and are still designed to model and solve the structural analysis problem in a single pass through the program. These programs may have features to deal with component specific problems such as special axisymmetric shell elements or crack elements. Often the analysis method is greatly simplified in order to reduce the cost of repeated analyses during a design process. In both cases analysis decisions such as local model refinement, or how and when to use linear and non-linear analyses are left to the analyst.

Either of these approaches, or a combination of them, may be adequate when a single, conservative, feasible design is acceptable. With the increasingly harsh thermo-mechanical environments expected for hot section components as engines are made lighter and more efficient, the structural limits of these components will be more severely challenged. Better, more efficient analysis methods must be developed in order to be able to assure in advance that hot section components will survive for their design lifetimes.

Under the HOST (HOT Section Technology) program, advanced component-specific modeling methods, with built-in analysis capability, will be developed separately for burner liners, turbine blades and vanes. These modeling methods will make maximum use of, but will not rely solely on, existing analysis methods and techniques, to analyze the three identified components. Nor will the complete structural analysis of a component necessarily be performed as a single analysis. The approach to be taken will develop complete software analysis packages with internal, component-specific, self-adaptive solution strategies. Each package will contain a set of modeling and analysis tools. The selection and order of specific methods and techniques within the set to be applied will depend on the specific-component, the current thermo-mechanical loading, and the current state of the component. All modeling and analysis decisions will be made internally based on developed decision criteria within the solution strategies; minimal user intervention will be required. In this way, the structural analyses of burner liners, turbine blades and vanes may actually be comprised of a series of global approximate analyses and local detailed analyses which lead to computationally-efficient, total structural analyses with assured accuracy, and without extensive, time-consuming user intervention.

The software packages will be modular and open-ended to allow the addition or substitution of new modeling and analysis methods and to allow modification to or change of the solution strategies. The primary structural analysis method will be the finite element method. However, additional methods such as approximate closed form analyses, semi-analytical solutions, and boundary integral methods will be considered to develop the complete solution strategies. Linear and non-linear solution capability for static and dynamic responses will be included in each component-specific model. Automatic options within the solution strategies will include remeshing with optimization, substructuring of the mass and stiffness matrices, and automatic load step and time step control. During development the accuracy and computational efficiency of the component-specific models will be verified by comparison with established solutions and data sets.

To support the advanced structural analysis capability developed herein, advanced thermo-mechanical load mission models will be developed to predict detailed time-related pressure and temperature distributions in the burner liner, turbine blade and vane. These distributions will accurately represent conditions experienced during an arbitrary commercial aircraft mission cycle. Also, as part of this load modeling effort, an advanced thermodynamic engine cycle model will be developed to predict the gross temperatures and pressures throughout the hot section of the engine as a function of the power lever setting. The advanced thermo-mechanical load modeling capability will be designed to interact with the component-specific structural models developed herein, as well as to supply detailed loading data for independent analysis programs. Acting together the structural and loads models will provide highly advanced capability to accurately predict the loading and structural response of the burner liner, turbine blade, and vane over an entire arbitrary mission cycle.

This capability will be further enhanced by developing methodology to synthesize the loading and structural response histories of the hot section components over an arbitrary mission cycle. The synthesis process will involve developing methods which can use sparse, pre-computed, component-specific structural and mission cycle data to construct these histories with minimal interaction with detailed analysis codes. This technology will significantly reduce the cost of predicting the loading and structural response histories of the burner liner, turbine blade and vane over complete cycles.

The thermo-mechanical load models and mission model synthesis methodology will be developed essentially independent of the component-specific structural models. As such, the progress of either part of the total program will not depend on the other. In this way, both parts can be designed more effectively to interact with independent loading or structural analysis programs.

COMPONENT-SPECIFIC MODELLING

PURPOSE

**DEVELOP ADVANCED HIGH-LEVEL STRUCTURAL
ANALYSIS CAPABILITY FOR HOT SECTION**

COMPONENTS

- 1) BURNER LINER**
- 2) TURBINE BLADE**
- 3) VANE**

TWO MAIN THRUSTS:

1) COMPONENT-SPECIFIC STRUCTURAL MODELLING

A) GEOMETRIC MODELLING AND DISPLAY

B) SELF-ADAPTIVE SOLUTION STRATEGIES

2) THERMO-MECHANICAL LOAD/MISSION MODELLING

A) ADVANCED THERMODYNAMIC ENGINE MODEL

**B) DETAILED TEMPERATURE AND PRESSURE
LOADING MODELS**

**C) MISSION MODEL "DECOMPOSITION/
SYNTHESIS" METHODOLOGY**

TASK STRUCTURE

BASE PROGRAM

SURVEY

DESIGN OF STRUCTURAL ANALYSIS ARCHITECTURE

THERMODYNAMIC ENGINE MODEL

SOFTWARE DEVELOPMENT

MISSION MODEL DECOMPOSITION

STRUCTURAL ANALYSIS METHODS EVALUATION

THERMO-MECHANICAL LOAD/MISSION MODEL DEVELOPMENT

COMPONENT-SPECIFIC MODEL DEVELOPMENT

202

OPTIONAL PROGRAM

SURVEY

STRUCTURAL ANALYSIS METHODS EVALUATION

COMPONENT-SPECIFIC MODEL DEVELOPMENT

MISSION MODEL DEVELOPMENT

VERIFICATION TESTING

3-D INELASTIC ANALYSIS METHODS FOR HOT SECTION COMPONENTS

BRIEF DESCRIPTION

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BACKGROUND

The most severe structural requirements imposed upon aircraft gas turbine engine components result from the extreme environmental conditions in the engine hot section. These conditions include very high temperatures with steep thermal gradients, high and fluctuating pressures, rapid transients, vibration, oxidation, corrosive and erosive atmospheres, and an assortment of structural loadings both from within the engine and as a result of engine/aircraft system interactions. Accurate prediction of structural response and life assessment of the components under these conditions require sound 3-D inelastic analytical methods. Present 3-D inelastic analysis methods usually rely on large volumes of input data to define the problem, frequent user intervention during the analysis process, and considerable care in assessing the accuracy and interpreting the results. Most of these methods are parts of general purpose structural analysis programs which were not intended for the complex 3-D inelastic analysis problems associated with gas turbine engine components. Thus highly-skilled technical manpower is required to set up the problems and frequently, to interpret the results.

OBJECTIVE AND APPROACH

The objective of this program is to develop advanced 3-D inelastic structural/stress analysis methods and solution strategies for more accurate yet more cost-effective analysis of components subjected to severe thermal gradients and loads in the presence of mechanical loads, with steep stress and strain gradients, and which include anisotropy and time and temperature dependent plasticity and creep effects. The approach is to develop four different theories, one linear and three higher order theories (polynomial function, special function, general function). The theories are progressively more complex from linear to general function in order to provide streamlined analysis capability with increasing accuracy for each hot section component and for different parts of the same component according to the severity of the local stress, strain and temperature gradients associated with hot spots, cooling holes and surface coating cracks. To further enhance the computational effectiveness, the higher order theories will have embedded singularities (cooling passages, for example) in the generic modeling region.

Each of the four theories consists of three formulation models derivable from independent theoretical formulations. These formulation models are based on (1) mechanics of materials, (2) special finite elements, and (3) an advanced formulation to be recommended by the contractor.

The mechanics of materials models shall be formulated for easily amenable solution (approximate calculations). The special finite elements will be formulated to be used as "stand-alone" modules and as modules integrated (using interfacing links) into general purpose structural analysis computer programs. The advanced formulation model shall be formulated to provide an alternate and complementary analysis capability to the special finite elements so that each theory can be used to check the other, and thereby minimize costly experiments that otherwise may be needed. In addition, each model shall be formulated to accommodate three different levels of constitutive

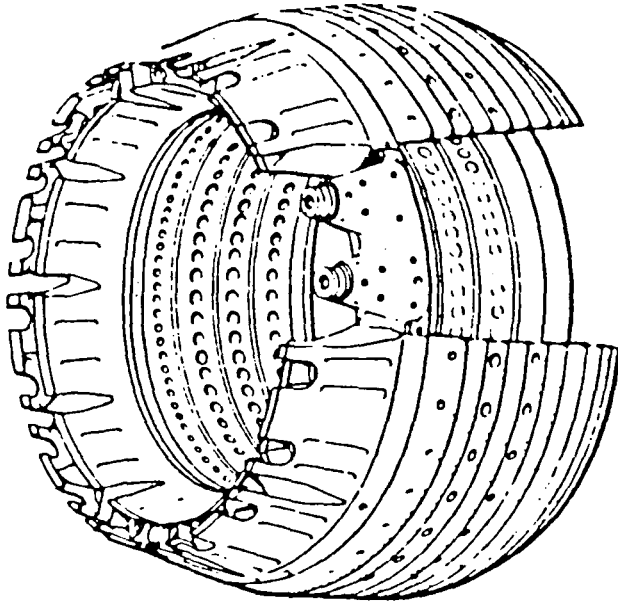
theory with progressive levels of complexity. The three different levels of constitutive theory are needed in order to: (1) provide formulation flexibility, (2) provide for modeling different material behavior in different parts of the component, and (3) increased accuracy to assess the validity of the approximations made. Appropriate solution strategies with self-adaptive features shall be developed along with numerical solution algorithms with self starting and dynamic incrementation to further enhance the computational effectiveness of these theories.

All theories, including models and constitutive relationships, will be validated with respect to accuracy and computational effectiveness using available analysis results from simulated and actual hot section components. In addition, the theories shall be verified using available experimental data and data generated under this program. The end product of these theories will be computer programs (modules) for stand alone use and for integration into other structural analysis programs. It is expected that the 3-D inelastic analysis capability being developed under this program will provide considerable flexibility for the solution of 3-D nonlinear structural problems; eventually it should lead to longer lifetimes and improved overall durability of the hot section components made from present and future materials. Also, this capability will provide enhanced capability to experimentally evaluate constitutive relationships.

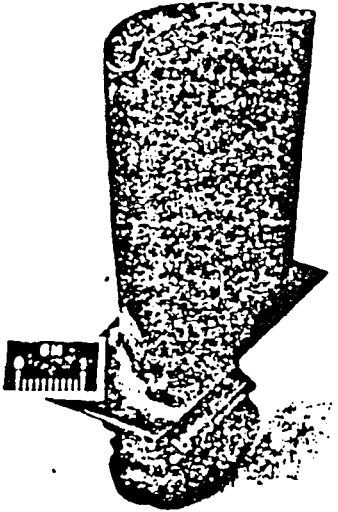
GENERAL SCOPE OF WORK

This program is a four year, 45,000 man-hour effort.

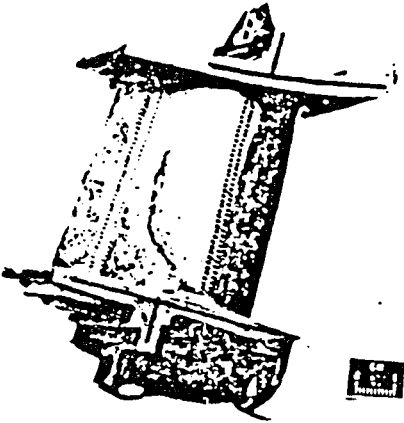
HOT SECTION COMPONENTS REQUIRING 3-D
INELASTIC ANALYSIS



COMBUSTOR LINER

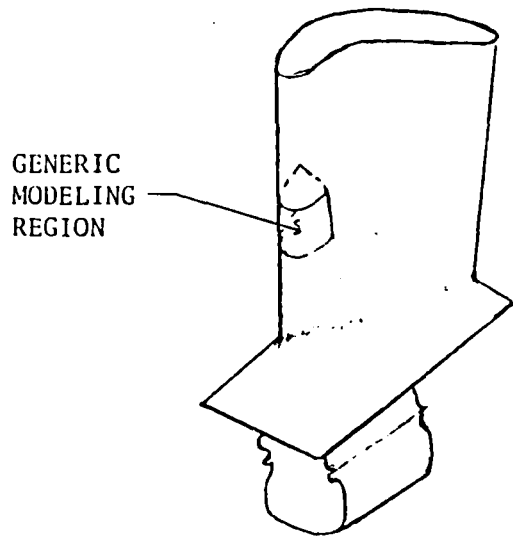


TURBINE BLADE

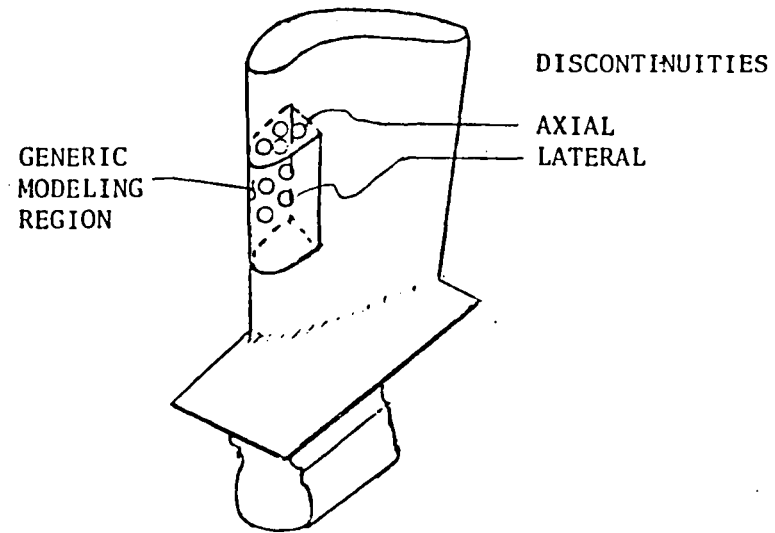


TURBINE VANE

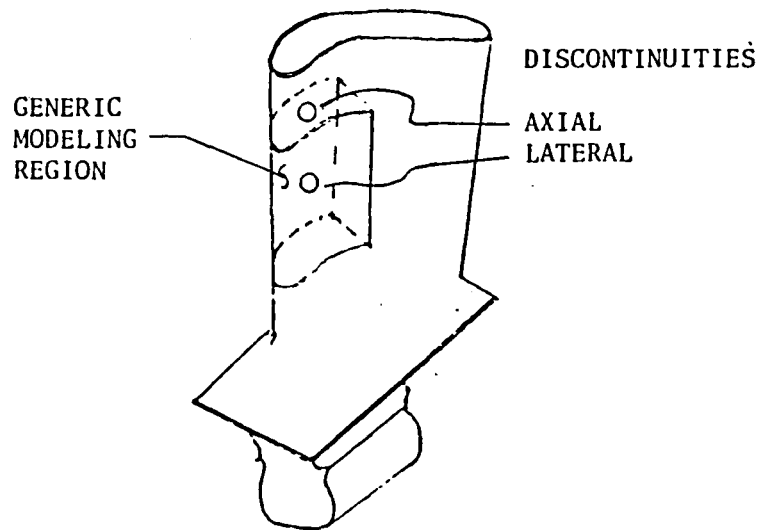
SCHEMATIC ILLUSTRATING REGIONS FOR THE FOUR
DIFFERENT ORDER THEORIES



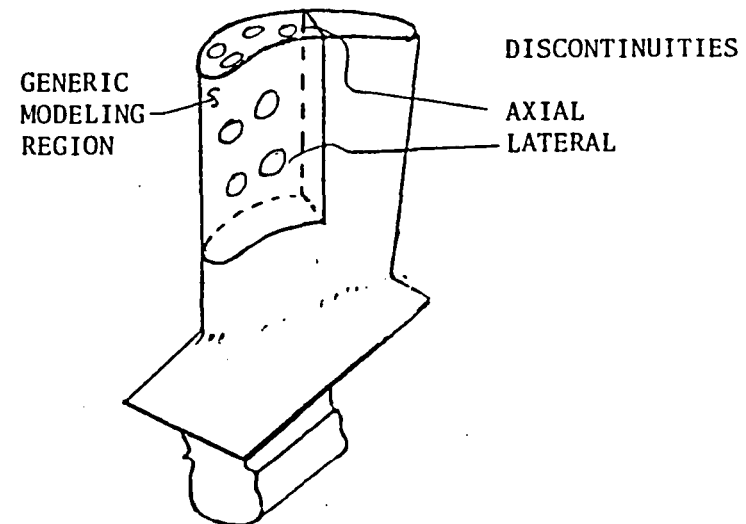
LINEAR



POLYNOMIAL



SPECIAL FUNCTIONS



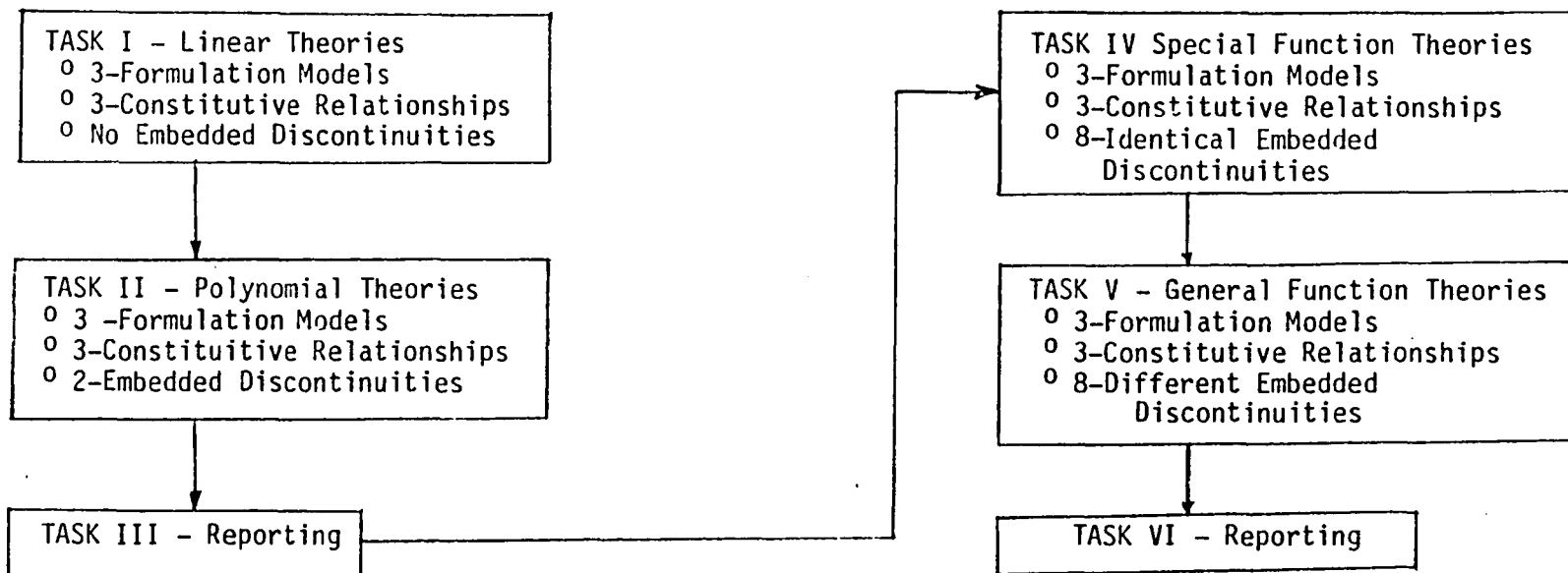
GENERAL FUNCTIONS

3-D INELASTIC ANALYSIS OF HOT SECTION COMPONENTS

FLOW CHART

Base Program

OPTION 1



207

NOTE: Each technical Task (I, II, IV and V) consists of three (3) subtasks describing the formulation models and constitutive relationships.

3-D INELASTIC ANALYSIS: TASK TIME SCHEDULE

TASK NO.	DESCRIPTION	PERCENT EFFORT	TIME FROM DATE OF CONTRACT YEARS				
			1	2	3	4	5
<u>BASE PROGRAM</u>							
I	Linear Theories	15	[Bar from Year 0 to 1]				
II	Polynomial Theories	22		[Bar from Year 0.5 to 2]			
A	Common Requirements for Tasks I and II						
III	Reporting (Base Program)	1	[Bar from Year 0 to 2.2]				
<u>Option 1</u>							
IV	Special Functions Theories	26		[Bar from Year 1.8 to 3.2]			
V	General Functions Theories	34		[Bar from Year 2.2 to 4.2]			
B	Common Requirements for Tasks IV and V						
VI	Reporting (Option 1)	2		[Bar from Year 1.8 to 4.8]			

208

Base Program (17,000 Man-hours) Option 1 (28,000 Man-hours)

LIFE PREDICTION AND CONSTITUTIVE BEHAVIOR -- OVERVIEW

Gary R. Halford
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

One of the primary drivers that prompted the initiation of the HOST Program was the recognized need for improved cyclic durability of costly hot section components. All too frequently, fatigue in one form or another was directly responsible for the less than desired durability and prospects for the future weren't going to improve unless a significant effort was mounted to increase our knowledge and understanding of the elements governing cyclic crack initiation and propagation lifetime. Certainly one of the important ingredients was the ability to perform accurate structural stress-strain analyses to determine the magnitudes of the localized stresses and strains since it is these localized conditions that govern the initiation and crack growth processes.

Consequently, the programs that evolved included high-temperature cyclic constitutive behavior work as well as cyclic life prediction methods development. Figure 1 lists the areas for which funding was sought. Our initial intent was to fund four programs, Life Prediction and Constitutive Modelling for Isotropic Materials and Life Prediction and Constitutive Modelling for Anisotropic Materials. The latter two have been combined into a single program for a number of technical and managerial reasons. Another change to our plans is the possibility of funding two contracts for the Constitutive Modelling of Isotropic Materials. This is largely because of the relative newness of this research area and the proliferation of competing theories.

Furthermore, additional funding has permitted us to pursue programs in the area of high temperature cyclic crack growth. Three such programs are in the planning stages: a contractual effort aimed at the problem of high temperature crack growth; and two University Grant activities, one being directed at the micromechanisms of high temperature crack growth, and the other involving an interdisciplinary approach to the overall problem of crack initiation, crack growth, and final fracture. Only the proposed contractual program will be discussed today. The University Grant programs will be reviewed next year. A milestone chart is shown in Figure 2 for the six contract and grant programs.

While the details of test programs will be given by the individuals that are intimately involved, I would like to emphasize the underlying objective of these programs: The development and verification of workable engineering methods for the calculation, in advance of service, of a) the local cyclic stress-strain response at the critical life governing location in typical hot section components, and b) the resultant cyclic crack initiation and crack growth lifetimes.

A contract has been in existence with Pratt & Whitney Aircraft since the first of June 1982, and the P&WA Project Manager, Vito Moreno, will be making the presentation covering that work. The other efforts will be described by the individuals responsible for creating the Request for Proposal Packages.

A Grant has been awarded to Professor H. W. Liu of Syracuse University for studies of the mechanisms of high temperature crack growth.

In addition, to the contract and grant programs, we are up-grading our in-house High Temperature Fatigue Laboratory capabilities as indicated in Figure 3.

FIG. 1

OVERVIEW

LIFE PREDICTION & CONSTITUTIVE BEHAVIOR

G. R. Halford, LeRC

- CYCLIC CRACK INITIATION
- CYCLIC CRACK GROWTH
- CYCLIC CONSTITUTIVE BEHAVIOR
- LeRC FATIGUE FACILITY UP-GRADING

FIG. 2

LIFE PREDICTION & CONSTITUTIVE MODELLING-
CONTRACTS & GRANTS

PROGRAM ELEMENT	FY83	FY84	FY85	FY86	FY87	FY88
ISOTROPIC CREEP-FATIGUE LIFE PREDICTION (C)	NAS 3-23288-PSMA (MORENO)					
ISOTROPIC CONSTITUTIVE MODELLING (C)	COMPETITIVE--POSSIBLY 2 AWARDS					
ANISOTROPIC LIFE PREDICTION/CONSTITUTIVE MODELLING (C)	COMPETITIVE					
FATIGUE CRACK GROWTH MECHANISMS (GRANT)	SYRACUSE UNIVERSITY (LIU)					
HIGH TEMPERATURE CRACK GROWTH (C)	COMPETITIVE					
INTERDISCIPLINARY CRACK LIFE (GRANT)	TO BE DETERMINED					

FIG. 3 FATIGUE FACILITY ENHANCEMENT

- **Servo-Controlled Testing Machine/Hi-Temp Crack Growth**
- **Servo-Controlled Tension/Torsion Machines(3)-Blaxial Studies**
- **Host/Satellite Computer Installation**
 - **Data Acquisition/Processing/Storage/Retrieval**
- **HCF/LCF Machines-Cumulative Damage Studies**

CREEP-FATIGUE LIFE PREDICTION FOR ENGINE HOT
SECTION MATERIALS (ISOTROPIC)

Vito Moreno
Program Manager
Pratt & Whitney Aircraft
Commercial Engineering

OBJECTIVES:

The objectives of this program are the investigation of fundamental approaches to high temperature crack initiation life prediction, identification of specific modeling strategies and the development of specific models for component relevant loading conditions.

PROGRAM DESCRIPTION AND APPROACH:

The sixty month technical program is divided into two sub-programs which contain a total of thirteen tasks. The basic program (Tasks I-IV) represents a 24-month effort. Task I includes a survey of the hot section material/coating systems used throughout the gas turbine industry. Two material/coating systems will be identified for the program. The material/coating system designated as the base system shall be used throughout Tasks I-XII. The alternate material/coating system will be used only in Task XII for further evaluation of the models developed on the base material. In Task II, candidate life prediction approaches will be screened based on a set of criteria that includes experience of the approaches within the literature, correlation with isothermal data generated on the base material, and judgements relative to the applicability of the approach for the complex cycles to be considered in the option program. The two most promising approaches will be identified. Task III further evaluates the best approach using additional base material fatigue testing including verification tests. Task IV consists of technical, scheduler, financial and all other reporting requirements in accordance with the Reports of Work clause. This activity concludes the basic program.

The optional program (Tasks V-XIII) represents a 36-month effort. Specific crack initiation prediction models will be developed within the various tasks to address various aspects associated with hot section life prediction. Task V considers the development of thermal-mechanical fatigue models for uncoated and coated structures. Task VI addresses multiaxial stress state effects. Task VII considers a cumulative loading model to address sub-cycle and block loading effects. A screening of available environmental and protective coating models is conducted in Task VIII. Also, the extent of the problem for thermal-mechanical cycling will be quantified in this task. Tasks IX and X consider the development of environmental attack and protective coating models. Task XI addresses the effects of mean stress in the creep-fatigue initiation process. In Task XII final verification of the model(s) developed in the previous tasks will be conducted. In addition, baseline isothermal, TMF and biaxial testing will be conducted in the alternate material/coating system to assess the applicability of the approaches and models developed on the base material. Task XIII consists of technical, scheduler, financial and all other reporting requirements in accordance with the Reports of Work clause.

PROGRAM STATUS:

- Basic program started 5-27-82.
- Cast B1900 + Hf designated as base material.
- Wrought IN718 designated as alternate material.
- Diffusion Aluminide (NiAl) and Overlay (MCrALY) coatings selected.
- Single heat (2500 lb.) of B1900 + Hf acquired.
- 24 bars cast for initial specimen fabrication.
- Review of life prediction approaches initiated.

CONSTITUTIVE MODEL DEVELOPMENT FOR ISOTROPIC MATERIALS

Albert Kaufman

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Cleveland, Ohio 44135

Abstract

The trend toward increased performance for aircraft gas turbine engines has resulted in higher turbine blade tip speeds and higher inlet gas temperatures and pressures. These more severe operating conditions have reduced the durability of hot section components and have demonstrated the need to improve upon the current analytical methods used in the design of these components.

Under a recently instituted contract effort undertaken as part of the HOST program, crack initiation life prediction methods will be developed for hot section components fabricated from isotropic materials. To apply these methods it is first necessary to determine the component structural response, specifically the stress-strain history at the critical cracking location. The structural analysis method must be capable of accounting for cyclic thermomechanical loading, plastic flow during thermal transients, creep and stress relaxation during steady-state operation and inelastic strain ratchetting and reversal due to repeated flight cycles.

In recent years, nonlinear finite element computer codes such as ANSYS and MARC have become available for cyclic analysis of components involving inelastic strains. These codes are based on classical plasticity theory and use uncoupled creep constitutive models. The classical methods utilize simplifying assumptions for computational convenience. Among these assumptions are (1) the definition of a specific yield surface with associated flow rules and hardening models, (2) the partitioning of inelastic strains into time-independent (plastic) and time-dependent (creep) strains and (3) the uncoupling of time-independent and time-dependent inelastic strain effects. That these classical methods and their assumptions do not realistically represent superalloy material behavior under cyclic loading have been demonstrated in two pre-HOST programs (the turbine blade durability study reported in NASA CR-165268 and the combustor liner durability study reported in NASA CR-165250).

The objective of this program is to develop a unified constitutive model for finite-element structural analysis of turbine engine hot section components. This effort constitutes a different approach for nonlinear finite-element computer codes which have heretofore been based on classical inelastic methods. A unified constitutive theory will avoid the simplifying assumptions of classical theory and should more accurately represent the behavior of superalloy materials under cyclic loading conditions and high temperature environments. Model development will be directed toward isotropic, cast nickel-base alloys used for aircooled turbine blades and vanes. The Contractor will select a Base Material for model development and an Alternate Material for verification purposes from a list of three alloys specified by NASA. The candidate alloys represent a cross-section of turbine blade and vane materials of interest to both large and small size engine manufacturers. Material stock for the Base and Alternate Materials will be supplied to the Contractor by the Government.

The contractual effort will be conducted in two phases, a Basic Program of two years duration and an optional follow-on program also of two years duration. In

the Basic Program, a unified constitutive model will be developed for the prediction of the structural response of isotropic materials for the temperatures and strain ranges characteristic of cooled turbine vanes in advanced gas turbine engines. A data base of uniaxial and multi-axial material properties required for the constitutive model will be obtained for the Base Material. The constitutive model will be incorporated into a finite-element computer code. An evaluation will be made of the capability of the analytical method to predict structural response for multi-axial stress states and nonisothermal conditions by conducting thermomechanical loading and benchmark notch verification experiments and analyses. As a final evaluation of the analytical methods, a structural analysis will be performed for a hot section component fabricated of the Base Material for simulated engine operating conditions. In the optional program entitled Option 1, further development will be undertaken to consider thermal history effects and to correct any deficiencies indicated in the constitutive model or in the computational algorithms in the code. The material property test procedure will be developed to minimize the amount of testing required, estimate the model material constants from conventional property data, and account for coating effects. In addition, the constitutive model development will be verified for an alternate material.

**CONSTITUTIVE MODEL DEVELOPMENT
FOR ISOTROPIC MATERIALS**

OBJECTIVE:

**TO DEVELOP A UNIFIED CONSTITUTIVE MODEL FOR
REPRESENTING CYCLIC INELASTIC BEHAVIOR OF
ISOTROPIC CAST NICKEL-BASE ALLOYS USED FOR
AIRCOOLED GAS TURBINE BLADES AND VANES.**

BASIC PROGRAM

SCREENING OF CANDIDATE CONSTITUTIVE MODELS

SPECIMEN FABRICATION AND TESTING

UNIAXIAL EVALUATION OF CONSTITUTIVE MODELS

BASIC PROGRAM

IMPLEMENTATION OF MODELS IN F. E. CODE

MULTIAXIAL EVALUATION OF CONSTITUTIVE MODELS

BENCHMARK NOTCH EXPERIMENTS

BASIC PROGRAM

COMPONENT DEMONSTRATION PROBLEM

DELIVERY OF COMPUTER CODE TO NASA

OPTION 1

DEVELOPMENT OF MAT. PROP. TEST PROCEDURE

FINAL DEVELOPMENT OF CONSTITUTIVE MODEL

MODEL VERIFICATION FOR ALTERNATE MATERIAL

LIFE PREDICTION AND CONSTITUTIVE MODELS
FOR ANISOTROPIC MATERIALS

Robert C. Bill
Propulsion Laboratory
AVRADCOM Research and Technology Laboratories
Lewis Research Center
Cleveland, Ohio

The trend toward improved engine efficiency and durability is placing increased demands on gas turbine materials, especially in the hot section. New materials and coatings are being developed to meet these demands. A particular area of challenge is in the turbine airfoil components. Here single (SC) and directionally solidified or recrystallized (DSR) polycrystalline materials are finding application. A difficulty impeding the full implementation of SC or DSR materials is the limited knowledge and understanding of failure (crack initiation) mechanisms and constitutive behavior.

The intent of this program is to develop a basic understanding of cyclic creep-fatigue deformation mechanisms and damage accumulation, a capability for reliable life prediction, and the ability to model the constitutive behavior of anisotropic SC and DSR materials employed in turbine airfoils. Four options comprise the program, and the work breakdown for each option reflects a distinct concern for two classes of anisotropic materials, SC and DSR materials, at temperatures encountered in the primary gas path (airfoil temperatures), and at temperatures typical of the blade root attachment and shank area. Work directed toward the higher temperature area of concern in the primary gas path includes effects of coatings on the behavior and properties of the materials of interest. The blade root attachment work areas will address the effects of stress concentrations associated with attachment features.

CONTRACT: CYCLIC CONSTITUTIVE MODELING AND LIFE PREDICTION METHODS
FOR ANISOTROPIC MATERIALS.

