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Fiber-Reinforced Ceramic Composites for Earth-to-Orbit Rocket Engine Turbines

Phase I - Final Report

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FOREWORD

The principal objectives of NASA-LeRC contract number NAS3-25468, "Fiber-Reinforced Ceramic Composites for Earth-to-Orbit Rocket Engine Turbines", are to identify the benefits and assess the potential for application of fiber-reinforced ceramic matrix composites (FRCMC) in future generation earth-to-orbit rocket engine turbines. The Rocketdyne Division of Rockwell International recently completed Phase I of this contract with the support of three sub-contractors, Willims International (WI), Societe Europeenne de Propulsion (SEP) and E. I. Du Pont De Nemours & Co., Inc. (DuPont). Phase I, with a start date of Nov. 1, 1988, was a 16-month effort with technical activity completed Feb. 28, 1990. Phase I scope includes identification of the key technical issues with regard to the intended applications of FRCMC, demonstration of critical sub-components, and development of a plan to address the key unresolved technical issues.

The authors express their appreciation to a number of personnel for their contributions to this program. They especially thank Joe Halada and the many personnel at WI as well as those at SEP who have enthusiastically supported a broad range of the program tasks. Within Rocketdyne, they thank Jim Tellier (Turbomachinery), Linsey Orr (Stress), Gary Tuttle (Design) and Al Martinez (Systems Engineering) as well as the other team members involved. Finally, a special thank you to Dr. T. P. Herbell and other key personnel at the NASA Lewis Research Center (NASA-LeRC) who have provided invaluable advice and support.

ABSTRACT

FRCMC are emerging materials systems that offer considerable potential for improvement of liquid rocket engines. Potential advantages of these materials in rocket engine turbomachinery include: higher performance due to higher turbine inlet temperature (TIT), reduced launch costs, reduced maintenance with associated cost benefits and reduced weight. FRCMC materials have not been fully characterized for rocket engine use; consequently, this program was initiated to assess the state of their development and to propose a plan for their implementation into liquid rocket engine turbomachinery.

A complete range of FRCMC materials was investigated relative to their development status and feasibility for use in the hot gas path of earth-to-orbit rocket engine turbomachinery. Of the candidate systems, carbon fiber-reinforced silicon carbide (C/SiC) was determined to offer the greatest near-term potential. Detailed evaluations were made of the feasibility and advantages for use of C/SiC in advanced earth-to-orbit turbomachinery. Critical hot gas path components were identified, and the first stage inlet nozzle and turbine rotor of the fuel turbopump for the liquid oxygen/hydrogen (LOX/H_2) propelled Space Transportation Main Engine (STME) were selected for conceptual design and analysis.

Substantial performance increases can be achieved by using FRCMC turbine components which allow higher turbine inlet temperatures without component cooling and resultant performance and flow losses. The critical issues associated with the use of FRCMC for these applications were identified, and sub-components (turbine blades) were designed, analyzed and fabricated illustrating FRCMC fabrication features.

The assessment of FRCMC status when compared to liquid rocket engine requirements resulted in the determination of key unresolved technical issues. The "Technology Development Plan" which was completed previously as Task V of this program provides a course of action for resolution of these issues and is included within this final report.

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FIGURES

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* - Novoltex is a registered trademark of SEP

ACRONYMS AND ABBREVIATIONS

FRCMC	Fiber-Reinforced Ceramic Matrix Composite
WI	Williams International
SEP	Societe Europeenne de Propulsion
DuPont	E. I. Du Pont De Nemours & Company (Inc.)
NASA-LeRC	National Aeronautics and Space Administration Lewis Research Center
TIT	Turbine Inlet Temperature
C/SiC	Carbon Fiber-Reinforced Silicon Carbide
LOX/H_2	Liquid Oxygen/Hydrogen
STME	Space Transportation Main Engine
I_s	Specific Impulse (lbs-f/lbs-m•sec)
SSME	Space Shuttle Main Engine
LOX/CH_4	Liquid Oxygen/Liquid Methane
STBE	Space Transportation Booster Engine
C/C	Carbon Fiber-Reinforced Carbon
FRS	Fiber-Reinforced Superalloys
RSR	Rapid Solidification Rate
ODS	Oxide Dispersion Strengthened
SiC/Ni_3Al	Silicon Carbide Fiber-Reinforced Nickel Aluminide
DS	Directionally Solidified
SC	Single Crystal
HEE	Hydrogen Environment Embrittlement
SAGBO	Stress Assisted Grain Boundary Oxidation
CVD	Chemical Vapor Deposition
Al_2O_3	Aluminum Oxide
Si_3N_4	Silicon Nitride
SiC/SiC	Silicon Carbide Fiber-Reinforced Silicon Carbide
SiC/Si_3N_4	Silicon Carbide Fiber-Reinforced Silicon Nitride
CVI	Chemical Vapor Infiltration
ALS	Advanced Launch System
Mk29F	Mark 29 Fuel Turbopump
ORNL	Oak Ridge National Laboratories
CAT	Computer-Aided Tomography
PAN	Polyacrylonitrile
SiO	Silicon Monoxide
SiO_2	Silicon Dioxide
C/HfC	Carbon Fiber-Reinforced Hafnium Carbide
ZrO_2	Zirconium Oxide
NDE	Non-Destructive Evaluation
HfC	Hafnium Carbide
NASA-MSFC	NASA George C. Marshall Space Flight Center

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SUMMARY

OBJECTIVES

The objectives of this contract are to identify the benefits and assess the potential for application of FRCMC in advanced earth-to-orbit liquid rocket engine turbines. Within Phase I several key issues were addressed with the objectives of assessing the state of development of FRCMC and determining their feasibility of application. Phase I tasks were planned to: select components for design, provide a design methodology for FRCMC, select preferred materials based on current technology, design and analyze components for selected engine systems, and demonstrate existing capabilities for the fabrication of sub-components. Principal program outputs have included sub-components which have been analyzed and a "Technology Development Plan", which identifies the effort required to resolve the key technical issues identified in this contract.

CONCLUSIONS

This contract effort has determined that FRCMC technology can provide significant performance benefits when used for rocket engine turbomachinery and that current technology is capable of demonstrating selected, critical components. Specifically, turbine rotors and nozzles are well-suited to FRCMC manufacturing capabilities and offer considerable potential performance benefit. FRCMC technology has advanced rapidly in recent years as demonstrated by successful FRCMC thruster tests, by manufacture of complex blisks and inlet nozzles, and by improved materials properties. Of the existing FRCMC systems, multifilament fiber-reinforced materials, especially C/SiC, have the best combination of properties (App. A) and near term producibility for the timely production of advanced liquid rocket engine components. Further improvement of existing FRCMC systems, development of new systems, and refinement of design and analysis methods are continuing.

Phase I of this program quantitatively assessed the potential benefits of FRCMC application to earth-to-orbit rocket engines (App. B). The status of FRCMC development was evaluated relative to rocket propulsion needs, and key technical issues were identified. Sub-components were fabricated which demonstrate critical design features and manufacturing feasibility. A plan was developed which provides a course of action for resolution of the remaining technical issues identified during the program.

Benefits from FRCMC use in liquid rocket engines are expected in several areas:

1. Increases in turbine inlet temperature (TIT) will result in significantly higher efficiency for gas generator engines. For example, an increase in TIT from a baseline of 870°C (1600°F) to an FRCMC compatible temperature of 1200°C (2200°F) with LOX/H_2 propellants results in a specific impulse (I_s) gain of over 5 sec. This performance improvement will significantly reduce the costs of transporting payloads into space and can be realized by: increasing payload for a specified vehicle configuration, reconfiguring the vehicle to reduce vehicle weight, or reconfiguring the engine to reduce nozzle exit area and engine weight.
2. Improvements in operating margin and component life will result in significant reductions in maintenance and refurbishment costs for reusable engines. For the staged combustion, Space Shuttle Main Engine (SSME) this benefit was shown to be

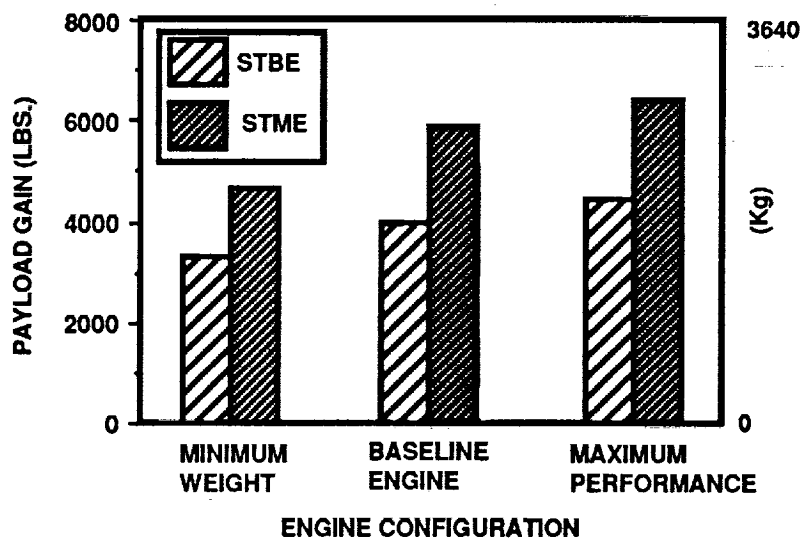
dramatic (Ref. 1), and analogous benefits would be expected for other engine cycles. The ability to achieve these improvements in component life by application of FRCMC requires verification by additional testing.

3. The high specific strength of FRCMC reduces component weight. This advantage was not quantified.

Advantages of increased TIT were quantified for various engine cycles and propellant combinations. Previous studies (Ref. 2) had shown that increasing TIT for staged combustion cycle, LOX/H_2 propellant, earth-to-orbit engines (e.g. SSME) would result in limited efficiency gains. Expander cycle engines do not require high TIT operation; therefore, benefits were not evaluated for this cycle. Parametric engine balances were run for gas generator cycle, earth-to-orbit engine configurations using both liquid oxygen/methane (LOX/CH_4) and LOX/H_2 propellants. These engine types are under consideration for advanced earth-to-orbit engines, including variants of the Space Transportation Booster Engine (STBE) and the STME. Significant gains in I_s were achievable with either propellant by increasing TIT from a baseline value of 870°C (1600°F) to an FRCMC allowable use limit of 1200°C (2200°F). These I_s gains resulted in potential payload gains (Fig. 1) with significant potential cost benefits.

FIG. 1

INCREASED TIT RESULTS IN HIGHER PAYLOADS



For each engine (STBE and STME) the payload gain was determined for the baseline configuration, for a minimum weight configuration and for a maximum performance configuration. The payload gains achieved with higher performance were significantly greater than engine weight savings. Consequently, greater emphasis was placed on the higher performance variants in order to gain the greatest payback.

Dramatic cost advantages were previously shown to be achievable through gains in component life for SSME-type engine turbine components (Ref. 1), and analogous benefits would be expected for other reusable engine systems. Life related properties data for existing FRCMC systems are insufficient to quantify these benefits, and additional data must be developed.

Reductions in individual component weights are specific to the detailed design and operating parameters for that component. These benefits were not quantified.

Achieving the benefits of FRCMC application will require a series of developments. The assessment of FRCMC status relative to rocket engine needs determined which developments are needed. These are specified within this plan as well as a course of action for their resolution.

ROCKETDYNE DEVELOPMENT PLAN

The development status assessment required identification of materials limitations with respect to liquid rocket engine needs. Based on these limitations, critical issues were defined which are addressed herein. It was determined that an integrated plan is needed to evaluate critical design, analysis, producibility and performance issues. The integrated plan will ultimately demonstrate FRCMC feasibility by full-up turbopump testing.

Phase II of the NASA FRCMC development program is the first step in the integrated test plan which will demonstrate a high temperature turbine. Phase II is a 44-month, $2M program which will design, fabricate and test a full-scale component in a simulated rocket engine environment to demonstrate operational capability under selected test conditions. The long term plan integrates the Phase II component with other critical FRCMC components into an operational turbopump to verify FRCMC feasibility and capabilities.

INTRODUCTION

OBJECTIVES

The overall objectives of this program are to identify the benefits and assess the potential for application of FRCMC in advanced earth-to-orbit liquid rocket engine turbomachinery.

PHASE I APPROACH

The technical plan for Phase I was structured into five technical tasks:

Task I: Environmental and Structural Requirements
Task II: Material Selection
Task III: Preliminary Design/Benefits Analysis
Task IV: Prototype Component Fabrication
Task V: Technology Development Plan

The approach taken to achieve the task requirements is shown in Fig. 2.

To assess critical material requirements and to choose materials and components for further study, an initial benefits analysis was performed (Task III) and was refined by iteration of Tasks I through III prior to component fabrication (Task IV). The inputs of Tasks I through IV were combined to assess the FRCMC development status which was compared to liquid rocket engine needs in order to identify the key technical issues which are addressed in the plan.

FIG. 2
FLOW DIAGRAM

TASK I (3.1.1)

Define Critical Material Requirements for Selected Components

TASK II (3.1.2)

(3.1.2.1) Choose Fiber/Matrix Systems*
(3.1.2.2) Update FRCMC Data Base*
(3.1.2.3) Recommend Component/Materials for Preliminary Design/Benefit Analysis*

TASK III (3.1.3)

(3.1.3.1) Develop Conceptual Designs*
(3.1.3.2) Assess Producibility and Quantify Benefits
(3.1.3.3) Recommend FRCMC System and Fabrication Methods; Identify Other Necessary Components; Select Engine/Propellant/ Components for Study

TASK IV (3.1.4)

(3.1.4.1) Select Supplier for Specified Component*
(3.1.4.2) Evaluate Component by Approved* Procedures
(3.1.4.3) Deliver Prototypes to NASA-LeRC

TASK V (3.1.5)

Formulate Technology Development Plan

() Contract SOW Numbers
* NASA-LeRC Approvals

To reach the objectives most efficiently, a team approach was taken. WI had been identified as a leader in the evaluation of FRCMC for air-breathing turbine applications and has participated in the areas of materials selection, design and analysis. SEP is the leading producer of FRCMC for rocket applications; and, through their U.S. licensee, DuPont, SEP provided materials for Phase I sub-component evaluations.

Rocketdyne's in-house materials properties data base was supplemented by both WI and SEP materials properties data. The materials systems evaluated were limited to those with current experience levels and near term expectations compatible with the program timing. Engine systems were limited to those which were currently under consideration and for which Rocketdyne had a baseline configuration and data base.

STATUS

An overview of the selections made in Tasks I through III is given in Tbl. 1. These selections were made based on consideration of a matrix of engine types and cycles, propellant combinations, components and materials systems (Tbl. 2). For quantitative assessments, the materials properties as detailed in a following section were used to provide operating limits for FRCMC components.

TBL. 1
FRCMC CONTRACT APPROVALS

TASK	ROCKETDYNE RECOMMENDATIONS
I. ENVIRONMENTAL AND STRUCTURAL REQUIREMENTS	COMPONENTS: First stage nozzle and rotor for STME PROPELLANTS: LOX/H_2 OPERATING TEMPERATURE: 1200°C (2200°F) STRESSES: Preliminary values consistent with C/SiC disk and blade stress limits TRANSIENTS: Comparable to SSME
II. MATERIAL SELECTION	PRIME CANDIDATE: C/SiC, fine filament structure, polar woven for both components ALTERNATE CANDIDATE: C/SiC, pseudo-isotropic lay-up for both components DATA BASE: Per attached
III. PRELIMINARY DESIGN/BENEFIT ANALYSIS	BENEFITS: Preliminary analyses of STBE (LOX/CH_4), STME (LOX/H_2), and SSME show principal advantages of elevated temperature for gas generator cycle (STME or STBE); payload weight trades indicate maximum advantage with maximum performance versus minimum weight configuration FABRICATION: Matrix infiltration by CVD/CVI of fiber pre-form

TBL. 2
ENGINE/COMPONENT/MATERIALS SELECTION MATRIX

ENGINE CYCLE	PROPELLANT COMBINATION	SPECIFIC ENGINE	COMPONENT	FIBER/MATRIX SYSTEM
GAS GENERATOR	***LOX/H_2***	***STME***	*	***C/SiC*** **
GAS GENERATOR	LOX/CH_4	STBE	*	**
STAGED COMBUSTION	LOX/H_2	SSME	*	**
STAGED COMBUSTION	LOX/CH_4 ***	SSME	*	**
EXPANDER CYCLE	N/A	N/A	N/A	N/A

ITALICS with underlining indicates candidate selections
* - Both rotating (rotor) and non-rotating (nozzle) components considered
** - C/SiC, SiC/SiC, SiC/Si_3N_4, C/C considered
*** - Benefits of LOX/CH_4 evaluated but CH_4 not actually used for SSME

C/SiC was selected as the principal materials system due to its combination of high strength at elevated temperature, good environmental resistance and relatively high level of development. Although polar woven architectures are desirable for selected components, the technology lags conventional 2D process technology. 2D layup properties meet program needs, are more highly developed, and are better suited to sub-component demonstration, particularly in the near term. Both the first-stage rotor and nozzle of the STME demonstrate many of the critical performance and production characteristics identified in Phase I, and both are considered for component demonstration in following stages of the plan.

The plans presented later in this report provide a step-wise progression, first through simulated rocket engine testing of a full-scale component (Phase II) and continuing through full-up demonstration of an advanced, high temperature turbopump.

CERAMIC MATRIX COMPOSITE BENEFITS
TO
EARTH-TO-ORBIT ROCKET ENGINE TURBINES

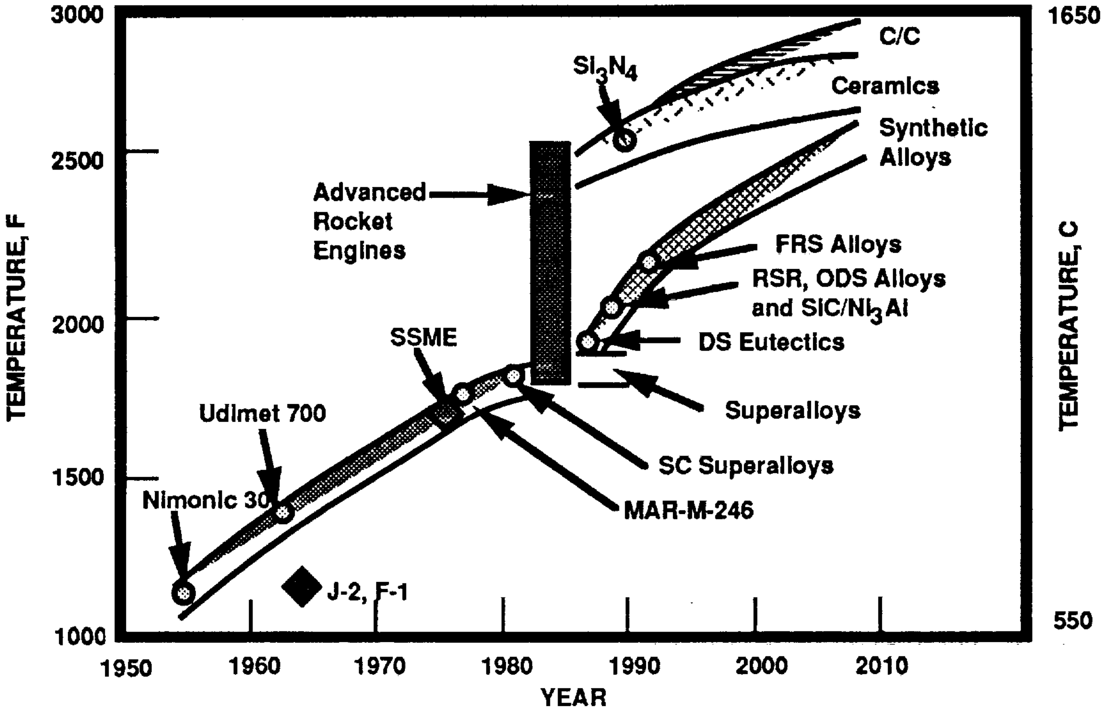
LIMITATIONS OF CONVENTIONAL MATERIALS

Conventional materials used in liquid rocket engine components include a broad array of highly engineered materials that continue to be refined and improved. Relative to potential application of FRCMC, hot gas path components are principally of interest. The primary materials used in the hot gas path of existing engines are superalloys. These alloys are gaining improved capabilities and reliability by alloy improvements, fiber reinforcement, and single crystal alloy development. Such improvements are limited in the gains achievable in temperature capability and environmental resistance and can do little relative to weight reduction due to their high density. As a result, new systems, such as FRCMC, need development to achieve dramatic gains.

Temperature

The maximum operating temperature of conventional superalloys is limited to about 870°C (1600°F) due to the rapid decrease in strength above this temperature (Ref. 3). Use of fiber reinforcement technology, recent developments in single crystal technology, and other superalloy improvements are expected to increase this by about 100°C (200°F). For further increases which, in turn, will result in significant efficiency gains, other materials must be developed (Fig. 3). Carbon/carbon (C/C) composites are usable at high temperatures but are subject to environmental attack unless protected by coatings, which have not yet been reliably demonstrated for sustained operating times and/or for repeated cycling in the rocket engine environment. Thus, ceramic matrix composites, which are environmentally stable and maintain mechanical properties to temperatures above 1100°C (2000°F), are of interest. (The ceramic matrix only partially protects the carbon fiber reinforcement, and additional coatings may be needed for long-term use, especially at temperatures which do not allow full closure of matrix microcracks, as discussed in following sections.)

FIG. 3
ROCKET ENGINE TURBINE BLADE MATERIAL
NEEDS AND CAPABILITIES



Superalloys are further limited by their reduced fatigue lives as operating stresses approach the high temperature yield stress. Both low cycle (e.g. from thermal transients or mission cycles) and high cycle fatigue must be considered. Added consideration must also be given to creep and stress rupture for long-life applications as the maximum use temperature is approached. In cases requiring use of conventional materials near their maximum operating temperature, useful operating life becomes severely limited, which results in significant inspection and maintenance needs for reusable engine components.

Atmosphere

LOX/H_2 propellant rocket engine operating environments can be highly degradative to superalloys. Hydrogen-rich environments are known to embrittle a wide range of metallic systems by various alloy specific and environment specific mechanisms often referred to as hydrogen environment embrittlement (HEE). This degradation severely restricts the range of applicable alloys and further restricts the life of certain of the alloys that are used.

Degradation is not limited to hydrogen attack or to attack of metallic systems. Oxidative conditions can occur with subsequent degradation, which in some cases can be catastrophic. Stress assisted grain boundary oxidation (SAGBO) has been observed in metallic systems. C/C, as noted above, is subject to oxidation with rapid weight loss and properties degradation at high temperatures in the combustion environment unless protected by coatings.

FRCMC, such as C/SiC, are less susceptible to degradation by the combustion environment. The SiC matrix microcracks upon cooling when processed, but these cracks close in use at temperatures near the processing temperature. Consequently, at high temperatures, the SiC matrix protects the embedded carbon fibers. At lower temperatures, oxidation of the carbon fibers can occur and should be guarded against with protective coatings. Also, care must be taken to avoid exposing fibers directly to the environment either by machining of or damage to the surface. The SiC matrix is not readily attacked by either hydrogen-rich or oxidizing environments at 1200°C (2200°F). NASA-LeRC thermodynamic studies under hydrogen-rich steam conditions show SiC to be relatively unstable but protected by surface oxides at these temperatures (Ref. 4). Experimental studies of SiC when exposed to a hydrogen environment indicated that impurities accumulated at grain boundaries are especially vulnerable to degradation. Lower purity materials showed some degradation at high temperatures, but this is not expected to be a significant problem for higher purity chemical vapor deposited (CVD) SiC matrix at 1200°C (2200°F). NASA thermodynamic studies indicated various other oxide matrices, including alumina (Al_2O_3), were relatively stable, but these systems lack the required thermal shock resistance and are not well characterized or highly developed as composite matrices.