OBJECTIVE: DEVELOP AND VERIFY CYCLIC MATERIALS CONSTITUTIVE MODELS AND
LIFE PREDICTION METHODS FOR COMPONENT SPECIFIC ANISOTROPIC
MATERIALS FOR USE IN STRUCTURAL ANALYSIS COMPUTER PROGRAMS.

DURATION: 5 YEAR, 35 MAN-YEAR EFFORT.

PROGRAM STRUCTURE: FOUR PHASE PROGRAMS, EACH PHASE AN OPTION

- o PHASE 1 - COATED SC AIRFOILS
- o PHASE 2 - COATED DS AIRFOILS
- o PHASE 3 - UNCOATED SC BLADE ROOT
- o PHASE 4 - UNCOATED DS BLADE ROOT

APPROACH:

- o SELECT MATERIALS AND COATINGS
- o SCREEN ADVANCED CONSTITUTIVE AND LIFE PREDICTION MODELS; SELECT BEST
- o INTEGRATE MODELS WITH STRUCTURAL ANALYSIS PROGRAMS
- o VERIFY IN SIMULATED COMPONENT TEST

PROGRAM WILL ADDRESS:

- o CREEP-FATIGUE
- o THERMOMECHANICAL FATIGUE
- o ORIENTATION EFFECTS
- o COATING/SUBSTRATE INTERACTIONS
- o BIAXIAL LOADING
- o ATTACHMENT STRESS CONCENTRATIONS

HOST High Temperature Crack Propagation

Thomas W. Orange
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

This program will attempt to build on the results of the Pre-HOST program I described yesterday, using the latest analytical and experimental fracture mechanics techniques. The anticipated program and its key features are summarized on the first slide. A flowchart is shown on the next slide, and I will describe the work to be done.

First we will attempt to extract additional information from the literature and from the Pre-HOST program. Specifically, nonlinear finite element (NLFE) analyses of the tubular specimens will be made using GAP ELEMENTS to calculate crack opening loads. Then, using these loads, the data will be re-analyzed to see if improved correlations result.

Various specimen configurations will be evaluated for elevated-temperature isothermal and TMF testing with tension-compression loading, with a new constraint being that now crack mouth opening displacement (CMOD) is to be measured. The most suitable specimen will be selected. Available methods for measuring CYCLIC CMOD at elevated temperatures on the selected specimen will be evaluated. Possible methods for measuring near-tip displacements are also to be identified. An "analog material" will be selected. This will be a material suitable for simulating high-temperature material behavior but at temperatures only a few hundred degrees F above ambient. The object of the analog material is to permit well-instrumented tests to be run at reasonable temperatures to ease the instrumentation and cost problems.

An experimental program similar in scope and approach to Task IV of NAS3-22550 will be run with additional instrumentation (at least CMOD) and more detailed analysis (a NLFE program with gap elements). The specimen, measurement methods, and material previously identified will be used, with both isothermal and simple (linear) TMF tests to be run. A limited series of tests using a different specimen configuration will be run to see if the resulting growth rate correlation is indeed specimen-independent.

We intend to evaluate several formulations which have been proposed for nonlinear fracture analysis in the presence of thermal gradients which result in material inhomogeneity. These will include the analyses of Blackburn et al, Ainsworth et al, Wilson & Yu, Kishimoto et al, and Atluri. Methods and strategies for performing the necessary calculations using a NLFE program will be considered. The five most promising formulations will be evaluated using the simple analytical model of an edge crack in a large plate, with a linear temperature gradient (and/or corresponding modulus variation) in the direction of crack propagation and a uniform distribution normal to the crack plane. The five formulations will be compared and two selected for further evaluation. One will be that judged to have the

most technical merit, the other the best compromise between technical merit and computational ease. Then these two formulations will be evaluated for further use by modeling the actual specimen geometry and temperature gradient to be used in later tests. Next we wish to determine whether the analytical formulations identified previously actually enable one to correlate nonlinear crack growth in the presence of thermal gradients. Specimens of the analog material will be tested under monotonic and cyclic load in the presence of a simple (nearly linear) thermal gradient.

At this point it will be NASA's option to proceed with the optional program as planned, to technically redirect the optional program by re-negotiation, or to terminate.

The optional program will consist of three main elements. We intend to extend the analytical effort to include a comparison of path-independent RATE integrals (or time derivatives of path-independent integrals), and this will be done in much the same manner as before. Then we will attempt to verify these analyses using the analog material at a somewhat higher temperature than before (into the creep range). Finally we will attempt to apply the knowledge gained using the analog material to predict and correlate crack propagation in a nickel-base alloy at temperatures typical of combustor liners.

ELASTOPLASTIC CRACK PROPAGATION AT ELEVATED TEMPERATURES

OBJECT: DEVELOP METHODS FOR CHARACTERIZING & PREDICTING CRACK GROWTH AT ELEVATED TEMPERATURES CONSIDERING NONLINEAR MATERIAL BEHAVIOR, THERMAL GRADIENTS & THERMOMECHANICAL CYCLING.

SCOPE: FOUR-YEAR TWO-PHASE CONTRACT.

FEATURES:

- * SURVEY & COMPARISON OF CURRENT PATH-INDEPENDENT INTEGRALS.
- * COMPUTER TEST OF ≥ 5 PATH-INDEPENDENT INTEGRALS USING SIMPLE PROBLEM.
- * EXTENSIVE USE OF CRACK DISPLACEMENT MEASUREMENTS FOR BETTER UNDERSTANDING.
- * ANALOG MATERIAL AT MODERATE TEMPERATURES ($\leq 500F$) FOR PHASE I: NI-BASE ALLOY AT $< 2000F$ IN PHASE II.
- * CYCLIC CRACK PROPAGATION TEST WITH TEMPERATURE GRADIENTS, THERMOMECHANICAL CYCLING & CREEP.

BASIC PROGRAM

RE-ANALYZE
PRE-HOST DATA

o EVALUATE SPECIMENS
o CMOD, CTOD METHODS
o ANALOGUE MATERIAL

SURVEY & EVALUATE
P-I INTEGRALS

EXPERIMENT:
ISO & TMF DA/DN

EXPERIMENT:
TEMPERATURE GRADIENT

OPTIONAL PROGRAM

EVALUATE P-I
RATE INTEGRALS

EXPERIMENT:
REPEAT WITH
NI-BASE ALLOY

EXPERIMENT:
DA/DT (SIMPLE)

EXPERIMENT:
DA/DT (COMPLEX)

EXPERIMENT:
COMPLEX WITH
NI-BASE ALLOY

SURFACE PROTECTION OVERVIEW

Stanley R. Levine
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

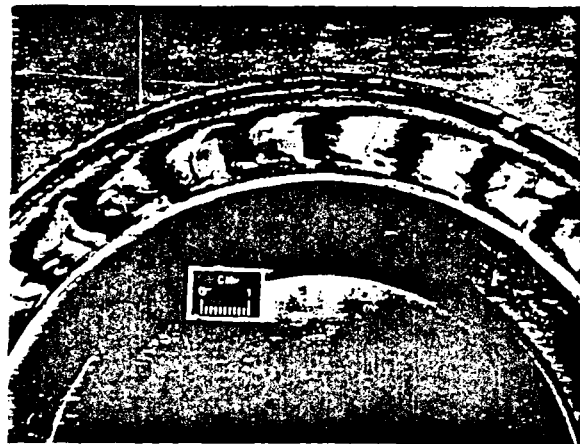
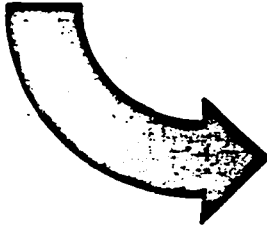
Turbine engine hot section materials are subjected to aggressive chemical and thermomechanical environments. High temperature environmental attack of dollar intensive turbine components reduces turbine efficiency and can limit life. The bottom line, of course, is that high temperature oxidation and hot corrosion attack costs you money. The objective of materials durability research at Lewis is to understand the mechanisms of alloy and coating attack, and the effects of interaction with the environment on mechanical behavior. This base of understanding provides the foundation for developing life prediction methods and identifying strategies for controlling attack via advanced metallic and ceramic coatings. The Turbine Engine Hot Section Technology Project (HOST) augments the life prediction area of our program.

Our objective under HOST is to develop a first-cut integrated environmental attack life prediction methodology for hot section components. Under HOST we are concerned with oxidation and hot corrosion attack of metallic coatings as well as their degradation by interdiffusion with the substrate. The effects of the environment and coatings on creep/fatigue behavior are being addressed through a joint effort with the Fatigue sub-project. Finally, an initial effort will attempt to scope the problem of thermal barrier coating life prediction. Verification of models will be carried out through benchmark rig tests including a 4 atm. replaceable blade turbine and a 50 atm. pressurized burner rig.

SURFACE PROTECTION OVERVIEW

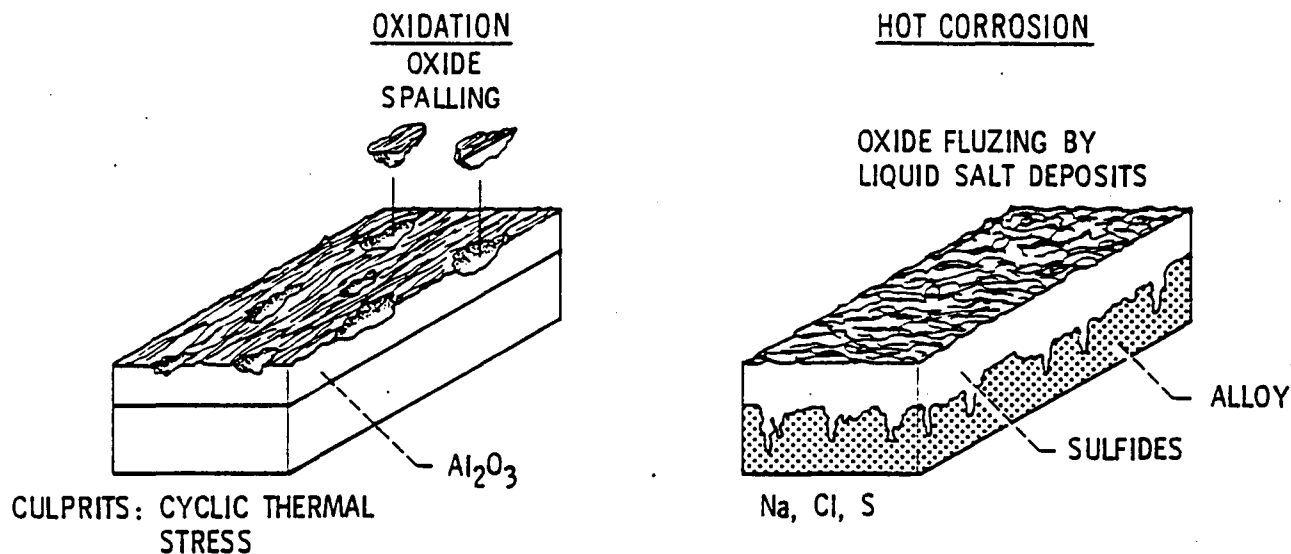
**S. R. LEVINE
MATERIALS DIVISION
MATERIALS DURABILITY BRANCH**

**HIGH TEMPERATURE
ENVIRONMENTAL ATTACK**



**REDUCES EFFICIENCY
LIMITS LIFE
COSTS \$**

SCHEMATIC OF MODES HIGH-TEMPERATURE ATTACK



SURFACE PROTECTION

OBJECTIVE: DEVELOP AN INTEGRATED ENVIRONMENTAL ATTACK LIFE PREDICTION METHODOLOGY FOR HOT SECTION COMPONENT LIFE

APPROACH:

- COMPILE FIELD FAILURE MODES DATA BASE
- MODEL EFFECTS OF ENVIRONMENTAL ATTACK AND COATINGS ON CRACK INITIATION
- CORROSION/~~EROSION~~ MODEL FOR AIRFOILS
- COATING OXIDATION/DIFFUSION MODEL
- COATING HOT CORROSION LIFE PREDICTION
- THERMAL BARRIER COATINGS
- MODEL VERIFICATION/INTEGRATION

COMMENTS:

- THIS WILL BE A FIRST CUT
- TIME, MANPOWER AND ARE INSUFFICIENT UNDER HOST TO DEVELOP A FULLY SATISFACTORY METHODOLOGY

SURFACE PROTECTION

PROGRAM ELEMENT	FY81	82	83	84	85	86	87	EXPECTED RESULTS
ENVIRONMENT/MECHANICAL PROPERTY INTERACTIONS (TASKS OF C/F CONTRACTS)			[Solid bar spanning FY 83-87]					MODEL FOR EFFECTS OF ENVIR. ATTACK & COATINGS ON CRACK INITIATION
AIRFOIL DEPOSITION MODEL	[Solid bar spanning FY 81-86]							MODEL TO PREDICT THE LOCATION & POTENTIAL SEVERITY OF CORROSION/ ATTACK OF TURBINE AIRFOILS
COATING LIFE PREDICTION	[Solid bar spanning FY 81-86]							CAPABILITY TO PREDICT COATING DEGRADATION ON BLADES, VANES, COMBUSTORS

SURFACE PROTECTION

PROGRAM ELEMENT	FY	81	82	83	84	85	86	87
ENVIRONMENT/MECH. PROP. INTERACTIONS				[Solid bar spanning FY 83-87]				
AIRFOIL DEPOSITION MODEL								
AIRFOIL MODEL		[Solid bar spanning FY 81-83]						
MODEL VERIFICATION (I)				[Dashed bar spanning FY 83-86]				
COATING LIFE PREDICTION								
OXIDATION/DIFFUSION (I)		[Solid bar spanning FY 81-83]						
HOT CORROSION SURFACE CHEMISTRY			[Dashed bar spanning FY 82-84]					
DUAL CYCLE ATTACK (I)				[Dashed bar spanning FY 83-86]				
LIFE PREDICTION VERIFICATION				[Dashed bar spanning FY 83-87]				
THERMAL BARRIER LIFE PRED.				[Dashed bar spanning FY 83-86]				[Wavy line]
RIG/ENGINE CORRELATION (I)				[Dashed bar spanning FY 83-87]				

AIRFOIL DEPOSITION MODEL

GOAL: DEVELOP THEORY TO PREDICT CORRODANT DEPOSITION ON TURBINE AIRFOILS

DURATION: 36 MONTHS

APPROACH:

- GRANT - DAN ROSNER, YALE
- EXTEND CHEMICALLY FROZEN BOUNDARY LAYER THEORY TO AIRFOILS
 - LAMINAR & TURBULENT FLOW
 - PRESSURE & TEMPERATURE OVER AIRFOIL
- MULTI-COMPONENT CORRODANTS

AIRFOIL DEPOSITION MODEL VERIFICATION

GOAL: VERIFY DEPOSITION MODEL AND INTEGRATE WITH ALLOY CORROSION RATE MODEL

DURATION: 48 MONTHS, IN-HOUSE

APPROACH: USING 4 ATM REPLACEABLE BLADE TURBINE RIG

- VERIFY AIRFOIL DEPOSITION MODEL
- VERIFY ALLOY RATE MODEL (FROM R&T BASE)
- INTEGRATE TO LOCATION/RATE MODEL

COATING OXIDATION/DIFFUSION LIFE PREDICTION

- GOAL: TO DEVELOP AN IMPROVED METHODOLOGY FOR PREDICTING OXIDATION LIFE OF METALLIC COATINGS
- DURATION: 36 MONTHS, IN-HOUSE
- APPROACH: SELECT, PROCURE COATED SPECIMENS
DETERMINE COATING LIFE VS TEMPERATURE
FURNACE
BURNER RIG
MEASURE SPALLING PARAMETERS FROM ISOTHERMAL OXIDATION
DETERMINE KINETICS OF DIFFUSIONAL DEGRADATION
DEVELOP CYCLIC OXIDATION MODEL WITH MODIFICATION FOR COATING DIFFUSIONAL DEGRADATION
TEST MODEL BY VARIATION OF CYCLE FREQUENCY

HOT CORROSION SURFACE CHEMISTRY

- GOAL: DETERMINE EFFECT OF SURFACE CHEMISTRY ON HOT CORROSION LIFE
- DURATION: 36 MONTHS
- APPROACH: • ANALYZE REPRESENTATIVE FIELD COMPONENTS
- DETERMINE COATING LIVES FOR VARIATIONS IN
ALLOY
COATING
COATING AGE
AGING METHOD
 - MODEL RESULTS/SUGGEST METHODOLOGY FOR LIFE PREDICTION
 - VERIFY METHODOLOGY

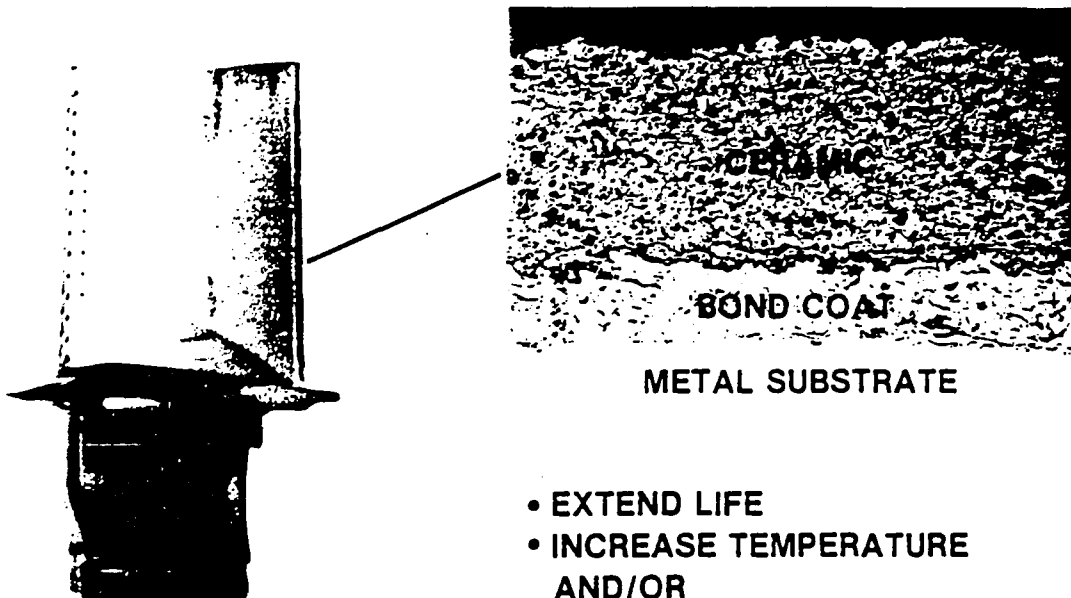
DUAL CYCLE ATTACK

- GOAL: CHARACTERIZE THE EFFECT OF COMBINED OXIDATION/HOT CORROSION CYCLIC EXPOSURE ON LIVES OF METALLIC COATINGS
- DURATION: 36 MONTHS (IN-HOUSE)
- APPROACH: DETERMINE MACH 0,3 BURNER RIG LIVES FOR COMBINED OXIDATION/HOT CORROSION CYCLIC EXPOSURE
- VARIATIONS IN DEPOSITION RATE
 - VARIATIONS IN TEMPERATURE LEVELS
 - VARIATIONS IN COATING AGE
- FIT RESULTS TO EMPIRICAL MODEL

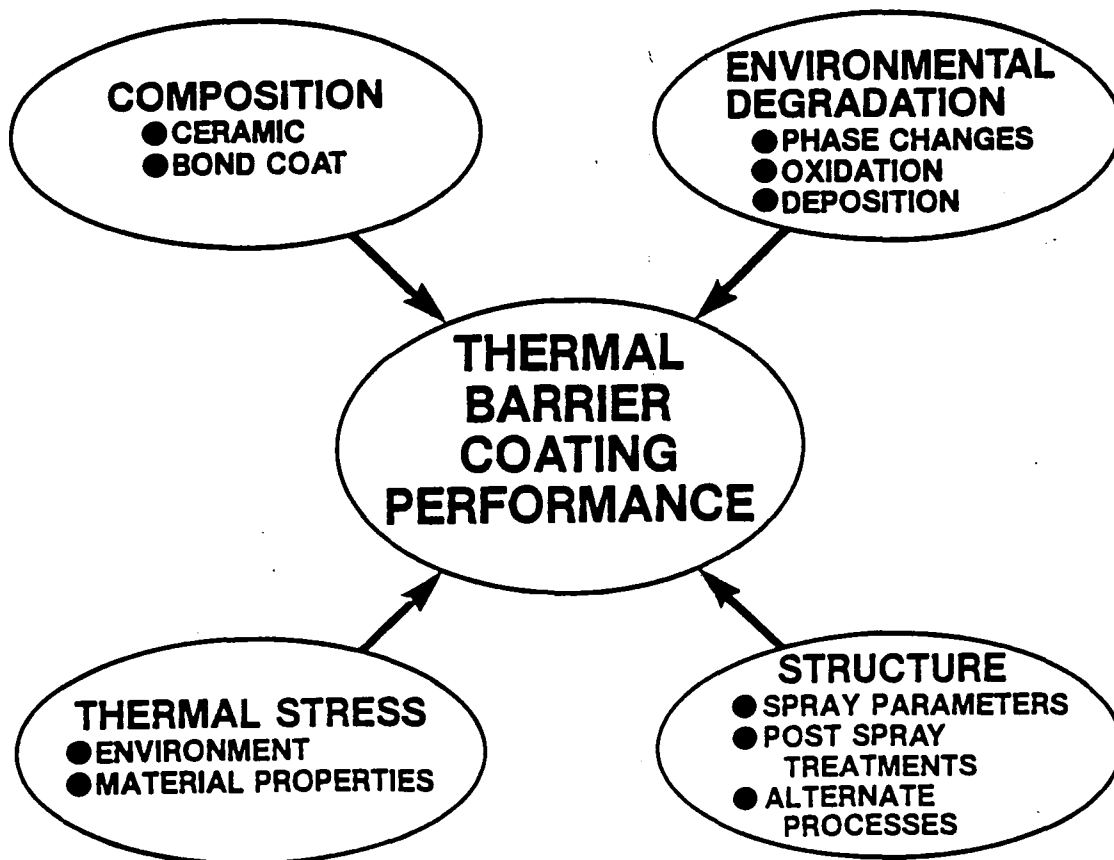
LIFE PREDICTION VERIFICATION

- OBJECTIVE: VERIFY COATING LIFE PREDICTION METHODOLOGY WITH BURNER RIG BENCHMARK TESTS
- DURATION: 30 MONTHS
- APPROACH: PREDICT COATING LIVES FOR SIMULATED LONG HAUL/SHORT HAUL COASTAL& INLAND MISSIONS
- SIMULATE MISSIONS&MEASURE COATING LIFE
- DIAGNOSE PREDICTIVE METHODOLOGY FOR DEFICIENCIES

THERMAL BARRIER COATINGS



- EXTEND LIFE
- INCREASE TEMPERATURE AND/OR
- REDUCE COOLING



TBC LIFE PREDICTION

GOAL: TO DEVELOP AN IMPROVED DESIGN/LIFE PREDICTION METHODOLOGY FOR
THERMAL BARRIER COATINGS

DURATION: 36 MONTHS

APPROACH: COMPILE COATING PROPERTY DATA
PREDICT COATING LIFE IN RIG & ENGINE
VARIATIONS IN:
COATING PROPERTIES
COATING THICKNESS
THERMAL CYCLE
TEST PREDICTIVE CAPABILITY BY RIG (& ENGINE) TEST

RIG/ENGINE CORRELATION

GOAL: VERIFY CORROSION, EROSION AND COATING LIFE MODELS AT NEAR
ENGINE CONDITIONS

DURATION: 36 MONTHS (IN-HOUSE)

APPROACH: VERIFY MODELS USING HIGH PRESSURE BURNER RIG
DEPOSITION/CORROSION
METALLIC COATING LIFE
THERMAL BARRIER COATING LIFE
DETERMINE MODEL DEFICIENCIES ATTRIBUTABLE TO
HIGH PRESSURE
HIGH HEAT FLUX

AIRFOIL DEPOSITION MODEL

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Lewis Research Center
Cleveland, Ohio 44135

Aircraft gas turbine failures associated with sea-salt ingestion and sulfur-containing fuel impurities focus attention on salt deposition and the attendant hot corrosion and fouling of gas turbine blades. However, in the past, quantitative understanding of deposition from gas turbine combustion gases has been impeded by the lack of a comprehensive yet tractable theoretical framework for organizing new deposition rate information. The present research program deals with the further development and exploitation of such a theory, and builds upon the foundation provided by previous NASA LeRC-sponsored research (Refs. 1-3 and references contained therein). The goal of this program is to develop the methodology to predict deposit evolution (deposition rate and subsequent flow of liquid deposits) as a function of fuel and air impurity content and relevant aerodynamic parameters for turbine airfoils. The program is carried out under a HOST-supported grant, "Theory of Mass Transfer from Combustion Gases" (NAG 3-201), with Professor Daniel E. Rosner and associates of the Chemical Engineering Department of Yale University.

The spectrum of deposition conditions encountered in gas turbine operations includes the mechanisms of vapor deposition, small particle deposition with thermophoresis, and larger particle deposition with inertial effects. In the present program the focus is on using a simplified version

of the comprehensive multicomponent vapor diffusion formalism to make deposition predictions for (1) simple geometry collectors and (2) gas turbine blade shapes, including both developing laminar and turbulent boundary layers. For the gas turbine blade the insights developed in previous programs are being combined with heat and mass transfer coefficient calculations using the "STAN 5" boundary layer code to predict vapor deposition rates and corresponding liquid layer thicknesses on turbine blades. A computer program is being written which utilizes the local values of the calculated deposition rate and skin friction to calculate the increment in liquid condensate layer growth along a collector surface. Preliminary results are now available for deposition and aerodynamic shear-driven flow of Na_2SO_4 on stationary cylinders and turbine blades.

Detailed results of progress to date appear in several papers and preprints (Refs. 4-9), copies of which can be obtained from Professor D. E. Rosner at Yale University, Department of Engineering and Applied Science, New Haven, CT 06520.

REFERENCES

1. D. E. Rosner, B.-K. Chen, G. C. Fryburg, and F. J. Kohl: "Chemically Frozen Multicomponent Boundary Layer Theory of Salt and/or Ash Deposition Rates from Combustion Gases," *Comb. Sci. and Tech.* 20, 87 (1979).
2. D. E. Rosner: "Thermal (Soret) Diffusion Effects on Interfacial Mass Transport Rates," *Physicochemical Hydrodynamics* 1, 159 (1980).
3. D. E. Rosner and K. Seshadri: "Experimental and Theoretical Studies of the Laws Governing Condensate Deposition from Combustion Gases," 18th Sympos. (Intl.) on Combustion, The Combustion Institute, 1981, p. 1385.
4. D. E. Rosner and J. Fernandez de la Mora: "Small Particle Transport Across Turbulent Nonisothermal Boundary Layers," *ASME Trans., J. Engin. for Power*, in press, 1982.
5. S. Gököglu and D. E. Rosner: "Correlation of Thermophoretically-Modified Small Particle Deposition Rates in Forced Convection Systems with Variable Properties, Transpiration Cooling and/or Viscous Dissipation," submitted to *Int. J. Heat and Mass Transfer*, 1982.
6. R. Israel and D. E. Rosner: "Use of a Generalized Stokes Number to Determine the Aerodynamic Capture Efficiency of Non-Stokesian Particles from a Compressible Gas Flow," *Aerosol Sci. and Tech.*, in press, 1982.

7. D. E. Rosner and J. Fernandez de la Mora: "Correlation and Prediction of Thermophoretic and Inertial Effects on Particle Deposition from Non-Isothermal Turbulent Boundary Layers," ASME Conf. on Particulate Laden Flows in Turbomachinery, St. Louis, MO, June 7-11, 1982.
8. D. E. Rosner, D. Günes, and N. Anous: "Aerodynamically-Driven Condensate Layer Thickness Distributions on Isothermal Cylindrical Surfaces," submitted to Chem. Engin. Commun., 1982.
9. D. E. Rosner, S. Gökçöglü, and R. Israel: "Rational Engineering Correlations of Diffusional and Inertial Particle Deposition Behavior in Non-Isothermal Forced Convection Environments," Engineering Foundation Int. Conference on Fouling of Heat Exchange Surfaces, White Haven, PA, Oct. 31-Nov. 5, 1982.

AIRFOIL DEPOSITION MODEL

GRANT NAG 3-201: "THEORY OF MASS TRANSFER FROM COMBUSTION GASES,"
WITH PROFESSOR D.E. ROSNER, CHE DEPARTMENT, YALE UNIVERSITY

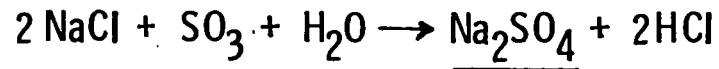
EMPHASIS: TRACE SALT VAPOR DEPOSITION AND CORRESPONDING SHEAR-
DRIVEN CONDENSATE LAYER FLOW

OBJECTIVE:

- OVERALL - DEVELOP MODEL TO PREDICT CORRODANT DEPOSITION ON
TURBINE AIRFOILS
- 1ST YEAR - MODEL DEPOSITION RATE FOR SEVERAL SIMPLE GEOMETRIES
- 2ND YEAR - PREDICT AND DISPLAY LIQUID LAYER EVOLUTION ON TURBINE
VANES AS A RESULT OF VAPOR DEPOSITION AND LIQUID LAYER FLOW

HOT CORROSION PROCESS

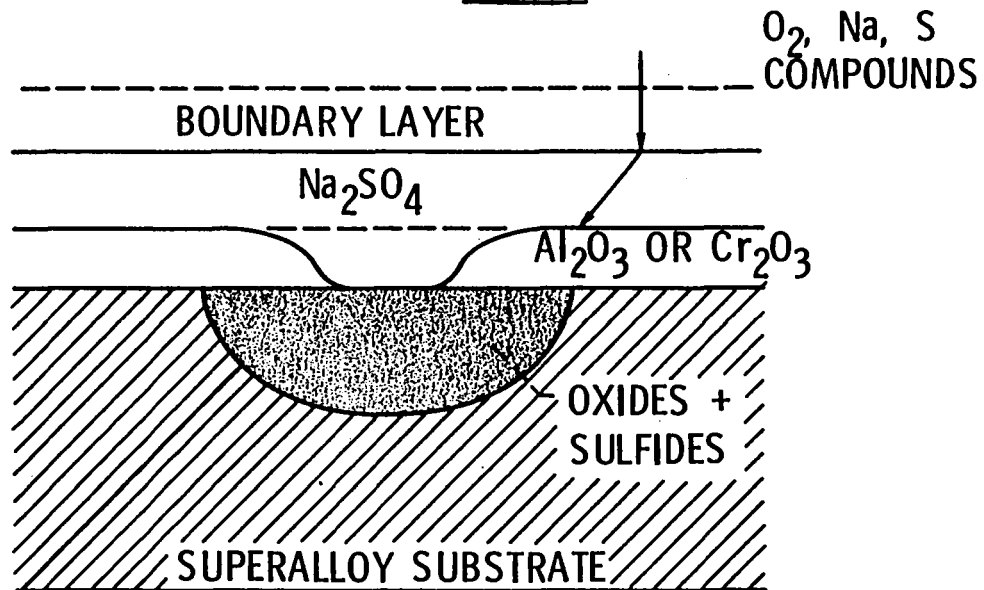
COMBUSTION
GASES



DEPOSITION



CORROSION



CHARACTERISTICS OF DEPOSITION FOR SPECTRUM OF PARTICLE SIZES

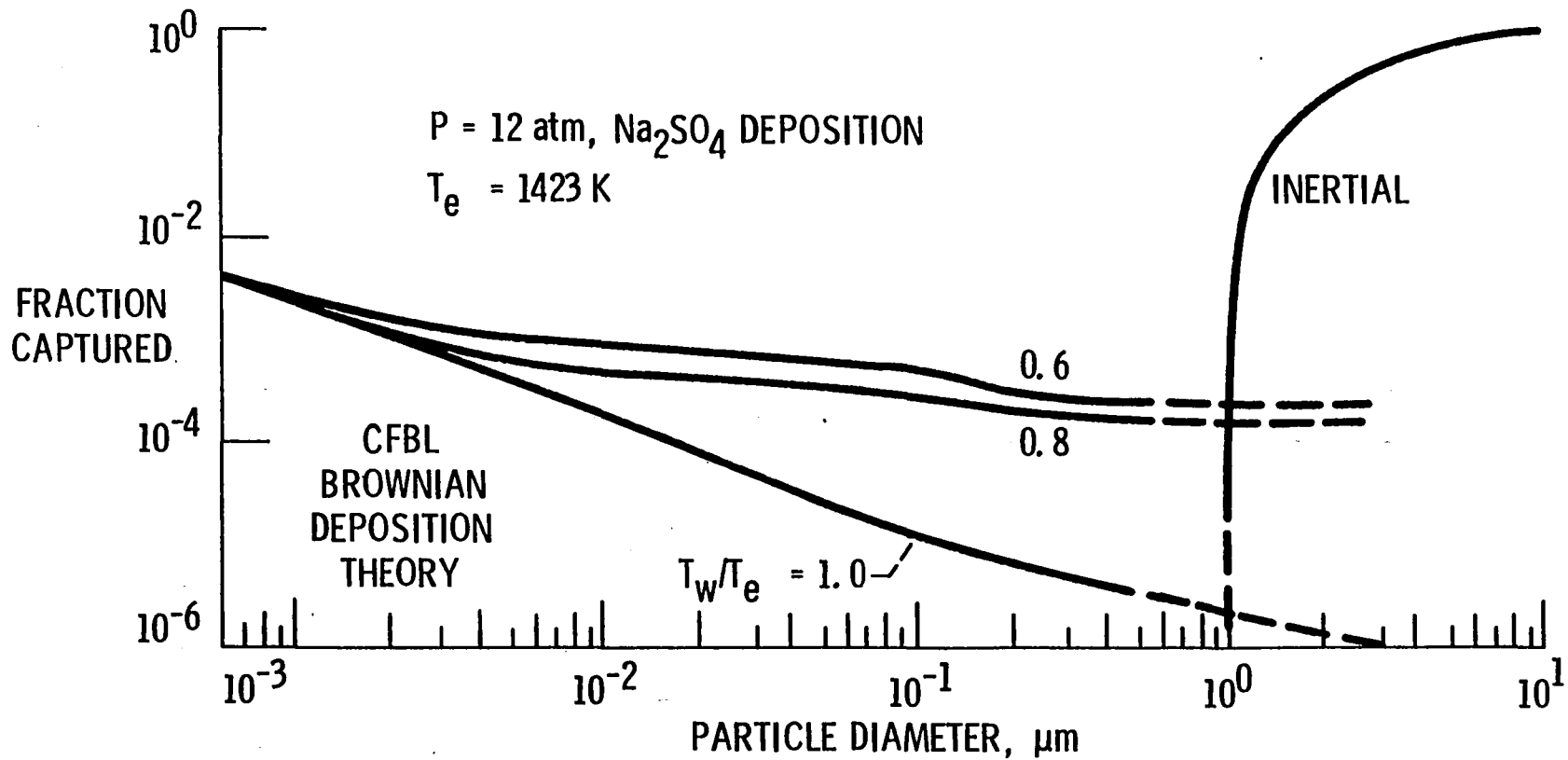
SIZE RANGE*	MASS TRANSPORT MODE	DEPOSITION SPECIES	TRANSPORT MECHANISM	DEPOSITION CHARACTERISTICS
$1-10\text{\AA}$	Vapor Diffusion	Atoms and Molecules (Vapors)	Fick Diffusion Soret Diffusion Eddy Diffusion	<ol style="list-style-type: none"> $T_{dp} < T_e$ Low η and deposition on side away from line-of-sight Low sensitivity to $T_e - T_w$ Rate levels off for $T_w \ll T_{dp}$
$10\text{\AA}-10^{-1} \mu\text{m}$	Vapor Diffusion Transition	Heavy Molecules (Condensate Aerosols, Clusters, Submicron Particles)	Brownian Diffusion Eddy Diffusion Thermophoresis	<ol style="list-style-type: none"> $T_{dp} = T_e$ Lowest η High sensitivity to $T_e - T_w$ Rate nearly linear with $T_e - T_w$
$10^{-1} - 100 \mu\text{m}$	Inertial	Macroscopic Particles	Inertial Impaction Eddy Impaction	<ol style="list-style-type: none"> No apparent T_{dp} Highest η Independent of $T_e - T_w$ Preferential deposition on side facing flow

* Mode of deposition is not fixed by particle size alone
 η = deposition or collection efficiency, T_{dp} = dew point temperature,
 T_e = gas mainstream temperature, T_w = wall temperature

CS-81-1182

PREDICTED DEPENDENCE OF SODIUM SULFATE DEPOSITION RATE ON PARTICLE SIZE

248



CS-81-1178

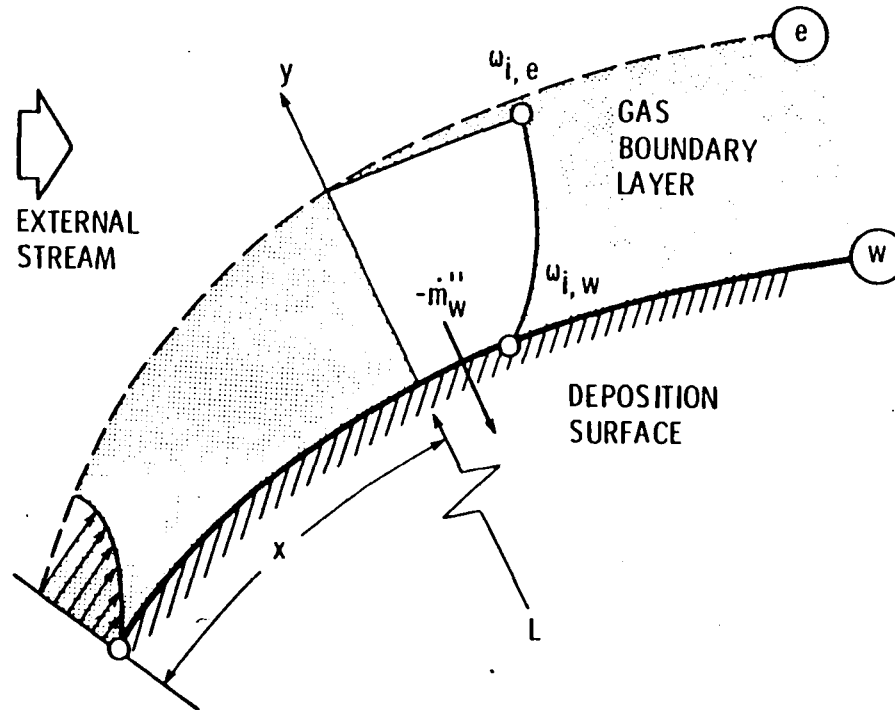
CHEMICALLY FROZEN BOUNDARY LAYER THEORY-CFBL

GOAL: PREDICT THE DEPOSITION RATE FOR TRACE INORGANIC SPECIES AS A FUNCTION OF SEED LEVEL, COLLECTOR GEOMETRY, THERMAL AND FLUID DYNAMIC PARAMETERS, ETC.

BASIC ASSUMPTIONS:

- 1. NO CONDENSATION OR CHEMICAL REACTION WITHIN THE MASS TRANSFER GASEOUS BOUNDARY LAYER**
- 2. CHEMICAL EQUILIBRIUM EXISTS AT THE VAPOR-CONDENSATE INTERFACE**
- 3. CHEMICAL SPECIES FOR TRANSPORT ACROSS THE BOUNDARY LAYER ARE VERY LOW IN CONCENTRATION**
- 4. TRANSPORT BY BODY FORCES AND PRESSURE DIFFUSION IS NEGLIGIBLE**

VAPOR DEPOSITION THEORY—CFBL

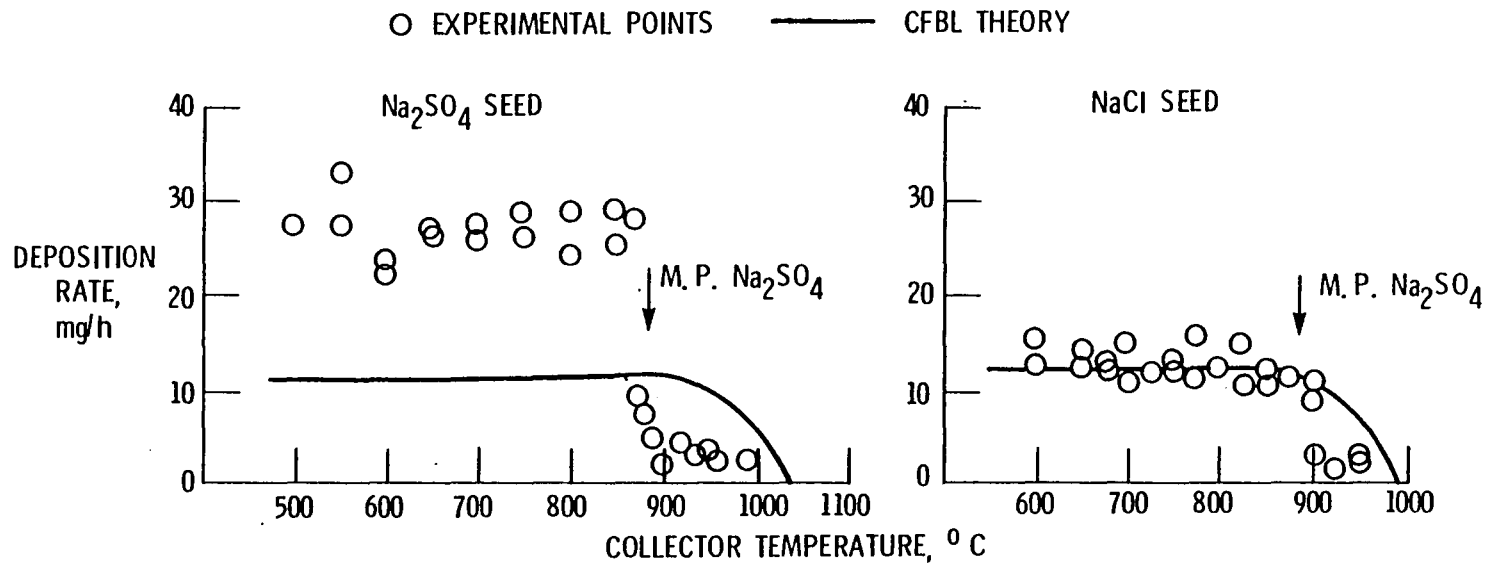


INTEGRATED DEPOSITION FLUX IS GIVEN BY

$$\dot{m}_i'' = \underbrace{\frac{(D_i \rho)_e}{L} \cdot F(\text{turb}) \cdot F_i(\text{Soret}) \cdot Nu_{m,i}(\text{Re}, Sc_i)}_{\text{TRANSFER COEFFICIENT}} \cdot \underbrace{\left[\Delta \omega_i + \frac{\tau_i}{F_i(\text{Soret})} \cdot \omega_{i,w} \right]}_{\text{DRIVING FORCE}}$$

(Fick) (Soret)

MACH 0.3 BURNER RIG DEPOSITION OF Na_2SO_4 ON ROTATING CYLINDRICAL COLLECTORS



LIQUID DEPOSIT LAYER FLOW

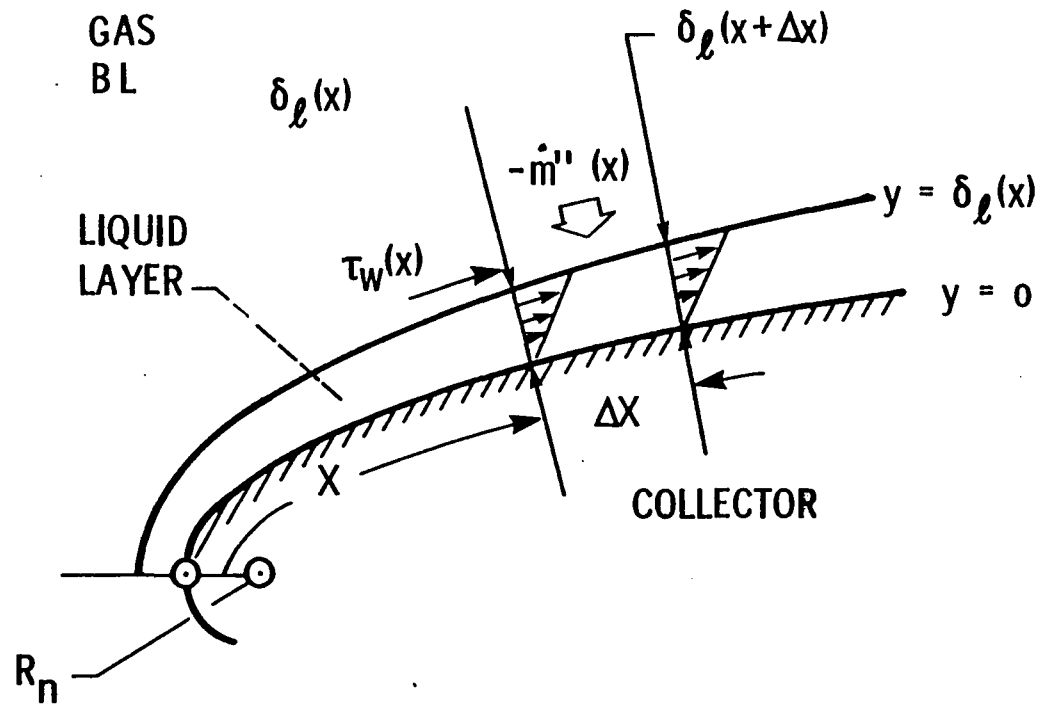
GOAL: PREDICT THE DISTRIBUTION OF LAYER THICKNESS WHEN AERODYNAMIC SHEAR IS THE DOMINANT MECHANISM OF CONDENSATE FLOW ALONG THE SURFACE

BASIC ASSUMPTIONS:

1. FILM IS THIN AND FLOW IS LAMINAR
2. LIQUID IS NEWTONIAN AND SHEAR STRESS ACROSS LAYER IS CONSTANT
3. OTHER BODY FORCES, SURFACE TENSION, ETC. ARE NEGLIGIBLE
4. SURFACE IS ISOTHERMAL

CS-82-2568

AERODYNAMICALLY-DRIVEN THIN CONDENSATE LAYER FLOW



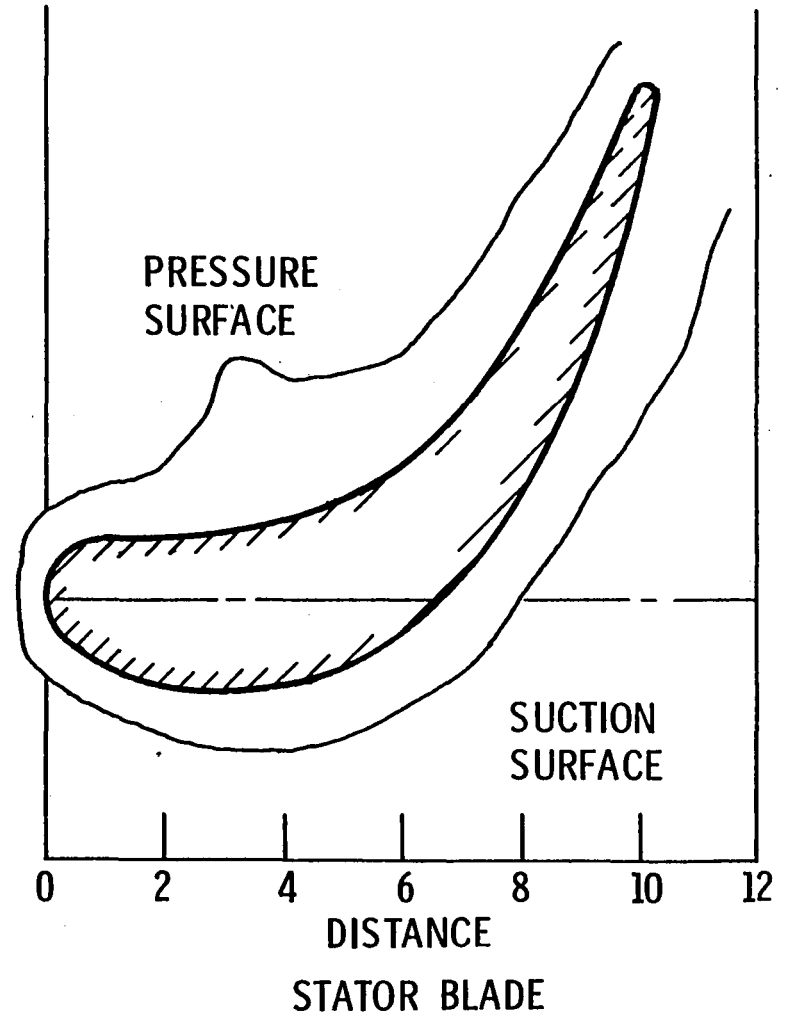
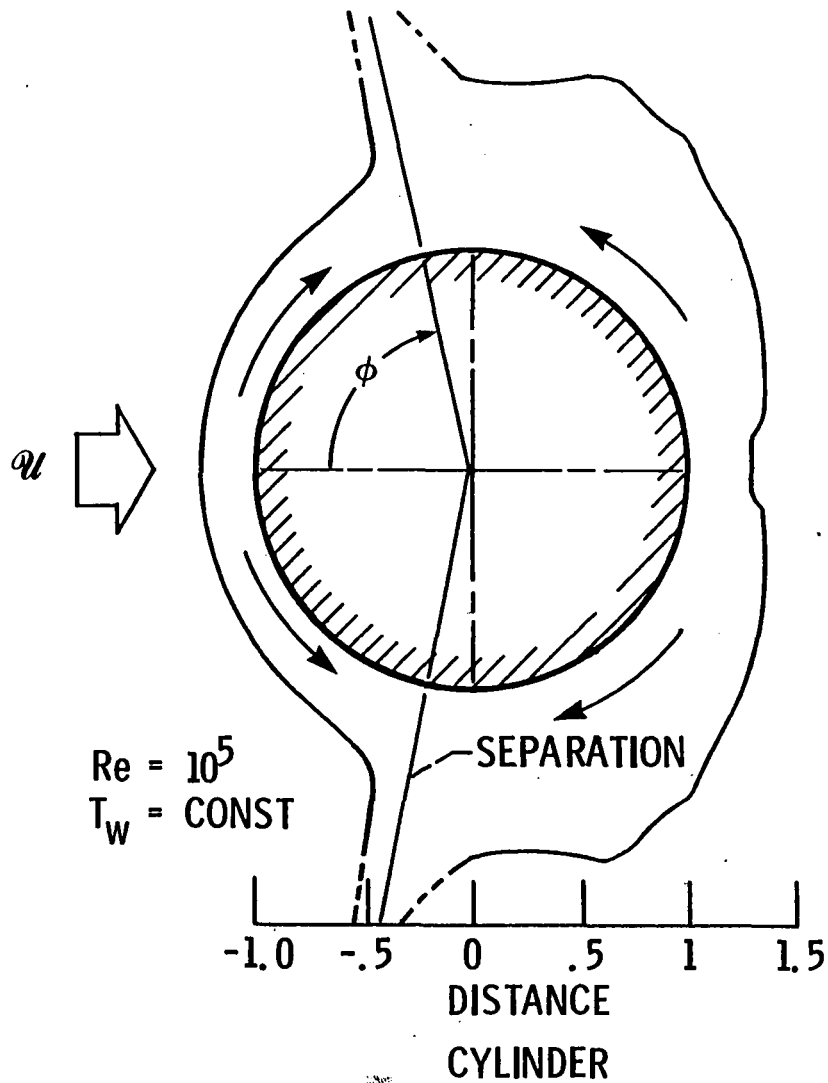
LIQUID LAYER THICKNESS, $\delta_\ell(x)$, IS GOVERNED BY

$$\frac{\partial \delta_\ell}{\partial t} + \frac{\partial}{\partial x} \left(\frac{\tau_w(x)}{2\mu_\ell} \cdot \delta_\ell^2 \right) = - \frac{\dot{m}''(x)}{\rho_\ell}$$

DEPOSIT THICKNESS AS A FUNCTION OF POSITION

VAPOR DEPOSITION AND LIQUID LAYER FLOW (NO SHEDDING)

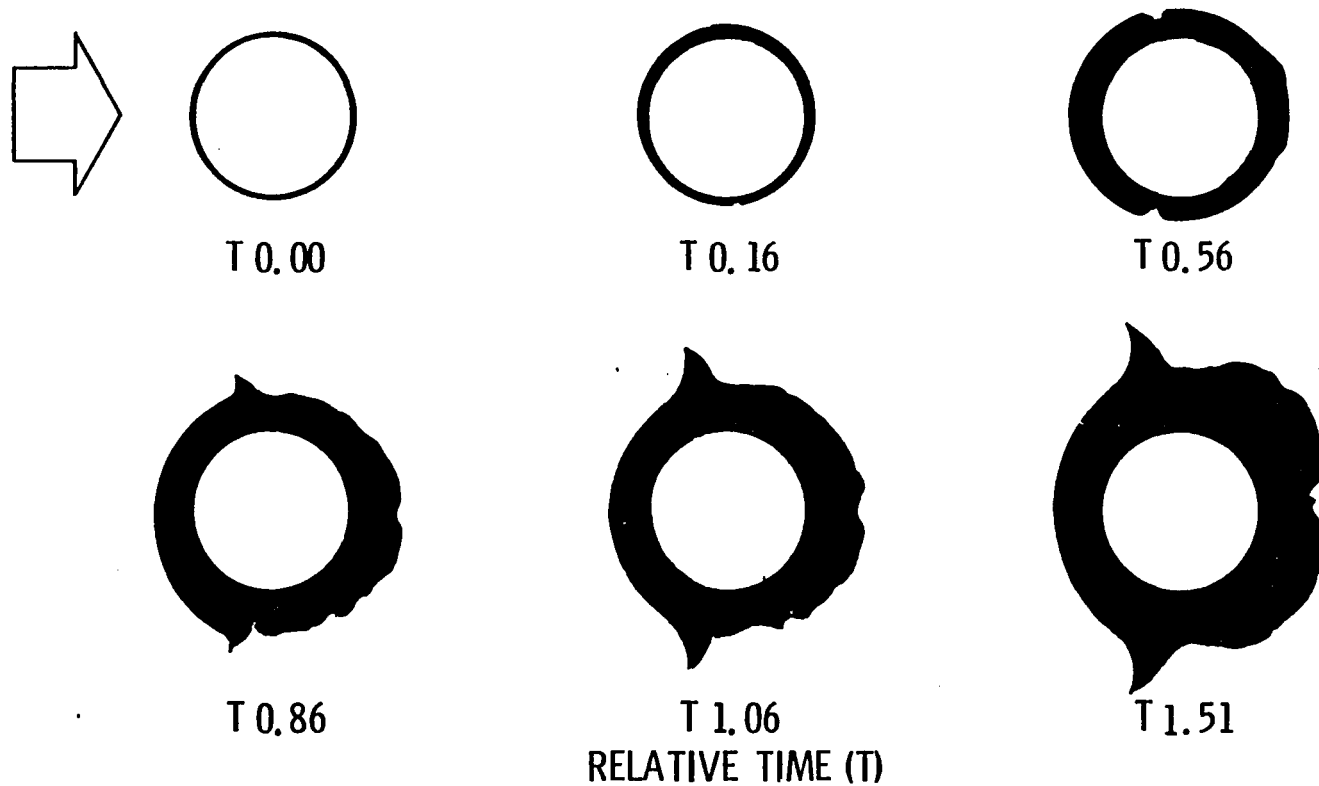
254



CS-82-2571

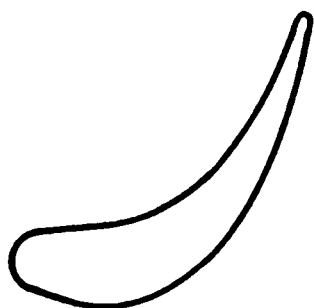
TRANSIENT EVOLUTION OF DEPOSIT LAYER STATIONARY CYLINDER IN CROSSFLOW

VAPOR DEPOSITION AND LIQUID LAYER FLOW (NO SHEDDING)
T WALL = CONSTANT

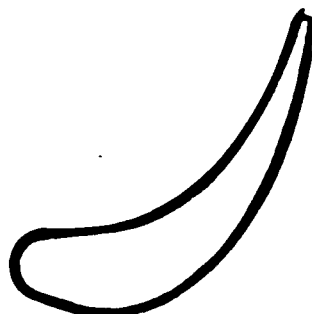


TRANSIENT EVOLUTION OF DEPOSIT LAYER STATOR BLADE

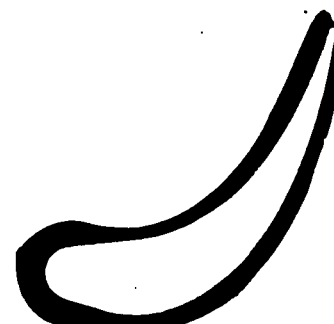
VAPOR DEPOSITION AND LIQUID LAYER FLOW (NO SHEDDING)
T WALL = CONSTANT



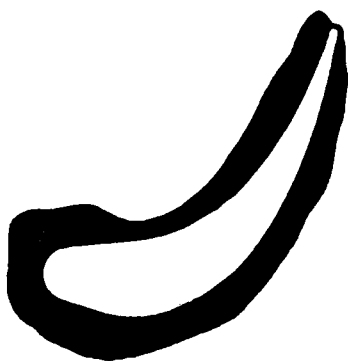
T 0.00



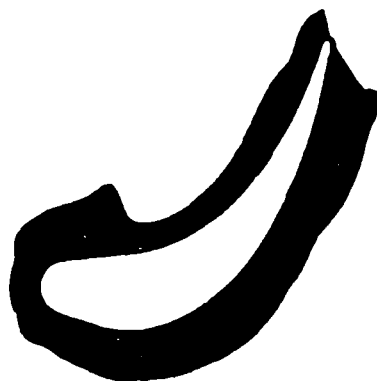
T 0.31



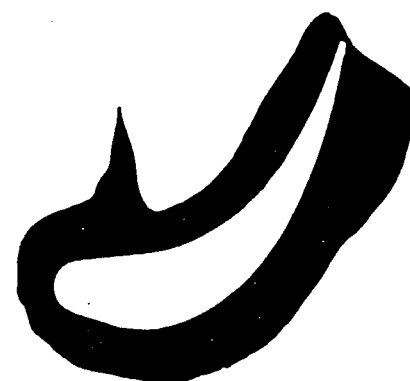
T 1.01



T 2.01



T 3.21
RELATIVE TIME (T)



T 4.81

FUTURE EMPHASIS OF DEPOSITION THEORY AND LIQUID LAYER FLOW

- BLADE ROTATION
- NONISOTHERMAL SURFACE TEMPERATURE DISTRIBUTION
- SALT SHEDDING; STEADY STATE
- MULTICOMPONENT VAPOR TRANSPORT
- SEED LEVEL TRANSIENTS
- ALTERNATE DEPOSITION MECHANISMS
 - A. CONVECTIVE DIFFUSION
 - B. THERMOPHORETIC ENHANCEMENT
 - C. PARTICLE IMPACTION

COATING LIFE PREDICTION

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Metallic coatings are widely used on hot section components of advanced gas turbine engines in order to take full advantage of the strength capabilities of turbine materials. Proper design to coating life limits can allow components to operate either for longer times or at higher temperatures, both of which are cost effective. However, costly engine inspections and component refurbishment or replacement are made many times on a conservative basis because component life and/or reliability are generally unknown. An analytical method for predicting life of metallic coatings on turbine airfoils should, therefore, result in substantial savings in maintenance and materials costs as well as providing an improved basis for initial design. The work to be discussed herein addresses itself to developing an improved methodology for predicting cyclic oxidation life of metallic coating on gas turbine airfoils.

A cyclic oxidation/spalling model was developed at LeRC that predicts long time cyclic furnace oxidation behavior of alloys. The computer inputs for the model are obtained from simple, short-time isothermal oxidation tests. In the present study, the model is being applied to an aluminide coating on U-700, a low pressure plasma sprayed (LPPS) NiCoCrAlY coating on U-700, and a monolithic LPPS NiCoCrAlY. An empirical diffusion model to account for coating degradation will be integrated with the oxidation/spalling model to predict coating life in cyclic furnace oxidation. The integrated model will then be verified/adjusted to predict cyclic burner oxidation. Further verification/adjustment will lead to a life prediction model for coated turbine airfoils. Preliminary results of isothermal and cyclic furnace oxidation of aluminide coated U-700 are presented.

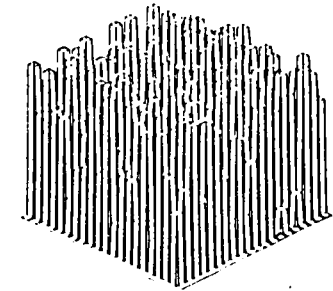
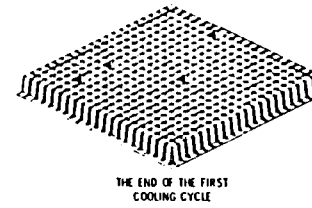
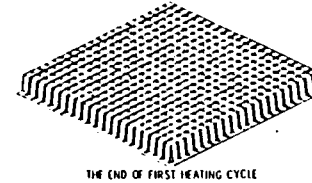
COATING LIFE PREDICTION

OBJECTIVE-

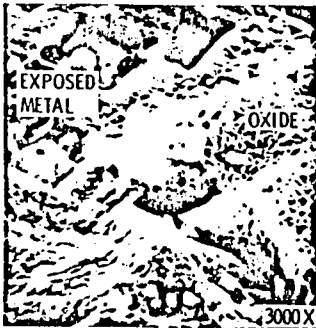
DEVELOP IMPROVED METHODOLOGY FOR PREDICTING CYCLIC OXIDATION LIFE OF METALLIC COATINGS ON GAS TURBINE AIRFOILS

CYCLIC OXIDATION VISUALIZATION

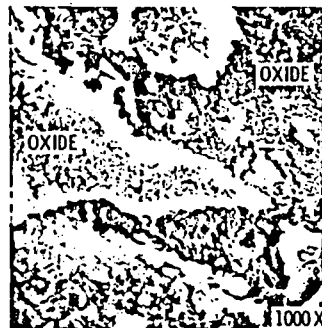
$$K_p = 0.12 \quad Q_p = 0.12 \quad N_s = 400$$



SEM MICROGRAPHS OF SPALLED Al_2O_3 SCALES

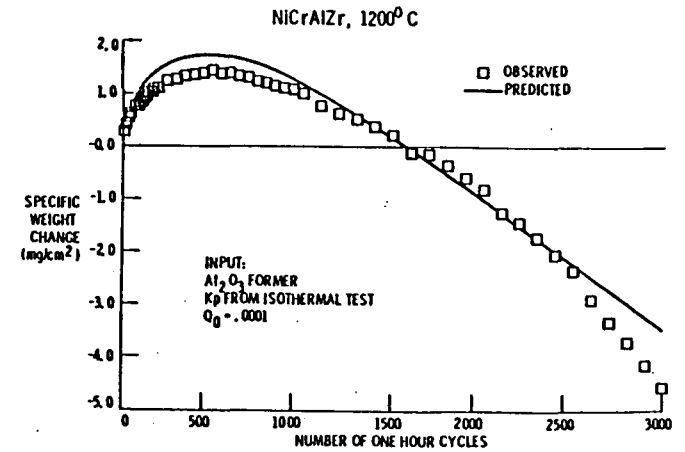


Ni-40Al, 1200° C, 217 CYCLES

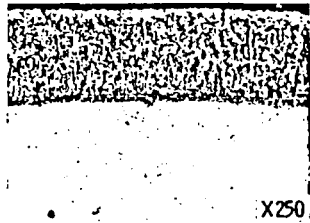


Ni-15Cr-24Al-0.3Zr, 1100° C, 500 CYCLES

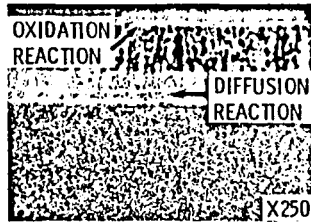
PREDICTIONS FROM ISOTHERMAL DATA AGREE WITH CYCLIC DATA



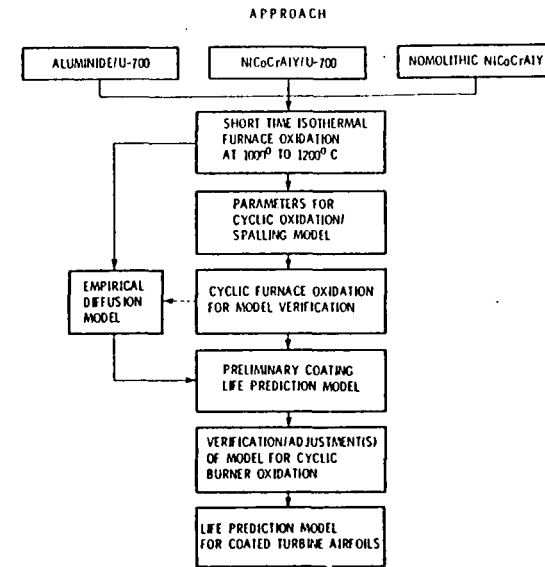
ENVIRONMENTAL AND SUBSTRATE REACTIONS DEGRADE COATINGS



AS-DEPOSITED NiCrAlY COATING



AFTER 200 hr AT 2000° F



261

COATING LIFE PREDICTION

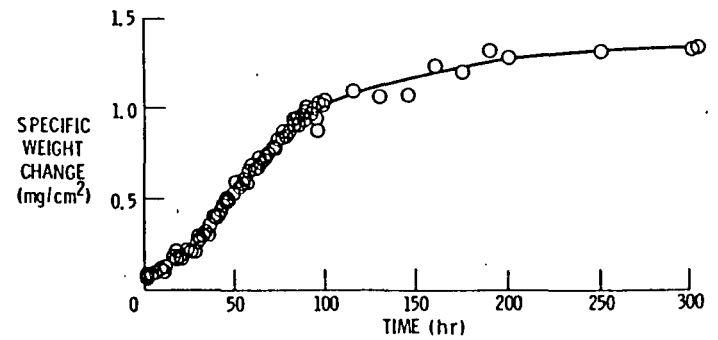
$$\text{LIFE} = F(f_o, f_D)$$

WHERE f_o = OXIDATION COMPONENT FOR GROWTH/SPALLING OF OXIDE SCALE

f_D = DIFFUSION COMPONENT FOR CRITICAL ELEMENT(S) OF THE COATING

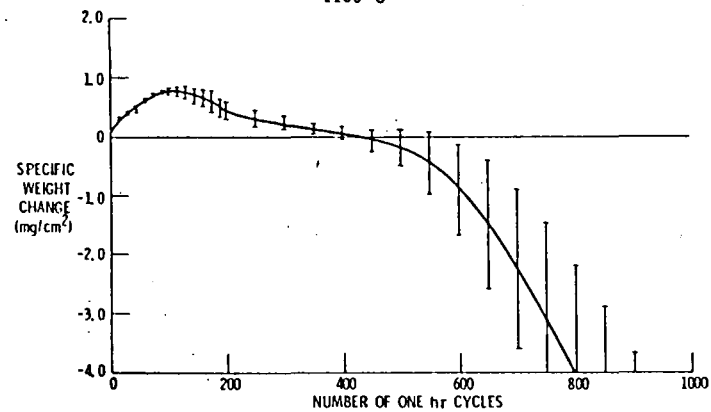
ISOTHERMAL FURNACE OXIDATION OF ALUMINIDE COATED U-700

1100° C

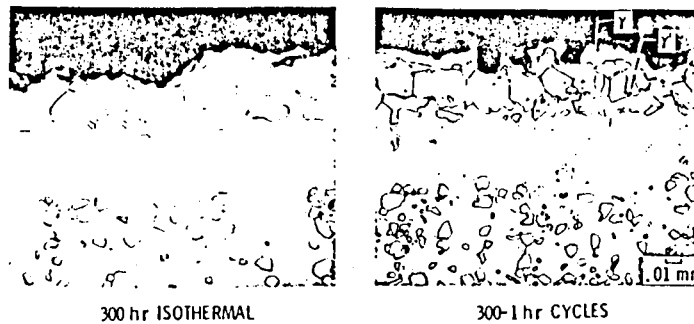


XXX11-3

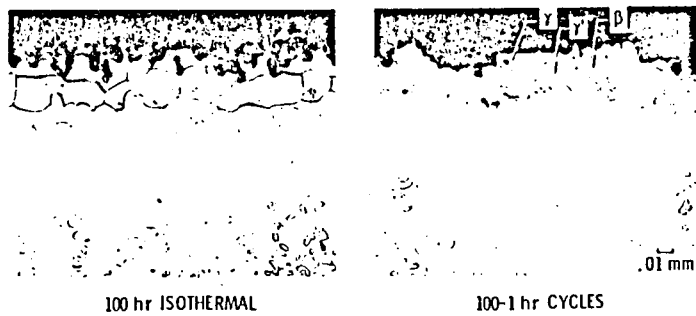
CYCLIC FURNACE OXIDATION OF ALUMINIDE COATED U-700
1100° C



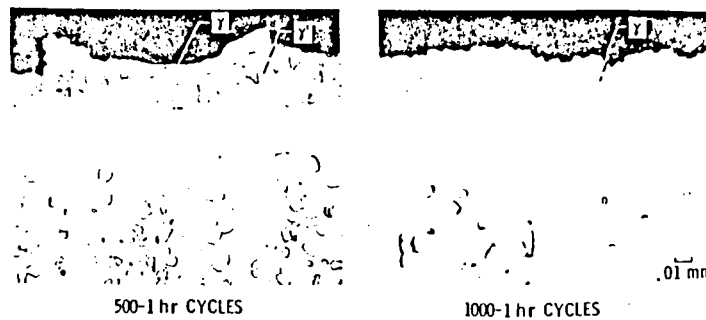
ALUMINIDE COATED U-700 AFTER FURNACE OXIDATION AT 1100° C



ALUMINIDE COATED U-700 AFTER FURNACE OXIDATION AT 1100° C



ALUMINIDE COATED U-700 AFTER FURNACE OXIDATION AT 1100° C



EFFECTS OF SURFACE CHEMISTRY ON HOT CORROSION LIFE

OVERVIEW

John Merutka
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

This program concentrates on analyzing a limited number of hot corroded components from the field and the carrying out of a series of controlled laboratory experiments to establish the effects of oxide scale and coating chemistry on hot corrosion life. This is to be determined principally from the length of the incubation period, the investigation of the mechanisms of hot corrosion attack, and the fitting of the data generated from the test exposure experiments to an empirical life prediction model. It is a six task program.