Density and Strength-to-Weight Ratio

The need for materials with high strength and strength-to-weight ratio at high temperatures has proven to be a significant limitation for advanced rocket engines (Ref. 5). Strength-to-weight ratio is especially important for rotating components because this ratio effectively limits the tip speeds achievable. Superalloys are especially limited. Fiber reinforcement improves the maximum temperature capability, but the reinforcement fibers are high density which mitigates the advantages to the strength-to-weight ratio. Advanced carbon/carbon (C/C) offers long-term potential, but the very high, anticipated strength-to-weight values for C/C are for selected systems not representative of typical currently available C/C. Present generation C/C composites are generally lower in strength than these advanced C/C composites. Also, the need for protective coatings limits C/C's near term utility, again emphasizing the need for FRCMC development.

SUMMARY OF FRCMC PROPERTIES

A broad range of FRCMC materials systems was initially considered including glass matrix and oxide matrix composites (Tbl. 3). Oxide matrix materials were generally considered incapable of enduring the severe thermal shock transients, and most were inadequately developed for consideration at this time. Glass matrix materials were unacceptable relative to minimum operating temperature requirements (1200°C (2200°F)). The non-oxide matrix systems combined preferred properties. Of these systems, SiC matrix and silicon nitride (Si_3N_4) matrix materials offered the best combination of mechanical properties, environmental resistance, thermal shock resistance and fabrication experience. In order to fabricate complex, near-net shapes, and to maintain acceptable off-axis and out-of-plane properties, it was necessary to use multifilament fiber-reinforced materials rather than monofilament, uniaxially reinforced materials. These restrictions limited the field to C/SiC, silicon carbide fiber-reinforced silicon carbide (SiC/SiC), and silicon carbide fiber-reinforced silicon nitride (SiC/Si_3N_4).

TBL. 3
SUMMARY OF FRCMC MATERIALS SYSTEMS

FIBER/MATRIX SYSTEMS EVALUATED:

SiC/Lithium-alumino-silicate
SiC/Magnesium-alumino-silicate
SiC/Calcium-alumino-silicate
SiC/Black glass
SiC/Borosilicate
SiC/Silica
C/Lithium-alumino-silicate
C/Borosilicate
C/Silica
C/Alumina
SiC/Silicon nitride
SiC/SiC
C/SiC
C/C

SCREENING CRITERIA USED:

Maximum operating temperature

Thermal shock resistance

Environmental resistance

Ultimate tensile strength

Fracture toughness

Fabricability

Maturity

CANDIDATE SYSTEMS SELECTED:

FIBER/MATRIX	TEMP.	THER. SHOCK	ENVIRO. RESIST.	MECH. PROPS.	MATURITY
SiC/Silicon nitride	1	1	1	4	3
SiC/SiC	1	1	1	2	2
C/SiC	1	1	2	1	1
C/C	1	1	2*	1-2	1

1-ACCEPTABLE 2-MARGINAL 3-NOT ACCEPTABLE 4-NOT KNOWN
*-COATING NECESSARY

The SiC/Si_3N_4 system appears promising for the future, but the development status of this system was considered too immature for timely inclusion within this program. Properties of the remaining candidates, C/SiC and SiC/SiC, were evaluated in detail and are compared to a typical current generation C/C in Tbl. 4. The properties shown are for materials produced by the chemical vapor infiltration (CVI) of a continuous fiber pre-form followed by a final CVD of the matrix which ensures protection of the fibers from the environment.

TBL. 4
SUMMARY OF FRCMC PROPERTIES

PROPERTY	MATERIAL		
	C/SIC	SIC/SIC	C/C
Density [g/cm3 (lb/in3)]	2.2(0.08)	2.50 (0.09)	1.94 (0.07)
UTS [MPa (ksi)] at 20°C (70°F)	317 (46) *	193 (28) **	150 (22) *
Strength/density [x10^5 cm(in)] at 20°C (70°F)	14.4(6.6) *	7.9 (3.1) **	8.0 (3.14) *
UTS [MPa (ksi)] at 1200°C (2200°F)	427 (62) *	214 (31) **	150 (22) *
Str./den. [x10^5 cm (in)] at 1200°C (2200°F)	19.4(7.75) *	8.6 (3.4) **	8.0 (3.14) *
Maximum Operating Temperature [°C (°F)]	1650 (3000)	1430 (2600)	>2200 (>4000)
Young's Modulus [GPa (msi)]	90 (13)	230 (33)	83 (12)
Tensile Elongation [%]	0.9	0.3	N/A
Fracture Tough. [MPa*$m^{0.5}$(ksi*$in^{0.5}$)]	35 (32)	30 (27)	N/A

* - 2D, in-plane
** - cross-ply, in-plane

The properties shown were used in Rocketdyne empirical formulas to determine turbine operating limits which were used in the benefits analyses discussed in the next section. These limits were subsequently verified by comparison to design allowable properties which had been determined by SEP based on their experience. For this program, it was found that the higher mechanical strength of C/SiC was needed to achieve performance targets for the highly stressed rotating components. SiC/SiC could still be considered for lower stress applications.

PERFORMANCE VS. TURBINE INLET TEMPERATURE

Benefits of FRCMC use were possible in several areas as discussed above. Improvements due to increased TIT can be readily quantified and were a natural consideration based on the high temperature capabilities of ceramics. Analyses were run for both LOX/H_2 and LOX/CH_4 propellant combinations with gas generator cycle Advanced Launch System (ALS) engines. These analyses are summarized here and discussed in detail in Appendix B. For staged combustion cycle, earth-to-orbit engines (e.g. SSME), analyses had been run previously (Ref. 2). Expander cycle engines do not use high TIT, and benefits for this cycle were not analyzed.

The staged combustion cycle, SSME analyses showed limited I_s gain with increased TIT for LOX/H_2 propellants. Greater incremental gains were possible with LOX/CH_4. The principal advantage to SSME was found to be for increased component life rather than higher performance, although performance (I_s) gains of approximately 2.5 sec. and 0.5 sec. were possible for LOX/CH_4 and LOX/H_2 respectively (Ref. 2).

For gas generator cycle, ALS engines, considerable gains in I_s were achievable by increasing TIT with either propellant combination (Tbl. 5). These gains were used to estimate payload gains (Fig. 1) for a specific configuration. Also, several engine variations (minimum weight engine configuration, baseline configuration, and maximum performance configuration) were evaluated. The payload gains resulting from I_s gains with either the baseline or the maximum performance versions far exceeded the weight savings possible with the minimum weight configuration. Consequently, the higher performance variations were considered to offer a better payback.

TBL. 5
PERFORMANCE CHANGES WITH INCREASED TURBINE INLET TEMPERATURE

PERFORMANCE INCREASES AT VACUUM FOR TIT INCREASE FROM 620-1400°C (1140-2540°F)		
ENGINE CONFIGURATION	**LOX/CH_4 STBE**	**LOX/H_2 STME**
Maximum Performance Engine I_s Gain, sec	10.3	6.6
Baseline Engine I_s Gain, sec	10.3	5.2
Minimum Weight Engine I_s Gain, sec	6.5	3.7

SEA LEVELPERFORMANCE INCREASES FOR TIT INCREASE FROM 620-1400°C (1140-2540°F)		
ENGINE CONFIGURATION	**LOX/CH_4 STBE**	**LOX/H_2 STME**
Maximum Performance Engine I_s Gain, sec	14.3	12.0
Baseline Engine I_s Gain, sec	9.7	4.8
Minimum Weight Engine I_s Gain, sec	6.7	3.3

Although LOX/CH_4 gave a higher incremental gain than LOX/H_2, the LOX/H_2 had greater overall performance capability and is the propellant currently under consideration for most STBE and STME engine configurations. Consequently, the LOX/H_2 environment was considered in following studies.

LIFE VS. OPERATING MARGIN

The properties data needed to design and analyze a turbine disk, or bladed disk (blisk), are summarized in Tbl. 6. In addition to fast fracture properties, delayed failure and reliability criteria are needed to estimate component life. Delayed failure data for FRCMC are limited in quantity and scope. SEP has performed a limited number of fatigue tests on their C/SiC and SiC/SiC materials. Tests were run to 1 million cycles at stresses to 60% of ultimate without failure and with retained strength after cycling equal to or greater than uncycled strength. This indicates that damage, if any, due to cycling was limited. This test does not simultaneously assess environmental degradation or low cycle, thermal stress effects. Tests have also been run to 10 million cycles at stresses to 30% of ultimate with similar results. Life tests on exit cones show very good environmental durability. Although encouraging, these data are insufficient for quantitative analysis. High cycle fatigue data are needed to 1 billion cycles, if life prediction models are not available, and simultaneous consideration must be given to environmental and low cycle fatigue effects.

TBL. 6
FRCMC MATERIAL CHARACTERIZATION
BLISK PROPERTIES DATA NEEDS

MECHANICAL PROPERTIES	
FAST FRACTURE	**DELAYED FAILURE**
ULTIMATE TENSILE STRENGTH	FATIGUE
Young's Modulus	High Cycle
Poisson's Ratio	Low Cycle
Strain-to-Failure	STRESS RUPTURE
Elastic Limit	CREEP
MODULUS OF RUPTURE	RELIABILITY
INTERLAMINAR SHEAR STRENGTH	
FRACTURE TOUGHNESS	

THERMAL/ENVIRONMENTAL PROPERTIES	
THERMAL	**ENVIRONMENTAL**
THERMAL CONDUCTIVITY	REACTIVITY (BY DIFFERENTIAL THERMAL ANALYSIS)
THERMAL EXPANSION COEFFICIENT	WEIGHT CHANGE (BY THERMOGRAVIMETRIC ANALYSIS)
SPECIFIC HEAT	OXIDATION STABILITY

COST ADVANTAGES

Areas of potential cost benefit include launch costs and maintenance costs. FRCMC component costs are currently higher than conventional counterparts, but higher production rates and improved processing are expected to reduce this differential in the future.

Launch Costs

Considering payload gains (Fig. 1) without consideration of other factors (e.g. vehicle reconfiguration) allows for a simplified estimate of launch cost advantages.

Current SSME launch costs are estimated between \$4.4K/Kg and \$8.8K/Kg (\$2K/lb and \$4K/lb). Targets for advanced, low cost launch systems have not been reached but are as low as \$660/Kg (\$300/lb). For a 2700Kg (6000 lb) payload increase, this results in a gain of from \$1.8M per launch based on low cost launch system targets to as high as \$12M to \$24M per launch based on current shuttle launch costs.

Maintenance Costs

Maintenance and refurbishment costs which are the result of the limited life of existing turbine blades (Ref. 1) are high. Improvements in component lifetimes would dramatically reduce these costs for reusable engines. The limited life data available for FRCMC, as discussed previously, preclude quantitative life estimates of FRCMC components. The potential for increased life remains encouraging but demands verification.

WEIGHT ADVANTAGES

Weight reduction can be realized by direct reductions in component weight or by reconfiguration to take advantage of higher performance. FRCMC densities are typically about one-third the densities of superalloys, and the elevated temperature strength-to-weight ratio of FRCMC is also much better than superalloys. Consequently, direct substitution of components would reduce weight with a resultant reduction in centrifugal loading of rotating components.

ADVANTAGES FOR SPECIFIC COMPONENTS

A review of the principal components in a turbopump (Tbl. 7) provides an overview of the potential advantages of FRCMC for specific components. For illustration purposes, a small gas generator cycle engine was reviewed; however, the recommendations would generally apply to other turbopumps. Based on this review, most turbine components could benefit in some manner from FRCMC application. Titanium inducers, impellers and spacers showed no apparent benefit because of the high specific strength of titanium alloys at cryogenic temperatures. The most realistic near-term opportunities for FRCMC are in the hot gas path which would especially benefit from FRCMC substitutions for the rotor and nozzle. With further development to reduce permeability, FRCMC turbine manifolds and tip seals would also be advantageous and are needed for the demonstration of a high temperature, flightweight turbopump.

TBL 7
CONVENTIONAL TURBOMACHINERY COMPONENTS

COMPONENT	CONVENTIONAL MATERIAL	COMPONENT	CONVENTIONAL MATERIAL
Turbine Rotor	Astroloy	Pump Side B'ring Sp.	410 CRES
Turbine Manifold	Haynes 188	Impeller	Inco 718
Nozzle	Haynes 188	Volute/Housing	Inco 718
Stator	Haynes 188	Slinger	Inco 718
Turbine Tip Seals	Inconel-X Honeycomb	Laby Ring	Inco 718
Turbine Seal Ring	P5N Carbon	Inducer	Titanium
Turbine Seal H's'ng	321 CRES	Backflow Deflector	Hastalloy B
Turbine Ht. Shield	Inconel	Inlet Housing	Inco 718
Tur. Side Mating Ring	Inconel-X, Chrome Pl.	Crossover	Inco 718
Laby Spacer Ring	410 CRES	Ring	Inconel-X
Laby Seal	410 CRES	Diffuser	Inco 718
Tur. Side B'ring Carr'r	Hastalloy B	Low P Seal, Carrier	321 CRES
Spring Pins	420 CRES	Low Pressure Seal	Barium
Shaft	Inco 718, Chrome Pl.	Pump S. B'ring Carr'r	Hastalloy B
Bearing Spacer	K-Monel	Spacer	Titanium
Ball Bearings	440C & Armalon		

FRCMC CONCEPTUAL AND DETAIL DESIGNS

ENGINE/COMPONENT SELECTION

The gas generator variant of the STME evaluated in this program was based on the Mark 29 fuel turbopump (Mk29F) used previously in the J2S program. A comparison of the Mk29F with the STME fuel turbopump is given in Appendix C1 with the Mk29F blade path and disk profile. The turbine parameters for the Mk29F and the STME fuel turbopumps are very similar such that designs based on the Mk29F will closely represent the STME design requirements.

Both the first stage nozzle and rotor were considered in the conceptual design stage as reviewed in Appendix C2. These components were compatible with FRCMC fabrication capabilities, and both represented a number of design, analysis and fabrication features critical to this program. The need for a high temperature capability manifold was also factored into the conceptual design. Due to permeability and fabrication constraints, FRCMC are not considered immediately viable for manifold applications, and manifolds were not considered for detailed design.

SUB-COMPONENT SELECTION

As discussed above, the first stage nozzle and rotor of the Mk29F were found to be viable candidates for component conceptual design. Detailed designs were to be performed on selected sub-components which for the selected components could be either nozzle vanes or turbine blades. In practice, it is expected that the nozzle would be made with integral vanes and the rotor would have integral blades for this turbopump. Consequently, a representative sub-component would be integral to the structure and would not incorporate attachment features, such as a firtree. The detailed design was, as a result, based on an integral coupon structure, and a rotor blade coupon was designed with multiple blades integral to a base to represent the component structure. This design has the added benefit of demonstrating the gas path features that could not be represented with discrete blades or vanes.

DETAIL DESIGN

The Mk29F turbine blade coupon which was selected for detailed design consisted of a 3-blade element with all blades integral to a common base as shown in Appendix D1. Coupon details and blade profiles correspond to the design constraints discussed in the following section on "Producibility". The associated specification (Appendix D2) had been developed as part of a Rocketdyne IR&D task for general use. This preliminary specification incorporates those features required for the detailed specification of FRCMC materials as discussed in the "Design" development status review. For the coupon structure, it was necessary to use a 2D lay-up rather than a polar weave. Polar weaving requires a larger, circular structure, such as a blisk. To machine smaller coupons from a larger disk of infiltrated, polar woven material would be cost prohibitive. However, polar weaving will still be considered for full-scale components.

SUB-COMPONENT FABRICATION

SELECTION AND SPECIFICATION

To demonstrate sub-component fabrication capabilities, an existing turbine blade coupon design was used (Appendix E1) that included features comparable to the Mk29F design. Specifically, an existing SEP design was used that was fabricated from partially infiltrated pre-forms that were already available. This approach minimized time required for sub-component fabrication. Materials specification for this coupon was the same as that discussed in Appendix D2.

FABRICATION

Fabrication of the coupons was by CVI of a 2D, plain woven, rectangular preform. Details of the CVI process are proprietary to SEP but consistent with specification requirements. Following machining of the infiltrated preform, a final CVD coating of the matrix material was applied. Both intermediate and final machining details are SEP proprietary but are consistent with specification requirements including needs to avoid exposure of the fibers to the environment.

SUB-COMPONENT CHARACTERIZATION

Sub-components were first inspected visually and by low magnification optical microscopy (Appendix E2). Surface quality and surface finish were very good with no apparent imperfections such as machining damage, exposed fibers, or surface-connected delaminations. Each of the 3 coupons were consistent in appearance, geometry, color and weight indicating reproducibility of the processing. Orientation of the outer weave layers was discernible visibly which allowed determination of fiber misalignment. Discrete, surface connected porosity was observed as expected from the CVI processing and was especially evident viewed from the rear of the blades.

Scanning electron microscopy on fracture surfaces of comparable materials from SEP indicated relatively uniform distribution of fibers, matrix and porosity. A high degree of fiber pullout was also observed at the fracture surface which would contribute to fracture toughness. Also observed was an apparent interfacial layer between the carbon fibers and the SiC matrix. Chemistry of this interface was not determined.

CAT scans were run on the coupons by DuPont at a frequency of 2.25 MHz and are shown in Appendix E3. The CAT scan data were supplemented by real-time microfocus x-ray. Neither technique revealed apparent defects or delaminations within the coupons.

Dimensional inspections were performed using low magnification optical microscopy. These inspections showed the samples to conform to overall geometry requirements and, again, showed a high level of consistency of fabrication. Fiber misalignment observed in the surface layers was less than 3° from nominal (within the specified 5° limit) for all coupons. Quality of leading and trailing edge surfaces was excellent with no apparent machining flaws.

FRCMC STATE OF DEVELOPMENT

FRCMC MATERIALS AND PROPERTIES

Tables 8 and 9 plus Appendix A constitute the detailed design properties used for the FRCMC components. A summary of these properties is given in Tbl. 4. The FRCMC data shown are typical for SEP materials produced by the CVI process. Design allowable properties were calculated from the ultimate properties as discussed previously and were used in the initial engine balances. These values were determined to be in good agreement with SEP's estimates of design limits. To utilize these properties for design, component operating conditions must be specified (Tbl. 10). Properties of SEP's 2D C/SiC were compatible with the component needs and were used in subsequent analyses.

For relatively thick (i.e. >6mm (0.25" thick)) laminates, 2D layups can have essentially orthotropic behavior with approximately the same in-plane strength as a polar woven reinforcement with equivalent radial and circumferential fiber contents. However, the high tensile hoop stress generated at the bore of a turbine rotor necessitates a high concentration of radial fibers at or near the bore to prevent burst. This is, in principle, best accomplished by polar weaving. Also, as polar weaving technology advances, it is expected to be more readily automated than 2D layup technology. In the near term, due to its higher level of production experience, 2D technology gives nominally equivalent performance to polar weaving with fewer production problems. Polar weaving is preferred in the longer term for selected components provided reliable weaving capability is demonstrated.

TBL 8
2D C/SIC PROPERTIES

			TEMPERATURE			
	DIRECTION	**CHARACTERISTIC**	20°C (70°F)	500°C (930°F)	1000°C (1830°F)	1500°C (2730°F)
TENSILE	1-2	Y. Mod., GPa (msi)	90 (13)	100 (15)	100 (15)	100 (15)
		S-Fail., MPa (ksi)	350 (51)	350 (51)	350 (51)	350 (51)
		S-Des., MPa (ksi)	250 (36)	250 (36)	250 (36)	250 (36)
	3	Y. Mod., GPa (msi)	20 (3)	20 (3)	20 (3)	20 (3)
		S-Fail., MPa (ksi)	20 (3)	20 (3)	20 (3)	20 (3)
		S-Des., MPa (ksi)	10 (1.5)	10 (1.5)	10 (1.5)	10 (1.5)
COMPRESSIVE	1-2	Y. Mod., GPa (msi)	120 (17)	120 (17)	120 (17)	110 (16)
		S-Fail., MPa (ksi)	580 (84)	600 (87)	600 (87)	700 (102)
		S-Des., MPa (ksi)	400 (58)	400 (58)	400 (58)	400 (58)
	3	Y. Mod., GPa (msi)	70 (10)	70 (10)	70 (10)	70 (10)
		S-Fail., MPa (ksi)	420 (61)	450 (65)	450 (65)	500 (73)
		S-Des., Mpa (ksi)	300 (44)	300 (44)	300 (44)	350 (51)
SHEAR	1-3 & 2-3	Y. Mod., GPa (msi)	15 (2)	15 (2)	15 (2)	15 (2)
		S-Fail., MPa (ksi)	35 (5)	35 (5)	35 (5)	35 (5)
		S-Des., Mpa (ksi)	20 (3)	20 (3)	20 (3)	20 (3)
THERMAL	---	Spec. Heat	620	1200	1400	1500
	1-2	Diffusivity	11	8	7	8
	3	Diffusivity	5	3	2	2
	1-2	Thermal Exp., /°C (/°F)	0.0 (0)	1.5 (0.8)	3.0 (1.7)	4.5 (2.5)
	3	Thermal Exp., /°C (/°F)	0.0 (0)	2.5 (1.4)	5.5 (3)	9.0 (5)

Poisson's Coefficients: 1-2 = 0.05; 1-3 = 2-3 = 0.25
Y. Mod. = Young's Modulus
S-Fail. = Ultimate Failure Stress; S-Des. = Design Allowable Stress
Spec. Heat = Specific Heat, J. /kg-K
Diffusivity = Thermal Diffusivity, $10^{-6} m^2/sec$
Thermal Exp. = Total Linear Thermal Expansion

TBL. 9
NOVOLTEX/SiC PROPERTIES

	DIRECTION	CHARACTERISTIC	20°C (70°F)	500°C (930°F)	1000°C (1830°F)	1500°C (2730°F)
			TEMPERATURE			
TENSILE	1-2	Y. Mod., GPa (msi)	75 (11)	80 (12)	85 (12)	70 (10)
		S-Fail., MPa (ksi)	80 (12)	90 (13)	100 (15)	100 (15)
		S-Des., MPa (ksi)	60 (9)	65 (9)	65 (9)	65 (9)
	3	Y. Mod., GPa (msi)	60 (9)	65 (9)	65 (9)	60 (9)
		S-Fail., MPa (ksi)	50 (7)	50 (7)	55 (8)	55 (8)
		S-Des., MPa (ksi)	35 (5)	35 (5)	35 (5)	35 (5)
COMPRESSIVE	1-2	Y. Mod., GPa (msi)	100 (15)	100 (15)	100 (15)	100 (15)
		S-Fail., MPa (ksi)	650 (94)	700 (102)	700 (102)	800 (116)
		S-Des., MPa (ksi)	500 (73)	500 (73)	500 (73)	500 (73)
	3	Y. Mod., GPa (msi)	100 (15)	100 (15)	100 (15)	100 (15)
		S-Fail., MPa (ksi)	650 (94)	700 (102)	700 (102)	700 (102)
		S-Des., Mpa (ksi)	500 (73)	500 (73)	500 (73)	500 (73)
SHEAR	1-3 & 2-3	Y. Mod., GPa (msi)	10 (1.5)	10 (1.5)	10 (1.5)	10 (1.5)
		S-Fail., MPa (ksi)	60 (9)	50 (7)	40 (6)	40 (6)
		S-Des., Mpa (ksi)	40 (6)	35 (5)	30 (4)	30 (4)
THERMAL	---	Spec. Heat	620	1200	1400	1500
	1-2	Diffusivity	12	6	5	5
	3	Diffusivity	9	5	4	4
	1-2	Thermal Exp., /°C (/°F)	0.0 (0)	0.9 (0.5)	2.1 (1.2)	4.0 (2.2)
	3	Thermal Exp., /°C (/°F)	0.0 (0)	1.0 (0.6)	3.0 (1.7)	5.5 (3.1)

Poisson's Coefficients: 1-2 = 0.25; 1-3 = 2-3 = 0.20
Y. Mod. = Young's Modulus
S-Fail. = Ultimate Failure Stress; S-Des. = Design Allowable Stress
Spec. Heat = Specific Heat, J. /kg-K
Diffusivity = Thermal Diffusivity, 10^{-6}m2/sec
Thermal Exp. = Total Linear Thermal Expansion

TBL. 10
FRCMC MATERIALS CHARACTERIZATION
OPERATING ENVIRONMENT PARAMETERS

PARAMETER	RANGE OF SPECIFIED VALUES
ATMOSPHERE	Nominal and expected variations
TEMPERATURE	Typical, maximum, dT/dt
PRESSURE	Typical and maximum
MECHANICAL STRESS	Nominal + and - cyclic
THERMAL STRESS	Transient and steady state
DURATION/LIFE	Total operating time, including test time
NO. OF CYCLES	Total no. of starts and stops, including tests

FRCMC VENDORS

Rocketdyne, in previous and on-going programs, has reviewed the capabilities of FRCMC vendors. Through these studies and through consultations with personnel at WI, it was determined that the leading supplier of FRCMC components is SEP.