GENERAL SCOPE OF WORK

Task I involves the analysis of six field components which were removed from service. The hot corrosion condition of these six will vary from slight to massive attack. Concurrent with the metallurgical analysis of field components in Task I, specimens of bare and coated alloys will be subjected in Task II to exposures in a high velocity burner rig (under conditions specified by NASA-LeRC) for not more than 1000 hours or until hot corrosion occurs. In Task III, the Contractor shall age specimens (bare and coated) in an inert atmosphere, in furnace oxidation (cyclic and isothermal), and in cyclic high velocity burner rig oxidation at 1100C (2012F) for 100, 300, 600, and 1000 hours. In Task IV, the Contractor shall determine the effect of the various aging treatments on the hot corrosion mechanisms involved under the burner rig conditions specified in Task II.

Throughout Tasks I through IV, the results should be viewed not only in terms of identifying a model for the actual materials and test conditions run, but from the point of view of identifying a methodology whereby a life prediction model for other materials can be developed based on the results of one or more simple laboratory tests. After all the test exposures, the Contractor will review all the data and provide: (1) a preliminary hot corrosion life prediction model and (2) a recommendation of other test parameters to be evaluated so that simple laboratory tests can be used to predict hot corrosion life. The methodology to develop a hot corrosion life prediction technique shall be submitted to the NASA Project Manager for review and approval.

In Task V, based on NASA Project Manager's approval, the Contractor shall complete an experiment to determine the capability of the suggested methodology to predict the hot corrosion life of selected alloys and coatings.

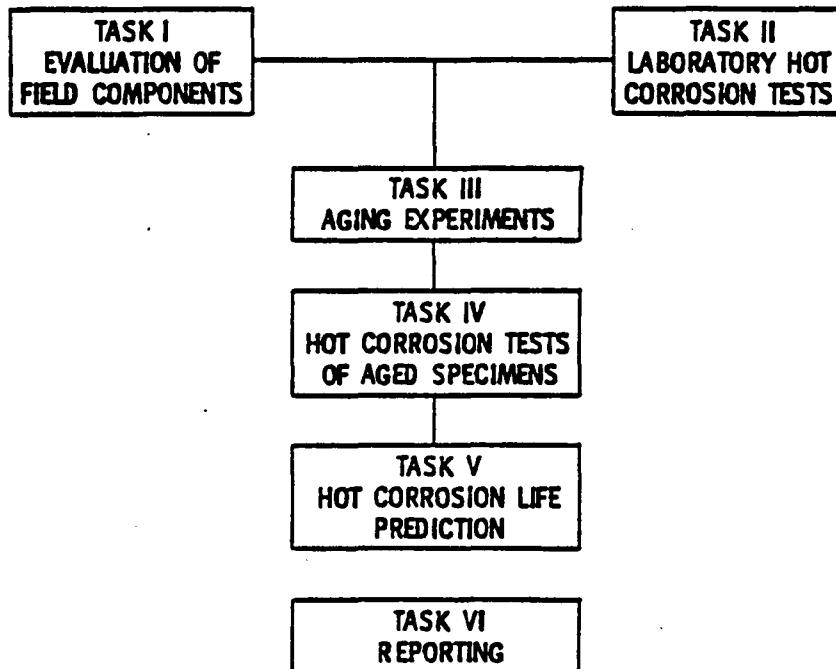
The last Task, VI, covers the reporting requirements.

RFP 3 - 412777

OBJECTIVE: DETERMINE EFFECTS OF SURFACE CHEMISTRY ON HOT CORROSION LIFE

BACKGROUND: PRIMARY MECHANISMS OF HOT CORROSION - FLUXING OF OXIDE SCALES BY LIQUID SALTS

- RIG TESTS GIVE INCONSISTANT RESULTS
- NEW TECHNIQUE DEVELOPED TO
 - DETERMINE THE INCUBATION THRESHOLD PERIOD
 - CARRY OUT REPRODUCIBLE HOT CORROSION TESTS



- ANALYSIS OF HOT CORROSION COMPONENTS FROM THE FIELD
- CONTROLLED LABORATORY EXPERIMENTS TO ESTABLISH EFFECTS OF SURFACE CHEMISTRY ON HOT CORROSION LIFE
- DEVELOPMENT OF EMPIRICAL LIFE PREDICTION MODEL BASED ON DATA GENERATED

EFFECTS OF SURFACE CHEMISTRY ON HOT CORROSION LIFE

TASK I EVALUATION OF FIELD COMPONENTS

SIX FIELD COMPONENTS (LITTLE TO MASSIVE CORROSION)
EVALUATION (METALLURGICAL AND CHEMICAL)

TASK II LABORATORY HOT CORROSION TESTS

BURNER RIG CONDITIONS: 0.3 MACH, PRE-CONDITIONED AIR
MATERIALS: U700 AND CONTRACTOR'S CHOICE - BARE AND COATED (DUPLICATES)
COATINGS: RT21 ALUMINIDE, LOW PRESSURE PLASMA NiCoCrAl_y, CONTRACTOR'S CHOICE
CYCLE: 60 min HOT, 6 min AIR BLAST COOL
SPECIMEN SURFACE TEMPERATURE: 900C (1750F)
TIME: 1000 HOURS OR UNTIL HOT CORROSION OCCURS
RUN ADDITIONAL SPECIMENS 100, 300, 500 HOURS: TIME NOT TO EXCEED 2/3rds OF THE TIME IN WHICH HOT CORROSION OCCURS
MONITOR: VISUAL AND INDUCTANCE EVERY 20 CYCLES
EVALUATION: METALLURGICAL AND CHEMICAL (OXIDE, ALLOY AND COATING COMPOSITION AND STRUCTURE)

TASK III AGING EXPERIMENTS

TEMPERATURE: 1100C (2012F)
MATERIALS: AS IN TASK II (TRIPLICATES)
AGING CONDITIONS:
TIME: 100, 300, 600, AND 1000 hrs
ENVIRONMENT: INERT; ISOTHERMAL FURNACE OXIDATION;
CYCLIC FURNACE OXIDATION; CYCLIC BURNER RIG OXIDATION
MONITOR: INDUCTANCE CHANGES AND WEIGHT CHANGES AS APPROPRIATE
CHARACTERIZATION: ONE SPECIMEN PER CONDITION AS IN TASK II

EFFECTS OF SURFACE CHEMISTRY ON HOT CORROSION LIFE

TASK IV HOT CORROSION TESTS OF AGED SPECIMENS

TEST CONDITION: AS IN TASK II, UNTIL HOT CORROSION OCCURS (DUPLICATES)

MONITOR: VISUAL AND INDUCTANCE EVERY 20 HOUR PERIOD

HOT CORROSION OCCURS: VISUAL SIGNS FOR THREE 20 hr PERIODS.

EVALUATION: (METALLURGICAL AND CHEMICAL)

PROPOSE: PRELIMINARY HOT CORROSION LIFE PREDICTION MODEL

SUGGEST: METHODOLOGY TO PREDICT HOT CORROSION LIFE BASED ON LAB EXPERIMENTS

TASK V HOT CORROSION LIFE PREDICTION

- VERIFY LIFE PREDICTION MODEL
- TEST METHODOLOGY

TASK VI REPORTING REQUIREMENTS

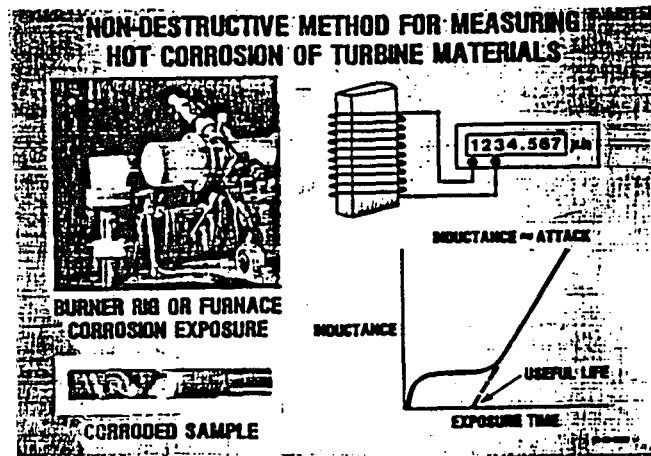
FINANCIAL

MONTHLY

ANNUAL

FINAL

ORAL PRESENTATIONS



COMBUSTION HOT SECTION TECHNOLOGY

David B. Ercegovic

Propulsion Laboratory
AVRADCOM Research and Technology Laboratories
Lewis Research Center
Cleveland, Ohio

The overall objective of the Turbine Engine Hot Section Technology Combustion Project is to develop and verify improved and more accurate analysis methods for increasing the ability to design with confidence the combustion system for advanced aircraft turbine engines. The analysis methods developed will be generically applicable to combustion systems and not restricted to one specific engine or manufacturer.

This projects approach is to first assess and evaluate existing combustor aerothermal analysis models by means of a contracted effort initiated during FY '82. This evaluation effort will quantify known models strengths and deficiencies. A balanced contract and in-house program will then be conducted to support, focus, and accelerate the development of new methods to more accurately predict the physical phenomena occurring within the combustor. This balanced program will include both analytical and experimental research efforts in the areas of aerothermal modeling and liner cyclic life.

It is expected that the combustor model development effort will generate improved understanding in the areas of: high pressure flame radiation characteristics, model numerical methods and solution schemes, complex geometrical boundary conditions, fuel spray - flow field interactions, combustion kinetics, flow and mixing of dilution jets, turbulence and heat transfer, and soot and carbon formation. The primary in-house effort in this area will be the determination of high pressure flame radiation characteristics in a full annular combustor. This experiment will be conducted in the NASA LeRC High Pressure Facility with the results compiled into a comprehensive flame radiation and liner heat flux model.

In the area of liner cyclic life, HOST will develop a test apparatus to economically determine combustor thermal strains and cyclic life. This test apparatus will be run in-house at NASA LeRC and will be the test vehicle for many of the advanced high temperature instruments developed under HOST sponsorship. The fundamental data generated in this project will be used to assess and develop current analytical liner life programs.

OBJECTIVE

TO DEVELOP IMPROVED ANALYTICAL MODELS OF THE INTERNAL COMBUSTOR FLOW FIELD AND LINER HEAT TRANSFER AS A MEANS TO SHORTEN COMBUSTOR DEVELOPMENT TIME AND INCREASE TURBINE ENGINE HOT SECTION LIFE.

APPROACH

- UTILIZE EXISTING MODELS - DETERMINE THEIR DEFICIENCIES
- CONDUCT SUPPORTING RESEARCH TO IMPROVE PHYSICAL MODELS
- REFINE MODELS TO IMPROVE NUMERICS AND NUMERICAL DIFFUSION
- INTEGRATE NEW AND IMPROVED ROUTINES INTO EXISTING MODELS AND VERIFY THEIR IMPROVED PREDICTIVE CAPABILITY

COMBUSTION

PROGRAM ELEMENT	FISCAL YEAR							EXPECTED RESULT
	81	82	83	84	85	86	87	
AEROTHERMAL MODELING ASSESSMENT		■						KEY MODEL AND DATA DEFICIENCIES IDENTIFIED
COMBUSTION MODELING DEVELOPMENT			▼	■	■	■	■	NEW PHYSICAL MODELS AND COMPUTING METHODS
MULTIPLE JET DILUTION MIXING		■	■	■	■	■	■	EXIT TEMPERATURE PROFILE PREDICTION TECHNOLOGY
FLAME RADIATION/HEAT FLUX	(IH)	■	■	■	■	■	■	HIGH PRESSURE FLAME RADIATION AND HEAT FLUX
DILUTION JET ANALYSIS	(IH)	■	■	■	■	■	■	JET MIXING MODEL
LINER CYCLIC RIG	(IH)	■	■	■	■	■	■	CYCLIC TEST FACILITY

271



AEROTHERMAL MODELING PROGRAM

PRINCIPAL INVESTIGATOR: S. K. SRIVATSA, GARRETT TURBINE ENGINE COMPANY, PHOENIX, ARIZONA

OBJECTIVE:

ASSESS THE CURRENT STATE-OF-THE-ART AND IDENTIFY THE DEFICIENCIES IN CURRENT AEROTHERMAL MODELS FOR GAS-TURBINE COMBUSTORS

S. K. Srivatsa
Garrett Turbine Engine Company
Phoenix, Arizona

AEROTHERMAL MODELING PROGRAM

GARRETT TURBINE ENGINE COMPANY
A DIVISION OF THE GARRETT CORPORATION
PHOENIX, ARIZONA



PROGRAM APPROACH OUTLINE

TASK 1

- 1.1 MODEL DEFINITION
- 1.2 DATA BASE GENERATION
- 1.3 BENCHMARK TEST CASE DEFINITION

TASK 2

- 2.1 MODEL EXECUTION
- 2.2 MODEL ASSESSMENT
- 2.3 PROGRAM PLAN FOR MODEL IMPROVEMENT



SUBTASK 1.1 - MODEL DEFINITION

DESCRIBE COMPUTER PROGRAMS TO BE USED FOR MODEL ASSESSMENT

COMBUSTOR PERFORMANCE MODEL (2-D AND 3-D)

NEAR-WALL MODEL

DESCRIBE PHYSICAL SUBMODELS TO BE ASSESSED

TURBULENCE MODELS

GASEOUS COMBUSTION MODELS

SPRAY COMBUSTION MODELS

SOOT FORMATION/OXIDATION MODEL

RADIATION MODELS



SUBTASK 1.2 - DATA BASE GENERATION

CLASSIFY AND REVIEW AVAILABLE DATA BASE

- o COMPLEXITY OF FLOW FIELD
 - o PARABOLIC FLOW
 - o STREAMLINE CURVATURE
 - o RECIRCULATING FLOWS W/O SWIRL
 - o SWIRLING FLOWS WITH AND W/O RECIRCULATION
- o PHYSICAL/CHEMICAL PROCESSES
 - o NON REACTING FLOWS
 - o REACTING FLOWS
 - o GASEOUS COMBUSTION
 - o SPRAY COMBUSTION
- o IDEALIZED FLOW ELEMENTS AND PRACTICAL COMBUSTORS



SUBTASK 1.3 - BENCHMARK TEST CASE DEFINITION

o TURBULENCE MODEL

- o FLOW OVER A FLAT PLATE: \bar{U} , $\overline{U'^2}$, $\overline{U'V'}$, K
- o TWO STREAM MIXING LAYER WITHOUT RECIRCULATION: \bar{U} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{W'^2}$, $\overline{U'V'}$
- o MIXING OF UNCONFINED COAXIAL JETS: \bar{U} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{U'V'}$
- o DEVELOPING PIPE FLOW: \bar{U} , \bar{V} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{U'V'}$
- o FLOW IN A CURVED DUCT: \bar{U} , \bar{V} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{W'^2}$, $\overline{U'V'}$
- o FLOW OVER A PLANE STEP: \bar{U} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{U'V'}$
- o MIXING OF CONFINED COAXIAL JETS WITH RECIRCULATION: \bar{U} , \bar{V} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{W'^2}$, $\overline{U'V'}$
- o MIXING OF UNCONFINED COAXIAL SWIRLING JETS W/O RECIRCULATION:
 \bar{U} , \bar{V} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{W'^2}$, $\overline{U'V'}$, $\overline{U'V'^2}$, $\overline{V'W'}$
- o SWIRLING FLOW IN A PIPE EXPANSION WITH RECIRCULATION:
 \bar{U} , \bar{V} , $\overline{U'^2}$, $\overline{V'^2}$, $\overline{W'^2}$, $\overline{U'V'}$, $\overline{U'W'}$, $\overline{V'W'}$



BENCHMARK TEST CASE DEFINITION (CONT'D)

o GASEOUS COMBUSTION MODEL

- o 1-D LAMINAR FLAT FLAME: FUEL, O₂, CO, CO₂, H₂O, H₂, T
- o 2-D LAMINAR DIFFUSION FLAME WITH RECIRCULATION: CH₄, CO₂, CO, H₂, H₂O, O₂, N₂,
VELOCITY, T
- o 2-D TURBULENT PREMIXED FLAME WITH RECIRCULATION: VELOCITY, CO, CO₂, C₃H₈, H₂O,
O₂, N₂, T
- o 2-D TURBULENT DIFFUSION FLAME W/O RECIRCULATION: VELOCITY, T, CH₄, H₂, CO₂, CO,
H₂O, O₂, N₂
- o 2-D TURBULENT DIFFUSION FLAME WITH RECIRCULATION: MIXTURE FRACTION O₂, N₂, CH₄,
CO, CO₂, H₂O, T
- o 2-D TURBULENT, SWIRLING DIFFUSION FLAME WITH RECIRCULATION: CO₂, CO, H₂, VELOCITY, T



BENCHMARK TEST CASE DEFINITION (CONT'D)

- o SPRAY COMBUSTION MODEL
 - o 2-D TURBULENT EVAPORATING SPRAY W/O RECIRCULATION: DROP SIZE DISTRIBUTION, T, MIXTURE FRACTION, GAS VELOCITY
 - o 2-D TURBULENT REACTING SPRAY W/O RECIRCULATION: T, CO, CO₂, O₂, N₂, H₂, H₂O, GAS VELOCITY
 - o 2-D TURBULENT REACTING SPRAY WITH RECIRCULATION: DROP NUMBER DENSITY, T, O₂, CO, CO₂, C_xH_y, GAS VELOCITY
 - o 2-D TURBULENT SWIRLING REACTING SPRAY WITH RECIRCULATION: DROPLET VELOCITY AND SIZE DISTRIBUTION, T, CO, CO₂, O₂, C_xH_y, GAS VELOCITY



BENCHMARK TEST CASE DEFINITION (CONT'D)

- o SOOT FORMATION/OXIDATION MODEL:
 - o 2-D LAMINAR DIFFUSION CH_4 -FLAME W/O RECIRCULATION: SMOKE CONCENTRATION
 - o 2-D TURBULENT DIFFUSION CH_4 , C_3H_8 FLAME W/O RECIRCULATION: SMOKE CONCENTRATION AND RADIATION
- o GAS-TURBINE COMBUSTORS - GARRETT IN-HOUSE DATA:
 - o COMBUSTOR DESIGN CRITERIA VALIDATION PROGRAM: VELOCITY, CO, CO_2 , C_xH_y , RADIATION
 - o UPRATE T-76 COMBUSTOR: PRIMARY ZONE: T, CO, C_xH_y
 - o TPE331-15 COMBUSTOR: LINER WALL TEMPERATURES



SUMMARY AND FUTURE WORK

- o SUMMARY
 - o DATA BASE FOR TURBULENCE AND GASEOUS COMBUSTION MODELS FAIRLY ADEQUATE
 - o DATA BASE FOR SPRAY MODELS LESS SATISFACTORY STATE
 - o DATA BASE FOR SOOT AND RADIATION MODELS RATHER INADEQUATE
- o FUTURE WORK
 - o COMPUTATIONS OF THE BENCHMARK CASES WITH AEROTHERMAL MODEL
 - o IDENTIFY MODEL DEFICIENCIES
 - o PREPARE PROGRAM PLAN FOR MODEL IMPROVEMENT.

AEROTHERMAL MODELING - PHASE I

A Progress Report

G. J. Sturgess
Commerical Engineering, Pratt & Whitney Aircraft Group

The objective of the program is to develop the computational fluid dynamics tools needed to improve combustor design, analysis and development. In the first phase, current models will be evaluated, shortcomings identified, and improvements recommended. These recommendations will be implemented in the second phase. The approach adopted is to evaluate state-of-the-art numerical code and physical models. The evaluation consists of a step-by-step procedure using benchmark experiments.

The program is divided into three major tasks: Task 1 is concerned with defining the models, establishing a data base, and identifying test cases from the data base; Task 2 involves running the model, evaluating its performance, and formulating a program plan to achieve the necessary improvements; Task 3 is concerned with management and reporting activities.

The contract with Pratt & Whitney Aircraft went into effect on 13 July 1982. Task 1 is nearly completed and Task 2 has been started.

Figure 1 outlines the calculation procedure. The modeling which has been selected represents the state of the art. The approach consists of a finite difference solution of the time-averaged, steady state, primitive variable, elliptic form of the Reynolds equations. Standard TEACH-type numerics are used to solve the resulting equations. These include hybrid differencing, SIMPLE algorithm for the pressure field, line-by-line iterative solution using the ADI method and the tri-diagonal matrix algorithm (TDMA). Convergence is

facilitated by using under-relaxation. The physical processes are modeled by a two-equation eddy viscosity model for turbulence; combustion is represented by a simple, irreversible, one-step chemical reaction whose rate is influenced only by the time scale of turbulence; the radiating medium is assumed to be gray and a flux method is used for radiation together with a gas emissivity obtained from a four gray gas model. The liquid fuel spray is treated by particle tracking using the PSIC technique, and turbulent diffusion of droplets is accounted for by a stochastic approach. Provision is made for the fuel to be either a pure substance or multi-component.

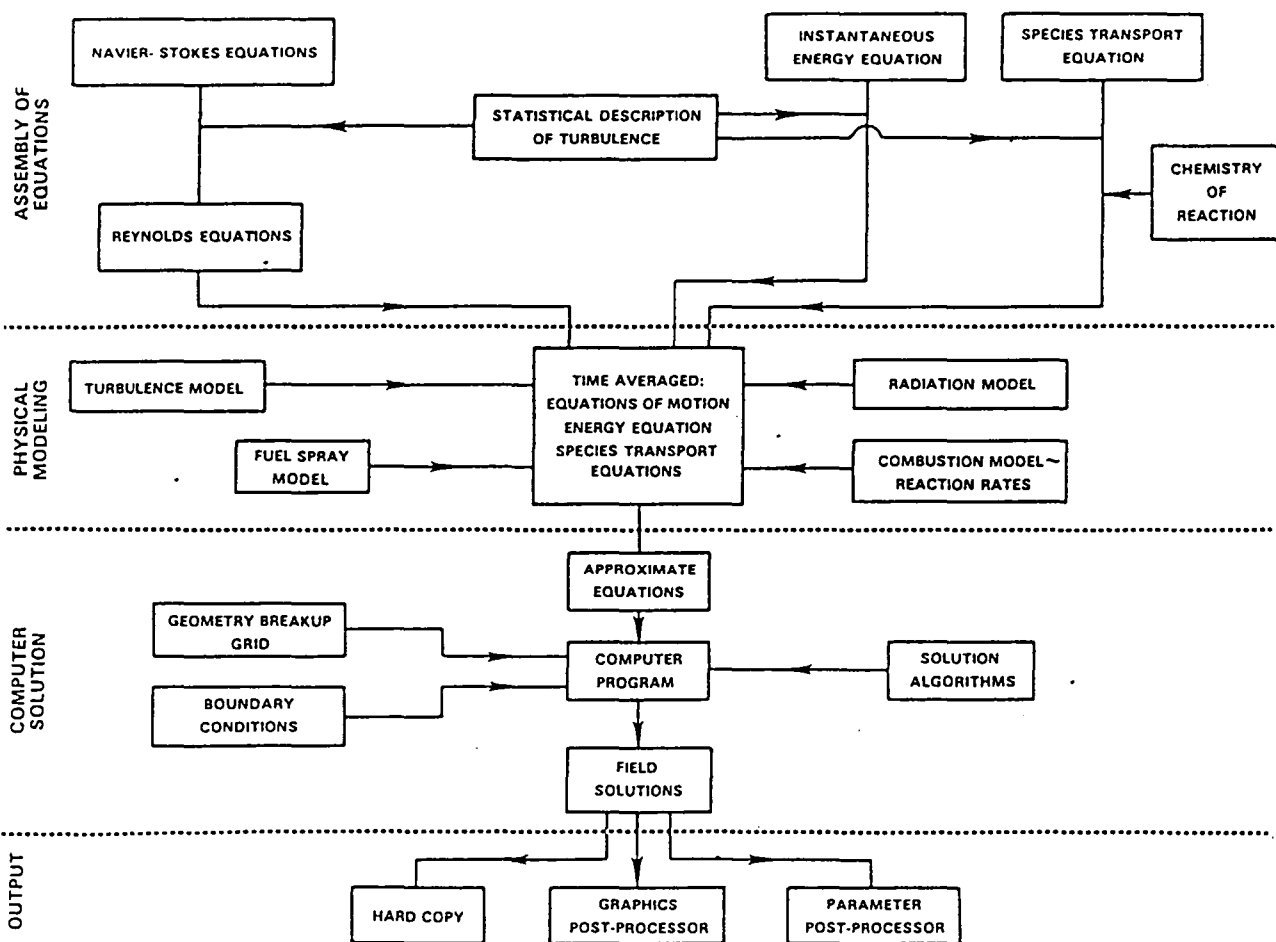


Figure 1 Flow Diagram of Calculation Procedure

The models will be evaluated against experimental data using a data base currently being prepared. In order to avoid difficulties in separating effects to assess the performance of individual models, wherever possible only benchmark quality experiments which deal with one physical process at a time are being considered. If real combustor flows are calculated, they will only be considered demonstrations of potential. Ideally, the comparisons will proceed from simple flows to complicated flows. Complicated flows will only be used to study the effects of interaction between different physical models.

Experiments are required to test each of the physical models. An "ideal experiment" has been defined, and experiments in the literature are being compared against this ideal to assess their qualification as benchmark test cases in the data base. An initial selection of test cases has been made. This selection covers co-axial jets with and without swirl in a confined sudden expansion, co- and counter-swirling co-axial jets, a widely-spaced co-axial jet bluff body diffusion flame, and a single jet in a crossflow. These cases represent component flows typical of those in the gas turbine combustor. Additional test cases are being selected to broaden the study.

Calculation of the initial selection of test cases has commenced, although it is too early at the present to comment on the results.