SEP's development of composites was started in response to their needs for improved organic matrix materials to be used as solid rocket motor casings. Using their background developed with these materials, SEP continued to develop other composite materials, including C/C and ceramic matrix composites which also supported their liquid rocket engine developments. By combining their capabilities for materials design, analysis and production, SEP has demonstrated the expertise essential to the engineering and production of FRCMC for advanced liquid rocket engine components. Other materials suppliers are developing improved capabilities but have not yet demonstrated component production capabilities equivalent to SEP.

PRODUCIBILITY

Producibility of FRCMC components, even by the most advanced producers, is constrained by a number of factors (Tbl. 11).

TBL. 11
FRCMC FABRICATION LIMITATIONS

* Hole required in disc center for polar woven parts
* Minimum leading and trailing edge radius of 0.15 to 0.20 mm (0.006 to 0.008")
* Blade height/blade gap < or = to 7/1 with minimum blade gap of 2 mm (0.08") for milling
* Minimum recommended trailing edge thickness of 0.4 mm (0.016")
* Fabrication of slightly tapered hub feasible
* Minimum 0.20 mm (0.008") fillet radius recommended at hub of blade; prefer larger
* Integral shrouds not feasible
* All machining in oxidizing areas must be done prior to final CVD to ensure fiber protection
* Final machining possible in non-oxidizing areas
* Rework capability severely limited (final CVD difficult to effectively repeat)
* Blade and vane shapes limited to positive taper with shadowed tips
* Cannot be used for absolute pressure containment
* 1 year lead time typical for prototypes

The size and uniformity (temperature and atmosphere) of available, production CVI reactors limit the component exterior dimensions. Maximum diameters of 0.93m (37") have been demonstrated to date for components produced in a 1.4m (55") diameter reactor. Maximum length is about 2-3m (6-8'). SEP is developing and installing a reactor with a nominal 2.5m (100") maximum diameter which is expected to be on-line in approximately 6 months.

Polar weave technology requires that a bore hole be maintained in the structure. Present weaving capabilities allow bore holes as small as 25mm (1") in a 150mm (6") diameter rotor. This 6:1 ratio is typical of current technology limits. Further, SEP indicates that the bore hole is advantageous for efficient processing. Reaction times and density/microstructure/porosity distribution limits in thick polar woven cross-sections (wall thicknesses of 25mm (1")) are improved by bore holes which aid infiltration of the reactant gases. Relatively thin 2D layups do not require a bore hole, but for thick sections bore holes may be needed for effective infiltration. Wall thicknesses much greater than 25mm (1") are not feasible due to non-uniformity and long reaction times (weeks to months).

Minimum radii are constrained largely by the fiber tow type and the ability to maintain composite mechanical properties in thin sections. Smaller tows facilitate finer structures but require longer pre-form processing times (and expense) and make infiltration between tows without delaminations more difficult. Currently, radii for woven structures are limited to a minimum of from 0.15 to 0.20mm (0.006 to 0.008") in order to maintain acceptable composite properties, and larger radii are preferred. These radii are achieved by in-process machining. Further consideration needs to be given to the effects of machining on properties, especially of fine structures, but these effects cannot be quantified at this time.

Blade height to gap ratio is limited by machinability constraints such that the ratio must be maintained at 7:1 or less. This is needed to allow tool access for machining blades and vanes. Also, due to machining requirements, a minimum blade spacing of 2mm (0.08") is recommended. Additional details of blade profiles are discussed in the following "Design" section.

Many of the geometry issues discussed must also be understood to be interactive. For example, a specific blade profile may require a greater blade spacing than that discussed. Consequently, detailed designs of specific components must be closely reviewed to determine conformance to the geometry constraints imposed by these interactions.

Balancing and machining in areas subject to environmental exposure must be done prior to final CVD to avoid exposing fibers directly to the environment. Areas not subject to environmental attack can be final machined after the last CVD step. The preferred balancing method is to balance following each of the intermediate CVI steps, but prior to the final CVD coating, such that only minor touch-up is required during the final processing stages.

Surface finish corresponding to an R_a value of 20 x 10^{-6}cm (8 x 10^{-6}") can be achieved between areas having surface connected porosity. Pores open to the surface result in large apparent surface discontinuities, the aerodynamic effects of which have not been analyzed.

Currently, repair or rework of parts determined to be flawed after CVD is not feasible for highly stressed rotating components but may be feasible for static structures. The final CVD coating relies upon infiltration into the surface pores for proper adherence. Consequently, patching of localized areas is not effective, but it may be possible to re-CVD the entire component.

The limited number of production reactors and the long processing times required result in long lead times, especially for prototypes. This is typically about one year. Considering the long lead times and the inability to rework, extra parts should be fabricated which, as discussed previously, are expensive. This is a lesser problem for production, rather than prototype, components. Developments are being pursued to reduce processing times, e.g. the chemical potential gradient processing technique developed at Oak Ridge National Laboratories (ORNL) is claimed to reduce CVI times from weeks to days. Thermal gradient processing and pressure gradient processing also offer potential for significant reductions of processing time.

Once produced, the FRCMC material contains high levels of residual porosity (typically up to about 15 volume % with some variation dependent upon materials and architecture) and is permeable to the combustion environment. This, combined with difficulties in fabricating complicated housing shapes, currently precludes successful fabrication of pressure vessels. Impermeable coatings usable in the rocket engine environment have not yet been demonstrated.

JOINING/ATTACHMENT

Joining of FRCMC both to similar and dissimilar materials is a very new technology. This technology has recently been reviewed by Cawley (Ref. 6).

For the production of large cross-section components or for unusual shapes, the possibility of joining multiple sub-assemblies by CVI while the sub-assemblies are held in integral contact in the reactor has been demonstrated. This provides no load-bearing fibers across the interface which must be factored into the stress analysis. Joining with fasteners is possible and has been demonstrated. Threads within the material have been demonstrated but are only recommended for low stress joints. Flanges can be incorporated and used with through fasteners when compatible with design constraints.

Attachment of FRCMC to surrounding hardware by mechanical means has been successfully demonstrated in engine assemblies. FRCMC rotors have been attached to metallic shafts by use of face splines. Clamps and compliant interface layers have been used to attach static FRCMC hardware to metallic hardware.

Brazing to dissimilar materials is possible for relatively low temperature joints.

PROCESS AND QUALITY CONTROL

Process controls can be and are employed at the earliest stages of the process, e.g. during weave preparation and lay-up of the pre-form. At these stages, critical control of fiber variables and fiber architecture occur. Such controls are detailed in the component specification. However, discussion here is focused primarily on controls implemented during and after the first CVI step, which are steps controlled by the CVI materials fabricator.

During processing, reactor conditions (gas chemistry, reactant temperatures, pressures, flow rates, etc.) are fixed based upon vendor experience with previously processed similar components. SEP's control points are proprietary, but their success with a wide range of components is indicative of their high level of experience. Most components require multiple infiltration steps. After each step it is necessary to machine the surface "skin" from the material to allow infiltration to proceed. While removed from the reactor, the in-process parts are inspected visually, are weighed to verify density, and are x-rayed to determine excessive density variations or apparent delaminations. Defective parts can be rejected at this stage to avoid added processing expense. Whether or not a defect is rejectable is principally an experience-based judgement dependent upon knowledge of the operating conditions (e.g. stress levels).

Final part inspection includes the methods noted above (visual, density, x-ray) and can, as required, employ additional checks. Laser holography has been used to determine distortion in stressed vs. unstressed components as an indication of weak or delaminated areas. Ultrasonic inspection can be used to map density distributions. Computer-aided tomography (CAT) has been demonstrated as an extremely effective method to quantitatively measure density gradients in thick, complex shape components. Recently developed CAT systems have the potential to identify weave irregularity and fiber misalignment within plies of a composite structure. Profilometry is used to verify surface finish. Chemistry can be verified as needed. Witness coupons may be prepared during component processing for verification of properties by destructive tests. These coupons can effectively simulate the microstructure and properties of an FRCMC component that has a uniform cross-sectional thickness, but simulation of complex shapes with variable cross-sections is difficult, if not impossible. Conformance of geometry to the drawing can be verified by standard measurement methods.

Design and analysis in combination with the non-destructive test methods noted, in some cases, would require further verification. This can be accomplished by proof-testing to ensure critical performance parameters are met. For turbine rotors this would typically be done by spin testing, preferably at temperature, at selected stress levels.

DESIGN

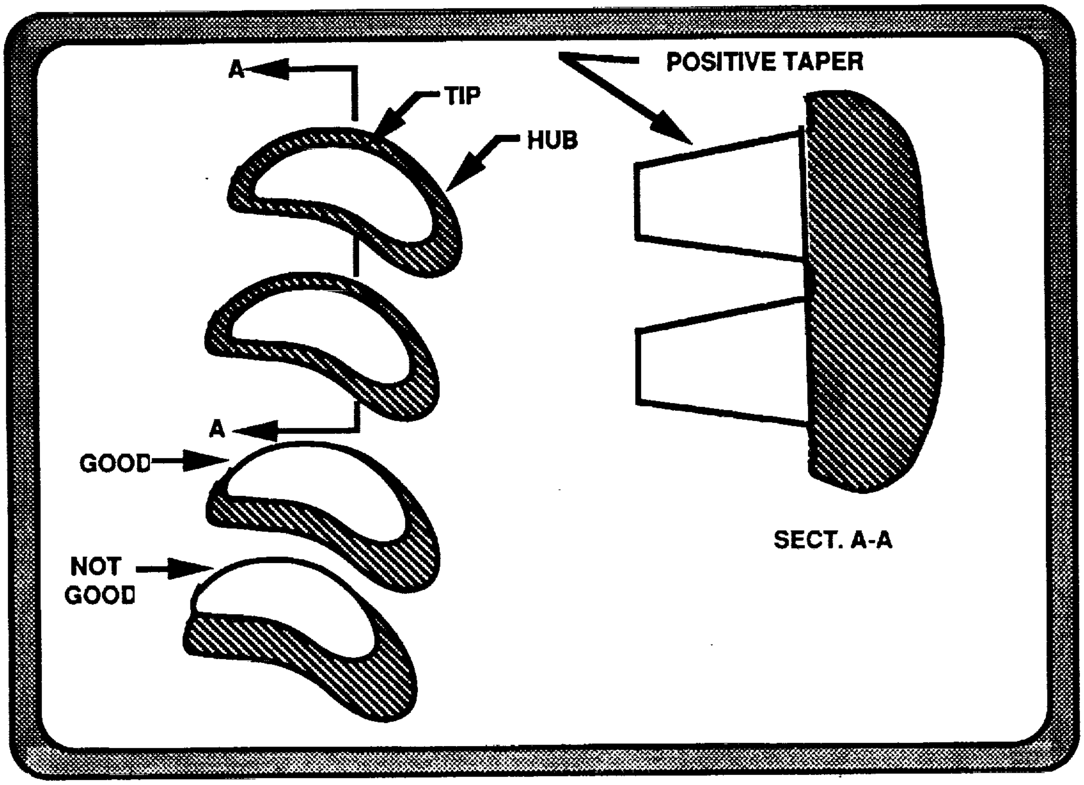
Proper design combines consideration of: the producibility issues addressed above, the need for attachment to other components, and the relationship between materials properties and component requirements as determined by analysis. To most effectively design with composites, it is necessary to "think composites" during the conceptual design stage, to perform initial composite analyses quickly using simple methods and finally to perform more detailed analysis and design using more sophisticated methods.

The primary design issues relative to the application of FRCMC as turbine and nozzle components are summarized in Tbl. 12. Relative to blades and vanes, machining requirements are such that blade profiles must conform to the generalized geometries shown in Fig. 4. Integral shrouds cannot be fabricated using current technology; and shrouds, if needed, must be fabricated separately.

TBL. 12
FRCMC DESIGN ISSUES

* TOLERANCING/FINAL MACHINING
  - Prior to final CVD - standard machine tolerance
  - After final CVD - minimal grinding only
  - Post CVD machining allowable in non-oxidizing areas only
* SURFACE FINISH
  - Standard machine finish possible
  - Polished surface finish to 20×10^{-6}cm (8×10^{-6}in)
  - Surface finish value does not factor in surface connected porosity
* ATTACHMENT METHODS
  - Rotor couplings - cooled drawbolt
    coarse male metal face spline
  - Flanges - Nuts/nut plates
    Inserts
    No parent mat'l threading in highly loaded areas
* ROTORS
  - Balancing good to 1 gram - inch (after final CVD)
  - Disc - Small positive taper from blade to hub
    No neck (recessed sections) allowed
  - Blades - positive taper only
    shadowed tips acceptable
* STATORS
  - Separable outer shrouds only
  - Full rings possible

FIG. 4
ACCEPTABLE FRCMC TURBINE BLADE PROFILES



The complexity of the composite structure and the number of phases involved require that the materials specification and associated drawings provide considerable detail. Typical drawing and specification requirements are shown in Tbl. 13.

TBL. 13
FRCMC COMPONENT DRAWING AND SPECIFICATION CONTENT

* Finished toleranced geometry only
* Metric and English dimensions
* Inspection criteria
* Format definition
* Part identification definition
* Surface finish requirements
* Component architecture definition
Fiber direction
Fiber allowable misalignment
Vol. fraction fiber in each direction
Filament type
Tow size and type
Homogeneity limits
Porosity limits
* Packaging requirements
* Disclosure statement

ANALYSIS

Analysis methods typically used for FRCMC are summarized in Tbl. 14. The overall approaches used by Rocketdyne, WI (Ref. 7) and SEP are similar. Various computer codes are capable of this level of analysis, such as the ANSYS code (Ref. 8) used at Rocketdyne. SEP uses non-linear properties data, which they have developed for their materials, to perform more detailed analyses. Probabilistic analysis methodologies for these non-linear, anisotropic materials require further development (Ref. 9).

TBL. 14
FRCMC STRUCTURAL ANALYSIS AND DESIGN

- Finite element methods used for analysis
- Combined failure modes considered - burst speed, blade disk dynamics, thermal fatigue
- Composite architecture selection based on experience
- 2D lay-up treated as orthotropic material
  - Elastic constants obtained by testing
  - No differentiation between disk and blade elastic properties
- Structural criteria
  - Combined static stresses (mechanical and thermal) must be < or = to design allowable stress
  - Fatigue (LCF and HCF) limits not rigorously established - considered covered by static limits
- Analytical treatment not rigorous
  - Material evaluated at macromechanical level only - WI capable of analysis of individual lamina
  - End effects not considered for leading and trailing edges including elastic constants and strength
  - Basic static stress criteria used
    - Strength only known in principal directions (off-axis data very limited)
    - WI analyzes combined (biaxial or multiaxial) stress using Tsai-Hill or modified Hill function
- Approach effective for simple designs in uniaxial or biaxial stress states
- Component or sub-component testing desirable to verify assumptions

Prior to analysis, the fiber architecture must be determined. This is an empirical selection based on component geometry, performance needs and experience. Using the selected material and fiber architecture and the corresponding materials properties, analysis is performed using finite element methods. For 2D materials, including polar woven structures, orthotropic properties are input. Anisotropic properties can be used in the analyses if necessary. Analyses are based on macromechanical principles. Detailed micromechanical models are not expected to be available for 5 to 10 years. Failure criteria are based on the limitation of cumulative static stress (mechanical plus thermal) to less than the design allowable limit. SEP's design limit for a blade is approximately 60% of their design limit for other structures. The lower limit for blades allows for high cycle fatigue effects on component life.

FRCMC DEVELOPMENT NEEDS

MATERIALS

FRCMC materials properties were reviewed in previous sections. FRCMC systems have a potentially beneficial range of capabilities for liquid rocket engine applications. The principal limitations found to date are due to the relative newness of these systems, and many of these limits are being met by ongoing developments. The limits found and the needs for development are reviewed below.

An FRCMC system consists of the fiber reinforcement, the matrix phase, the fiber/matrix interface, porosity, and (when used) protective coatings. The chemistry, microstructure, and distribution of these phases all influence the final FRCMC properties. The presence of multiple, interactive phases complicates FRCMC specification and development vs. conventional, monolithic materials. Consequently, a fundamental understanding of the material at a microstructural level must be gained; and related, fundamental studies are ongoing. Current understanding is limited to the macrostructural level.

Fibers

There are a number of development needs for the fiber reinforcement. Carbon fibers, prepared from both pitch and polyacrylonitrile (PAN) precursors, have been available for some time. They are readily, commercially available in various grades (e.g. high modulus, high strength), tow sizes (filament counts), surface treatments, etc. Commercial grades are well characterized and reproducible, and new varieties continue to be developed. Carbon fibers are resistant to very high temperatures but are susceptible to environmental attack (either oxidizing or hydrogen environment attack) at high temperatures and must be protected. Protection can be provided by either the matrix or by protective coatings. Carbon fibers have very low (near zero) coefficients of thermal expansion along their axes and are anisotropic, with the result that a large thermal expansion mismatch with the ceramic matrix occurs. Therefore, high levels of stress occur in the matrix when cooled from the process temperature which cause microcracking. As the microcracks are stressed and open, the degree of fiber protection afforded by the strain intolerant matrix is lowered. Coatings, such as glasses, that flow at the use temperature help to seal these microcracks. Development of improved coatings that can tolerate the thermal shock and stress imposed by rocket engine use are needed. Also, fibers with improved environmental resistance are needed, including improved SiC fibers. Existing replacements for carbon fibers lack the high strength and strength-to-weight ratio of carbon. Since the fibers act as the principal load bearing element, newly developed fibers must have improved mechanical properties as well as better environmental resistance.

There are two main types of SiC reinforcement fibers now in use. Multifilament (e.g. Nicalon, Nippon Carbon Co., Tokyo, Japan and Dow Chemical Co., Midland, Michigan) fibers have a high level of oxygen and other impurities which limit their maximum use temperature due to internal oxidation. Monofilament (e.g. SCS-6, Avco Corp., Wilmington, Massachusetts) fibers, prepared by CVD, are higher purity with higher strength and higher operating temperature than Nicalon but cannot be formed into complex preform geometries and are not readily, commercially available. Production of complex shapes from monofilament reinforced matrices currently requires high levels of final machining which exposes fibers and often leaves irregular surfaces which would cause substantial aerodynamic problems. Use of coatings to protect the exposed fibers is anticipated but has not been demonstrated. Additional multifilament fiber development continues to be needed for production of complex shapes. Among current development programs is SEP's program with fiber producers to develop a low oxygen content, multifilament SiC fiber. Also, NASA-LeRC has been working with Dow Corning to develop a small diameter, stoichiometric SiC fiber.

Other fiber chemistries are being developed, both oxides and non-oxides. Development of such fibers with various matrices is in progress. Among fibers under development, in addition to multifilament SiC, are polymer derived Si_3N_4 and Al_2O_3 fibers.

Matrices

SiC ceramic matrices have been developed and demonstrated in combustion atmospheres to temperatures as high as 1650°C (3000°F). SiC has good thermal shock resistance, resists oxidation by the formation of an adherent, protective surface oxide, and is compatible with both carbon and SiC fiber reinforcement. However, SiC is more thermal shock sensitive than Si_3N_4 (Ref. 10); it is thermodynamically unstable in high temperature, hydrogen combustion environments (Ref. 4); and, at low oxygen partial pressures, it is subject to active oxidation due to the formation of volatile silicon monoxide (SiO) vs. the stable silicon dioxide (SiO_2) which forms at higher oxygen partial pressures.

Si_3N_4 has potential advantages for severe thermal shock applications, but development of CVI processed Si_3N_4 is very new. The limited data available for Si_3N_4 matrix composites are encouraging, but feasibility has not been demonstrated on a component scale. Maximum use temperature of Si_3N_4 is well above the 1200°C (2200°F) target.

Very high temperature matrices have been demonstrated on a small scale. Carbon fiber-reinforced hafnium carbide (C/HfC) was produced by Refractory Composites Inc. (RCI) and successfully tested by NASA-LeRC as a small, uncooled thruster. Although the very high temperature capabilities of these newer materials are not essential to this program, they are of interest for further gains in the future and offer the possibility of even further improvements in operating margin and reliability at the temperatures of interest here.

Oxide matrices (e.g. Al_2O_3 and stabilized zirconia, ZrO_2) also show promise but are not known to be sufficiently developed to meet program schedule requirements.

Improved non-destructive evaluation (NDE) techniques are needed as well as a better understanding of the information provided by existing techniques. NDE results (x-ray and ultrasound) are empirically interpreted to determine whether or not rejectable flaws exist. Use of laser interference images to find areas of excessive strain while under stress appears to be a promising development for quantifying defect effects. CAT also offers potential and continues to be developed as a composite evaluation tool. Acoustic microscopy has promise for detection of near-surface defects.

Even with improved NDE, proof testing of components is expected to be necessary for some time. Improved proof testing capabilities, which better simulate actual operation, including temperature and atmosphere effects, are needed.

Interfaces

The proper fiber-matrix interface is essential to achieving desired composite behavior. Strong bonding or interaction between fiber and matrix results in low levels of fiber pullout and limited crack deflection with low toughness and a potentially catastrophic failure mode. Pullout and a more "graceful" failure mode are promoted by weaker bonding which can be achieved by coating the fibers prior to CVI. Surface treatments for this purpose are usually proprietary to the respective vendors. Development of improved interfaces is ongoing.

Porosity

Internal porosity and some surface connected porosity are inherent to the CVI process. As the infiltration progresses, a "skin" forms on the preform surface. This skin is intermittently removed by machining. The "stop and start" of the CVI process is time-consuming and expensive. Optimization of fabrication time with product properties, such as strength and toughness, results in a preferred level of residual porosity in the final product which for the principal materials discussed here typically falls between 10 and 15 volume %. Effects of surface connected porosity on aerodynamic performance have not been analyzed.

Coatings

Coatings for environmental protection and for reduction of permeability continue to be developed. SiC coatings and CVD Si_3N_4 coatings have been demonstrated for oxidation protection. Recent work has demonstrated hafnium carbide (HfC) coatings or HfC mixed with SiC for higher temperature oxidation protection than achievable with SiC alone. Glassy sealants are being used to heal cracks. Other work is on-going for hydrogen permeation prevention.

PROCESSING

Needs for process improvement cannot be separated from the materials needs discussed above. Current processing is slow which, in turn, results in long lead times and high costs. Available processing equipment is limited both by size and by number of reactors. Processing equipment is capital intensive. Thus, development of faster processes that achieve equal or improved quality is essential to economical production. Developments, such as ORNL's chemical potential gradient processing, are intended to markedly reduce process times.

Experience based process control methods for parts that are similar to previously produced parts are acceptable. However, as new configurations are produced, this approach will result in added process development time. The use of on-line control and closed-loop process control is preferred. Some development of these methods is ongoing.