CONTENTS

- **Objectives**
- **Approach**
- **Status**
- **Major results**

OBJECTIVES

Develop computational fluid dynamics tools needed to improve combustor design, analysis and development, by the following means:

- **Define aerothermal models in the combustor design process**
- **Establish a suitable data-base against which to test models**
- **Identify shortcomings in the data-base**
- **Evaluate performance of models and identify their limitations**
- **Recommend future work to complete the data-base**
- **Recommend a course to result in improved models**

APPROACH

- **Use state-of-the-art numerical code and physical models**
- **Evaluate physical models using a step-by-step approach utilizing benchmark-quality experiments**
- **Be cognizant of the influence of numerical diffusion**

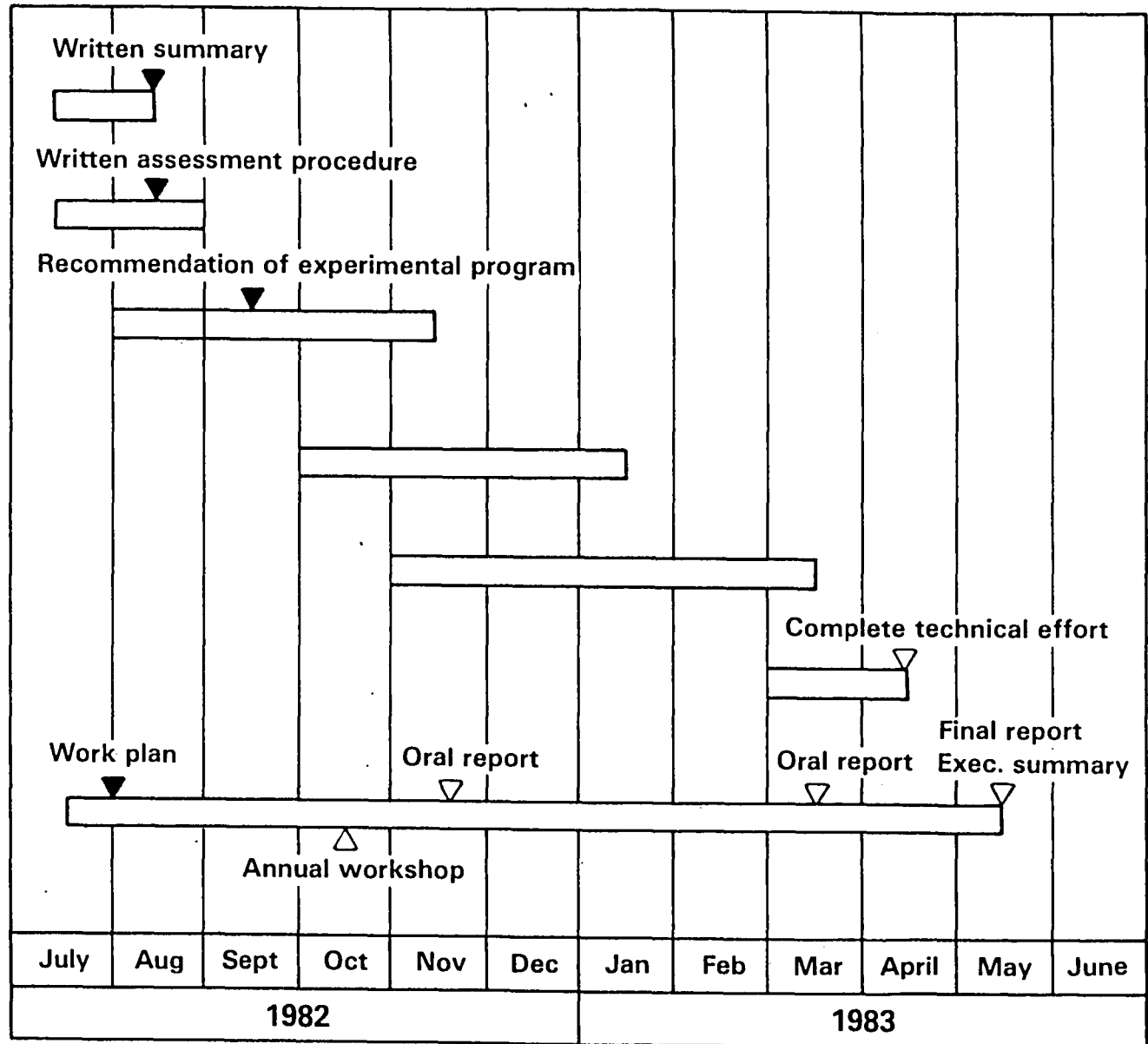
STATUS

- **Contract went into effect 13 July, 1982 (NAS3-23524)**
- **Work is on schedule and budget**
- **There are no current problems of a technical nature**

PROGRAM SCHEDULE

290

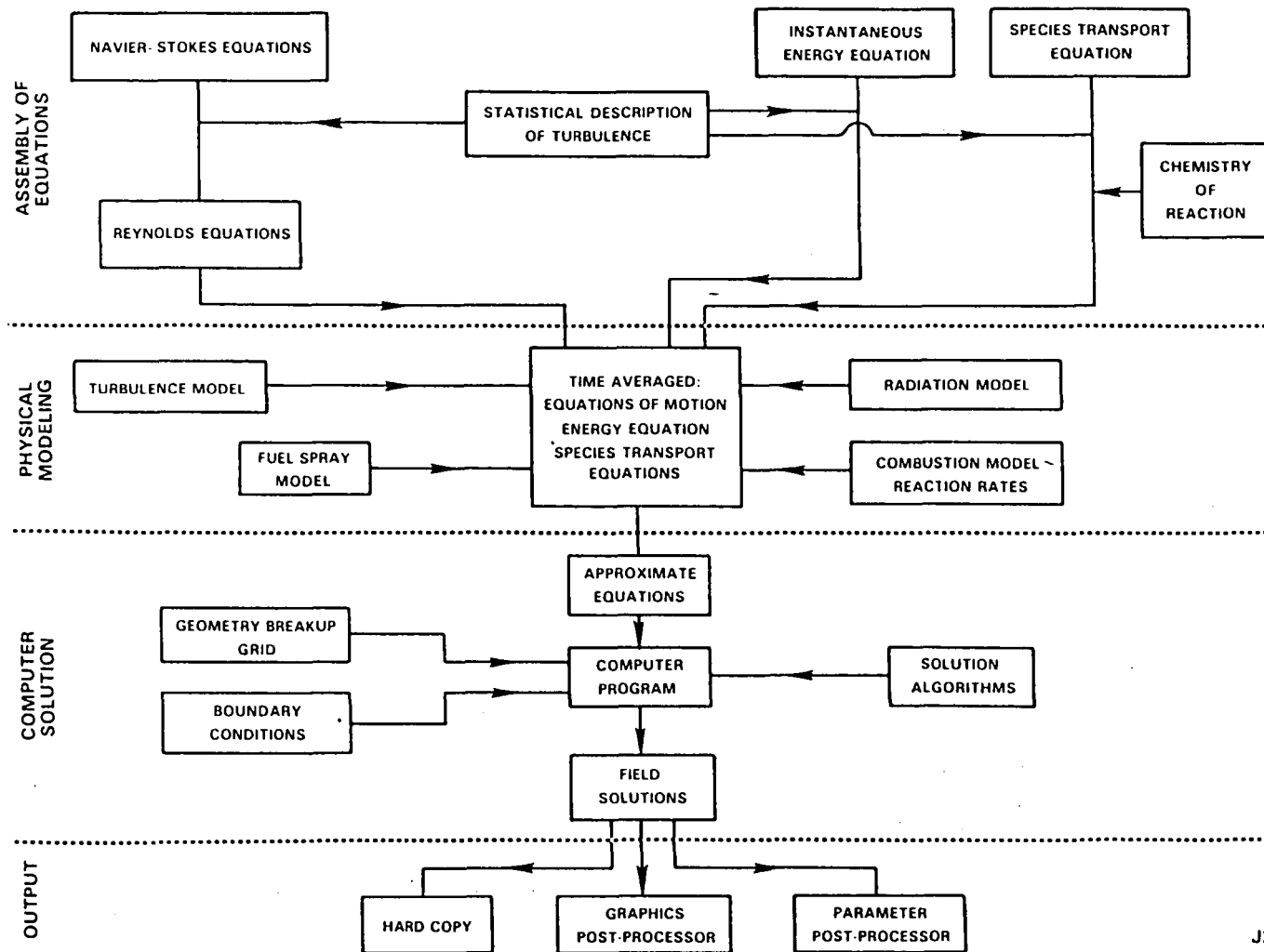
- Task 1
- 1.1 Model definition
- 1.2 Data definition
- 1.3 Test case definition
- Task 2
- 2.1 Run model
- 2.2 Assess model
- 2.3 Program plan
- Task 3



MAJOR RESULTS

(TASK 1.1)

FLOW DIAGRAM OF CALCULATION PROCEDURE



(TASK 1.1)

MODELING SELECTED AS FOLLOWS:

- **TEACH numerics — hybrid differencing, SIMPLE, ADI, TDMA, under-relaxation**
- **K- ϵ turbulence model**
- **Fuel spray treatment — particle tracking using PSIC for a thick spray, with a stochastic approach for turbulent diffusion of droplets**
- **Combustion model — simple, irreversible, one-step chemical reaction, with eddy breakup burning rate**
- **Radiation — grey gas transport equation containing particle terms, with Barteld's 4 grey gas emissivity modified for collision broadening at high pressure**

TASK 1.2

EVALUATION PROCEDURE:

- **To avoid difficulties in successfully separating effects to assess individual model performance, work with benchmark-quality experiments dealing with only one physical process at a time**
- **Experiments will be required in the areas of:**
 - **Thermal radiation**
 - **Fuel spray development and flow field interactions**
 - **Combustion reaction and chemical kinetics**
 - **Fluid mechanics**
 - **Soot formation**
- **Proceed from simple flows to complicated flows. Complicated flows will only be used to study interaction effects between different physical models**
- **Calculations of three-dimensional flows and/or real combustor flows should only be considered as demonstrations of potential**

(TASK 1.2)

CRITERIA FOR THE IDEAL EXPERIMENT

- **Minimum flow dimensionality**
- **Well-behaved flows**
- **Continuous variation of test parameters**
- **Progression of flow complexity**
- **Extensive instrumentation**

(TASK 1.3)
RECOMMENDATIONS WITHIN THE SCOPE
OF THE PRESENT PROGRAM TO FILL VOIDS
IN THE EXPERIMENTAL DATA-BASE:

- **Although there are many voids in the current data base the nature of the ideal benchmark experiments required to fill them with respect to time, instrumentation, facilities, and cost far exceed the scope of the present contract**

- **It is recommended that no experimental work be carried out as part of this contract**

(TASK 1.3)

INITIAL SELECTION OF TEST CASES

- Johnson (UTRC)
 - Co-axial jets in confined sudden expansion (NAS3-22771)
 - Co-axial jets with swirl in confined sudden expansion
- Gouldin (Cornell)
 - Co and counter-swirl co-axial jets (NSF-R ANN-GI-36538/NSG-3019)
- Roquemore (APL)
 - Widely-spaced co-axial jet diffusion flame
- Greber (Case Western Reserve)
 - Jet in a cross-flow (NGR-36-027-008)

(TASK 2.1.) MODEL TESTING

Calculation of the initial test cases has been started

**Working is currently proceeding on Johnson,
Gouldin and Roquemore's experiments**

AEROTHERMAL MODELING PROGRAM

Milton Kenworthy
General Electric Co.
Evendale, Ohio

General Electric began work on this effort in August, 1982. Rather than describing progress, therefore, this presentation will indicate some significant features of the planned approach.

Figure 1 schematically shows the individual computerized models utilized in General Electric's combustor aero design approach. The preliminary design module provides the overall envelope definition of the burner. The diffuser module provides the detailed contours of the diffuser and combustor cowl region, as well as the pressure loss characteristics into each of the individual flow passages into the dome and around the combustor. The flow distribution module provides the air entry quantities through each of the apertures and the overall pressure drop. The heat transfer module provides detailed metal temperature distribution throughout the metal structure as input to stress and life analysis that are not part of the aerothermo design effort.

The internal flow module, which entails 3-D elliptic flow field calculations, is not at present, used in General Electric's design method. It is planned, however, that it will be incorporated in the very near future to the extent that its usefulness is demonstrated. It is expected to be initially useful in providing improved hot gas side inputs to the heat transfer module and to help guide the development of combustor exit pattern factor. The capabilities include analysis of the mixing of the dilution region without chemical reaction, the treatment of fuel insertion, and the chemical reaction zone itself. While the phenomena in each of these regions is developed to a different extent of rigor, they all utilize the same basic 3-D elliptic framework.

General Electric's internal flow module, INTFLOW, has a basic core structure that can interchangeably use either the 3-D combustor performance model from the Combustor Design Criteria Validation Program prepared by Garrett Corporation

(HC Mongia, RS Reynolds, and TW Bruce, 1979) or the newer Northern Research and Engineering Corporation's version of this same type of 3-D elliptic code. These core packages are supplemented by special input routines and output plotting routines that have been prepared at General Electric. Figure 2 is the type of calculated output plots that are available from INTFLOW. The length of the arrows represent the relative velocity of the flows within this annular combustor, which includes swirl cups and a complex dilution pattern.

In the Aerothermal Modeling Program, comparisons are to be made with benchmark quality test data to permit evaluation of the accuracy of the modules and to identify the sources of error or inaccuracy within the modules. Data from the F101 series of General Electric combustors have been selected as a major basic source of this benchmark data. Extensive combustor liner metal temperature data and combustor exit gas temperature pattern data are available. These combustion aerothermodynamic data are particularly significant as these measured results are the important inputs in the stress and life analysis of the combustor structure and the turbine nozzles and vanes.

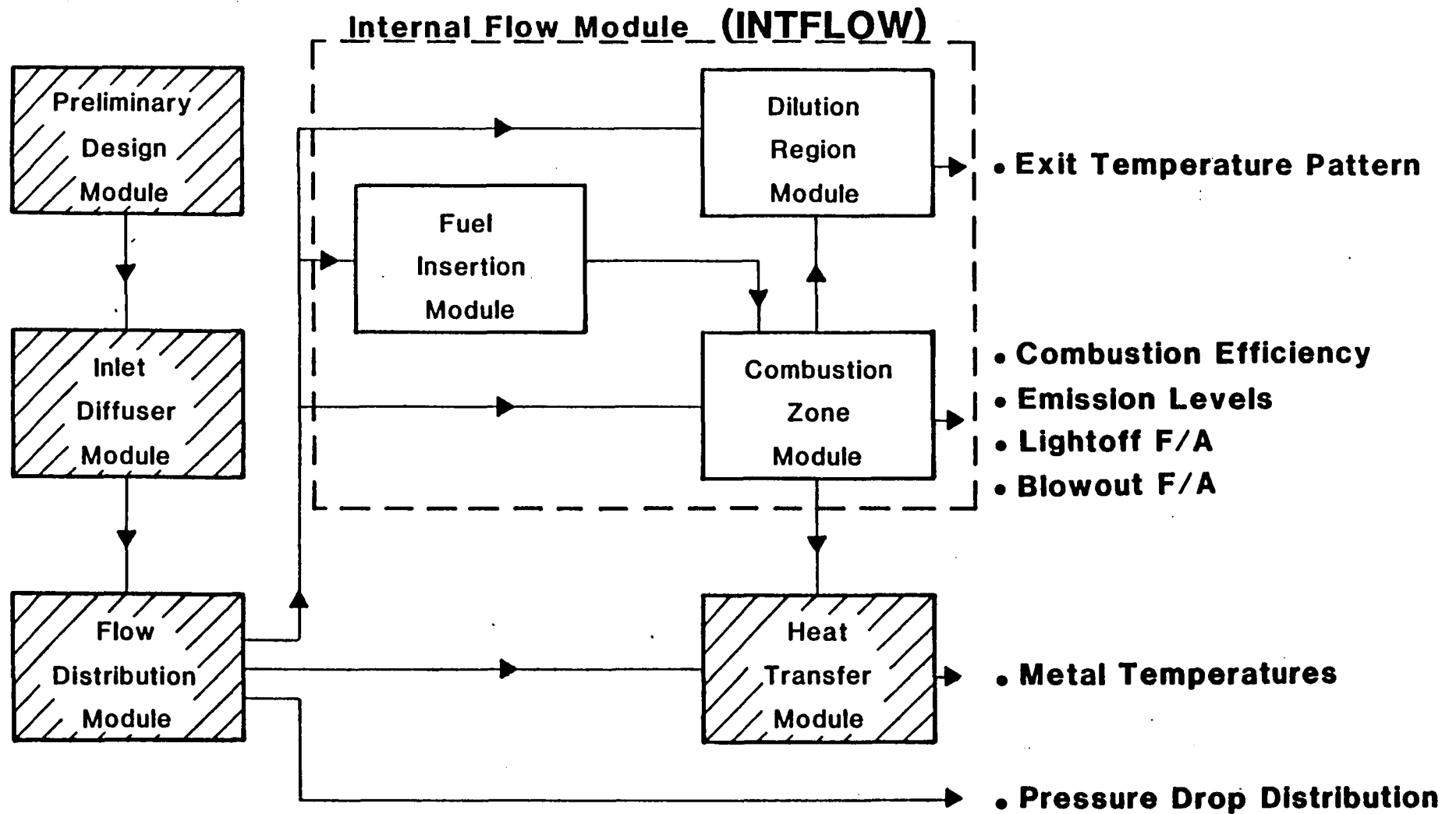
In addition, data from laboratory experiments, with more detailed flow data, will be utilized to help evaluate detailed features within the 3-D elliptic module. At General Electric, alternate computerized treatments are available as 2-D axisymmetric elliptic codes for: turbulence model modifications, numeric changes, kinetics treatment, and time fluctuation treatment. Hence, axisymmetric data will be selected for some of the computerized studies.

Also, a set of experiments will be conducted at General Electric with a test combustion sector having dilution hole characteristics like the General Electric F101 combustor, but with wall boundaries compatible with the current 3-D elliptic code model capabilities. At present, the 3-D elliptic codes available at

General Electric require that the combustor wall boundaries must be modeled along the cylindrical and axial grid lines; an axially curved wall such as the contraction at the aft end of most combustors cannot be accurately modeled. Thus, these experiments will be done with a combustor having a flat dome and cylindrical walls to correspond to the current model capabilities and permit the separation of the boundary shape effects. In addition, the experiments will be done with a series of tests of increasing complexity to help evaluate the error or inaccuracy due to each step in complexity.

Figure 3 indicates the steps in the exploration. A test will be done without burning and without swirl cups utilizing uniform dome flow. By utilizing a different temperature for the dilution flow, the dilution mixing can be documented with thermocouple measurements. Swirl cup flow is then introduced to examine the adequacy of modeling this complexity. The dilution holes complexity will also be varied. Burning tests will then be conducted first with gaseous fuel to avoid the question of spray drop size and vaporization and then with a liquid atomizing nozzle. A total of 15 different test setups are planned including either different configurations or different temperature traced regions.

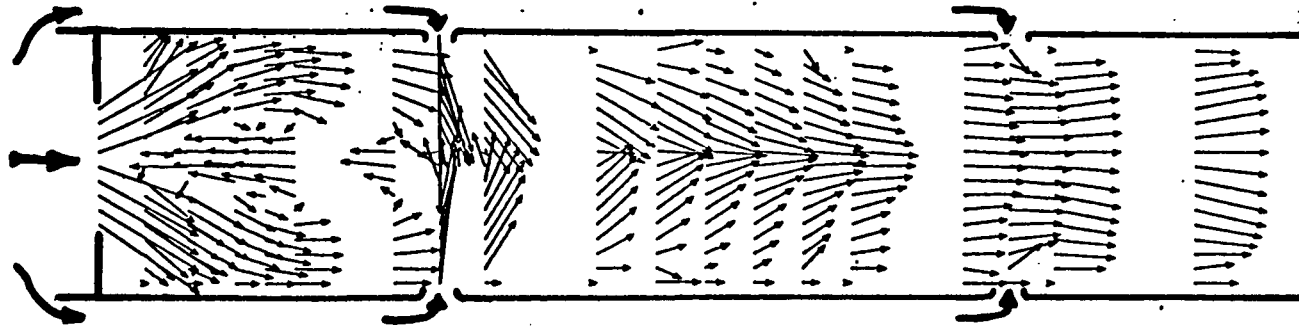
Through comparison studies of model calculations with the type of data indicated, a program plan to improve the overall aerothermo model will be defined that will address the model deficiencies.



- Shaded boxes are modules in current use in combustor design and development work.
- Open boxes are modules in General Electric's aerothermal model planned for use in the future in design and development work after adequate accuracy is demonstrated or developed.

Figure 1. Internal flow module.

CALCULATED VELOCITIES



PLANEXY7- 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20

LENGTH IN X OR Y DIRECTION, 1 IN=100 M/SEC

CALCULATED RADIAL TEMPERATURE PROFILES

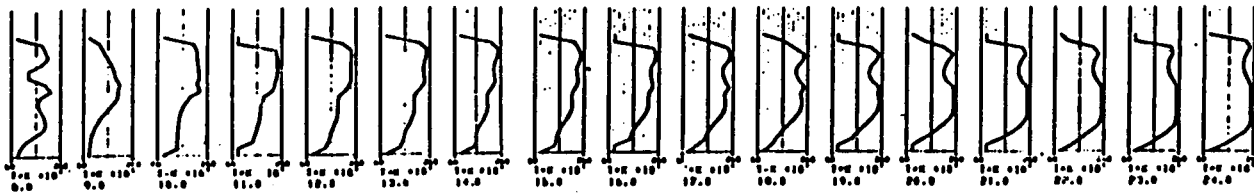


Figure 2. Illustration of computer plotted output for General Electric's INTFLOW module.

GENERAL ELECTRIC TEST PROGRAM WITH COMBUSTOR TEST CONFIGURATIONS COMPATIBLE
WITH ELLIPTIC MODEL

NONBURNING TESTS: DOME FLOW TRACED WITH GAS AT DIFFERENT TEMPERATURE
THAN DILUTION AIR

- o PERFORATED PLATE DOME & F101 DILUTION HOLE PATTERN
- o F101 TYPE SWIRL CUP DOME
 - STANDARD F101 LINER DILUTION HOLE PATTERN
 - ONLY ONE ROW OF DILUTION HOLES
 - DILUTION HOLE PLACEMENT TO MODIFY HOT STREAKS
 - DILUTION HOLE TOLERANCE EFFECTS

BURNING TESTS

- o GASEOUS FUEL
- o LIQUID FUEL WITH PRESSURE ATOMIZING NOZZLE

A TOTAL OF 15 SET UPS PLANNED INCLUDING EITHER CONFIGURATION CHANGES
OR TRACE REGION CHANGES

Figure 3. General Electric test program.

DILUTION ZONE MIXING STUDIES

by J.D. Holdeman

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Motivated by considerations of dilution zone mixing in gas turbine combustion chambers, NASA sponsored, in 1972 - 1975 contract and grant studies of the mixing characteristics of a row of jets injected normally into a duct flow of a different temperature (references 1, 2, & 3). Based on the favorable response to these studies, and the areas for further work which they identified, NASA Lewis is currently conducting a balanced program of contract, grant, and in-house research on various aspects of the 'jet in a confined crossflow' problem. Included in these are: 1) development of interactive computer codes for analysis of dilution jet mixing, and 2) extension of the experiments on jets in a confined crossflow.

From the data of References 1 & 2, an empirical model was developed (refs. 4 & 5) to describe the observed temperature distributions. The current interactive code provides a 3-D pictorial representation of the temperature, as given by these correlations, for any user-specified downstream location, flow, and orifice parameters. Although calculations can be performed for (almost) any flow and geometric conditions of interest, they are, of course, most reliable for conditions within the range of the experiments. These codes will be improved and extended, and options added, as new data become available.

The experiments in References 1 to 3 dealt primarily with a single row of jets mixing into an isothermal flow in a constant cross-section duct. Variations in the mixing were observed as a function of jet-to-mainstream momentum ratio, orifice size, and spacing. The current experiments examine perturbations of this problem characteristic to gas turbine combustion chambers, namely: flow area convergence, non-isothermal mainstream flow, and opposed in-line and staggered injection.

Papers discussing grant studies on free-stream turbulence effects and reverse flow geometries, and in-house analytical calculations of jets in crossflow will be presented at the Combustion Fundamentals Research Conference.

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3. Kamotani, Y. and Greber, Isaac: Experiments on Confined Turbulent Jets in Cross Flow. NASA CR-2392, 1974.
4. Walker, R.E. and Eberhard, R.G.: Multiple Jet Study Data Correlations. NASA CR-134796, 1975.
5. Holdeman, J.D. and Walker, R.E.: Mixing of a Row of Jets with a Confined Crossflow. AIAA Journal, vol.15, no.2, Feb.1977, pp243-249.

DILUTION ZONE MIXING STUDIES

OBJECTIVE - TO CHARACTERIZE DILUTION
ZONE MIXING IN SUFFICIENT
DETAIL TO:

- * IDENTIFY AND UNDERSTAND THE
DOMINANT PHYSICAL MECHANISMS
GOVERNING THE MIXING PROCESS
- * REFINE AND EXTEND EMPIRICAL
MODELS TO PROVIDE A NEAR-TERM
COMBUSTOR DESIGN TOOL
- * PROVIDE A DATA BASE FOR
VERIFICATION OF ANALYTICAL
MODELS

DILUTION ZONE MIXING STUDIES

- * EXPERIMENTS ON EFFECTS OF FREE-STREAM TURBULENCE ON A JET IN CROSSFLOW (GRANT)
- * EXPERIMENTS ON DILUTION JETS IN REVERSE FLOW COMBUSTOR GEOMETRIES (GRANT)
- * DEVELOPMENT OF INTERACTIVE CODES FOR EVALUATION OF DESIGN ALTERNATIVES (HOST: IN-HOUSE)
- * EXPERIMENTS ON JETS IN A CONFINED CROSSFLOW (HOST: CONTRACT)
- * ANALYTICAL CALCULATIONS OF JETS IN CROSSFLOW (IN-HOUSE)

DILUTION ZONE DESIGN COMPUTER PROGRAMS

OBJECTIVE: DEVELOPMENT OF INTERACTIVE COMPUTER CODE FOR ANALYSIS OF MIXING OF JETS WITH A CONFINED CROSSFLOW

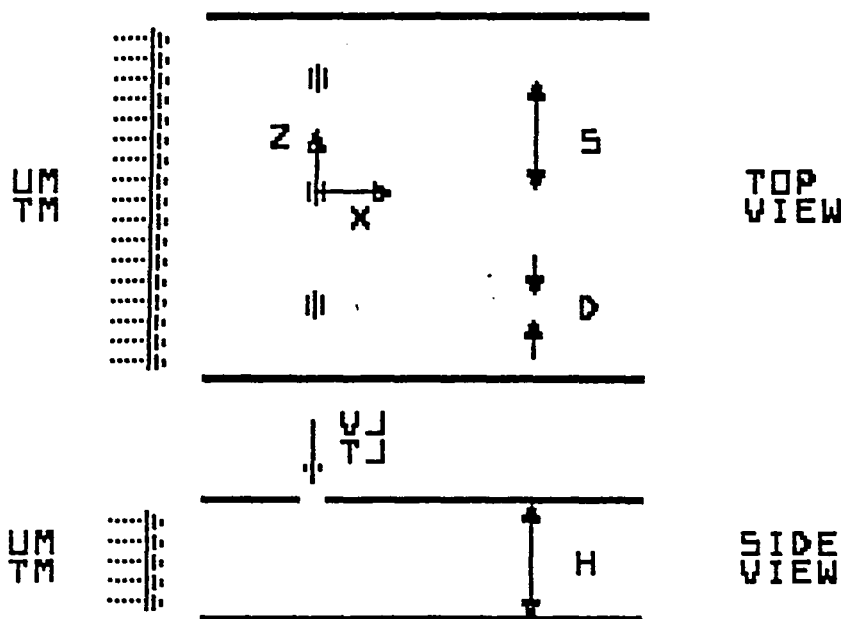
FEATURES: PROVIDES A 3-D PICTORIAL REPRESENTATION OF THE TEMPERATURE FIELD

PURPOSE: EVALUATE EFFECTS OF VARYING FLOW AND GEOMETRY

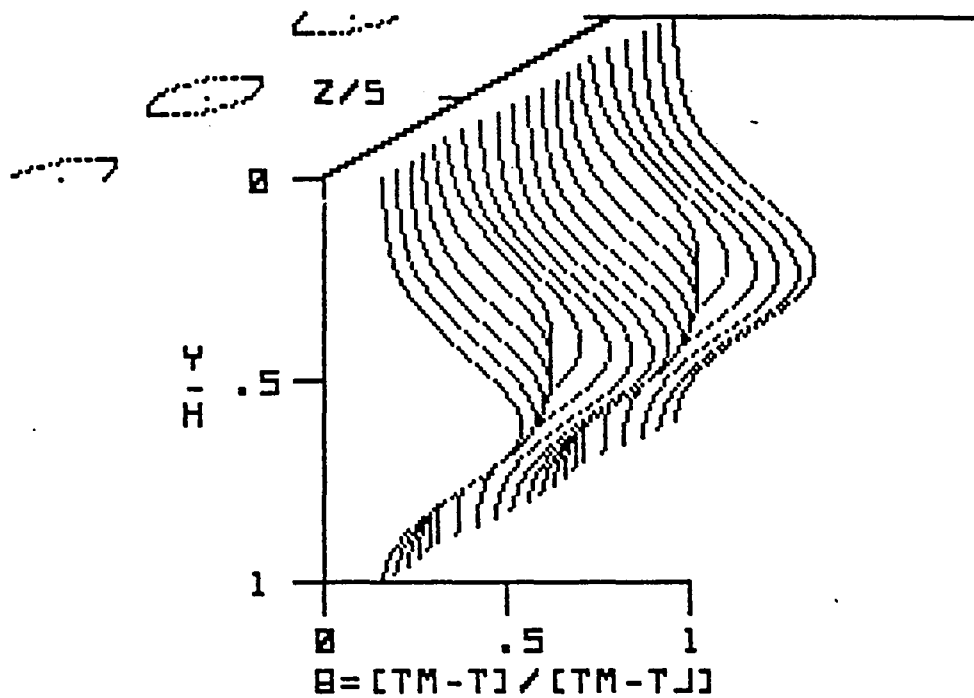
GUIDE DESIGN TO REDUCE DEVELOPMENT TIME AND COST

STATUS: CODES WILL BE IMPROVED AND OPTIONS ADDED AS NEW DATA BECOME AVAILABLE

FLOW SCHEMATIC



Jet in a Confined Crossflow

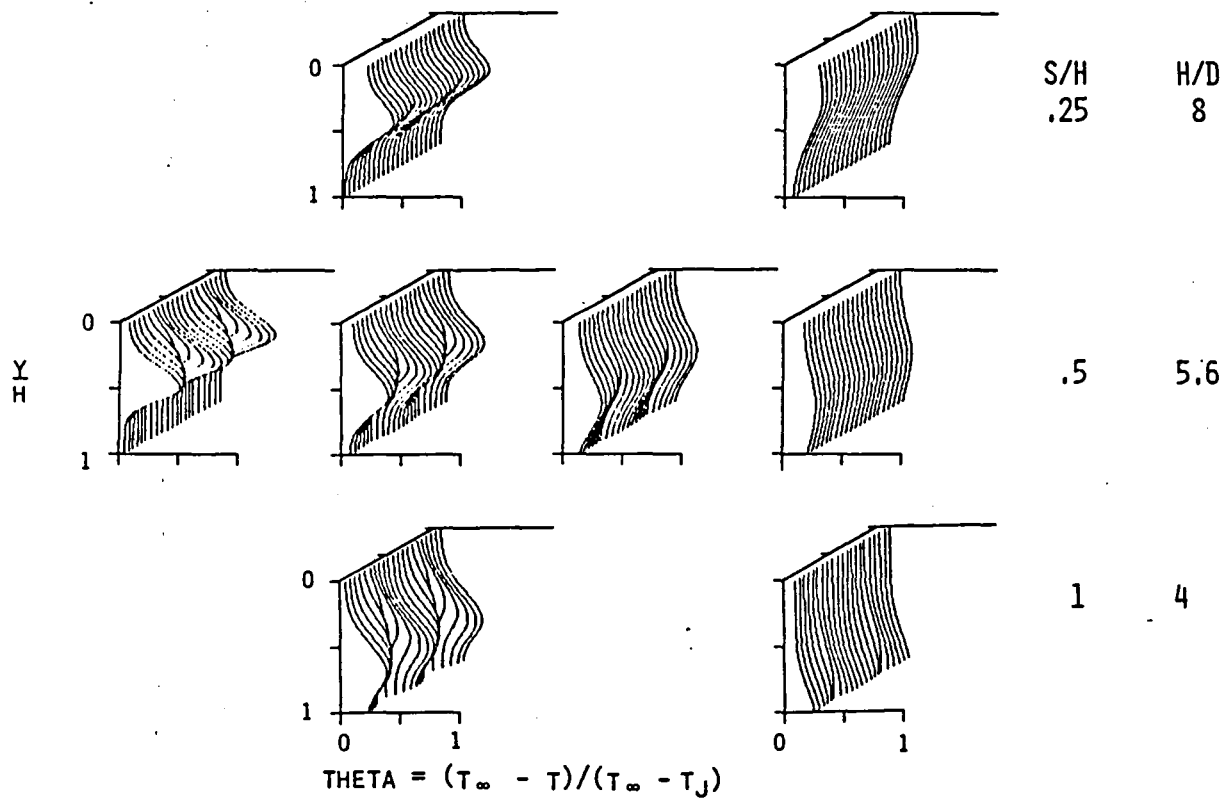


$S/D=2$ $H/D=4$ $J=32$ $DR=2$ $CD=.64$
 $X/H=.5$ $MR=.5$ $TB=.33$ $PF=.34$

Typical Temperature Profile Distribution in Y - Z Plane

$$M_J/M_\infty = .25 \quad (J = 32; (s/D)(H/D) = 16)$$

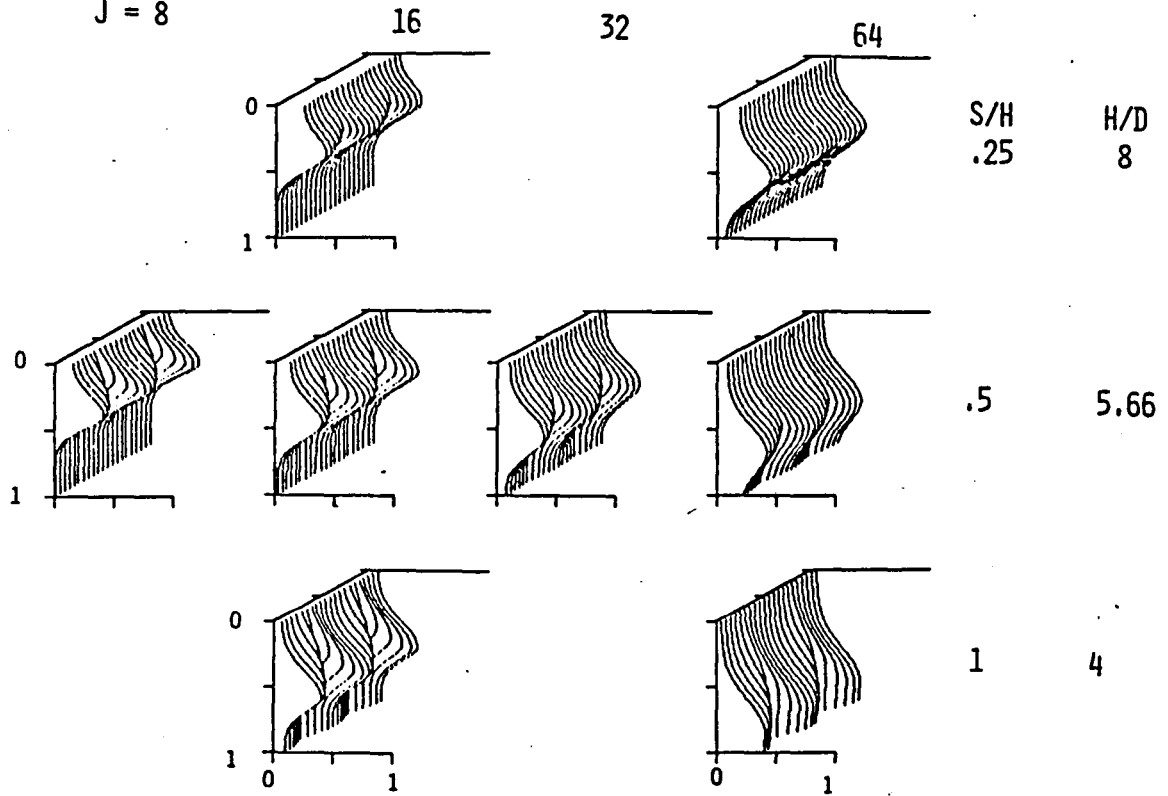
$$X/H = .25 \quad .5 \quad 1 \quad 2$$



Variation in Temperature Distributions with Downstream Distance

$$X/H = .5; (S/D)(H/D) = 16$$

J = 8



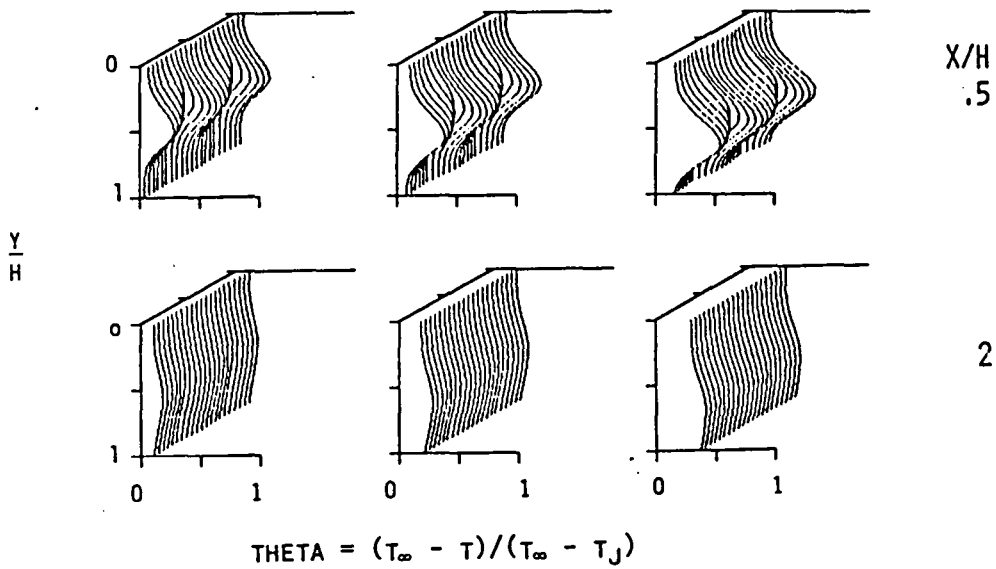
$$\text{THETA} = (T_{\infty} - T)/(T_{\infty} - T_J)$$

Varations in Temperature Distributions with Momentum Ratio

H/D = 8
 $\dot{M}_J/\dot{M}_\infty = .125$

5.66
.25

4
.5



Variation in Temperature Distributions with Orifice Size
at Constant Spacing to Height Ratio ($S/H = .5$; $J = 32$)



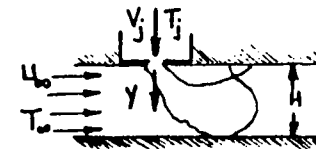
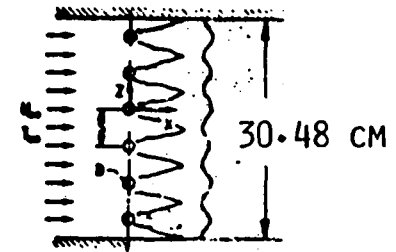
NASA DILUTION JET MIXING - PHASE I

OBJECTIVE:

- 0 COLLECT A DATA BASE ON MIXING OF A ROW OF JETS WITH A CONFINED CROSS FLOW
- 0 DEVELOP EMPIRICAL JET MIXING CORRELATIONS

PARAMETERS INVESTIGATED:

- 0 MOMENTUM RATIO (J), H/D , s/D , q_j/q_∞
- 0 NON-UNIFORM CROSS-STREAM TEMPERATURE AND VELOCITY PROFILES
- 0 COLD/HOT JET INJECTION
- 0 CROSS-STREAM FLOW AREA CONVERGENCE

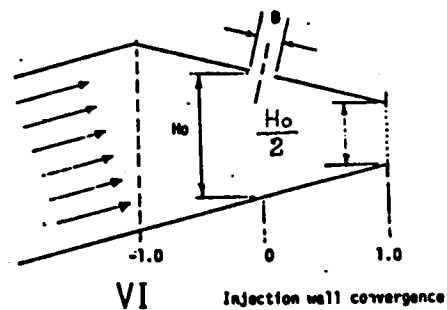
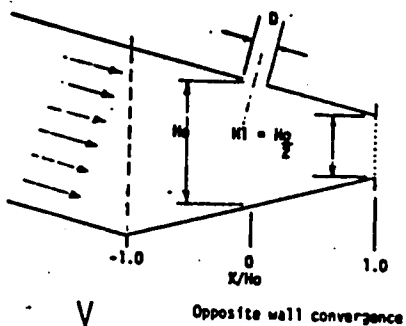
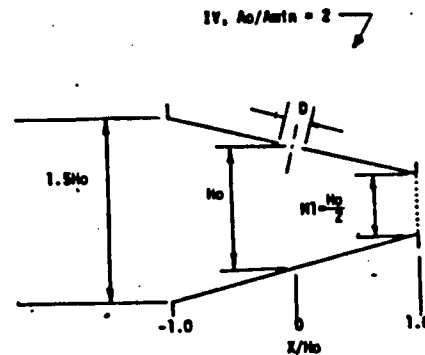
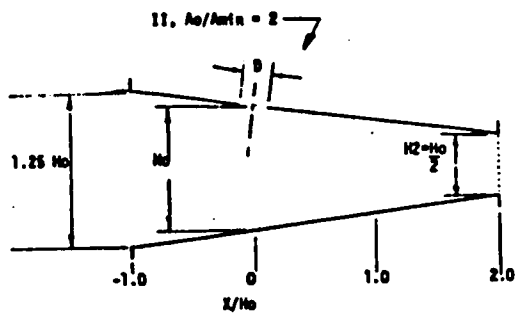




PHASE 1 TEST SECTION CONFIGURATIONS

TEST SECTIONS:

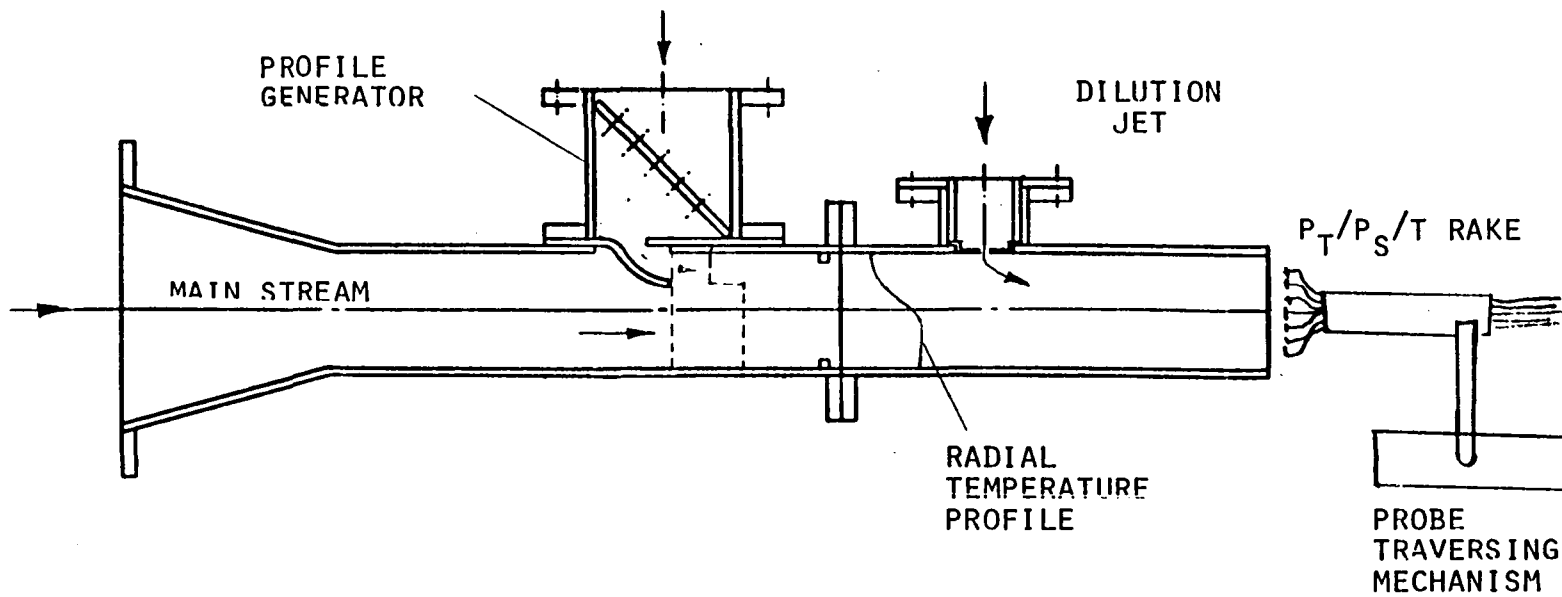
$$H_0 = 10.16 \text{ cm}$$



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PHOENIX, ARIZONA



SCHMEMATIC OF THE DILUTION JET MIXING TEST RIG



317

AIRESEARCH MANUFACTURING COMPANY OF ARIZONA

A DIVISION OF THE GARRETT CORPORATION



NASA DILUTION JET MIXING - PHASE II

OBJECTIVE:

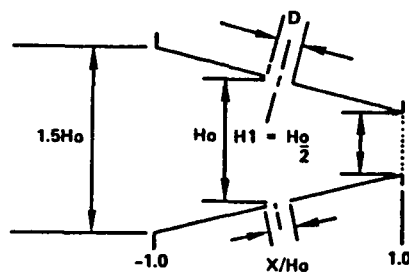
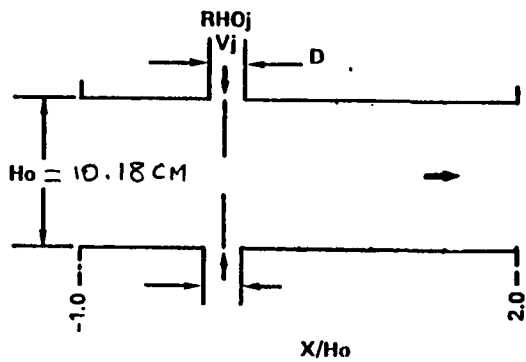
- 0 EXTEND THE DATA BASE ON MIXING OF SINGLE-SIDED ROW OF JETS WITH A CONFINED CROSS FLOW,
- 0 COLLECT DATA BASE ON MIXING OF TWO-SIDED ROW OF JETS WITH A CONFINED CROSS FLOW
- 0 DEVELOP EMPIRICAL JET MIXING CORRELATIONS

PARAMETERS INVESTIGATED:

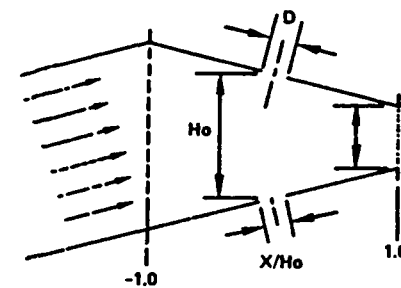
- 0 CIRCULAR VS SQUARE ORIFICES, TWO-DIMENSIONAL SLOT
- 0 MOMENTUM RATIO (J), H/D , s/D
- 0 IN-LINE AND STAGGERED ORIFICE CONFIGURATIONS
- 0 NON-UNIFORM CROSS-STREAM TEMPERATURE AND VELOCITY PROFILES
- 0 CORSS-STREAM FLOW AREA CONVERGENCE



PHASE II TEST SECTIONS AND ORIFICE CONFIGURATIONS

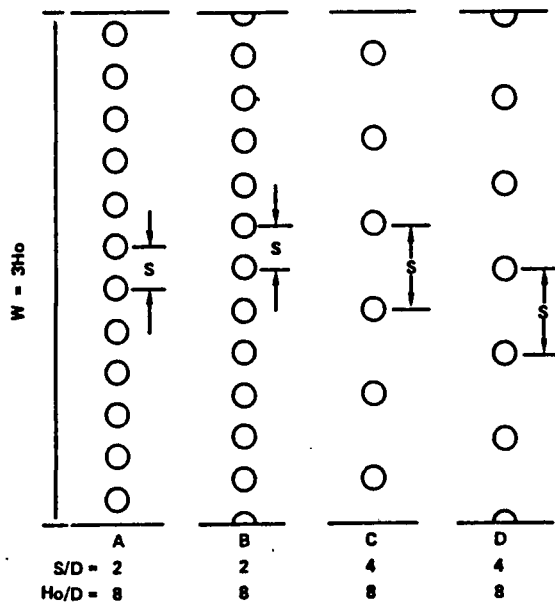


SYMMETRIC CONVERGENCE



ASYMMETRIC CONVERGENCE

Test Sections



Orifice Configurations.



PHASE II SERIES 1 TEST CONDITIONS

U_M = 15 M/S

T_M = 645°K

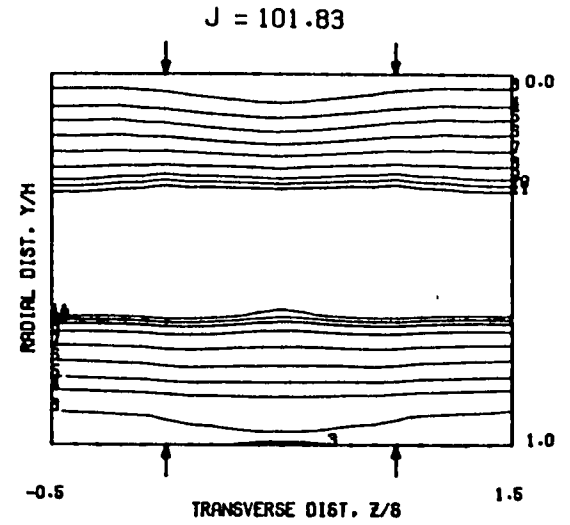
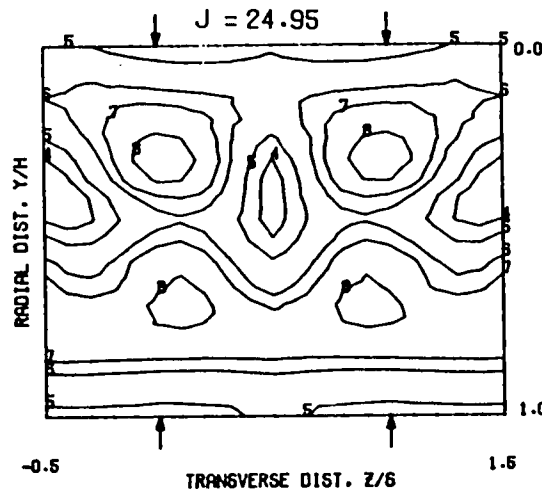
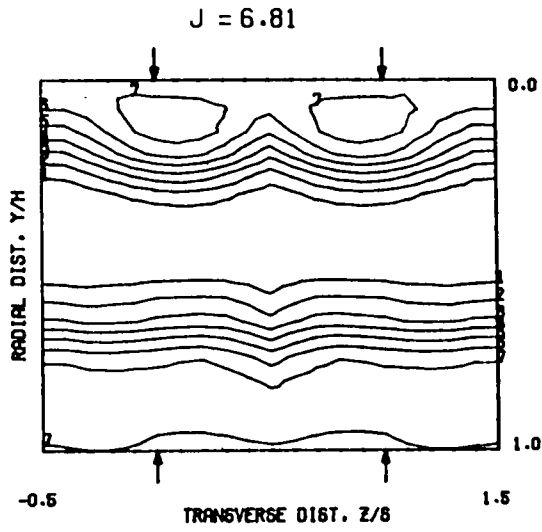
H₀ = 10.16 Cm

H/D	S/D	CONFIGURATION	J _{TOP}	J _{BOTTOM}
8	2	IN-LINE	6.81	6.88
		INJECTION	25.0 101.8	24.8 101.9
8	2	STAGGERED	6.53	6.54
		INJECTION	25.2 99.3	24.7 99.6
8	4	IN-LINE	7.85	7.81
		INJECTION	27.9 108.3	27.3 107.0
8	4	STAGGERED	5.98	6.14
		INJECTION	25.7 103.1	25.7 104.3

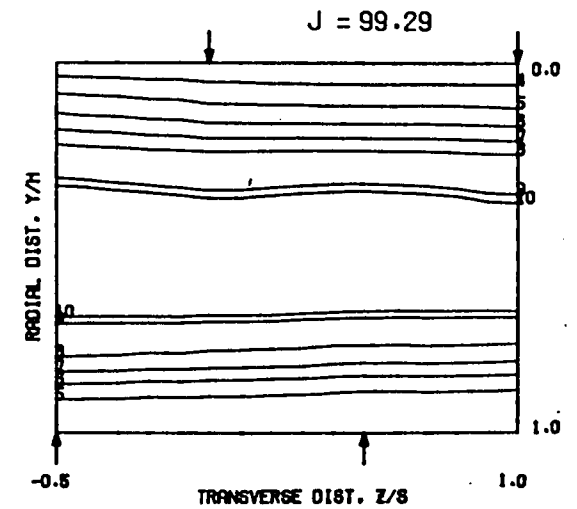
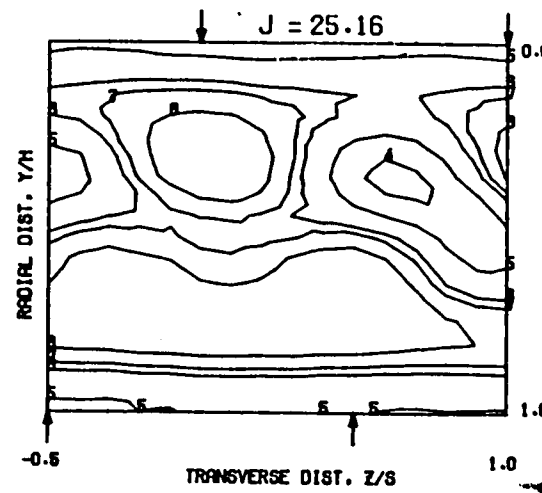
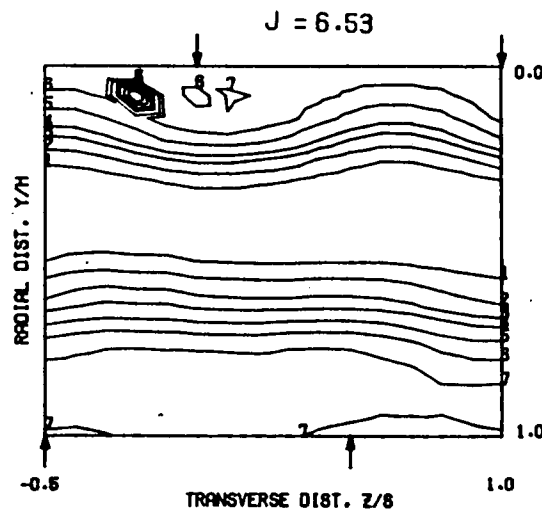


MEASURED THETA DISTRIBUTIONS FOR $S/D = 2$, $H/D = 8$, $X/H_0 = 0.5$

INLINE ORIFICE CONFIGURATION



STAGGERED ORIFICE CONFIGURATION

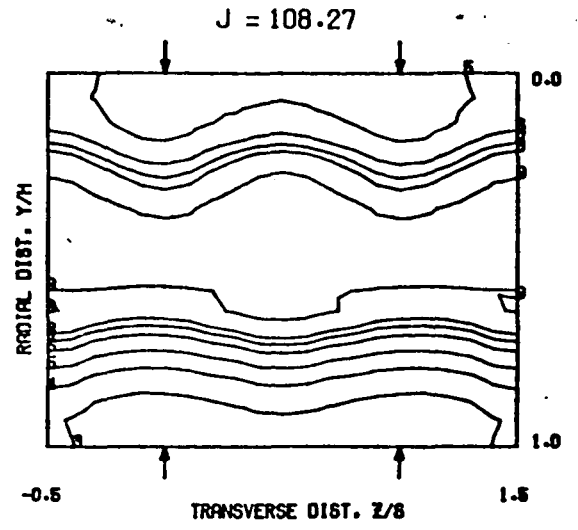
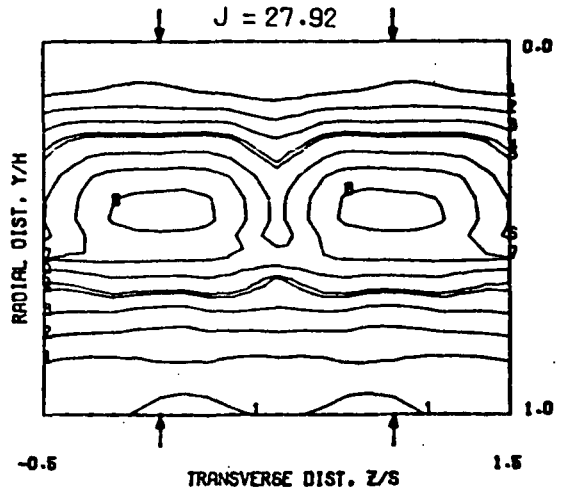
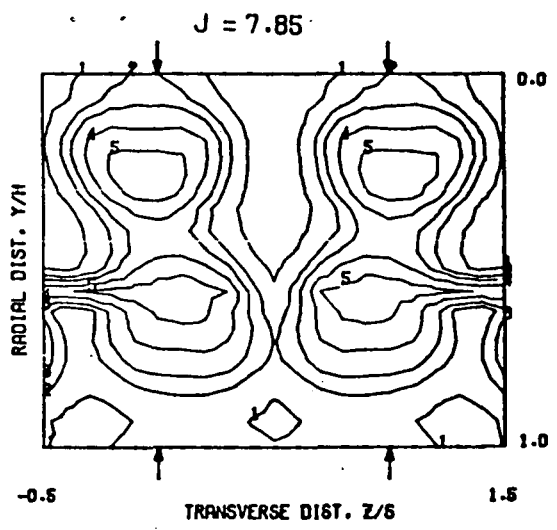


$$\theta = (T_M - T)/(T_M - T_J)$$

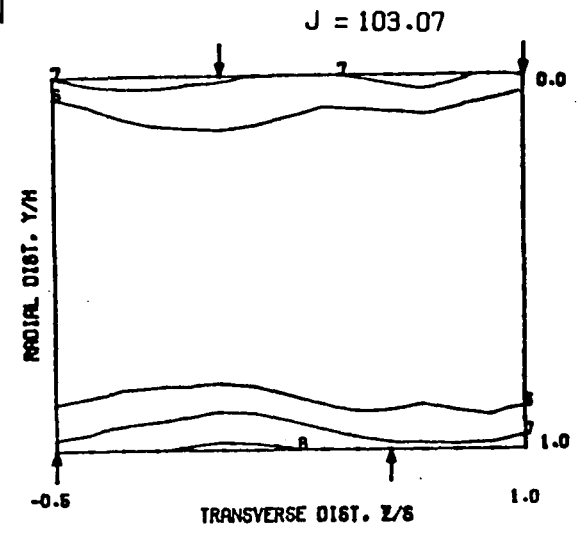
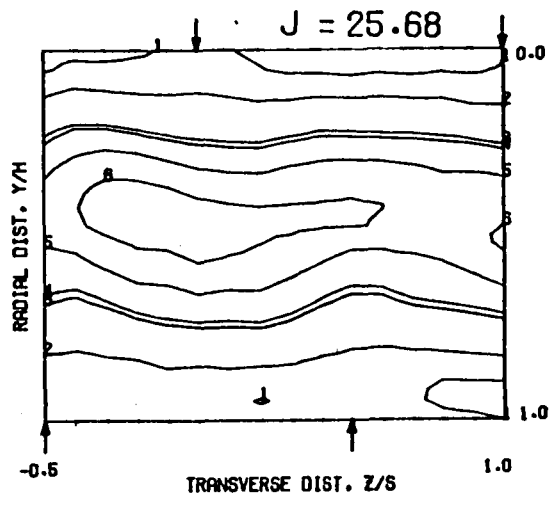
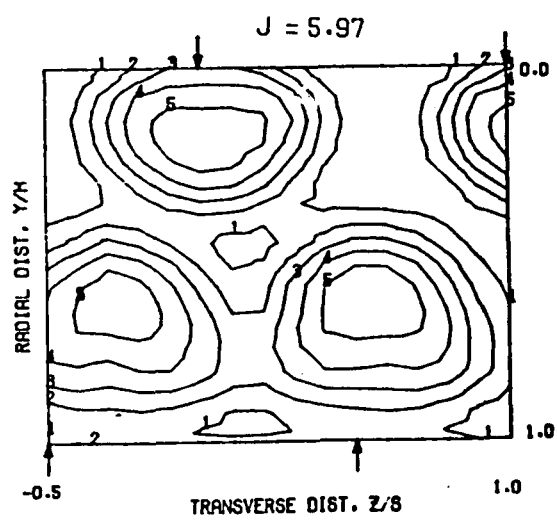


MEASURED THETA DISTRIBUTIONS FOR $S/D = 4$, $H/D = 8$, $X/H_0 = 0.5$

IN-LINE ORIFICE CONFIGURATION



STAGGERED ORIFICE CONFIGURATION

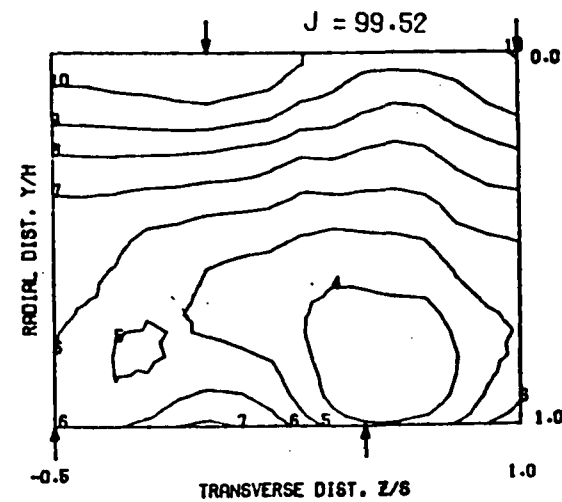
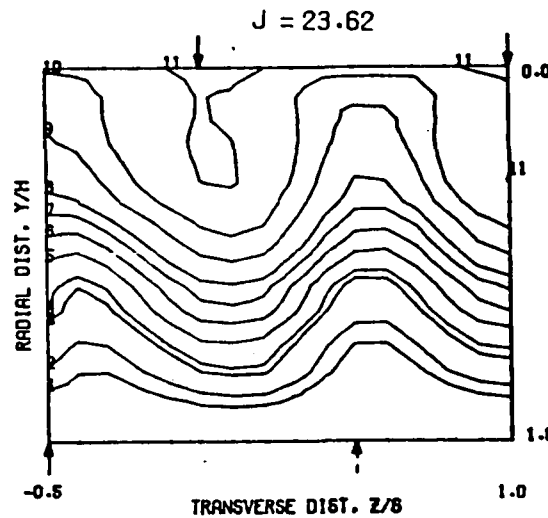
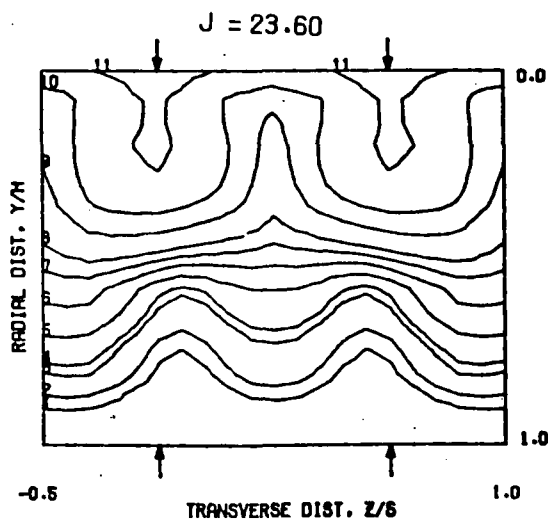
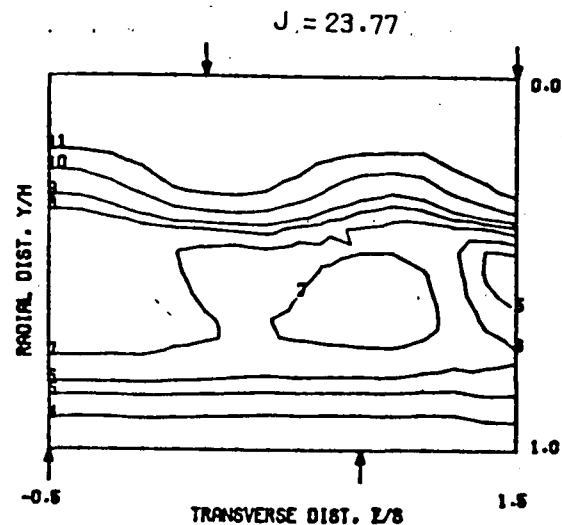
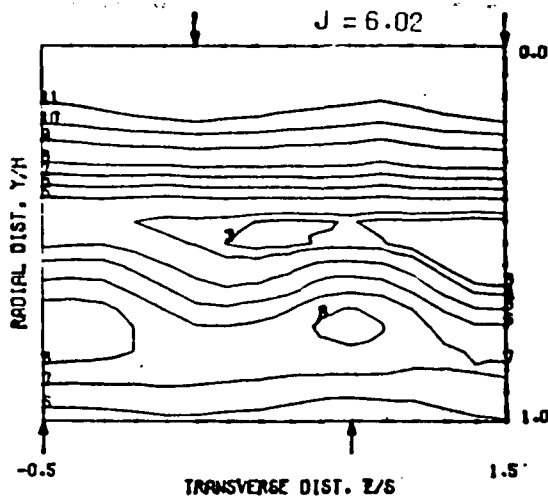
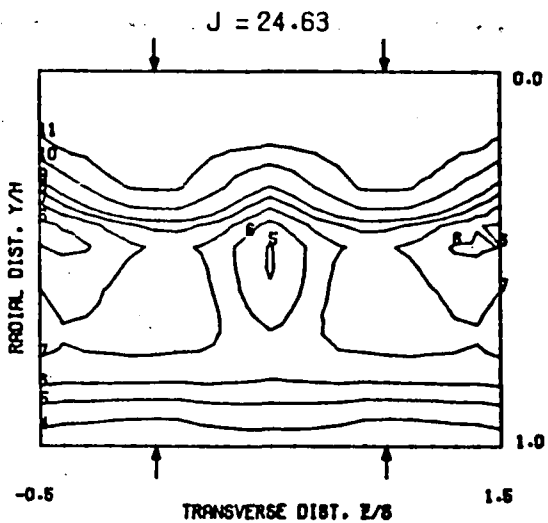


$$\theta = (T_M - T)/(T_M - T_J)$$



MEASURED THETA DISTRIBUTION FOR PROFILED MAINSTREAM

$S/D = 2, H/D = 8$



$S/D = 4, H/D = 8$
 $\theta = (T_{MAX} - T)/(T_{MAX} - T_J)$



PHASE II SERIES 2 TESTS

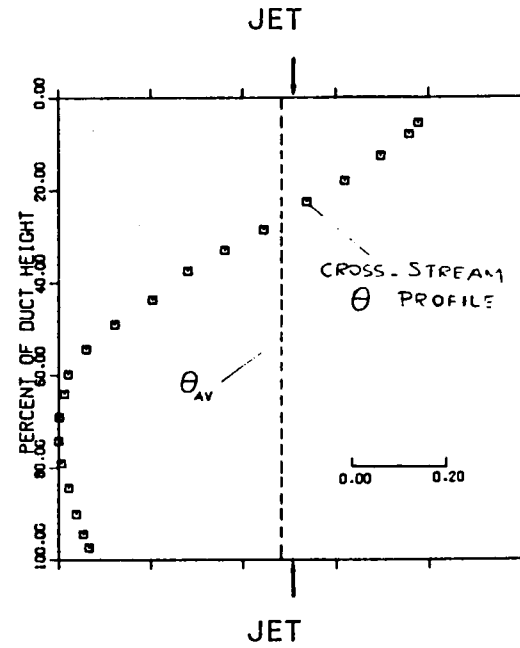
SCOPE:

O TWO-SIDED JET INJECTION WITH PROFILED
CROSS-STREAM

TEST CONDITIONS:

$U_m = 15 \text{ M/S}$ $H = 10.16 \text{ CM}$, $T_J = 300^{\circ}\text{K}$

H/D	S/D	CONFIGURATION	J _{TOP}	J _{BOTTOM}
8	2	IN-LINE	24.6	24.7
		STAGGERED	6.02	6.21
		STAGGERED	23.8	23.4
8	4	IN-LINE	23.6	24.2
		STAGGERED	23.6	24.1
		STAGGERED	99.5	99.3



$$\theta = (T_{MAX} - T) / (T_{MAX} - T_J)$$



FUTURE TEST PLAN ON PHASE II

- o ONE-SIDED JET INJECTION
 - o TWO - DIMENSIONAL SLOT
 - o SQUARE HOLES

- o TWO-SIDED JET INJECTION
 - o NON-UNIFORM CROSS-STREAM TEMPERATURE PROFILES
 - o UNEQUAL JET INJECTION RATES
 - o CONVERGENT TEST SECTIONS (SYMMETRIC AND ASYMMETRIC)