JOINING/ATTACHMNENT

Existing joining and attachment methods were discussed above. The ability to both analyze and fabricate fasteners requires improvements. Joining of in-process parts through simultaneous CVI of separate sub-assemblies appears encouraging for similar materials. Brazing has potential for lower temperature areas. Of particular need is the development of sound attachments between FRCMC and metals in highly loaded, high temperature areas, e.g. rotor to shaft attachments.

REPAIR

Development of repair methods, both for rework to salvage in-process parts and for completed parts damaged during assembly or use, is highly desirable. However, existing repair methods are extremely limited in their capabilities for ceramic composites.

DESIGN

The status of design issues was discussed previously and has been reviewed recently in the literature (Ref. 8). Of initial interest to design is selection of preform fiber architecture which is currently done empirically. Development of a structured or analytical approach is preferred, especially for the design of unique, new components. Remaining design issues are principally a function of materials characteristics and producibility issues which have been discussed throughout this report.

ANALYSIS

As noted above, the current analytical approach is based on macroanalysis and will necessarily be limited to such for several years until a fundamental, microanalytical understanding is gained. A microanalytical approach is preferred, especially for the analysis of sections in which retention of composite macroproperties is questionable (e.g. fine leading and trailing edges). With sufficient materials properties data available, current analytical codes are capable of analyzing anisotropic, non-linear materials. Details of the non-linear materials response characteristics are not generally available, but SEP has a detailed, proprietary data base which includes these characteristics for selected materials systems. Analysis for life-related properties (e.g. creep, stress rupture, fatigue) cannot be accomplished using available macromechanical models. Limited experimental data are available, and both additional data and micromechanics models are needed. These are being developed for specific systems.

FUNCTIONAL CHARACTERIZATION

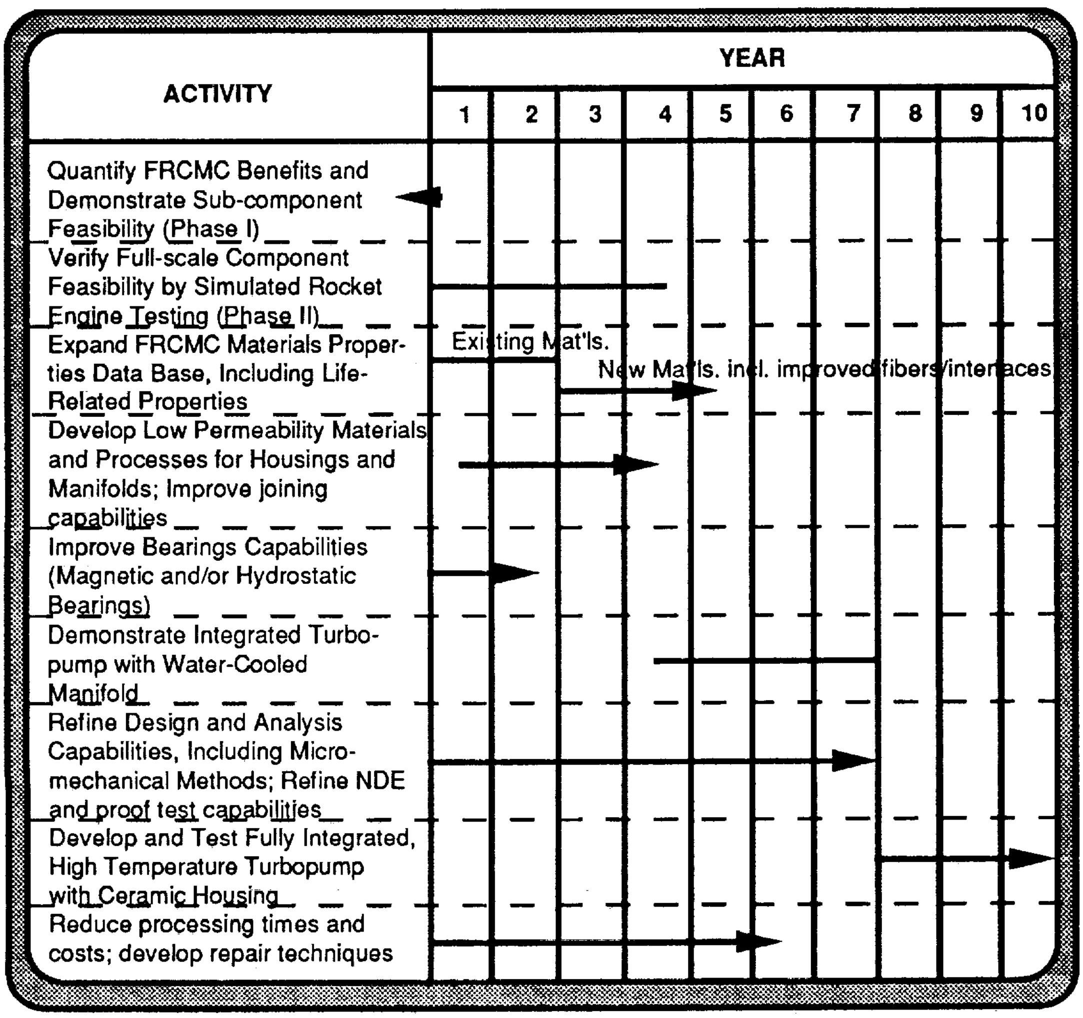
Based on the needs for micromechanic models and further experimental data, it is currently preferred to characterize materials relative to their specified function. Ultimate properties are relatively well characterized for selected systems (such as 2D C/SiC). Life-related properties need added characterization. For turbine rotors this will include characterization of both high and low cycle fatigue properties at temperatures and stresses characteristic of the identified turbine operating conditions.

DEVELOPMENT PLAN AND SCHEDULE

OVERALL PROGRAM

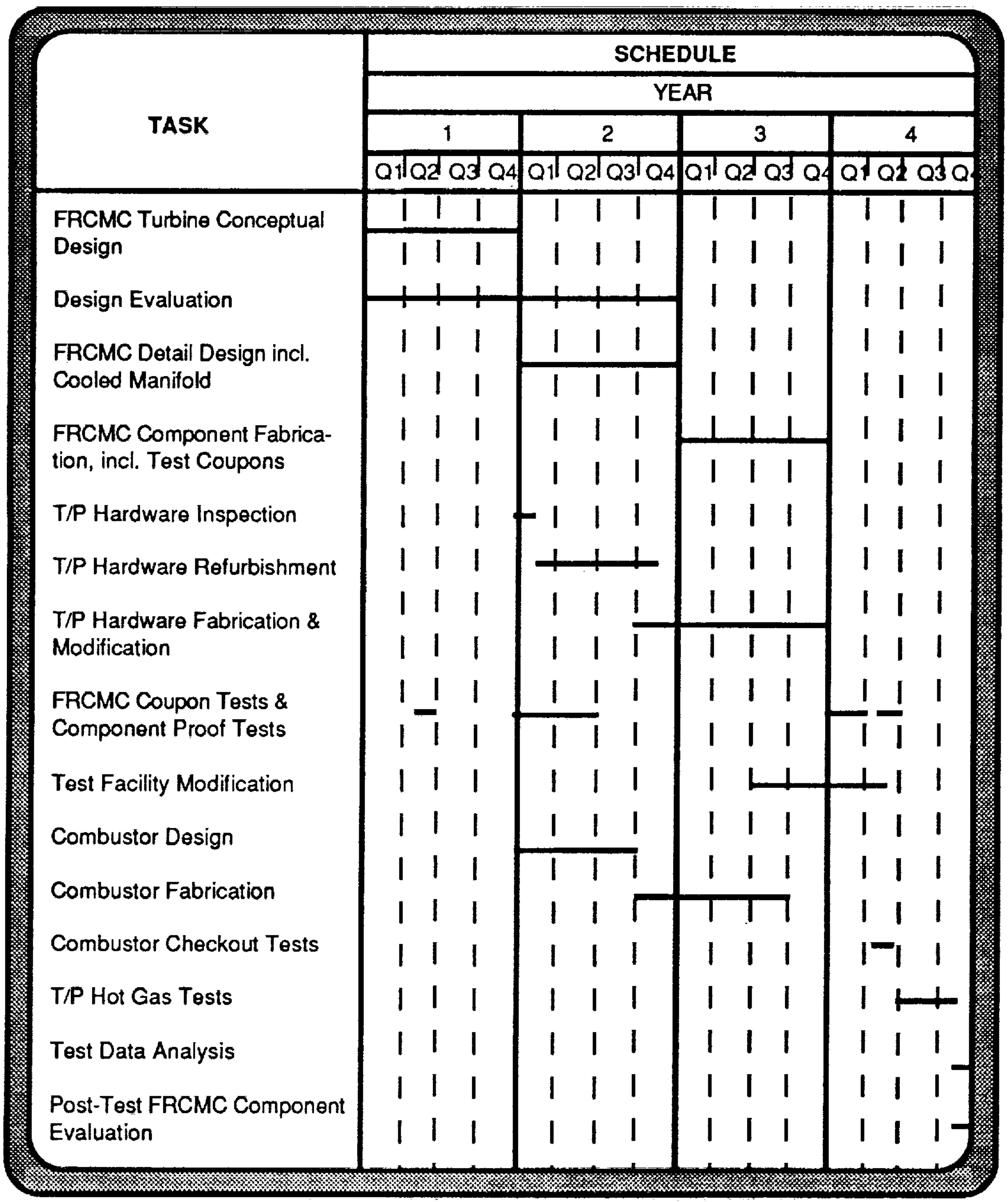
The long-term plan (Fig. 5) addresses the technical issues discussed for FRCMC and includes development of improved bearings as required to take full advantage of the potential of FRCMC components. This plan is based on a continuation of effort following completion of Phase I. Also considered is the development of low permeability materials which can be processed into complex shapes for housings that are essential for the demonstration of a high temperature, flightweight turbopump. Testing using a water-cooled manifold is proposed only as an interim verification and demonstration step in the total program. Integration of selected activities would reduce overall time and cost as discussed below.

FIG. 5
LONG-TERM PLAN FOR DEVELOPMENT OF FULLY INTEGRATED FRCMC EARTH-TO-ORBIT ROCKET ENGINE TURBOPUMP



An integrated program for full-up turbopump testing to demonstrate the feasibility and benefits of FRCMC in an advanced turbopump is outlined in Fig. 6. The outlined plan would integrate the Phase II component into an operational turbopump which would include other critical components essential to the nominal 1200°C (2200°F) operating temperature which has been proposed. As outlined, the plan would require an estimated 4 years to complete with a total program cost of approximately $4-6 M., which is inclusive of the Phase II program costs.

FIG. 6
ADVANCED TURBOPUMP DEVELOPMENT PLAN



Essential features of the plan relative to the previously identified critical technical issues are:

1. A conceptual design of the entire turbopump would be completed which identifies critical components essential to high temperature operation.

2. Following conceptual design evaluation, detail designs of key components would be completed. The plan as outlined would include a cooled manifold made of conventional materials rather than an uncooled FRCMC manifold. As noted, high temperature manifold development is needed, but this is not expected to be available in the near term due to producibility and permeability limitations of existing systems.

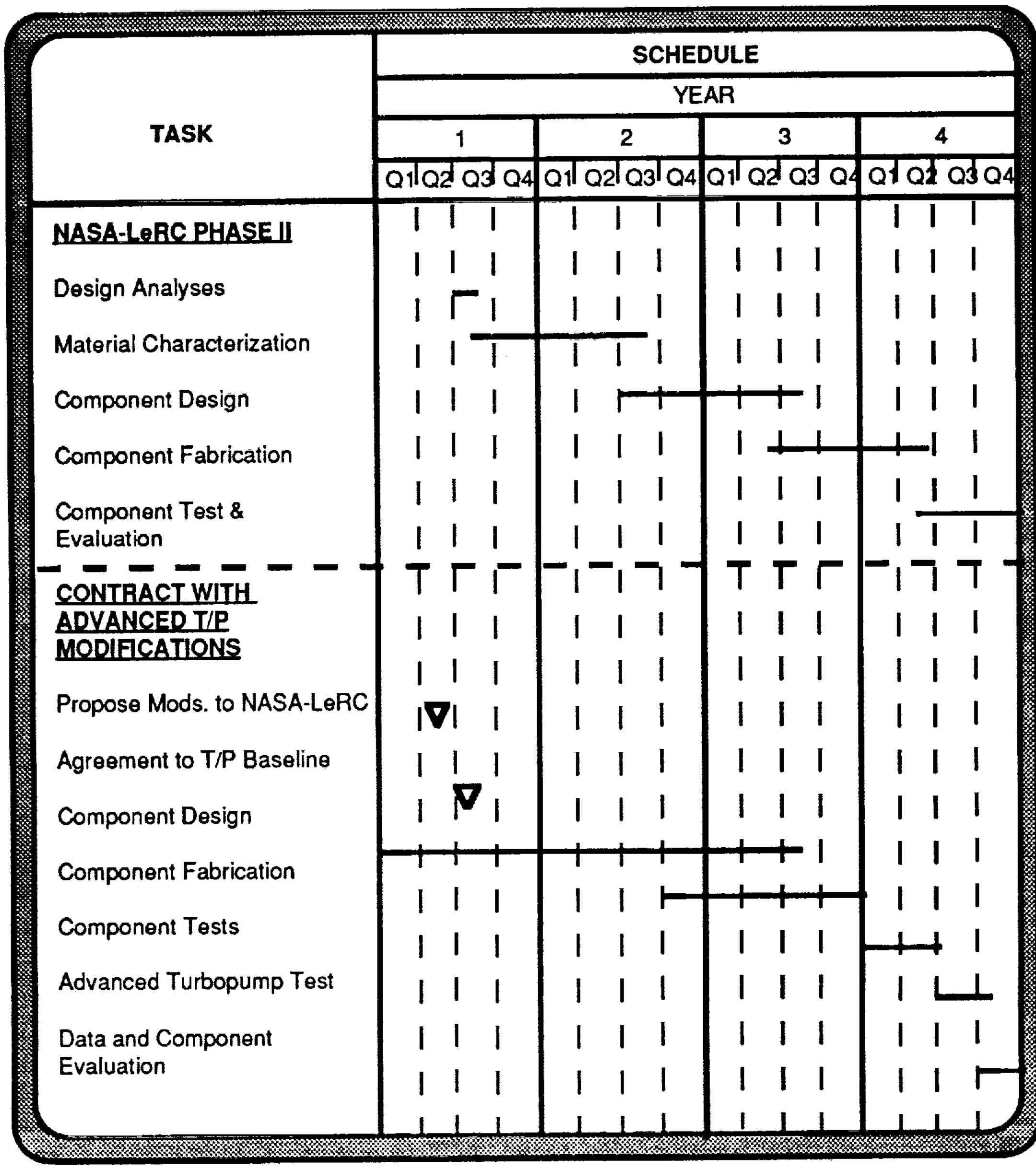
3. Following fabrication of the critical hardware, proof and destructive tests would be run to ensure properties conformance to requirements identified during the design study. Coupons would also be tested to confirm properties of the specified FRCMC system(s). Coupon testing emphasizes life-related properties (creep, stress rupture and fatigue) which would be evaluated at conditions comparable to the specified conditions of actual operation.

4. Modifications of existing test facilities, including combustor modifications, would be needed to achieve the elevated operating temperature.

5. Finally, hot fire engine tests would be run and test results analyzed to verify performance benefits and component reliability.

The program as outlined minimizes total program cost and accelerates timing as opposed to a series program requiring Phase II completion prior to initiation of the full-up test plan. A parallel effort could be run with Phase II integrated as shown in Fig. 7 which maintains many of the cost and schedule advantages of the fully integrated program yet maintains a discrete Phase II program.

FIG. 7
INTEGRATION OF PHASE II WITH TURBOPUMP PLAN



Either of the programs assumes continuation of the ongoing efforts discussed, such as development of improved analytical methods and development of improved materials systems but does not rely on completion of these efforts to achieve near-term program goals. Rocketdyne continues to monitor and evaluate new developments through internal research and development programs. The program as outlined would use the preferred materials systems currently available as well as existing design and analysis methods to demonstrate an advanced turbopump in the near-term. Improvements in analytical methods could be included to refine analyses as the improved methods become available. New materials could be utilized to upgrade components when verified materials properties become available for these new materials. However, such improvements are not essential to the program and are not considered in the outlined program scope or cost estimates.

PHASE II PROGRAM

The Phase II program is a 44-month, nominal $2M program and is shown as part of the integrated program outlined in Fig. 7 with details of the originally proposed Phase II program presented in Appendix F.

Phase II will further characterize the essential materials properties identified in Phase I for a selected material system. The material system chosen is a 2D C/SiC produced by the CVI of a continuous carbon fiber preform. Properties which will be verified include life-related properties necessary for the long-term, cyclic operation of an advanced turbopump at 1200°C (2200°F).

The preferred codes for FRCMC analysis will be identified; their applicability will be verified for the selected component (e.g. first-stage inlet nozzle for the STME); and existing codes will be modified, if necessary, to perform the analyses. Details of the codes are proprietary; but approach, capabilities, limitations, and input/output parameters will be disclosed.

The STME fuel turbopump nozzle was selected for demonstration in Phase II because it is exposed to the highest turbine temperatures and because it effectively demonstrates a number of key design, fabrication and test issues. The nozzle is a complex geometry which would be fabricated with integral vanes demonstrating the key machining features noted previously, such as surface finish, minimum radii, critical vane spacings, fiber alignment, etc. It is an essential component for demonstration of a full-up turbopump. A smaller, lower cost turbine could be effectively used for an integrated, comprehensive turbopump test program. Also, the nozzle includes critical sub-elements that can be evaluated on a prototype scale. For example, nozzle vanes could be fabricated with platforms that would be tested for thermal shock resistance using a LOX/H_2, gas generator thermal cycling rig such as that available at the NASA Marshall Space Flight Center (NASA-MSFC).

The Phase II demonstration of a full-up component in the simulated rocket engine environment will be combined with Phase I results to generate a detailed plan for the implementation of FRCMC into a future generation liquid rocket engine. This plan would be expected to follow the general outline shown in Figs. 5 to 7.

CONCLUSIONS

Utilization of FRCMC in future generation earth-to-orbit rocket engine turbines offers a number of potential advantages:

- Increasing the TIT of gas generator cycle, LOX/H_2 propelled engines, such as the STME, from a baseline temperature of 870°C (1600°F) to an FRCMC realizable temperature of 1200°C (2200°F) will result in significant (>5 sec) gains in I_s.
- Based on these gains in I_s, significant (approx. 2700Kg (6000lbs)) increases in payload can be achieved without reconfiguration of the vehicle, or the vehicle could be reconfigured to take advantage of the improved performance.
- These payload gains have a significant potential cost benefit of from $2M per launch based on projected low cost launch costs to as a high as $24M per launch based on current SSME launch costs.
- FRCMC also offer considerable potential for improvements in operating margin and component life which would provide major cost benefits, especially for reusable engines. However, this potential needs to be verified by testing.

The preferred FRCMC system for use in highly stressed turbine components is CVI C/SiC, which combines good environmental resistance with high strength and near-term applicability. For lower stress applications, SiC/SiC offers even better environmental resistance but sacrifices some strength. Either system could be produced using polar woven, 2D, or multiaxial fiber architectures as appropriate to the specific component. Alternative materials and processes have not demonstrated near-term capability for the production of complex components, such as those required for rocket engine turbomachinery, but may have long-term potential.

Further development is needed to successfully apply FRCMC technology to liquid rocket engine turbomachinery as addressed in this report. Development needs include:

- improvements in reinforcement fibers, matrix materials and interfaces,
- refinement of process control and NDE methods,
- reduction of process costs and production lead times,
- advancement of analysis capabilities to include micromechanical analysis,
- improvements in protective coatings,
- expansion of the materials properties data base to include life-related properties data,
- improvement of polar weaving technology and automation of 2D technology.

Finally, it is necessary to demonstrate FRCMC capabilities in an integrated, high temperature turbopump. This can be done initially by using a water-cooled, metallic manifold with internal FRCMC components as needed in the hot section. However, it will ultimately be necessary to develop flightweight, high temperature manifolds. FRCMC fabrication technology has not demonstrated near-term ability for the production of manifold shapes, and further development of low permeability coatings is needed for successful pressure vessel applications. Alternatively, new materials systems will have to be developed for manifold structures. Rocketdyne's plan addresses demonstration of FRCMC in the near-term and provides a longer-term development perspective for the implementation of FRCMC into earth-to-orbit rocket engine turbomachinery.

APPENDICES

A - FRCMC PROPERTIES DATA

B - DETAILED BENEFITS ANALYSIS

C - CONCEPTUAL DESIGN
- C1 - ENGINE/COMPONENT SELECTION
- C2 - FRCMC TURBOPUMP CONCEPTUAL DESIGN

D - DETAILED DESIGN AND SPECIFICATION
- D1 - FRCMC BLADE COUPON DESIGN
- D2 - PRELIMINARY MATERIALS SPECIFICATION

E - SUB-COMPONENT FABRICATION DETAILS
- E1 - SEP COUPON DESIGN
- E2 - FRCMC COUPON PHOTOMICROGRAPHS
- E3 - CAT SCAN RESULTS

F - PHASE II PROGRAM

APPENDIX A

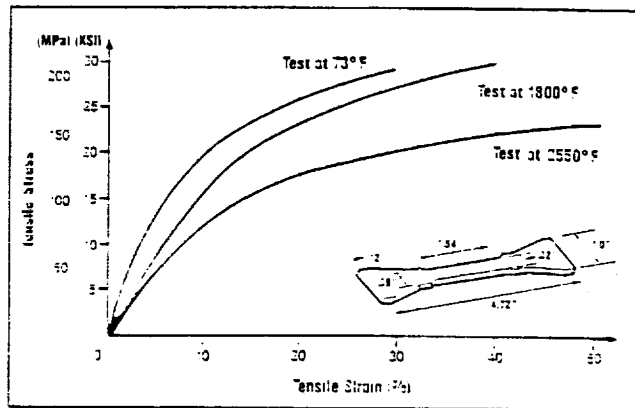
FRCMC PROPERTIES DATA

PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Stress Strain Curves

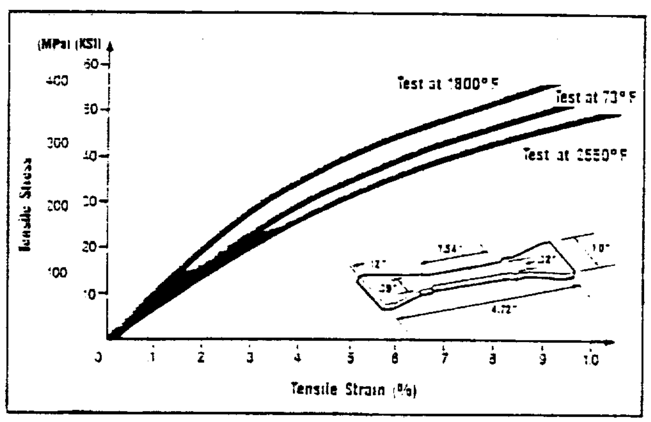
Tensile Stress vs. Strain SiC/SiC

Stress-Strain curve for SiC/SiC laminate using 0°/90° balanced Nicalon fabric.



Tensile Stress vs. Strain C/SiC

Stress-Strain curve for C/SiC laminate using 0°/90° balanced T-300 fabric.



Nicalon is a registered trademark of Nippon Carbon Co. Ltd. Japan
Thornel T-300 is a reg. trademark of Amoco Performance Products Inc. U.S.A.