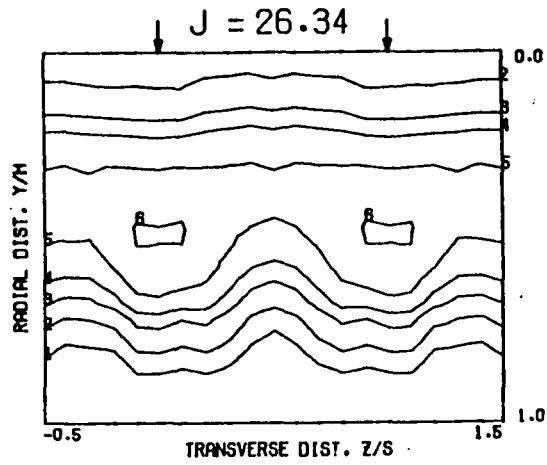
SCHEDULED COMPLETION DATE ON PHASE II TESTS: DECEMBER 1982

GARRETT TURBINE ENGINE COMPANY
A DIVISION OF THE GARRETT CORPORATION
PHOENIX, ARIZONA

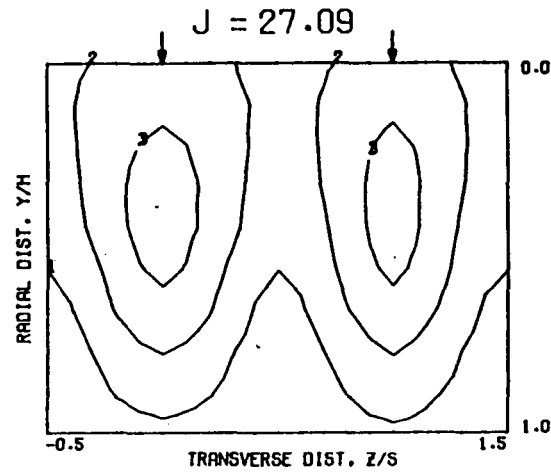


MEASURED THETA DISTRIBUTIONS WITH FLOW AREA
CONVERGENCE FOR $S/D = 4$, $H/D = 8$ AT $X/8 = 1$

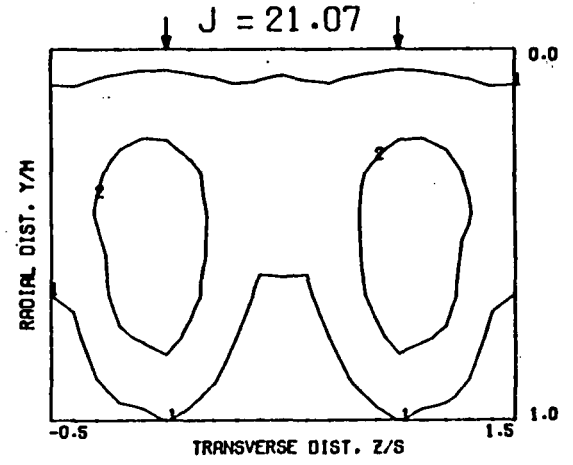
$\phi = 90^\circ$, $A_1/A_2 = 1$



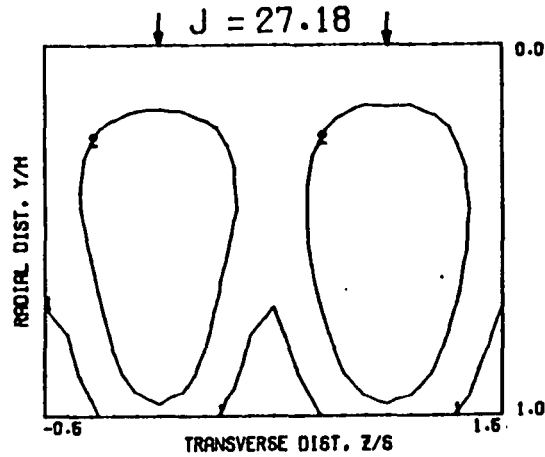
$\phi = 97^\circ$, $A_1/A_2 = 1.33$



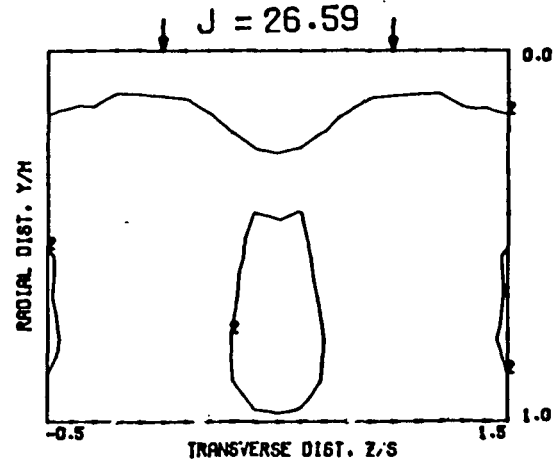
$\phi = 104^\circ$, $A_1/A_2 = 2$



$\phi = 90^\circ$, $A_1/A_2 = 2$



$\phi = 117^\circ$, $A_1/A_2 = 2$





COMPARISON BETWEEN ONE-SIDED AND TWO-SIDED JET INJECTION

S/D = 2, H/D = 8

PARAMETER	M_{JET}/M_{MAIN}	TWO-SIDED INJECTION		ONE-SIDED INJECTION	
		IN-LINE	STAGGERED	H/D = 8	H/D = 4
MOMENTUM RATIO, J	0.23	6.81	6.53	25.3	5.74
$\theta_{EB} = (T_M - T_{EB}) / (T_M - T_J)$		0.198	0.189	0.169	0.176
MAX JET PENETRATION, Y_c/H		0.2	0.19	0.40	0.36
JET HALF WIDTH, $W_{1/2}^+/H$, AT $X/H = 1$		0.33	0.25	0.28	0.20
JET HALF WIDTH, $W_{1/2}^-/H$, AT $X/H = 1$		0.0	0.0	0.12	0.13
θ/θ_{EB} AT $X/H = 1$		1.82	1.94	1.89	2.06
MOMENTUM RATIO, J	0.47	25.0	25.2	107.8	22.0
$\theta_{EB} = (T_M - T_{EB}) / (T_M - T_J)$		0.318	0.319	0.302	0.271
MAX JET PENETRATION, Y_c/H		0.35	0.35	0.66	0.60
JET HALF WIDTH $W_{1/2}^+/H$ AT $X/H = 1$		0.	0.	0.19	0.24
JET HALF WIDTH, $W_{1/2}^-/H$ AT $X/H = 1$		0.2	0.28	0.26	0.27
θ_c/θ_{EB} AT $X/H = 1$		1.08	1.10	1.17	1.44



COMPARISON BETWEEN ONE-SIDED AND TWO-SIDED JET INJECTION

S/D = 4, H/D = 8

PARAMETER	M_{JET}/M_{MAIN}	TWO-SIDED INJECTION		ONE-SIDED INJECTION	
		IN-LINE	STAGGERED	H/D = 8	H/D = 4
MOMENTUM RATIO, J	0.125	7.85	5.97	26.3	6.14
$\theta_{EB} = (T_M - T_{EB})/(T_M - T_J)$		0.112	0.102	0.105	0.107
MAX JET PENETRATION, Y_c/H		0.37	0.24	0.54	0.54
JET HALF WIDTH, $W_{1/2}^+/H$, AT $X/H = 1$		0.05	0.13	0.27	0.20
JET HALF WIDTH, $W_{1/2}^-/H$, AT $X/H = 1$		0.29	0.11	0.23	0.24
θ/θ_{EB} AT $X/H = 1$		2.64	1.77	1.63	2.54
MOMENTUM RATIO, J	0.23	27.9	25.7	109.0	26.7
$\theta_{EB} = (T_M - T_{EB})/(T_M - T_J)$		0.190	0.189	0.181	0.192
MAX JET PENETRATION, Y_c/H		0.50	0.50	1.0	1.0
JET HALF WIDTH $W_{1/2}^+/H$ AT $X/H = 1$		0.0	0.0	0.0	0.0
JET HALF WIDTH, $W_{1/2}^-/H$, AT $X/H = 1$		0.24	0.27	0.33	0.45
θ_c/θ_{EB} AT $X/H = 1$		1.43	1.37	1.41	1.75

COMBUSTION SYSTEM FOR RADIATION INVESTIGATIONS

J. D. Wear

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

Description of Combustion System Hardware

The combustion system consists of an inlet interface flange, inlet diffuser, fuel struts and nozzles, combustor liner, liner housing and exhaust flange. The system will be installed in an existing test facility that can furnish combustion air at the conditions listed below. The system was designed for operation at 40 atmospheres inlet pressure, 900 K inlet temperature, and air flow to 80 kg/sec.

Six penetrations are provided in the outer pressure housing.

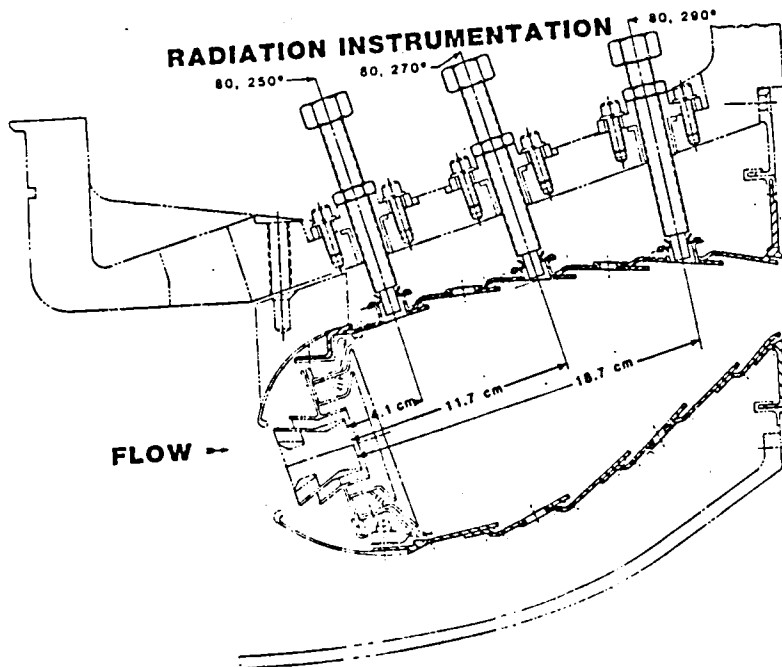
Adapters at the penetrations, permit use of various types of radiation instrumentation, such as total radiometers, spectral radiometers, porous plug and heat flux gages.

The primary zone of the combustor will have hardware modifications that will permit operation at different primary zone fuel-air ratios with constant overall fuel-air ratio.

Rotating exhaust instrumentation will be used to determine combustor performance in addition to the radiation data.

Hardware of an existing high temperature combustion system was modified to accept radiation instruments. Five total radiation radiometers and two heat flux gages were installed.

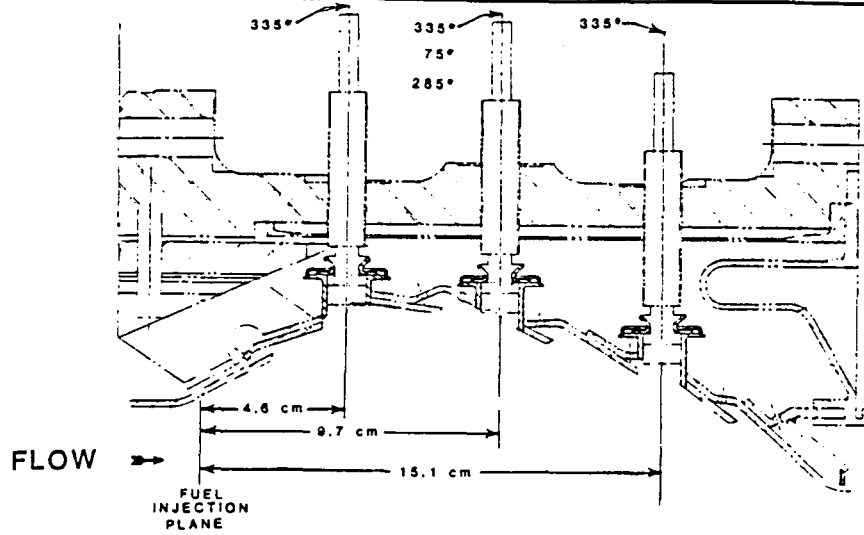
Data are presented showing total radiation at three axial positions of the combustor, and comparison of total radiation with data from a heat flux gage.



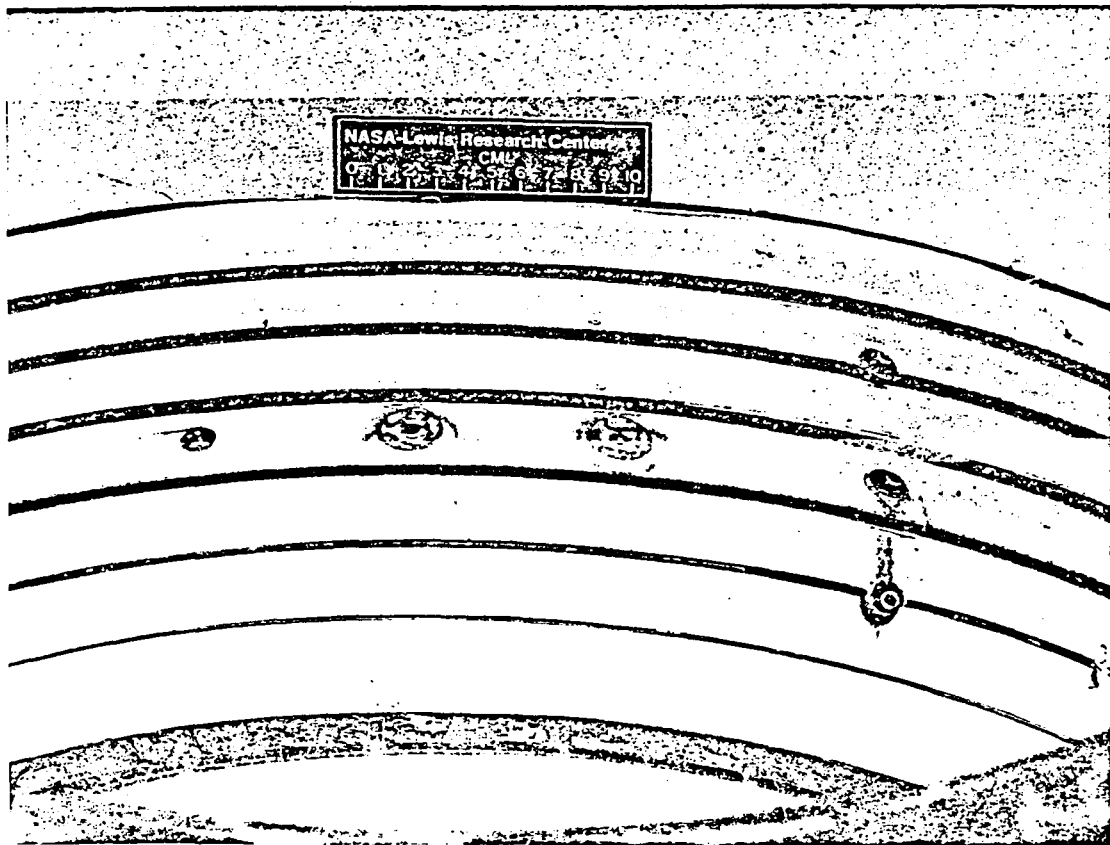
DESIGN CYCLE CONDITIONS

	<u>INLET TOTAL</u>		<u>DIFFUSER</u>	<u>COMBUSTOR</u>	
	Press.	Temp.	INLET	F/A	Exhaust
	MPa	K	MACH		Temp.
			NO.		K
TAKEOFF	4.05	889	0.328	0.0275	1779
CLIMB	3.47	849	.331	.0248	1669
CRUISE	1.72	815	.333	.0257	1674
TAXI IDLE	.50	517	.337	.0114	972

RADIATION INSTRUMENTATION



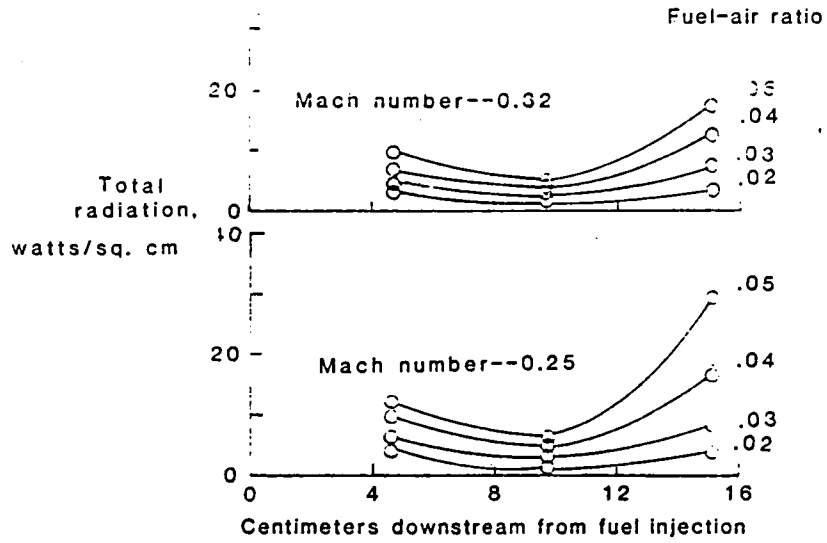
1-42-4463



TOTAL RADIATION

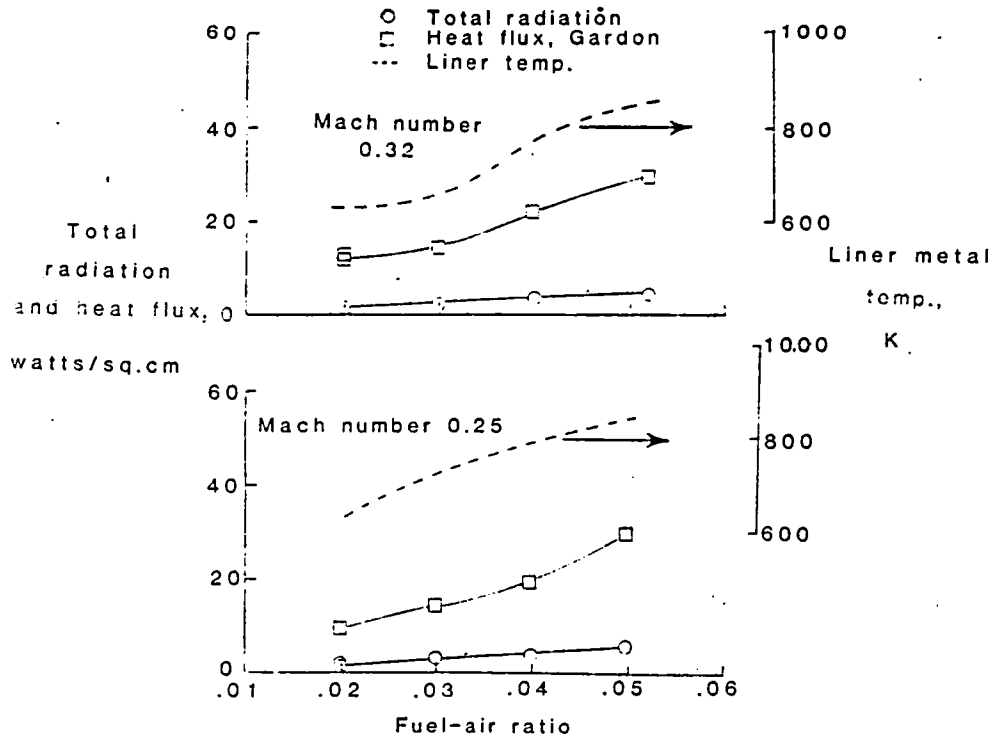
Combustor inlet

Pressure--0.69 MPa
Temp.--560 K



RADIATION AND HEAT FLUX

Press.--0.69 MPa
Temp.--560 K



USING THE IN-HOUSE LINER CYCLIC RIGS

Robert L. Thompson
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

The objectives of the HOST Burner Liner Cyclic Rig test program are basically twofold: (1) to assist in developing predictive tools needed to improve design analyses and procedures for the efficient and accurate prediction of burner liner structural response; and (2) to validate these predictive tools by comparing the predicted results with the experimental data generated in the tests. The data generated will include measurements of the thermal environment (metal temperatures) as well as the structural (strain) and life (fatigue) responses of simulated burner liners and specimens under controlled boundary and operating conditions. These data will be used to validate, calibrate and compare existing analytical theories, methodologies and design procedures, as well as improvements in them, for predicting liner temperatures, stress-strain-responses and cycles to failure. Comparison of analytical results with experimental data will be used to show where the predictive theories, etc. need improvements. In addition, as the predictive tools; as well as the tests and test methods, are developed and validated, a proven, integrated analytical/experimental method will be developed to determine the cyclic life of a burner liner.

Figure 1 includes a list of the test rigs under consideration and the basic liner segments or components to be tested in each rig. Each succeeding test rig and the tests to be conducted in that rig are increasingly more complex than the preceding one, beginning with a flat plate, then a tube, then a subelement, and finally a full-scale liner test. Correspondingly, the structural analysis becomes more complex in this progression of test configurations. The Quartz Lamp Box Rig, from which experimental temperature and strain data will be obtained, will also serve as the test rig configuration for the evaluation of special instrumentation under development in the HOST program; for example, infrared camera for temperature mapping, thin-film thermocouples, thin-film strain gauges, laser speckle techniques, etc. The instrumentation with the greatest potential will be incorporated and used in the other rigs.

Test conditions and variables to be considered in each of the test rigs and test configurations, and also used in the validation of the structural predictive theories and tools, will include: thermal and mechanical load histories (simulating an engine mission cycle, different boundary conditions, specimens and components of different dimensions and geometries, different materials, various cooling schemes and cooling hole configurations, several advanced burner liner structural design concepts, and the simulation of hot streaks. Based on these test conditions and test variables, the test matrices for each rig and configurations will be established with the intent to verify the predictive tools over as wide a range of test conditions as possible using the simplest possible tests. An illustrative flow chart for the thermal/structural analysis of a burner liner and how the analysis relates to the tests is shown schematically in Figure 2. The chart shows that several nonlinear constitutive theories are to be evaluated.

Preliminary structural analyses in which several viscoplastic (unified) theories are being evaluated are underway for a flat plate, an axisymmetric combustor liner (tube) segment and a three-dimensional simulated combustor liner segment, each of which will be tested in its appropriate test rig. The basic elements required for a structural analysis of a flat plate are outlined in Figure 3. Analysis of the axisymmetric liner is just beginning. A representative finite element ring model and an imposed transient temperature distribution are shown in Figure 4. Analysis of a 3-D combustor liner constructed from stacked sheet metal louvers has been initiated. The construction of this combustor liner is shown in Figure 5. A representative symmetric finite element model of one of the segments with cooling holes is shown in Figure 6. In this example, less than 1° or 1/360th of the inner liner is being modeled. Typical temperature inputs to the structural analysis code, both steady-state and transient distributions obtained from measured data and a thermal analysis, are shown in Figure 7. The predicted strains will be compared with the experimental strains in order to validate the predictive theories for each of these test configurations. In addition, these types of preliminary analyses will also be useful in determining where both thermocouples and strain gauges should be located on the specimens in order to ensure that regions of steep thermal gradients and high stress concentrations are captured in the test measurements. A tentative schedule for completing the structural analyses of these test specimens is shown in Figure 9. As other test configurations are identified in this study, they will be added to this list.

**STRUCTURAL ANALYSIS
OF COMPONENTS TO BE TESTED IN HOST LINER CYCLIC RIGS**

1. QUARTZ LAMP BOX RIG
FLAT PLATE
2. QUARTZ LAMP ANNULAR RIG
SUBELEMENT OF COMBUSTOR LINER
TUBE
3. LOW PRESSURE CYCLIC CAN RIG
FULL-SCALE COMBUSTOR LINER

FIGURE 1

NONLINEAR THERMAL/STRUCTURAL ANALYSIS OF ADVANCED COMBUSTOR LINERS

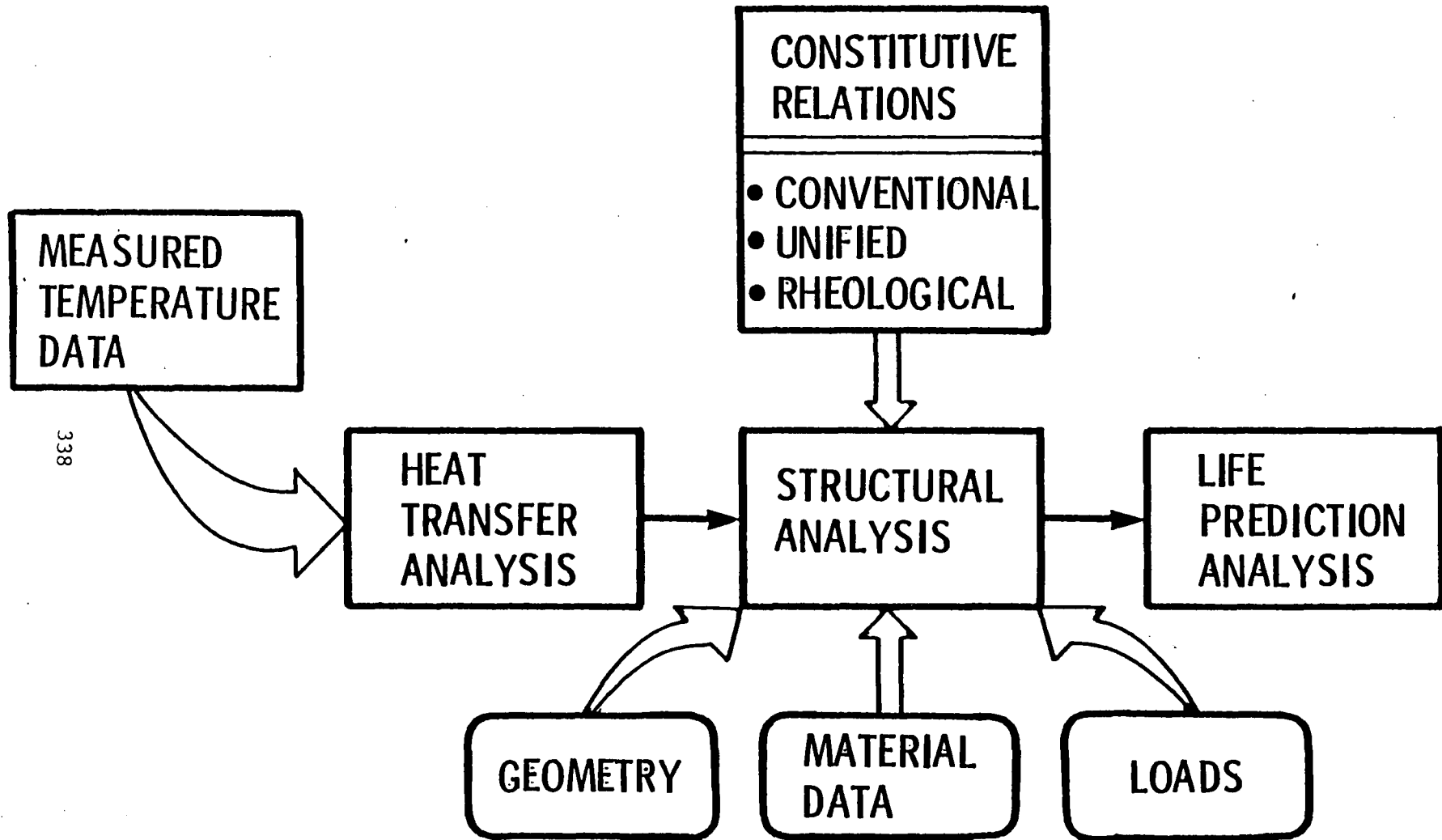
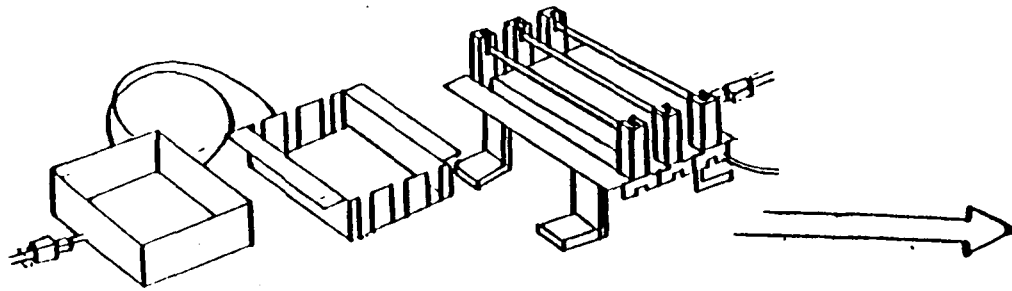


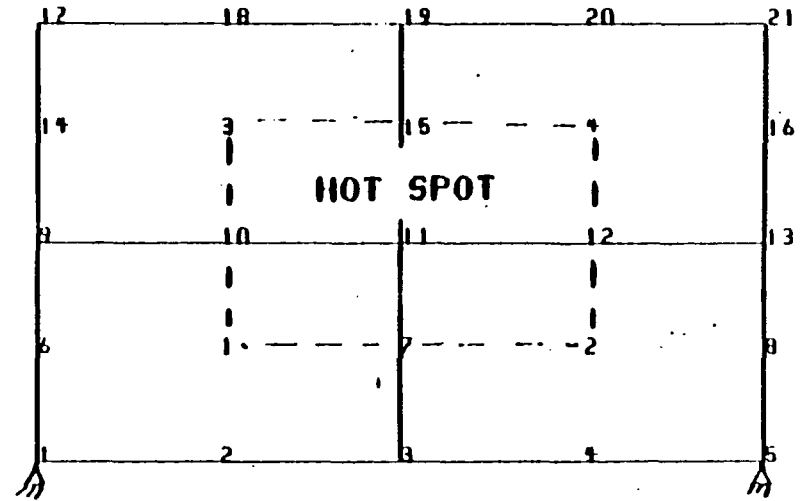
FIGURE 2

COMBUSTOR LINER ANALYSES AND TESTS (HOST)

QUARTZ LAMP BOX RIG



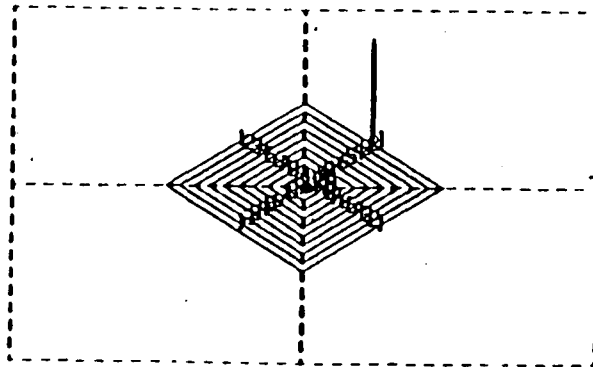
STRUCTURAL ANALYSIS OF FLAT PLATE



339

- 1 = 1.35E3
- 2 = 1.38E3
- 3 = 1.40E3
- 4 = 1.43E3
- 5 = 1.46E3
- 6 = 1.48E3
- 7 = 1.51E3
- 8 = 1.53E3
- 9 = 1.56E3
- 10 = 1.59E3

IMPOSED TEMPERATURE HISTORY



- 1 = -5.34E4
- 2 = -4.50E4
- 3 = -3.67E4
- 4 = -2.83E4
- 5 = -2.41E4
- 6 = -1.16E4
- 7 = -3.25E3
- 8 = 5.11E3
- 9 = 1.35E4
- 10 = 2.18E4