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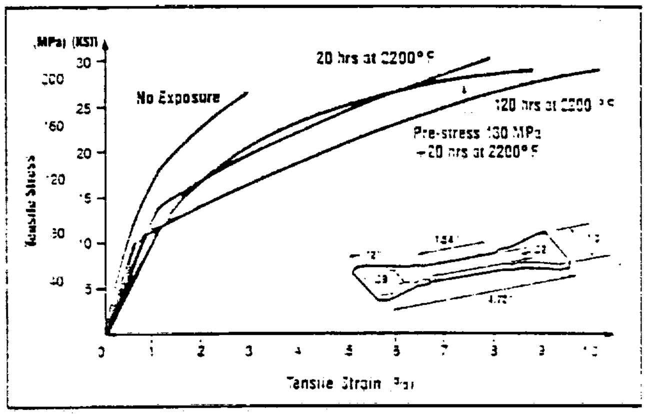
PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Stress Strain Curves

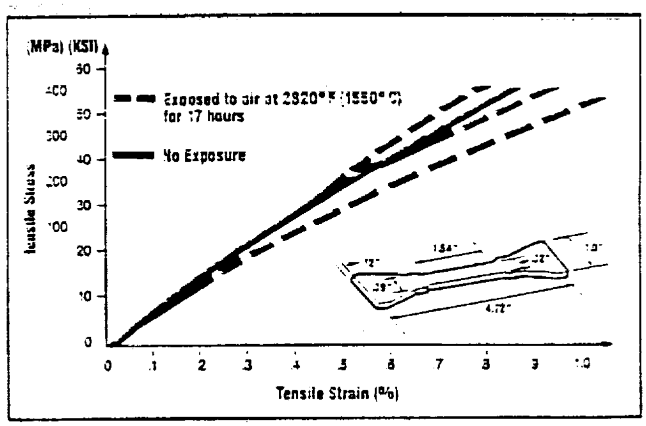
Tensile Stress vs. Strain Oxidized SiC/SiC

Stress strain curves for SiC/SiC laminate with 0°/90° balanced Nicalon® fabric. exposed to air at 2200°F for various times and tested at room temperature. Note the sample which was pre-stressed to 75% of ultimate to show the effect of microcracking on oxidation.



Tensile Stress vs. Strain Oxidized C/SiC

Stress-Strain curves for C/SiC laminate with 0°/90° balanced T-300 carbon fabric exposed to air for 17 hours at 2820°F. There is no notable degradation in tensile properties.



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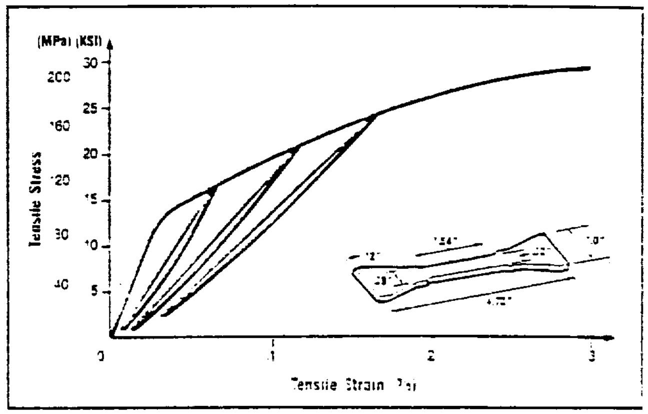
PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Stress Strain Curves

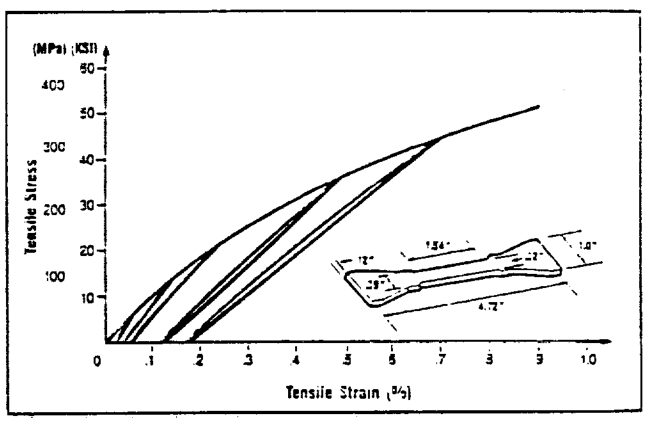
Cyclic Tensile Stress vs. Strain SiC/SiC

Stress-Strain curve for SiC/SiC laminate using 0°/90° balanced Nicalon® fabric. Tested at room temperature. Sample was unloaded three times during loading then brought to failure.



Cyclic Tensile Stress vs. Strain C/SiC

Stress-Strain curve for C/SiC laminate using 0°/90° balanced T-300® fabric. Tested at room temperature. Sampled was unloaded five times during loading then brought to failure. Note that the modulus stays nearly constant. Loading and unloading does not distort the total stress-strain curve.



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PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Physical Properties for SiC/SiC Laminates

PROPERTY	UNITS		TEMPERATURE					
			73°F	(23°C)	1832°F	(1000°C)	2552°F	(1400°C)
Fiber Content	%			40		40		40
Specific Gravity				2.5		2.5		2.5
Porosity	%			10		10		10
Tensile Strength	KSI	(MPa)	29	(200)	29	(200)	22	(150)
Elongation (Tensile)	%			0.3		0.4		0.5
Initial Young's Modulus (Tensile)	MSI	(GPa)	33	(230)	29	(200)	25	(170)
Poisson's Ratio								
V12	—			.05	NA		NA	
V13	—			.18	NA		NA	
Flexural Strength	KSI	(MPa)	44	(300)	58	(400)	41	(280)
Compressive Strength								
In Plane	KSI	(MPa)	84	(580)	70	(480)	44	(300)
Thru the Thickness	KSI	(MPa)	61	(420)	55	(380)	36	(250)
Shear Strength (Interlaminar)	KSI	(MPa)	5.8	(40)	5.1	(35)	3.6	(25)
Thermal Diffusivity								
In Plane	$10^{-6}FT^2/S$	$(10^{-6}m^2/S)$	130	(12)	54	(5)	54	(5)
Thru the Thickness	$10^{-6}FT^2/S$	$(10^{-6}m^2/S)$	65	(6)	20	(2)	20	(2)
Coefficient of Thermal Expansion								
In Plane	$10^{-6}/°F$	$(10^{-6}/°K)$	1.7	(3)	1.7	(3)	NA	
Thru the Thickness	$10^{-6}/°F$	$(10^{-6}/°K)$	.9	(1.7)	1.9	(3.4)	NA	
Fracture Toughness	$KSI\sqrt{IN}$	$(MPa\sqrt{m})$	27	(30)	27	(30)	27	(30)
Specific Heat	BTU/lbm °F	(J/kg °K)	.15	(620)	.29	(1200)	NA	
Total Emissivity				0.8		0.8		0.8
Thermal Conductivity								
In Plane	BTU/HR Ft·°F	$(Wm^{-1}\cdot K^{-1})$	11.0	(19.0)	8.8	(15.2)	NA	
Thru the Thickness	BTU/HR Ft·°F	$(Wm^{-1}\cdot K^{-1})$	5.5	(9.5)	3.3	(5.7)	NA	

These test coupons were tested both in inert atmosphere and in air for very short exposure time.
The samples were made with 0/90 balanced Nicalon' fabric, see page with test specimen description.

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PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Physical Properties for C/SiC Laminates

PROPERTY	UNITS		TEMPERATURE					
			73°F	(23°C)	1832°F	(1000°C)	2552°F	(1400°C)
Fiber Content	%			45		45		45
Specific Gravity				2.1		2.1		2.1
Porosity	%			10		10		10
Tensile Strength	KSi	(MPa)	51	(350)	51	(350)	48	(330)
Elongation (Tensile)	%			0.9	NA		NA	
Initial Young's Modulus (Tensile)	MSi	(GPa)	13	(90)	15	(100)	15	(100)
Poisson's Ratio								
V12	—			0	NA		NA	
V13	—			0	NA		NA	
Flexural Strength	KSi	(MPa)	73	(500)	102	(700)	102	(700)
Compressive Strength								
In Plane	KSi	(MPa)	84	(580)	87	(600)	102	(700)
Thru the Thickness	KSi	(MPa)	61	(420)	65	(450)	73	(500)
Shear Strength (Interlaminar)	KSi	(MPa)	5.0	(35)	5.0	(35)	5.0	(35)
Thermal Diffusivity								
In Plane	10^{-6}FT²/S	(10^{-6}m²/S)	118	(11)	75	(7)	86	(8)
Thru the Thickness	10^{-6}FT²/S	(10^{-6}m²/S)	54	(5)	20	(2)	20	(2)
Coefficient of Thermal Expansion								
In Plane	10^{-6}/°F	(10^{-6}/°K)	1.7	(3)	1.7	(3)	NA	
Thru the Thickness	10^{-6}/°F	(10^{-6}/°K)	2.8	(5)	2.8	(5)	NA	
Fracture Toughness	KSi $\sqrt{IN}$	(MPa$\sqrt{m}$)	32	(35)	32	(35)	32	(35)
Specific Heat	BTU/lbm °F	(J/kg °K)	.15	(620)	.33	(1400)	NA	
Total Emissivity				0.8		0.8		0.8
Thermal Conductivity								
In Plane	BTU/HR Ft·°F	(Wm⁻¹·K⁻¹)	8.3	(14.3)	11.9	(20.6)	NA	
Thru the Thickness	BTU/HR Ft·°F	(Wm⁻¹·K⁻¹)	3.8	(6.5)	3.4	(5.9)	NA	

These test coupons were tested both in inert atmosphere and in air for very short exposure time.
The samples were made with 0/90 balanced T-300' fabric. see page with test specimen description.

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PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Physical Properties for Novoltex®/SiC Laminates

PROPERTY	UNITS	TEMPERATURE					
		73°F	(23°C)	1832°F	(1000°C)	2552°F	(1400°C)
Fiber Content	%		24		24		24
Specific Gravity			2.3		2.3		2.3
Porosity	%		12		12		12
Tensile Strength	KSI (MPa)	12	(80)	15	(100)	20	(140)
Elongation (Tensile)	%		0.5		0.6		0.8
Initial Young's Modulus (Tensile)	MSI (GPa)	11	(75)	12	(85)	10	(70)
Poisson's Ratio							
V12	—		0.2	NA		NA	
V13	—		0.1	NA		NA	
Flexural Strength	KSI (MPa)	44	(300)	NA		NA	
Compressive Strength							
In Plane	KSI (MPa)	94	(650)	102	(700)	116	(800)
Thru the Thickness	KSI (MPa)	107	(740)	107	(740)	112	(770)
Shear Strength (Interlaminar)	KSI (MPa)	7.5	(50)	5.8	(40)	5.8	(40)
Thermal Diffusivity							
In Plane	10^{-6} FT^2/S (10^{-6} m^2/S)	130	(12)	54	(5)	54	(5)
Thru the Thickness	10^{-6} FT^2/S (10^{-6} m^2/S)	97	(9)	43	(4)	43	(4)
Coefficient of Thermal Expansion							
In Plane	10^{-6}/°F (10^{-6}/°K)	.9	(1.7)	1.8	(3.2)	2.2	(4)
Thru the Thickness	10^{-6}/°F (10^{-6}/°K)	1.3	(2.3)	2.3	(4.2)	2.9	(5.3)
Fracture Toughness	KSI $\sqrt{IN}$ (MPa$\sqrt{m}$)	NA		NA		NA	
Specific Heat	BTU/lbm °F (J/kg °K)	.15	(620)	.33	(1400)	NA	
Total Emissivity			0.8		0.8		0.8
Thermal Conductivity							
In Plane	BTU/HR Ft·°F (Wm^{-1}·K^{-1})	10.1	(17.5)	9.2	(15.9)	NA	
Thru the Thickness	BTU/HR Ft·°F (Wm^{-1}·K^{-1})	7.5	(13.0)	7.3	(12.6)	NA	

These test coupons were tested both in inert atmosphere and in air for very short exposure time.
The samples were made with Q3-D reinforced Novoltex® preform, see page with test specimen description.

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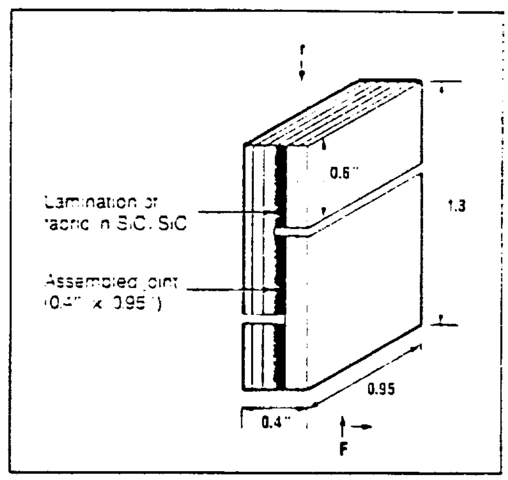
PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES

Effect of Test Temperature on Bond Shear Strength 2D-SiC/SiC as Produced vs. Bonded Specimens

Test Temperature °F	Shear Strength* (KSI)	
	As-produced composite	Bonded specimen
73	5.8	4.2
1800	5.1	3.0
2550	3.6	3.6

*Measured on bonded specimen as shown.



SiC/SiC Bond Shear Strength Retention After High Temperature Exposure in Air (measured at room temp.)

Conditions	Shear Strength* (KSI)
Control—No exposure	4.2
10 hours @ 2000°F 50 hours @ 2000°F	4.2 4.6
10 hours @ 2550°F 50 hours @ 2550°F	3.6 4.2

PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES

Fatigue Properties

A number of fatigue tests have been run on both C/SiC and SiC/SiC at both room and elevated temperatures. Like most composite materials these ceramic matrix composites show excellent fatigue properties.

C/SiC has been tested at the following conditions with *NO Failures:*

- Tension-Compression ± 22KSI at 70°F, 1 x 10^6 cycles.
- Tension-Tension 4–40 KSI at 2000°F, 1 x 10^6 cycles.
- Vibration 15 KSI mean load, ± 5KSI at 150 Hz at 70°F, 1 x 10^6 cycles.

SiC/SiC has been tested at the following conditions with *NO Failures:*

- Tension-Tension 1-10 KSI at 73°F, 1 x 10^6 cycles.
- Tension-Tension 1.5-15 KSI at 73°F, 1 x 10^6 cycles.

Thermal Shock and Cycling

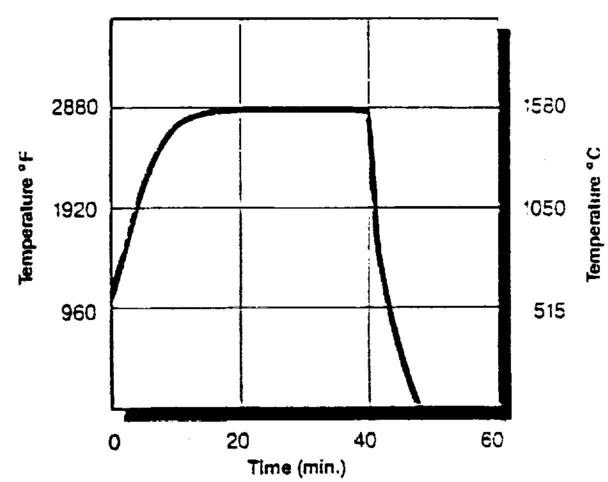
Both C/SiC and SiC/SiC have been tested in thermal shock with excellent results.

C/SiC was exposed to the cycle shown eight times with less than 2% loss in strength.

SiC/SiC had 100% strength retention after being exposed to the same cycle, with a 2200°F hold temperature, thirty times while being loaded to 15 KSI before each cycle.

5000°F Thermal Shock

C/SiC and Novoltex® /SiC plates with holes in them have been exposed to solid rocket exhaust gases in excess of 5000°F for short periods of time with no failures.



Thermal Shock Cycle for C/SiC

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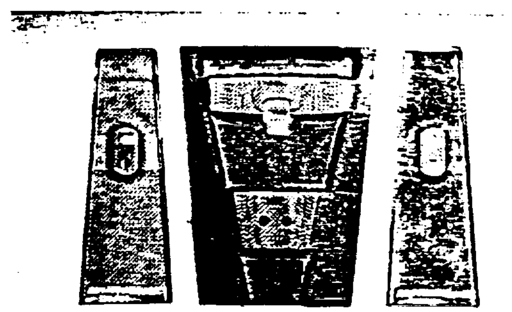
PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Component Experience: Engine Flaps

- Both inner and outer flaps have been tested in an engine for over 300 hours with excellent results.
- These flaps have been tested in acoustic fatigue to 4 x 10^6 cycles with no failures.
- A Mirage 2000 with a SNECMA M-53 Engine with these flaps will fly at the Paris Air Show in 1989.



SiC/SiC Inner Flaps. These flaps have been tested for hundreds of hours in a SNECMA M-53 Engine.

SiC/SiC Inner Hot Flaps (shown)

- SiC/SiC solid reinforced construction.
- Two part flap and seal configuration shown.
- Tested in an engine at 1550° F for 300 hours with higher temperature afterburner cycles.

C/SiC Outer Flaps (see picture on brochure cover)

- C/SiC complex hollow construction.
- Advanced Metal/Ceramic Fittings to mount flap to engine.
- Tested in an engine at 950° F for 300 hours with higher temperature afterburner cycles.

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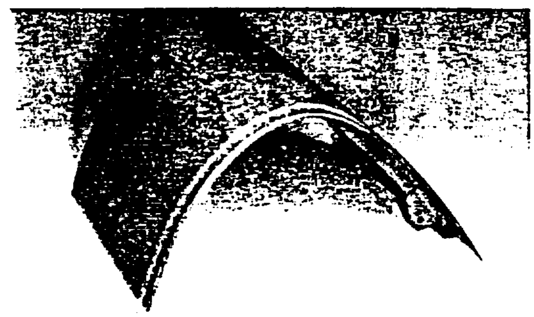
PRELIMINARY ENGINEERING DATA

CERAMIC MATRIX COMPOSITES
Component Experience: Hermes Leading Edge

- Lightweight structural component fabricated in one piece with C/SiC.
- Part size (18"x18") shows large complex shape capability.
- Successfully tested under the following conditions:
  - 16 cycles with a 20-minute hold at 2800° F.
  - 2 cycles with 12- and 16-minute holds at 3200° F.
  - All cycles performed in air with tension/compression loading.
- Excellent strength retention.



C/SiC Leading Edge. This ½ scale Hermes leading edge section has been tested at 3200° F.

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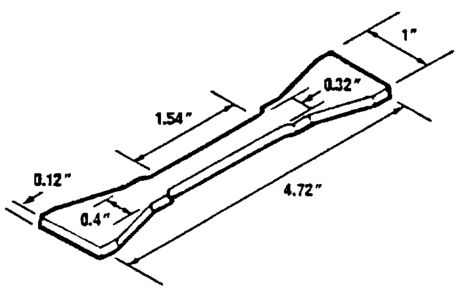
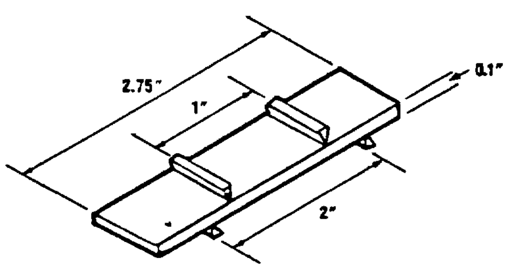
PRELIMINARY ENGINEERING DATA

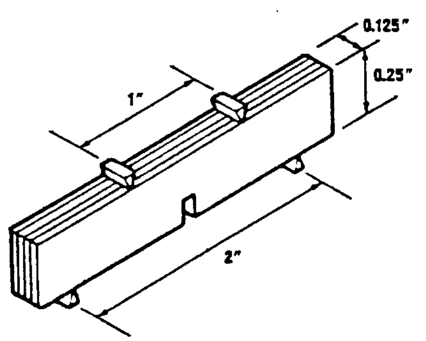
CERAMIC MATRIX COMPOSITES
Test Specimen Description



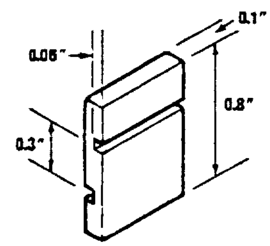
Tensile



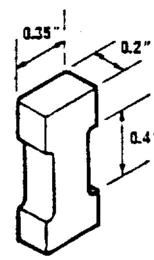
Flex



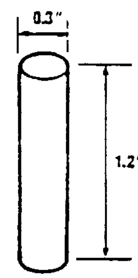
Fracture Toughness



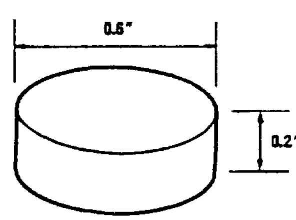
Shear Strength



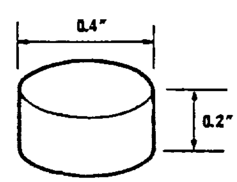
Compressive Strength



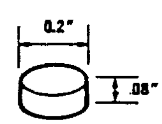
Thermal Expansion



Total Emissivity



Thermal Diffusivity



Specific Heat

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APPENDIX B

DETAILED BENEFITS ANALYSIS

INTRODUCTION

A study was made to assess the benefits of ceramic turbine blades in two typical and current conceptual engine systems: the Space Transportation Main Engine (STME) and the Space Transporation Booster Engine (STBE). The first is a LOX/H2 system while the second is LOX/CH4. Their corresponding engine flow schematics are shown in Figure 1 and Figure 2 respectively. The current baselines for each were used. The study adhered to current technology and engine design ground rules used in all engine subsystems except the turbines. Table 1 and Table 2 are the summary of the important design groundrules and turbopump operating limits for STME and STBE engines. In the study, the improvements in engine specific impulse, engine weight, and chamber pressure of using ceramic turbine blades in the turbine inlet temperature range of 1600-3000R were established.

SUMMARY

Engine vacuum specific impulse was shown to increase as turbine inlet temperature increased from 1600-3000R; more significantly for LOX/CH4 (approximately 10 seconds) than for LOX/H2 (approximately 5 seconds), see Table 3.

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Table 3. Performance Increase at Vacuum from 1600-3000 R.

	LOX/CH4 STBE	LOX/H2 STME
Maximum I_s Engine, sec	10.3	6.6
Baseline Engine, sec	10.3	5.2
Minimum Weight Engine, sec.	6.5	3.7

Table 4. Performance Increase at Sea Level from 1600-3000 R.

	LOX/CH4 STBE	LOX/H2 STME
Maximum I_s, sec	14.3	12.0
Baseline Engine, sec	9.7	3.2
Minimum Weight Engine, sec	6.7	4.8

Vacuum Performance

The higher specific impulse increase with turbine inlet temperature at vacuum for the LOX/CH4 engine was caused by the larger reduction of gas-generator-induced specific impulse losses when compared to the LOX/H2 engine. This characteristic makes LOX/CH4 engines have a higher performance improvement potential with ceramic turbine technology than LOX/H2. This tend is true not only for the maximum specific impulse cases but also for the fixed Pc cases, i.e. baseline Pc engine and the minimum weight engine cases (Table 3). The larger reduction in gas generator performance loss for LOX/CH4 is due to the interaction of three factors: methane's higher gas generator flow fraction, methane's larger change in gas generator flow fraction with temperature, and methane's lower gas generator ideal specific impulse faction loss ($\Delta I_s gg/I_s$ ideal). These differences in turn relate directly to the poorer gas generator gas properties of LOX/CH4, and the better gas generator gas properties for LOX/H2.

Sea Level Performance

The sea level performance (Table 4) has all the same trends with turbine inlet temperature as the vacuum performance discussed above. In addition, sea level performance is affected by the nozzle exit area induced drag at sea level. This drag is directly proportional to nozzle expansion area ratio (ε) and inversely proportional to chamber pressure. Chamber pressures for the LOX/CH4 cases were generally higher than the LOX/H2 cases, and area ratios generally lower, leading to lower exit area drag losses at sea level. This interplay of chamber pressure and area ratio led to the performance increase differences between the two engines indicated in Table 4. Performance again is generally higher for LOX/CH4 than for LOX/H2.