STRESS/STRAIN CONTOUR PLOTS

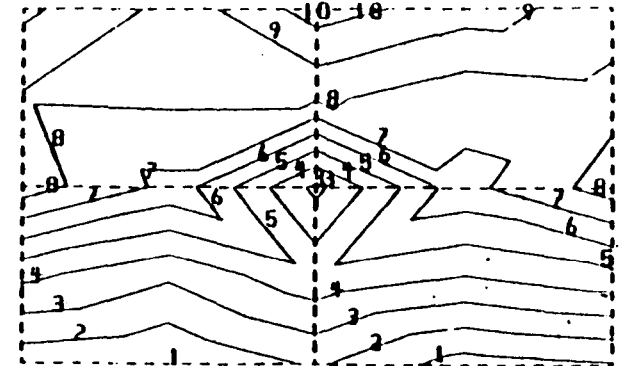


FIGURE 3

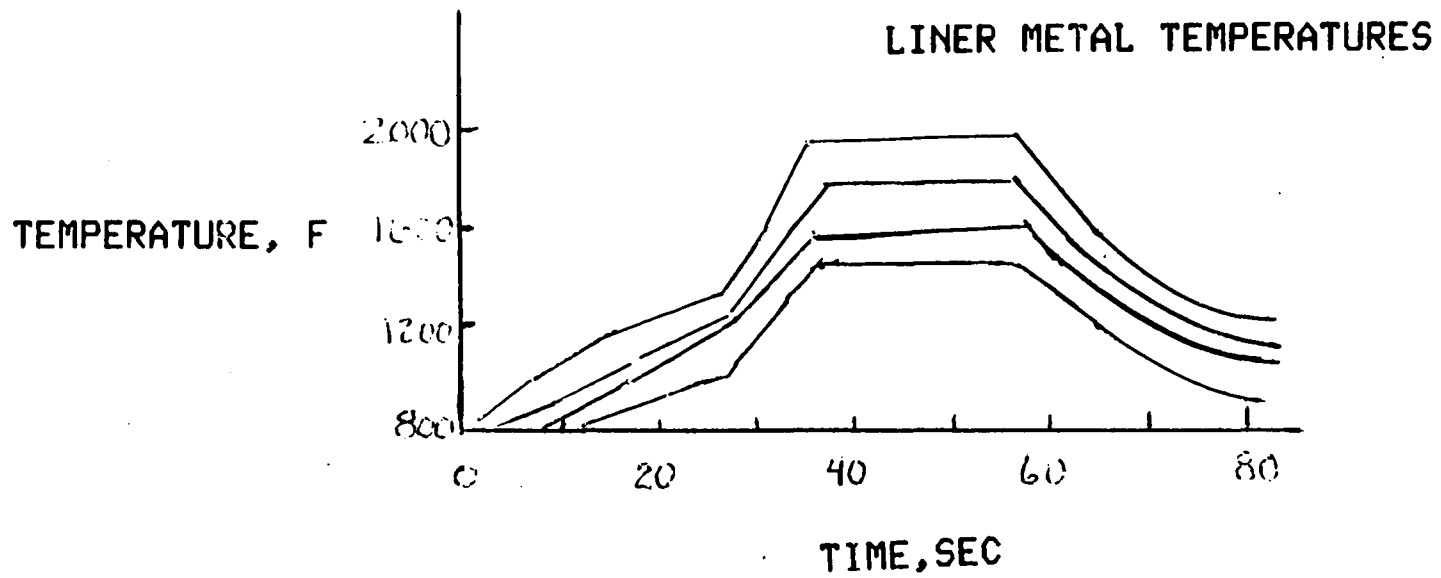
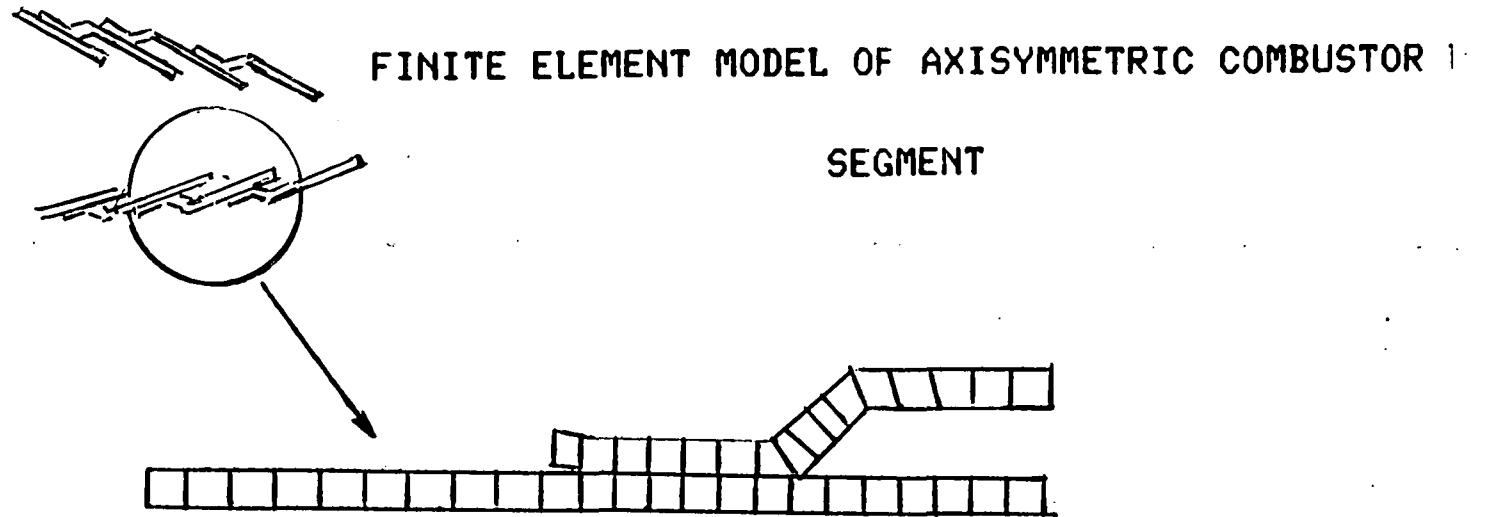


FIGURE 4

TYPICAL LOUVER COMBUSTOR CONSTRUCTION

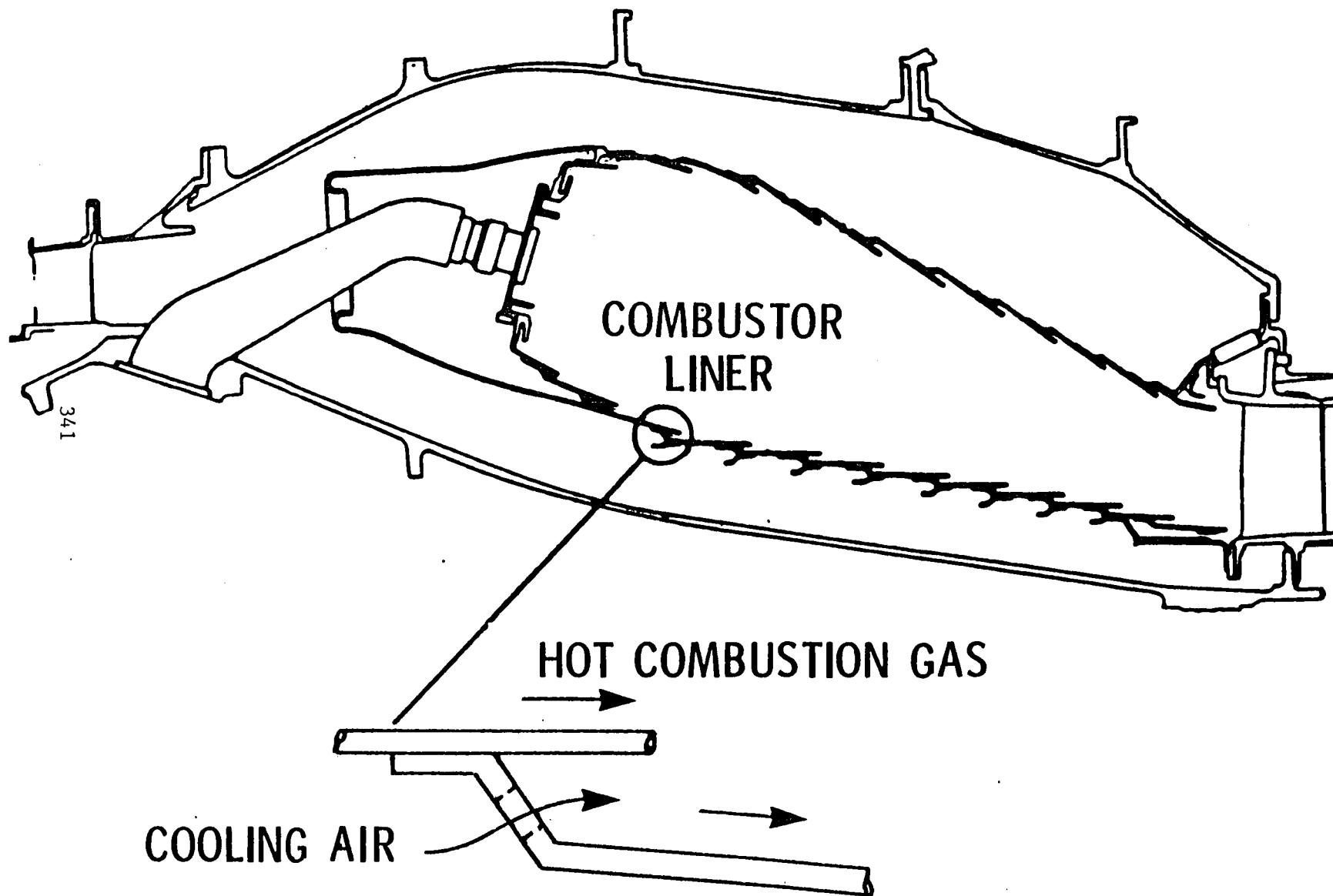


FIGURE 5

LOUVER METAL TEMPERATURES

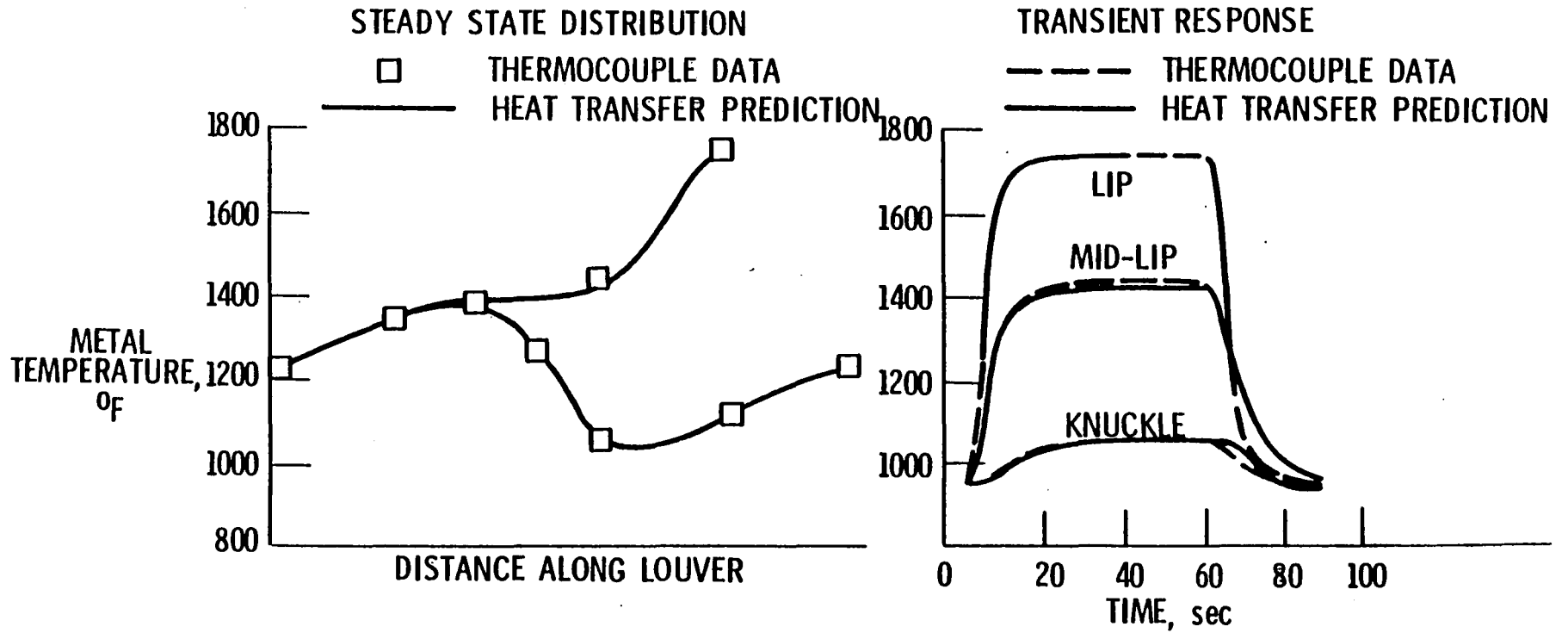
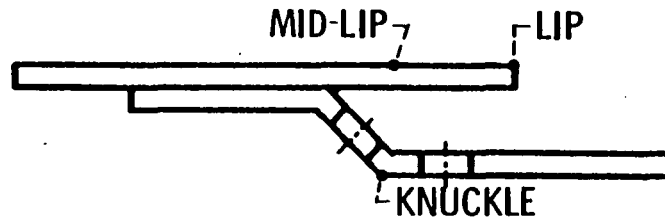
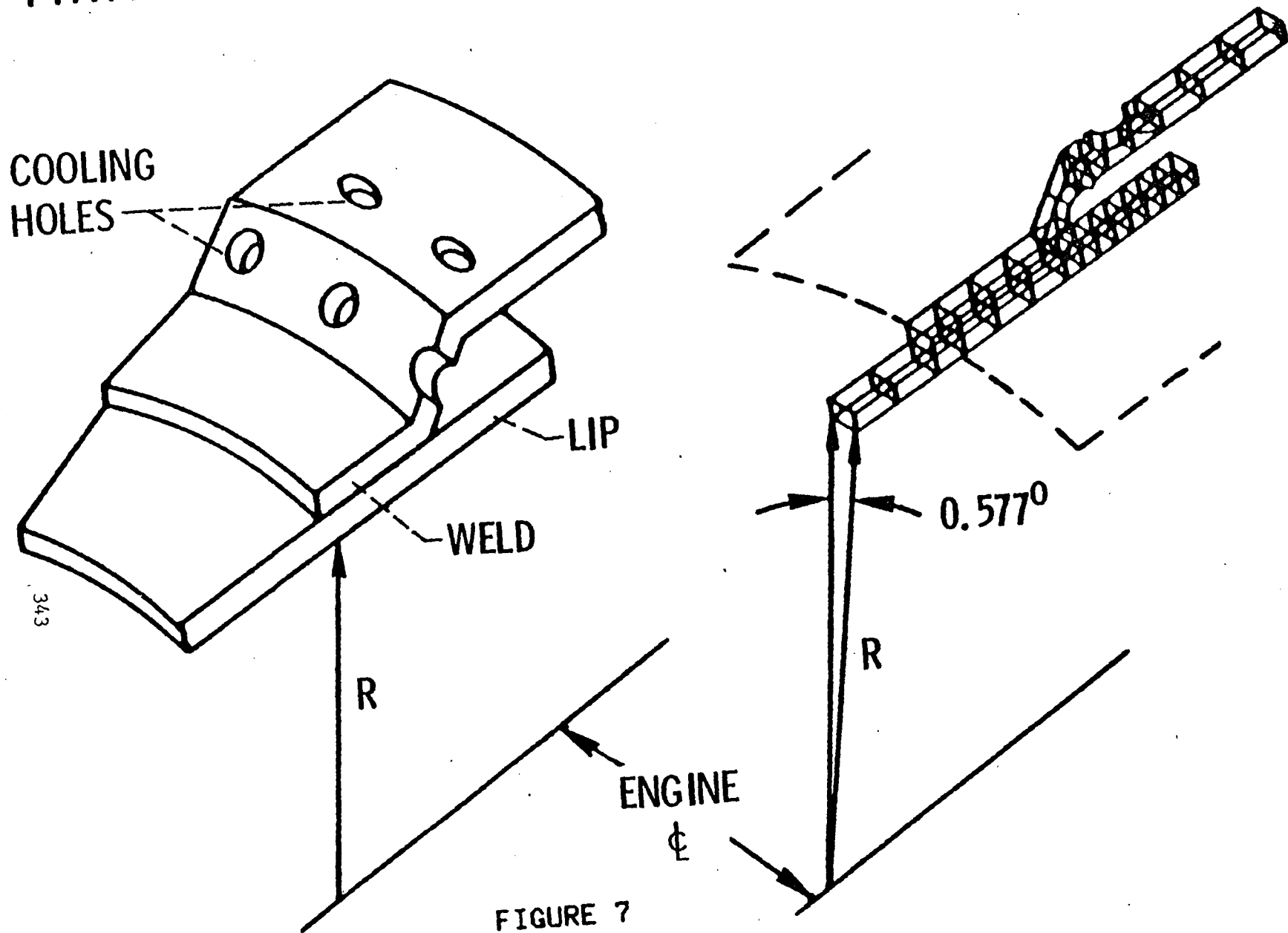


FIGURE 6

FINITE ELEMENT MODEL OF COMBUSTOR SEGMENT



SCHEDULE FOR STRUCTURAL ANALYSIS OF COMPONENTS

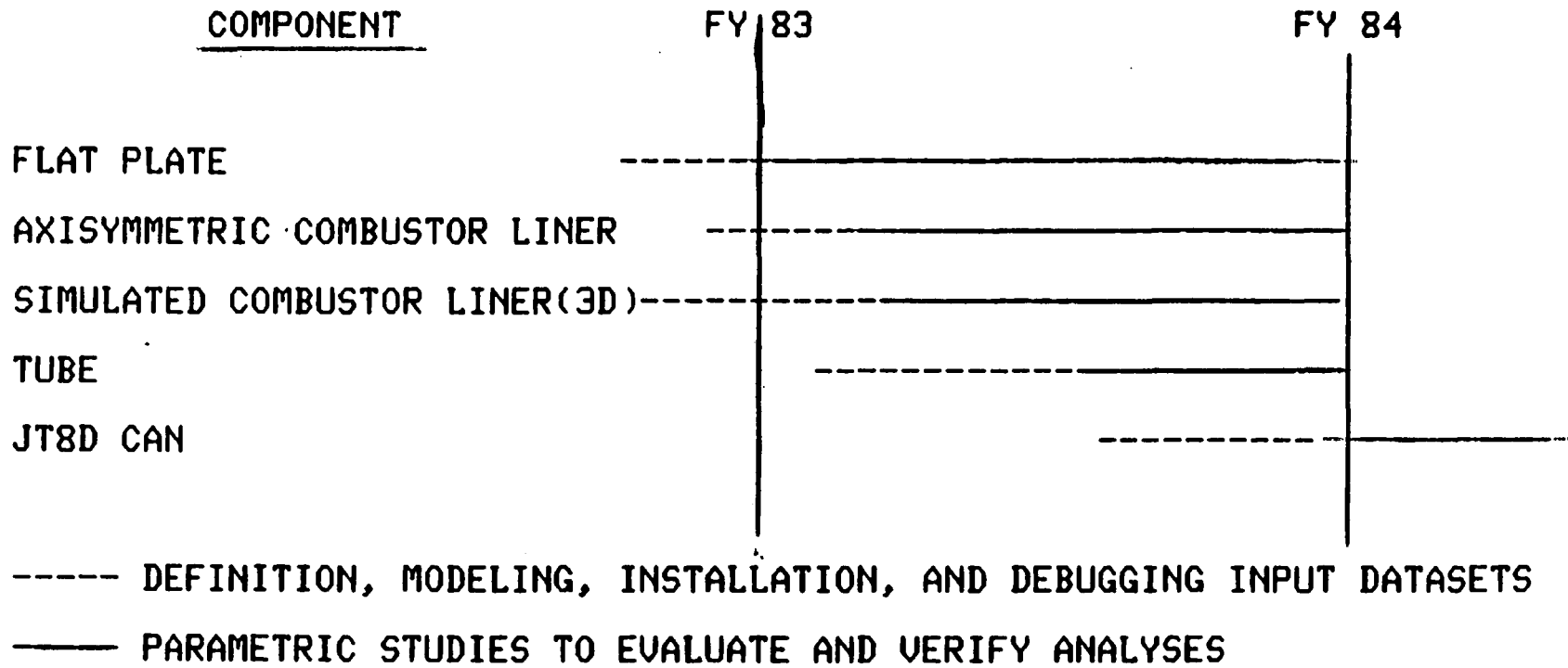


FIGURE 8

HOST Liner Cyclic Facilities

Facility Description

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Three liner cyclic test apparatus are presently planned. The first is a simple 5 x 8 inch rectangular box incorporating four 6KVA quartz lamps for cyclic heating of the test specimen. This apparatus will include a silicon window viewport for IR camera temperature measurement. The quartz lamp box will be used to verify simple liner configurations and to evaluate strain and temperature measuring techniques. This evaluation is scheduled to begin in June 1983.

The second facility (shown schematically in Figure 1), is a twenty inch diameter annular O.D. liner simulator which is a joint NASA Lewis - United Technology Research Center program. One hundred and twelve 6 KVA quartz lamps will be cycled. Power levels will be adjusted to simulate typical liner heat loadings. Air will be supplied to provide typical liner film and backside cooling. Two or three liner designs will be evaluated to compare with the modeling results. The indicated modeling effort will be conducted independently by UTRC and NASA Lewis.

This apparatus in addition to 672 KVA of 480 Volt power, requires 7.5 lb/sec of 1000^oF air at 30 psia, 3.5 lb/sec of ambient temperature air at 5 psig, 1.5 lb/sec of ambient temperature air at 1 psig and 64 GPM of cooling water. This apparatus is scheduled to go under testing in April 1984.

The third facility; the low pressure liner cyclic can rig (shown schematically in figure 2); is being designed to fit the same test leg as the annular quartz lamp rig. It will utilize a JT8D size can combustor operating on Jet-A or possibly propylene fuel. (Propylene being a gaseous fuel of similar percent hydrogen as Jet-A, would significantly reduce fuel injector problems associated with cyclic operation while retaining similar flame radiation characteristics.) The test section of this rig would operate at about 2 1/2 atmospheres absolute pressure. A vitiated preheater will be used to supply 2.4 - 4.8 lbs/sec of 800 - 1000^oF air to the test section. A torch ignitor will be used to minimize ignition problems associated with the cyclic operation. Testing of this apparatus is scheduled for November 1984.

Special Test Instrumentation

Liner cold side temperatures will be measured using an IR-TV camera system. This will permit several hundred temperature measurements to be made in a relatively small area. Liner hot side temperatures will be measured with thin film thermocouples. New technology high temperature strain gauges will be used to obtain local strain measurements.

Preliminary Small Scale Tests

December 1981 saw the initial testing of a three lamp quartz lamp apparatus pictured in figure 3. Limited success was obtained with this rig. A test plate temperature of 2000°F was achieved. Lamp life, however, appeared to be limited for the standard commercial quartz lamps then available. Redesigned lamps will be used in the two new quartz lamp facilities which should overcome the problems of the earlier lamps.

A preheater test is scheduled for December 1982 to design vitiated and non-vitiated preheaters required for the quartz lamp annular rig and the cyclic can rigs.

HOST LINER CYCLIC FACILITIES

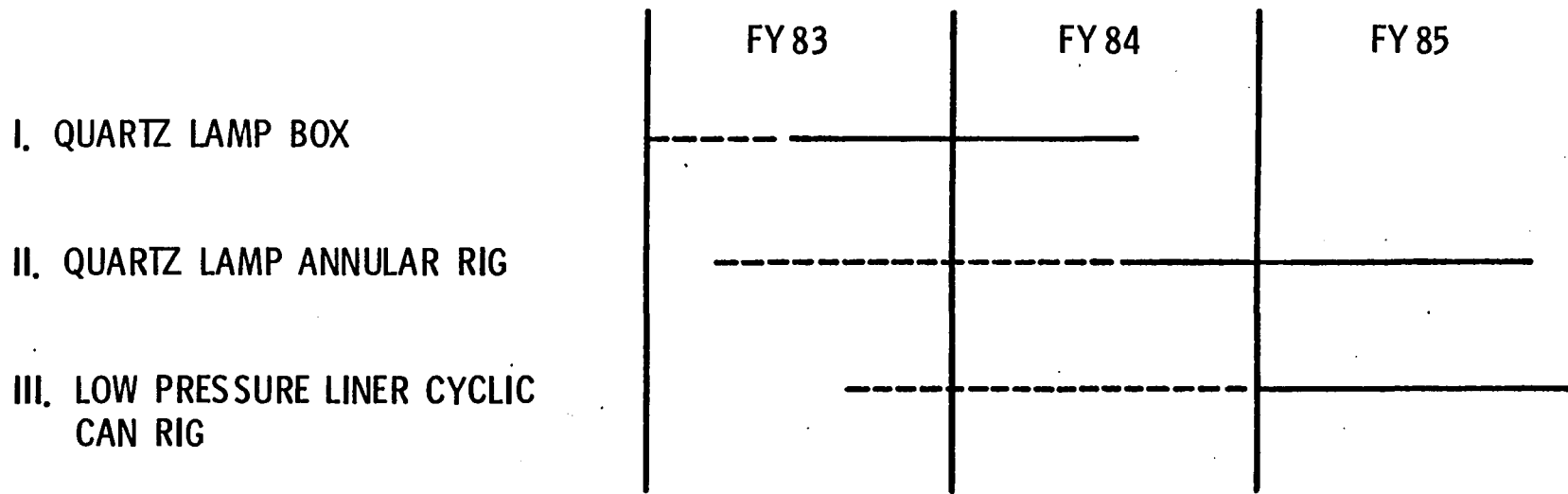
I. QUARTZ LAMP BOX

II. QUARTZ LAMP ANNULAR RIG

III. LOW PRESSURE LINER CYCLIC CAN RIG

HOST CYCLIC LINER PROGRAM

SCHEDULE



----- DESIGN, PROCUREMENT, INSTILLATION
————— TEST

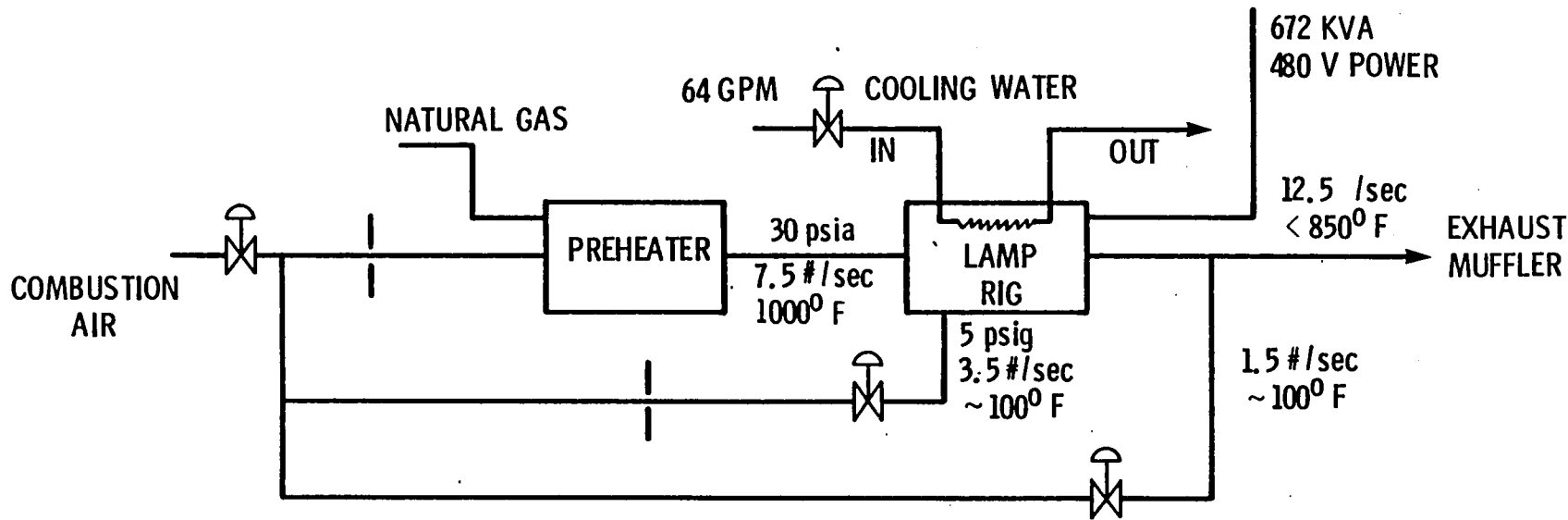
FIRST QUARTZ LAMP BOX RIG



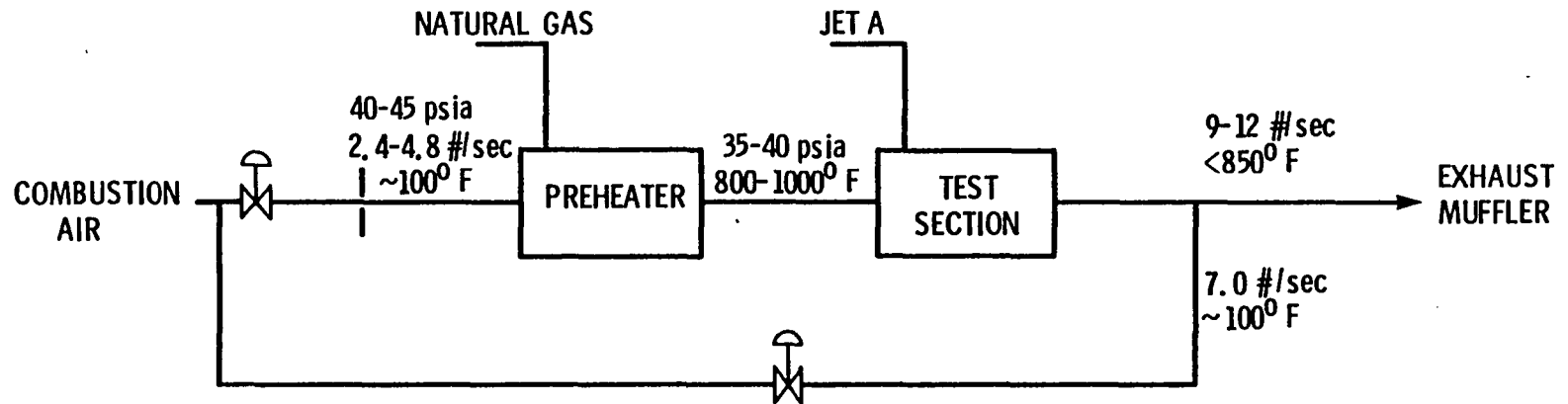
FIRST QUARTZ LAMP BOX RIG

- **OBTAINED 2000° F TEST PLATE TEMPERATURE**
- **LIMITED LAMP LIFE**

HOST QUARTZ LAMP ANNULAR RIG SCHEMATIC-ECRL-1



HOST LOW PRESSURE LINER CYCLIC CAN RIG SCHEMATIC-ECRL-1



SPECIAL INSTRUMENTATION

I. IR - TV MONITORING OF LINER TEMPERATURES

II. THIN-FILM THERMOCOUPLES

III. LASER STRAIN GAUGE

IV. HIGH TEMPERATURE STRAIN GAUGES

FACILITY PREHEATER TEST

OBJECTIVE:

1. DETERMINE IF A NATURAL GAS FIRED VITIATED PREHEATER IS CLEAN ENOUGH FOR USE WITH THE ANNULAR LAMP RIG.
2. EVALUATE PERFORMANCE OF LOW PRESSURE LINER CYCLIC CAN RIG VITIATED PREHEATER.

THE FUTURE OF HOST

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REPORTING

- 0 ANNUAL CONTRACTOR WORKSHOP
- 0 ANNUAL REPORTS
- 0 FINAL REPORTS

RESOURCES

0 FY81: \$2.4M

0 FY82: \$4.0M

0 FY83: \$5.6M

0 FY84-89: \$7.5-\$12M/YEAR (?????)

FUTURE DIRECTIONS

- 0 CONTINUING HOT SECTION DURABILITY RESEARCH
- 0 SELECTED INTERDISCIPLINARY GRANTS
- 0 STRONG NASA LEWIS IN-HOUSE RESEARCH EFFORTS
- 0 COMPATIBILITY WITH DOD
- 0 ANALYSIS METHODS IS THE CORNERSTONE

NEXT YEAR

ANNUAL CONTRACTOR WORKSHOP II

- 0 OCTOBER 25, 26, 27, 1983
- 0 RESULTS, RESULTS, RESULTS
- 0 FORMAL PAPERS/PROCEEDINGS (FEDD)
- 0 WORKSHOP SIDE SESSIONS ???

1. Report No. NASA TM-83022	2. Government Accession No.	3. Recipient's Catalog No.	
4. Title and Subtitle TURBINE ENGINE HOT SECTION TECHNOLOGY (HOST)		5. Report Date October 1982	6. Performing Organization Code 533-04-1A
		8. Performing Organization Report No. E-1458	10. Work Unit No.
7. Author(s)		11. Contract or Grant No.	
9. Performing Organization Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135		13. Type of Report and Period Covered Technical Memorandum	
		14. Sponsoring Agency Code	
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		15. Supplementary Notes	
16. Abstract A two-day workshop on the research and plans for turbine engine hot section durability problems was held on October 19 and 20, 1982, at the NASA Lewis Research Center. Presentations were made during six sessions, including structural analysis, fatigue and fracture, surface protective coatings, combustion, turbine heat transfer, and instrumentation, that dealt with the thermal and fluid environment around liners, blades, and vanes, and with material coatings, constitutive behavior, stress-strain response, and life prediction methods for the three components. The principal objective of each session was to disseminate the research results to date, along with future plans, in each of the six areas. Contract and government researchers presented results of their work. This publication contains extended abstracts and visual material presented during the workshop.			
17. Key Words (Suggested by Author(s)) Turbine Engine Technology (HOST); Life prediction; Combustor liners; Turbine airfoils; Material behavior; Aircraft engines; Durability analysis methods		18. Distribution Statement Foreign distribution excluded. Source of Availability: NASA Industrial Applications Centers STAR Category 39	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 364	22. Price A16

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