DISCUSSION

The data generated is summarized in Table 5 and Table 6 for the LOX/CH4 engine and in Table 7 for the LOX/H2 engine. Plots of the data appear in Figs. 3 through 8 for LOX/CH4 and Figs. 9 through 14 for LOX/H2. They consist of engine vacuum specific impulse (I_s), sea level I_s, weight, chamber pressure, engine diameter, and engine length. Detailed pump and turbine parameters for the STBE engine study are shown in Table 8 and Table 9.

Mixture ratio shifting effect

"Open-cycle" type of engines like GG cycle engines, their turbine exhaust gasses are not introduced through the main injector but through the downstream expansion in the nozzle. It does have a performance penalty on the secondary GG flow. It also causes thrust chamber mixture ratio differ from engine mixture ratio as shown in Fig. 15. In this study only fuel-rich gas generators are used. The reason for this is that the specific heat of the fuel-rich gases is higher than that for oxidizer-rich gases and therefore is more energetic. The secondary flowrate for fuel-rich turbine gas is much less and the engine performance is much higher.

As shown in Fig. 15, the thrust chamber MR is always higher than engine MR at a given temperature and the difference is inversely proportional to the turbine inlet temperature. For a given engine MR, the thrust chamber MR at low turbine inlet temperature is always higher than the high temperature one. (See Fig. 15). This thrust chamber MR shifting effect will affect the thrust chamber performance and, consequently, the engine performance.

Figure 16 shows the thrust chamber theoretical performance (ODE) vs. thrust chamber MR for LOX/CH4 system. The optimal performance occurs at a MR of 3.55. If the engine MR is 3.14, the corresponding thrust chamber MR is 3.25 for a temperature at 3000° R and is 4.0 for a 1600° R temperature. The high temperature one shows the advantage in the thrust chamber performance. A reversed trend is observed for a engine MR at 2.9 (see Fig. 16). However, if the engine MR is above the optional MR (= 3.55) then the high temperature engines always show benefit in thrust chamber performance. The reason for this is that the high temperature thrust chamber MR is always closed to the optimal MR. A similar trend is also observed for LOX/H2 engines as shown in Fig. 17, but its optimal performance occurs at a MR of 4.75. Table 10 and Table 11 summarize this effect for LOX/CH4 and LOX/H2 engines respectively.

Performance Trends

As discussed in the summary, the performance trends and differences between the LOX/CH4 and LOX/H2 engines have to do with the thrust chamber MR, the gas generator (GG) gas properties differences (vacuum performance) and with these GG gas properties, chamber pressure and area ratio differences for the sea level performance. All cases (maximum I_s, baseline Pc, and minimum weight) have similar trends and differences. The LOX/CH4 cases were run at constant thrust chamber mixture ratios. Later on, the constant Pc baseline case was run at constant engine mixture ratio. Trends with the latter case indicated I_s performance with LOX/CH4 peaks at a turbine inlet temperature near 2500 R for the engine mixture ratio of 2.82 (Fig. 3), at 2800 R for the engine mixture ratio of 3.14, and at near 3000 R for an engine mixture ratio of 3.37. This upward movement of the peak performance indicates engine mixture ratio approaches thrust chamber mixture ratio as the turbine temperature increases because less gas generator flow is required at higher temperatures. The thrust chamber ratio of 3.5 is the mixture ratio for maximum performance of LOX/CH4 propellants (Fig. 16); and the optimum condition to operate the engine. A higher turbine inlet temperature allows operation at this optimum thrust chamber mixture ratio for the thrust level chosen. Similar trends occur at sea level (Fig. 4).

Engine Weights

Engine weights are shown in Fig. 5 for constant Pc, maximum I_s, optimum I_s and minimum weight cases. The minimum weight case takes advantage of the chamber pressure (Pc) at which engine weight minimizes to insure a minimum weight as turbine inlet temperature is raised above 1600 R. This is usually a Pc near 2500 psia. For the LOX/CH4 this low Pc penalizes specific impulse as can be seen from Fig. 3. The maximum I_s case penalizes engine weight by seeking to increase chamber pressure to maximize I_s. The optimum I_s drawn on Fig. 5 from data in Fig. 3, uses the baseline Pc (3253 psia) to arrive at a compromise in weight savings and I_s gains. From the standpoint of payload and since I_s impacts payload more significantly than engine weight does (1375 lbs of engine weight for a second of I_s for STME Engine, and 510 lbs for STBE Engine) the optimum I_s curve of Fig. 3 represents a preferred approach.

Ceramic Material

Due to hub-to-tip ratio problem, the fuel turbine pitchline velocity is limited at 1055 ft/sec for STBE engine at turbine inlet temperature of 1600° R. this pitchline velocity value is below the regular TMPA 286 material strength (1450 ft/sec at 1600° R). Using ceramic material will not show any benefit for STBE engine at 1600° R temperature. For STME engine at same temperature, the turbine pitchline velocity is 1630 ft/sec which is above the TMPA286 material strength. Therefore, ceramic material does show benefit for STME engine at 1600° R temperature. At temperature above 2000° R, ceramic materials start to benefit the STBE engine.

Table 1

Study Groundrules for Ceramic Reinforced STME Engine

• Thrust Chamber	
propellant:	LOX/H2
Thrust (vacuum)	580 klb
chamber pressure:	2250 psia (baseline)
MR (engine)	6.0
Nozzle exit pressure	3.11 psia
ηc^*	0.98
Nozzle % length	80%
Cycle	gas generator cycle
Regen coolant:	H2
Coolant split	(% of engine flow)
MCC	18%
Nozzle	32%
• Gas Generator	
GG topoff u/s of main valves	
L/L injector	
Fuel rich gas generator	
• Turbopump	
Tank pressure	LH2 = 24.5 psia LOX = 47.0 psia
No boost pumps	
No kick pumps	
Main pump stages	(centrifugal type)
LH2	2 stage
LOX	1 stage
Impeller tip speed	LH2 = 1900 ft/sec LOX = 900 ft/sec
Inlet/outlet diameter ratio	LH2 = .75 LOX = .75
Suction specific speed	LH2 = 58631 LOX = 40 049
Turbopump arrangements	Dual shaft - series turbines with fuel turbine upstream
Turbine type	
LH2	2 stage velocity compound
LOX	1 stage pressure compound

Table 1 (continued)

Turbine pressure ratio (overall)	15.0
Turbine inlet temperature	= 1600 ~ 3000 R
Turbine blade AN^2	see attached curve
Turbine pitchline velocity	see attached curve
Turbine hub/tip ratio	max = 0.92 min = 0.60

- System pressure drops

GG Pc = MCC Pc (to maximize Isp)		
Injector LOX ΔP	0.15 Pc	
LH2 ΔP	0.1 Pc	(MCC & GG)
Line ΔP	0.1 Pc	
Control Valve ΔP = 0.06 x Pc	(MOV, GGFV, GGOV)	
On-Off valve ΔP = 0.06 x Pc	(MFV)	

Table 2

STUDY GROUNDRULES FOR CERAMIC REINFORCED STBE ENGINE

• Thrust Chamber		
Propellants	:	LOX/CH4
Thrust (sea level)	:	750 K lb
Chamber Pressure	:	3253 psia (baseline case)
MR (Thrust Chamber)	:	3.5
Nozzle exit pressure	:	6 psia
η_{c^*}	:	0.98
Nozzle % length	:	80
Cycle	:	Gas Generator Cycle
Regen Coolant	:	CH4
Coolant Split	:	(% of engine flow)
MCC	:	30%
Nozzle	:	36%
• Gas Generator		
GG tapeoff U/S of main valves		
L/L injector		
Fuel rich gas generator		
• Turbopump		
Tank Pressure	:	CH4 = 45 psia LOX = 65 psia
No boost pumps		
No kick pumps		
Main pump stages CH4 = 2 stgs LOX = 1 stgs	:	(Centrifugal type)
Impeller tip speed	:	CH4 = 1476 ft/sec LOX = 900 ft/sec
Inlet/outlet diameter ratio	:	CH4 = 0.75 LOX = 0.75
Suction specific speed	:	rpm x $Gpm^{1/2}$ /$ft^{3/4}$ CH4 = 40500 LOX = 37500
Turbopump arrangements	:	
Dual shaft - series turbines with Fuel turbine U/S		
Turbine type	:	CH4 = 2 Spc LOX = 2 Spc
Turbine pressure ratio (over-all)		= 20
Turbine inlet temperature		= 1600 ~ 3000°R
Turbine blade AN^2	:	See attached curve
Turbine pitchline velocity	:	See attached curve
Turbine hub/tip ratio	:	
Max	:	0.92
Min	:	0.60
Hydrostatic bearing		

Table 2 (continued)

- System Pressure drops

GG Pc	=	MCC Pc (to maximize Isp)
Injector Δp	=	0.2 x Pc (MCC & GG)
Line Δp	=	0.1 x Pc
Control valve Δp	=	0.1 x Pc (MOV, GGFV, GGOV)
ON - off valve Δp	=	0.05 x Pc (MFV)

Table 5

CERAMIC REINFORCED TURBINE FOR STBE ENGINE
LOX/CH4 - GG CYCLE, REGEN COOLED

Fsl=750 Klb, MR(T/C)=3.5, Pe=6 psia

	PARAMETERS	1600	2200	2600	3000
Baseline PC	Engine MR	2.821	3.140	3.298	3.372
	Chamber Pressure	3253	3253	3253	3253
	Area Ratio	54.5	53.8	53.2	53.0
	Engine Vac. Isp	346.9	353.6	356.2	357.2
	Engine S.L. Isp	304.4	310.6	313.2	314.1
	Engine Length	152.7	153.1	153.3	153.3
	Engine Diameter	95.3	94.9	94.5	94.4
	Engine Weight	8284	8078	8015	7990
Maximize Isp	Engine MR	2.978	3.129	3.245	3.329
	Chamber Pressure	2620	3340	3922	4024
	Area Ratio	45.5	55.0	61.9	62.9
	Engine Vac. Isp	347.9	353.7	356.9	358.2
	Engine S.L. Isp	303.0	310.9	315.8	317.3
	Engine Length	159.6	152.3	147.7	147.1
	Engine Diameter	98.1	94.5	92.0	91.5
	Engine Weight	8005	8109	8330	8336
Minimize Weight	Engine MR	3.042	3.251	3.353	3.396
	Chamber Pressure	2337	2396	2448	2464
	Area Ratio	41.5	42.1	42.6	42.8
	Engine Vac. Isp	347.7	351.9	353.8	354.2
	Engine S.L. Isp	301.5	305.7	307.7	308.2
	Engine Length	163.3	162.7	162.1	161.8
	Engine Diameter	99.8	99.2	98.7	98.6
	Engine Weight	7952	7841	7789	7763

Table 6

CERAMIC REINFORCED TURBINE FOR STBE ENGINE

LOX/CH4 - GG CYCLE, REGEN COOLED

Fsl=750 Klb, Pe=6 psia (Constant Engine MR)

	PARAMETERS	1600	2200	2600	3000
Baseline PC MReng=3.14	T/C MR	3.948	3.5	3.32	3.247
	Chamber Pressure	3253	3253	3253	3253
	Area Ratio	60.5	53.8	50.5	49.6
	Engine Vac. Isp	344.9	353.6	354.9	355.1
	Engine S.L. Isp	298.5	310.6	313.8	314.5
	Engine Length.	161.4	153.1	150.6	149.2
	Engine Diameter	100.4	94.9	92.3	91.6
	Engine Weight	8271	8078	8050	8027
Baseline PC MReng=2.821	T/C MR	3.5	3.118		2.904
	Chamber Pressure	3253	3253		3253
	Area Ratio	54.5	48.3		45.4
	Engine Vac. Isp	346.9	349.2		348.0
	Engine S.L. Isp	304.4	310.1		310.6
	Engine Length	152.7	147.4		145.2
	Engine Diameter	95.3	90.5		88.5
	Engine Weight	8284	8274		8291

Table 7

CERAMIC REINFORCED TURBINE FOR STME ENGINE

LOX/H2-GG CYCLE, REGEN COOLED
(F = 580 Klb, MR = 6.0)

	PARAMETERS	1600	2200	2600	3000
BASELINE	Chamber pressure	2250	2250	2250	2250
	Area Ratio	62.0	62.0	62.0	62.0
	Engine Vac. Isp	435.9	439.1	440.2	441.1
	Engine S.L. Isp	345.9	347.9	348.6	349.1
	Engine Length	165.1	165.6	165.8	165.9
	Engine Diameter	101.8	102.1	102.2	102.3
	Engine Weight	6172	6180	6222	6191
MAXIMIZE Isp	Chamber Pressure	2511	2992	3185	3238
Pe = 3.11 psia	Area Ratio	67.3	76.6	80.3	80.8
	Engine Vac. Isp	436.4	440.2	441.8	443.0
	Engine S.L. Isp	349.5	356.9	359.8	361.5
	Engine Length	160.8	155.4	153.2	152.3
	Engine Diameter	100.0	97.4	96.5	96.1
	Engine Weight	6167	6343	6435	6753
MINIMIZE WEIGHT	Chamber Pressure	1966		1981	1978
	Area Ratio	54.58		53.6	53.3
	Engine Vac. Isp	435.1		438.3	438.8
	Engine S.L. Isp	343.7		347.9	348.5
	Engine Length	166.9		165.5	165.2
	Engine Diameter	102.7		101.8	101.6
	Engine Weight	6116		6112	6111

Table 8

PUMP PARAMETERS FOR CERAMIC REINFORCED STBE ENGINE LOX/CH4-GG CYCLE,REGEN COOLED (Fsl=750 klb,Pe=6 psia)

	PARAMETERS	1600		2200		2600		3000	
		CH4	LOX	CH4	LOX	CH4	LOX	CH4	LOX
Baseline PC	Flowrate	645	1819	583	1832	557	1838	546	1842
	Discharged Pressure	6311	4554	6311	4554	6311	4554	6311	4554
	Volumetric Flowrate(GPM)	10970	11463	9923	11543	9479	11582	9292	11606
	Headrise (FT)	34203	9075	34203	9075	34203	9075	34203	9075
	RPM	16770	10800	17662	10615	17063	10751	16816	10754
	Specific Speed	1175	1244	1177	1227	1111	1244	1084	1296
	Impeller Diameter	14.2	16.2	13.5	16.4	13.9	16.2	14.0	16.2
	Impeller Tip Speed	1038	762	1039	760	1032	762	1029	762
	Efficiency	79.02	79.63	79.04	79.48	78.35	79.63	78.04	79.65
	HP	50742	37690	45890	38021	44220	38080	43523	38151
	DN , mm*rpm*10^{6}	1.44	0.98	1.44	0.97	1.40	0.98	1.37	0.98
Maximize Isp	Flowrate	622	1853	584	1828	559	1816	546	1818
	Discharged Pressure	4684	3668	6562	4676	8474	5491	8858	5633
	Volumetric Flowrate(GPM)	10587	11679	9938	11520	9519	11442	9289	11457
	Headrise (FT)	25323	7284	35573	9322	46010	10970	48105	11257
	RPM	17086	10579	17641	10773	17998	10825	18223	10796
	Specific Speed	1473	1450	1142	1219	940	1080	909	1057
	Impeller Diameter	12.3	15.1	13.7	16.4	15.0	17.4	15.1	17.7
	Impeller Tip Speed	920	696	1056	770	1177	824	1200	833
	Efficiency	81.05	80.94	78.69	79.42	75.90	77.99	75.32	77.70
	HP	35348	30321	48012	39010	61659	46427	63398	47881
	DN , mm*rpm*10^{6}	1.31	0.89	1.47	0.98	1.61	1.05	1.64	1.06
Minimize Weight	Flowrate	616	1872	577	1876	560	1878	554	1880
	Discharged Pressure	4047	3272	4175	3354	4291	3428	4326	3449
	Volumetric Flowrate(GPM)	10472	11799	9820	11826	9528	11833	9418	11850
	Headrise (FT)	21844	6484	22544	6650	23180	6798	23368	6842
	RPM	17203	10671	17765	10661	18035	10657	18141	10649
	Specific Speed	1648	1604	1609	1574	1576	1548	1567	1541
	Impeller Diameter	11.6	14.3	11.3	14.5	11.3	14.6	11.3	14.6
	Impeller Tip Speed	868	666	879	673	888	679	891	680
	Efficiency	81.72	81.58	81.60	81.48	81.48	81.38	81.45	81.35
	HP	29912	27054	28992	27843	28964	28516	28877	28751
	DN , mm*rpm*10^{6}	1.26	0.87	1.28	0.87	1.30	0.88	1.30	0.88

Table 9

TURBINE PARAMETERS FOR CERAMIC REINFORCED STBE ENGINE LOX/CH4-GG CYCLE,REGEN COOLED (Fsl=750 klb,Pe=6 psia)

	PARAMETERS	1600		2200		2600		3000	
		CH4	LOX	CH4	LOX	CH4	LOX	CH4	LOX
Baseline PC	TYPE	2SPC	2SPC	2SPC	2SPC	2SPC	2SPC	2SPC	2SPC
	FLOWRATE	178.7	178.6	119.2	119.1	92.6	92.7	83.6	83.6
	INLET TEMPERATURE	1600	1325	2200	1833	2600	2130	3000	2411
	ADMISSION	1.0	1.0	0.61	1.0	0.52	1.0	0.36	1.0
	PRESSURE RATIO	5.091	3.968	4.864	4.153	4.725	4.276	4.234	4.771
	VELOCITY RATIO	0.287	0.339	0.293	0.313	0.239	0.262	0.284	0.230
	EFFICIENCY	74.71	77.63	70.30	76.70	65.93	73.33	68.55	69.91
	PITCH DIAMETER	14.4	22.3	16.7	25.9	16.2	24.9	20.1	23.6
	PITCHLINE VELOCITY	1055	1050	1289	1201	1208	1167	1475	1106
	HUB/TIP RATIO	0.92	0.85	0.92	0.91	0.91	0.92	0.92	0.92
	AN**2 (in*rpm)**2x10^{-10}	1.90	3.08	2.82	2.42	2.61	2.11	3.42	2.01
Maximize Isp	TYPE	2SPC	2SPC	2SPC	2SPC	2.SPC	2SPC	2SPC	2SPC
	FLOWRATE	132.6	132.5	123.0	123.0	117.7	117.7	111.1	111.1
	INLET TEMPERATURE	1600	1342	2200	1828	2600	2084	3000	2355
	ADMISSION	1.0	1.0	0.45	1.0	0.38	1.0	0.42	1.0
	PRESSURE RATIO	4.559	4.431	4.956	4.076	5.091	3.968	4.992	4.046
	VELOCITY RATIO	0.288	0.366	0.328	0.341	0.307	0.287	0.274	0.272
	EFFICIENCY	74.61	77.60	70.51	77.37	69.53	75.27	68.62	74.20
	PITCH DIAMETER	13.7	25.6	18.8	27.6	20.1	26.2	18.7	26.2
	PITCHLINE VELOCITY	1025	1181	1450	1298	1583	1238	1487	1234
	HUB/TIP RATIO	0.92	0.90	0.91	0.92	0.92	0.92	0.92	0.92
	AN**2 (in*rpm)**2x10^{-10}	1.70	2.58	3.79	2.51	4.26	2.29	4.00	2.28
Minimize Weight	TYPE	2SPC	2SPC	2SPC	2SPC	2SPC	2SPC	2SPC	2SPC
	FLOWRATE	115.0	115.1	81.6	81.7	68.0	68.1	68.4	68.3
	INLET TEMPERATURE	1600	1349	2200	1861	2600	2181	3000	2523
	ADMISSION	1.0	1.0	0.35	1.0	0.35	1.0	0.64	1.0
	PRESSURE RATIO	4.363	4.63	4.325	4.671	3.833	5.27	3.723	5.425
	VELOCITY RATIO	0.290	0.335	0.33	0.287	0.25	0.197	0.194	0.158
	EFFICIENCY	74.66	77.40	69.39	75.27	66.62	65.13	60.10	57.73
	PITCH DIAMETER	13.5	23.6	18.1	24.7	15.1	20.2	12.2	17.3
	PITCHLINE VELOCITY	1017	1098	1407	1149	1188	940	967	804
	HUB/TIP RATIO	0.92	0.89	0.89	0.92	0.87	0.90	0.87	0.86
	AN**2 (in*rpm)**2x10^{-10}	1.63	2.55	4.23	2.22	3.67	1.91	2.27	2.10

Table 10: STBE Engine Performance vs Turbine Inlet Temperature
LOX/CH4 @ Pc = 3253, MR_{Eng} = 2.821

T (°R)	1600	2200	3000
$MR_{T/C}$	3.5	3.12	2.90
$(I_{sp})_{T/C}$	363.2	359.2	354.2
$(I_{sp})_{gg}$	139.0	161.8	182.7
$(I_{sp})_{Eng}$	346.9	349.2	348.0

Table 11: STME Engine Performance vs Turbine Inlet Temperature
LOX/H2 @ Pc = 2250, MR_{Eng} = 6.0

T (°R)	1600	2200	2600	3000
$MR_{T/C}$	6.763	6.497	6.396	6.312
$(I_{sp})_{T/C}$	442.1	443.8	444.3	444.7
$(I_{sp})_{gg}$	265.0	290.7	305.1	315.5
$(I_{sp})_{Eng}$	435.9	439.1	440.3	441.1

STME GAS GENERATOR CYCLE

REGEN BASELINE

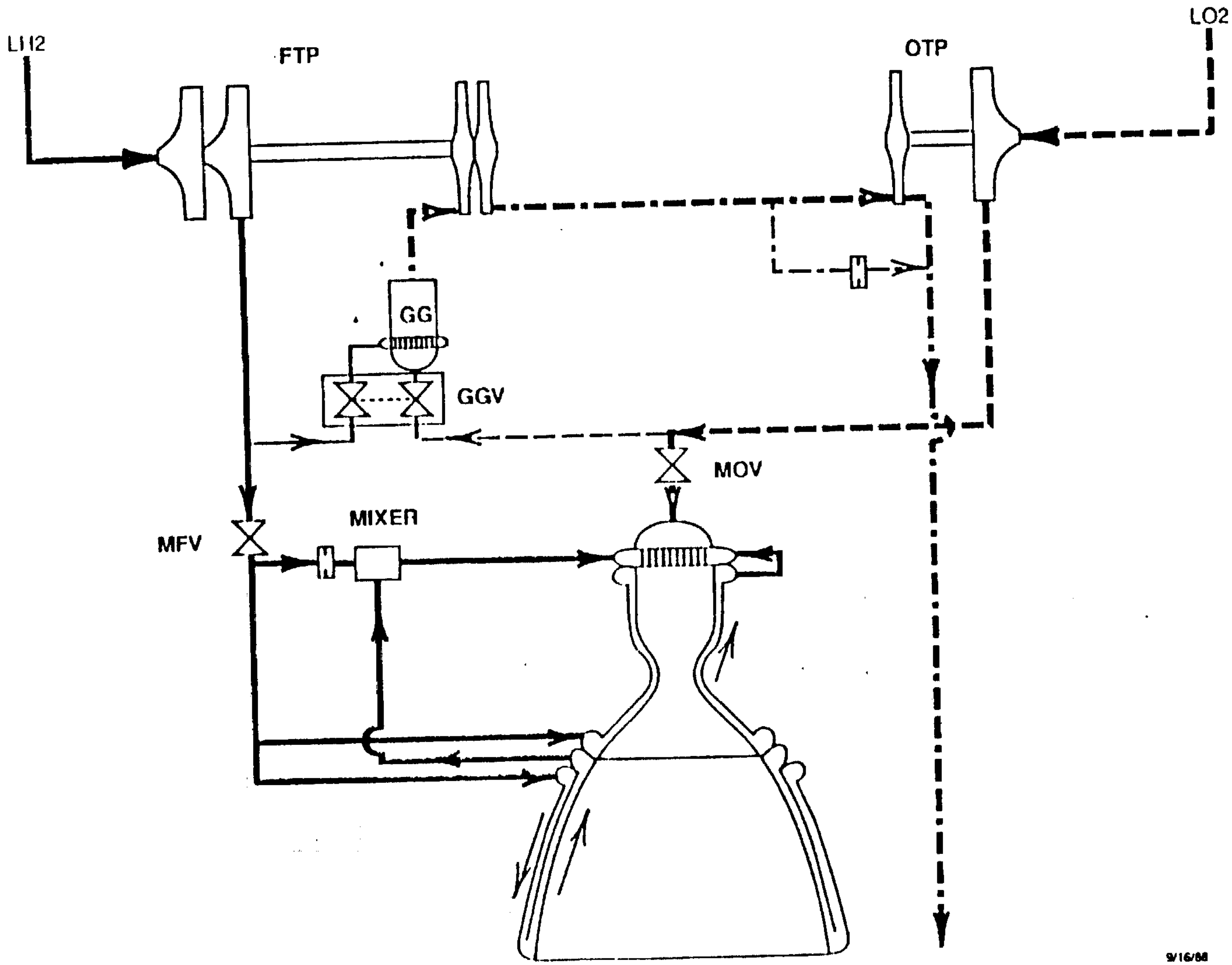

Figure 1

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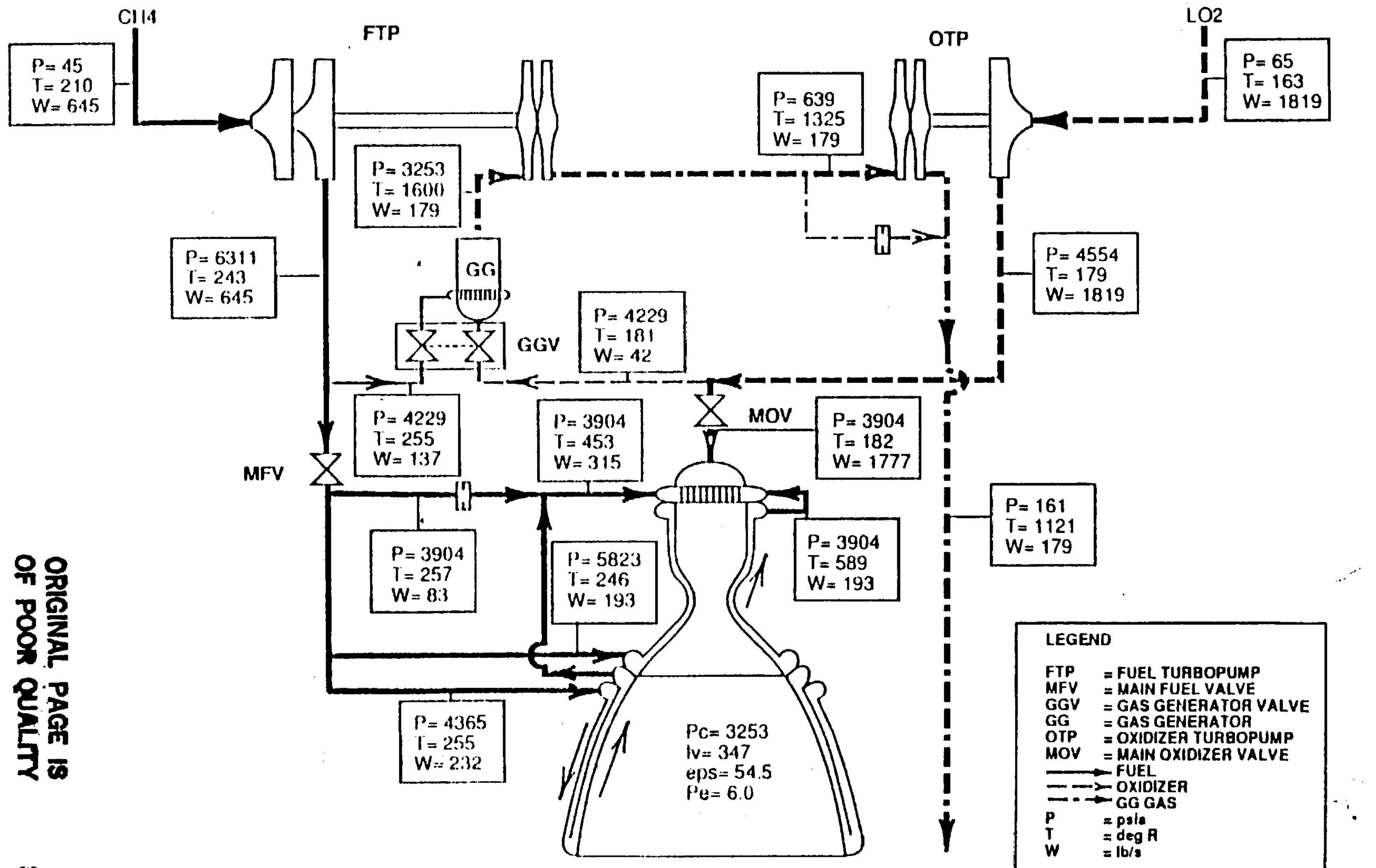
STBE LOX/CH4 GAS GENERATOR CYCLE

REGEN BASELINE 750 K SEA LEVEL



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Rockwell International
Rocketdyne Division

Figure 2

9/8/88

D88-821

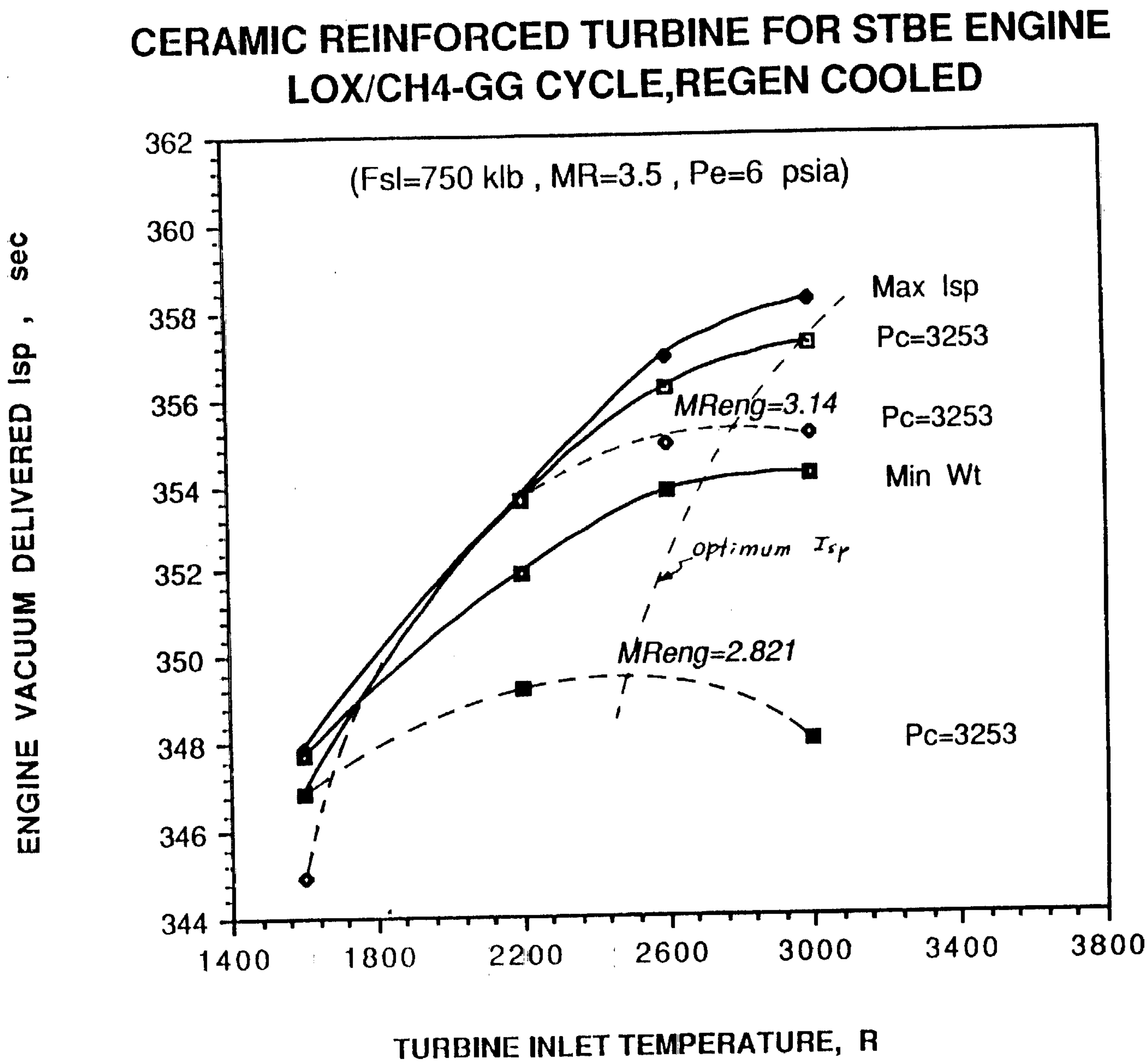

CERAMIC REINFORCED TURBINE FOR STBE ENGINE
LOX/CH4-GG CYCLE,REGEN COOLED
(Fsl=750 klb , MR=3.5 , Pe=6 psia)
Max Isp
Pc=3253
MReng=3.14
Pc=3253
Min Wt
optimum Isp
MReng=2.821
Pc=3253
ENGINE VACUUM DELIVERED Isp , sec
TURBINE INLET TEMPERATURE, R
362
360
358
356
354
352
350
348
346
344
1400
1800
2200
2600
3000
3400
3800

Figure 3

CERAMIC REINFORCED TURBINE FOR STBE ENGINE
LOX/CH4-GG CYCLE,REGEN COOLED

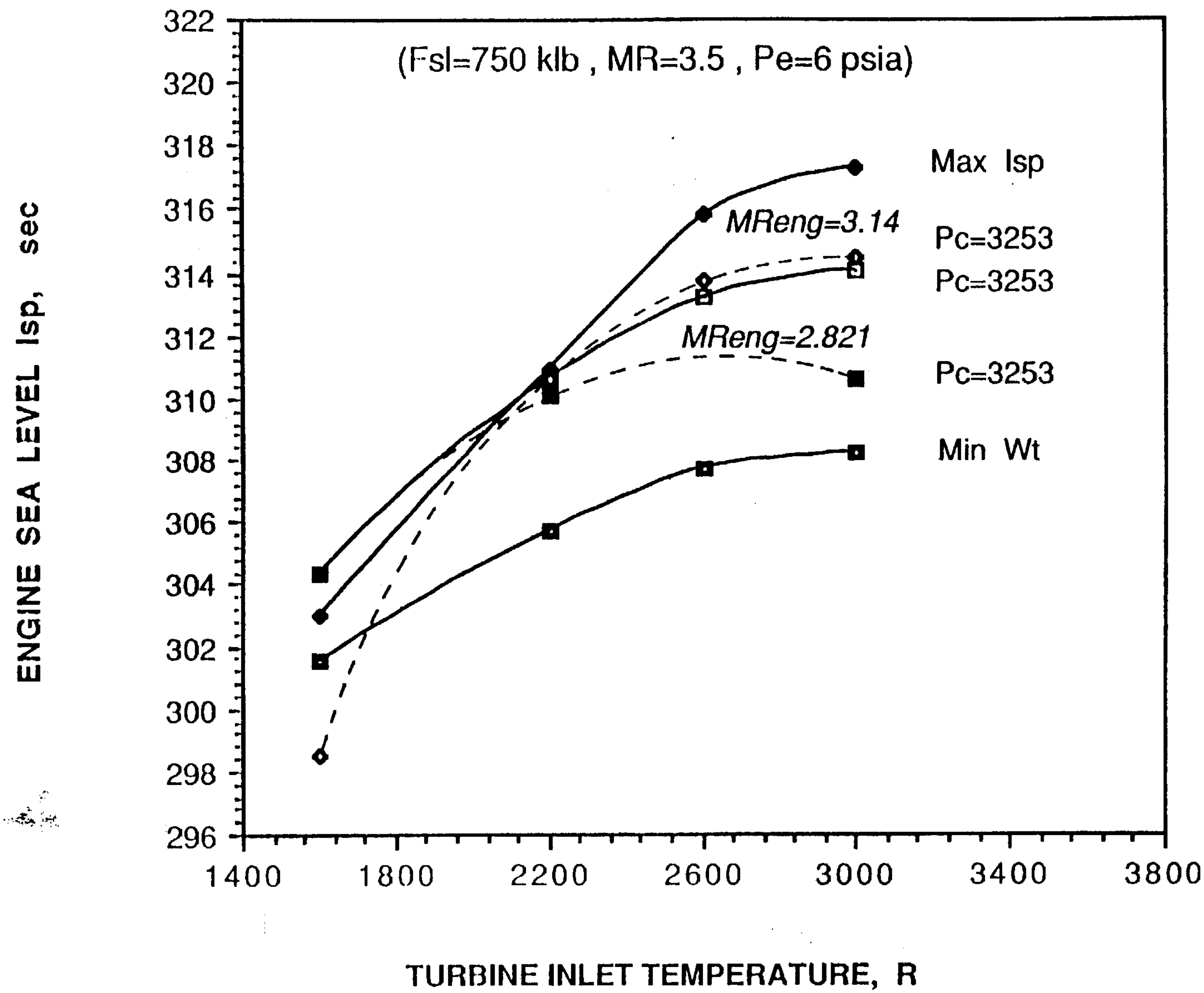
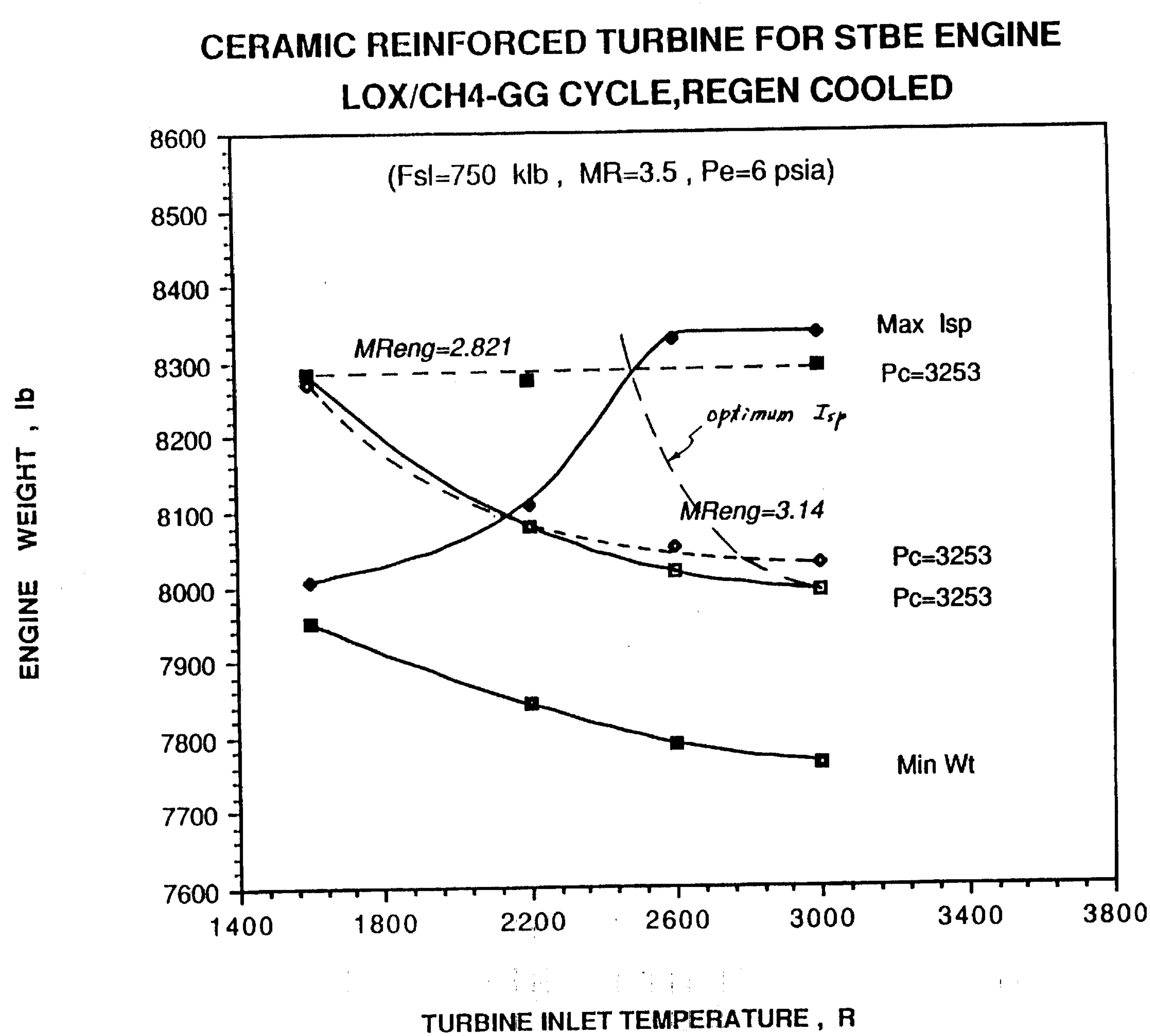
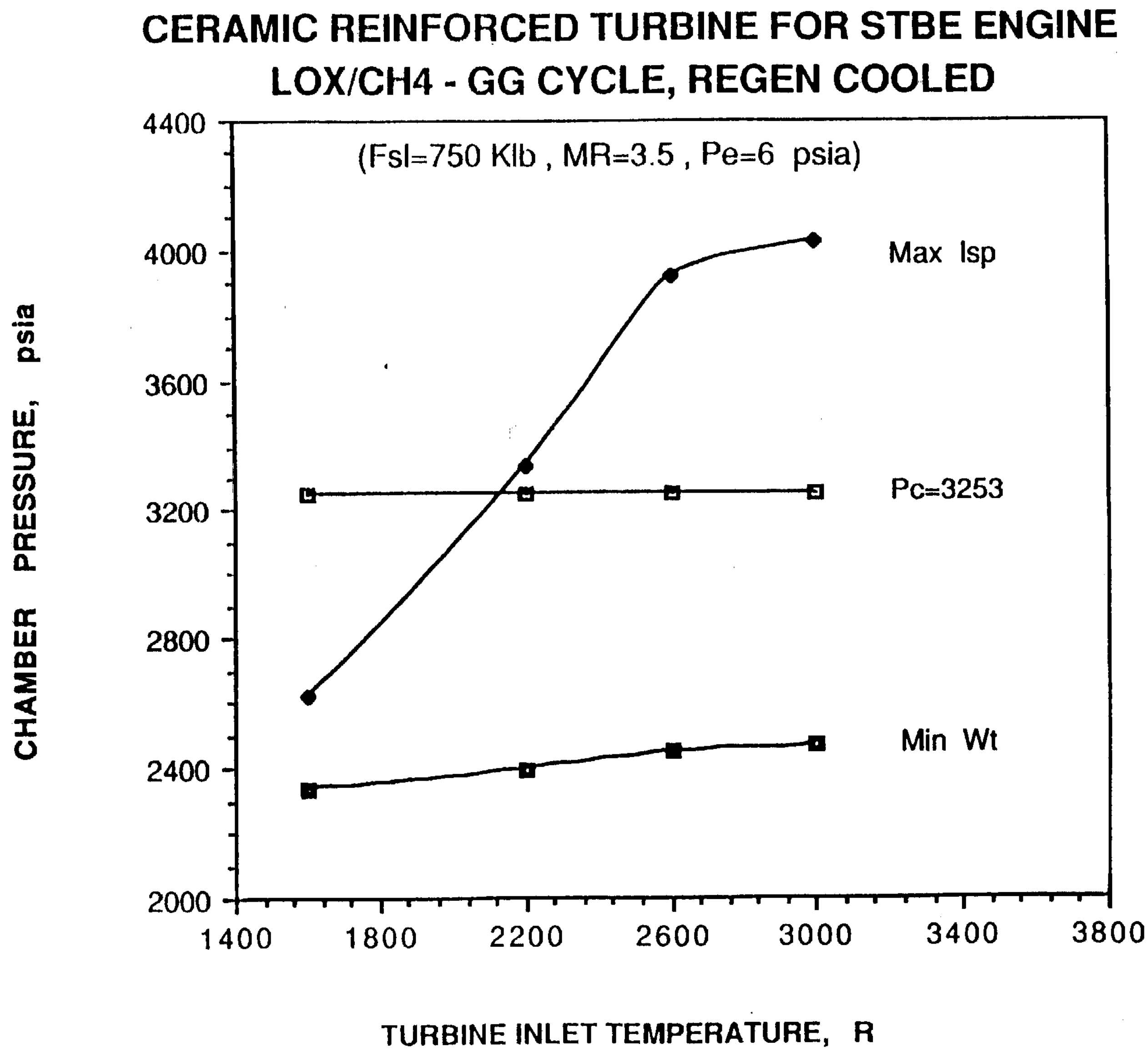


Figure 4



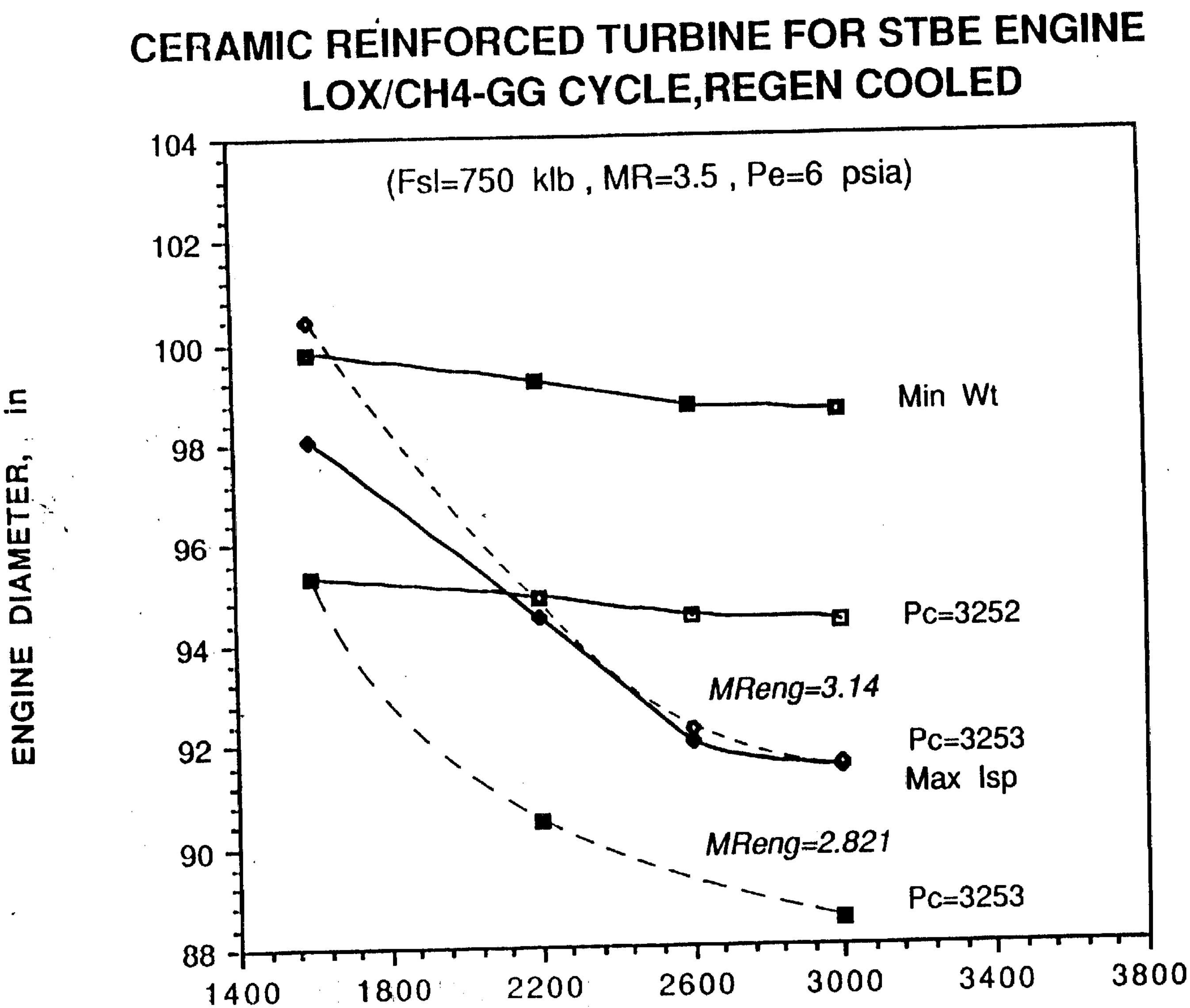
CERAMIC REINFORCED TURBINE FOR STBE ENGINE
LOX/CH4-GG CYCLE,REGEN COOLED
(Fsl=750 klb , MR=3.5 , Pe=6 psia)
MReng=2.821
MReng=3.14
optimum Isp
Max Isp
Pc=3253
Pc=3253
Pc=3253
Min Wt
ENGINE WEIGHT , lb
8600
8500
8400
8300
8200
8100
8000
7900
7800
7700
7600
1400
1800
2200
2600
3000
3400
3800
TURBINE INLET TEMPERATURE , R

Figure 5



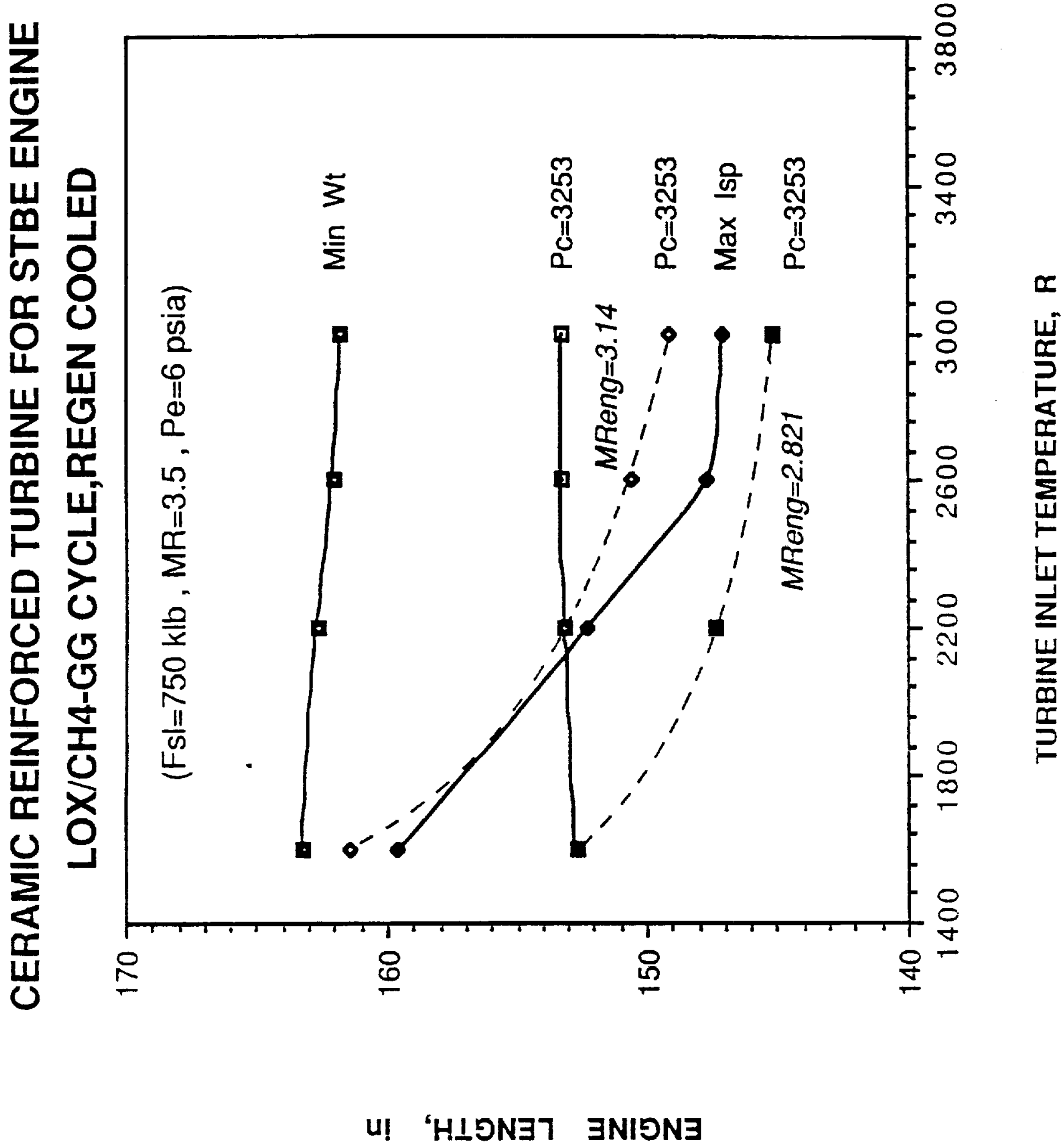
CERAMIC REINFORCED TURBINE FOR STBE ENGINE
LOX/CH4 - GG CYCLE, REGEN COOLED
(Fsl=750 Klb , MR=3.5 , Pe=6 psia)
Max Isp
Pc=3253
Min Wt
CHAMBER PRESSURE, psia
4400
4000
3600
3200
2800
2400
2000
1400
1800
2200
2600
3000
3400
3800
TURBINE INLET TEMPERATURE, R

Figure 6



CERAMIC REINFORCED TURBINE FOR STBE ENGINE
LOX/CH4-GG CYCLE,REGEN COOLED
(Fsl=750 klb , MR=3.5 , Pe=6 psia)
Min Wt
Pc=3252
MReng=3.14
Pc=3253
Max Isp
MReng=2.821
Pc=3253
ENGINE DIAMETER, in
TURBINE INLET TEMPERATURE, R
104
102
100
98
96
94
92
90
88
1400
1800
2200
2600
3000
3400
3800

Figure 7



CERAMIC REINFORCED TURBINE FOR STBE ENGINE
LOX/CH4-GG CYCLE,REGEN COOLED
(Fsl=750 klb , MR=3.5 , Pe=6 psia)
Min Wt
Pc=3253
MReng=3.14
Pc=3253
Max Isp
Pc=3253
MReng=2.821
ENGINE LENGTH, in
TURBINE INLET TEMPERATURE, R
170
160
150
140
1400
1800
2200
2600
3000
3400
3800

Figure 8

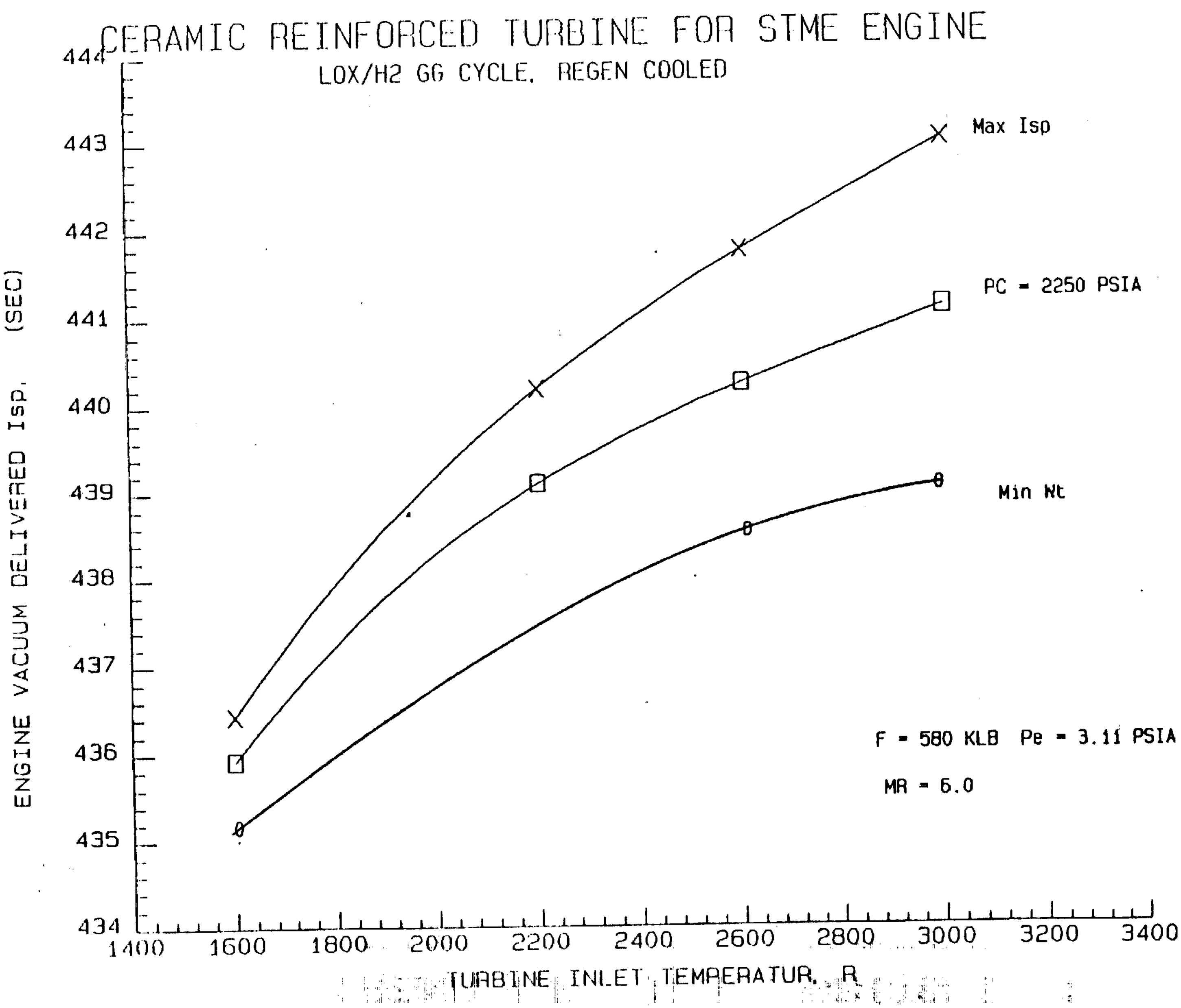
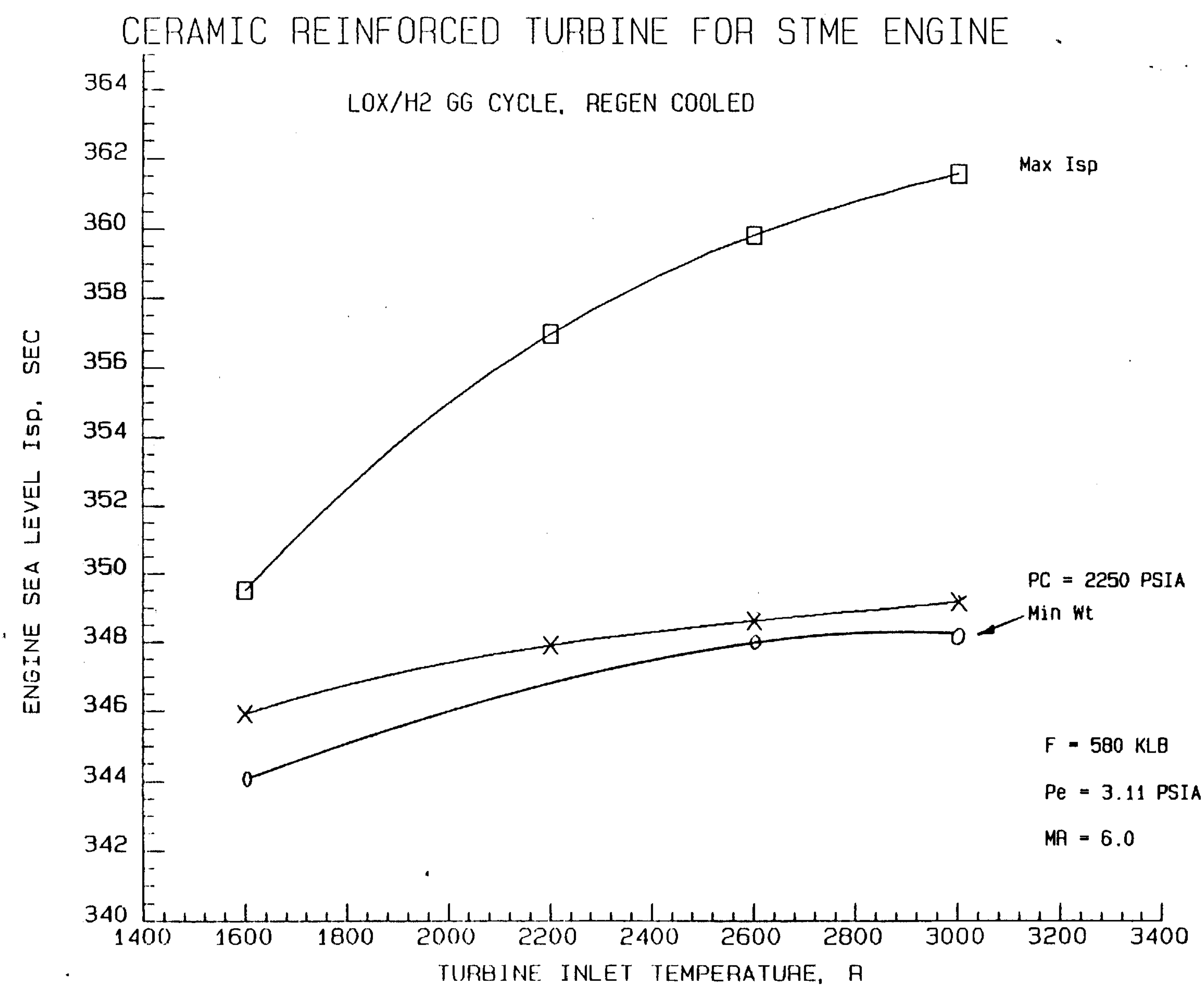

CERAMIC REINFORCED TURBINE FOR STME ENGINE
LOX/H2 GG CYCLE, REGEN COOLED
Max Isp
PC = 2250 PSIA
Min Wt
F = 580 KLB Pe = 3.11 PSIA
MR = 6.0
ENGINE VACUUM DELIVERED Isp. (SEC)
TURBINE INLET TEMPERATUR, R
434
435
436
437
438
439
440
441
442
443
444
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200
3400

Figure 9



CERAMIC REINFORCED TURBINE FOR STME ENGINE
LOX/H2 GG CYCLE, REGEN COOLED
Max Isp
PC = 2250 PSIA
Min Wt
F = 580 KLB
Pe = 3.11 PSIA
MR = 6.0
ENGINE SEA LEVEL Isp, SEC
364
362
360
358
356
354
352
350
348
346
344
342
340
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200
3400
TURBINE INLET TEMPERATURE, R

Figure 10

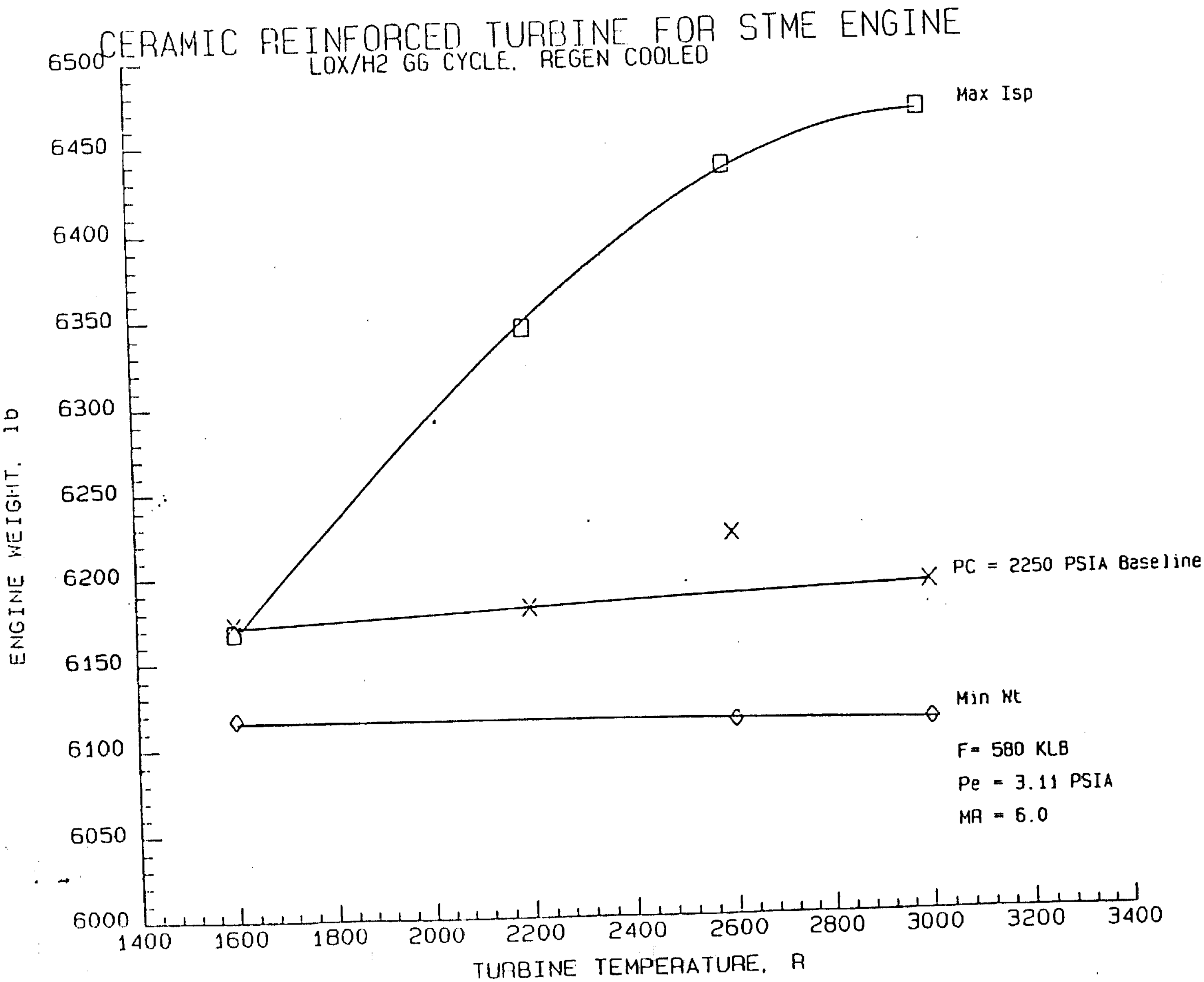

CERAMIC REINFORCED TURBINE FOR STME ENGINE
LOX/H2 GG CYCLE. REGEN COOLED
Max Isp
PC = 2250 PSIA Baseline
Min Wt
F= 580 KLB
Pe = 3.11 PSIA
MR = 6.0
ENGINE WEIGHT, lb
6500
6450
6400
6350
6300
6250
6200
6150
6100
6050
6000
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200
3400
TURBINE TEMPERATURE, R

Figure 11

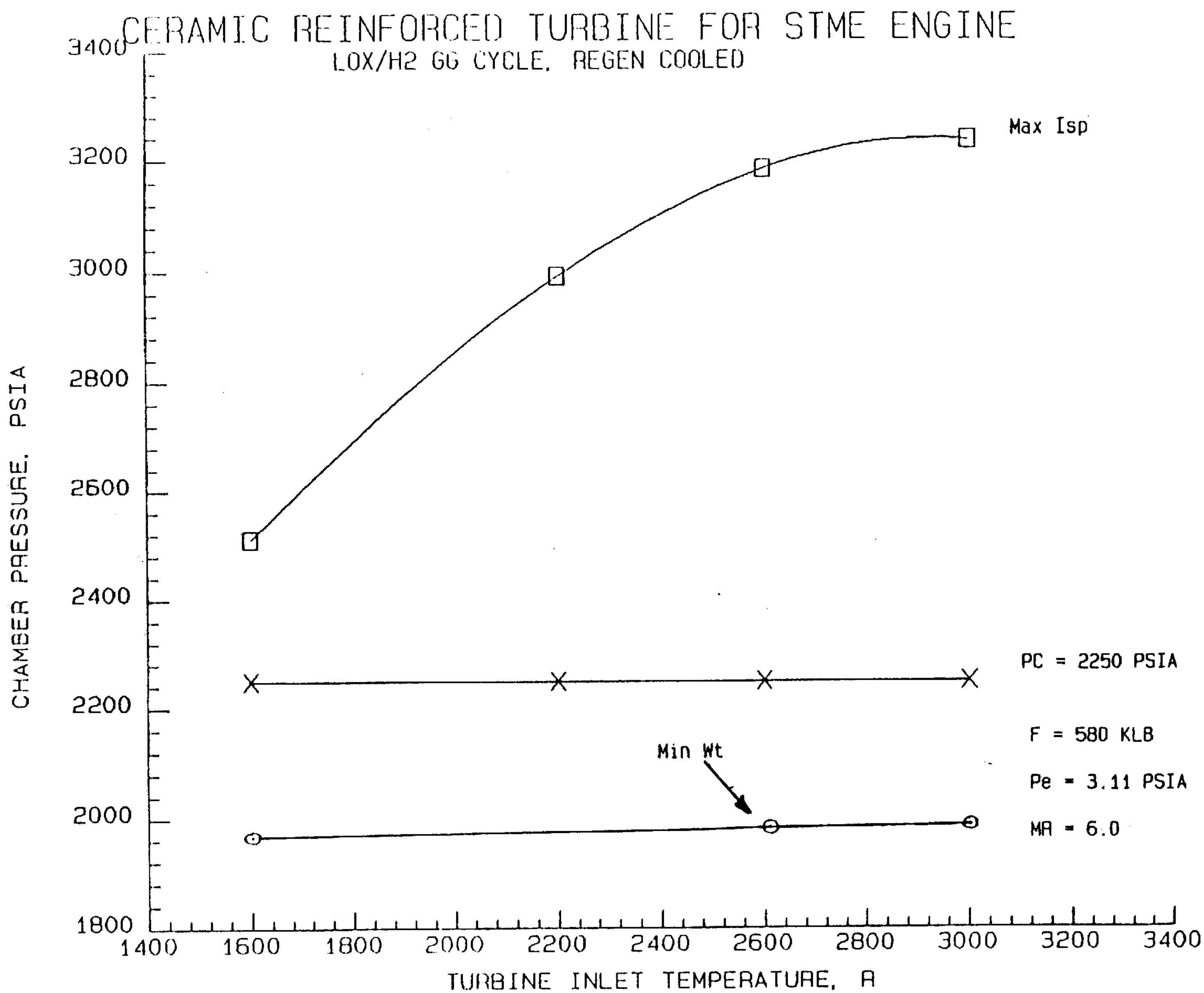

CERAMIC REINFORCED TURBINE FOR STME ENGINE
LOX/H2 GG CYCLE, REGEN COOLED
CHAMBER PRESSURE, PSIA
3400
3200
3000
2800
2600
2400
2200
2000
1800
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200
3400
TURBINE INLET TEMPERATURE, R
Max Isp
PC = 2250 PSIA
F = 580 KLB
Pe = 3.11 PSIA
MR = 6.0
Min Wt

Figure 12

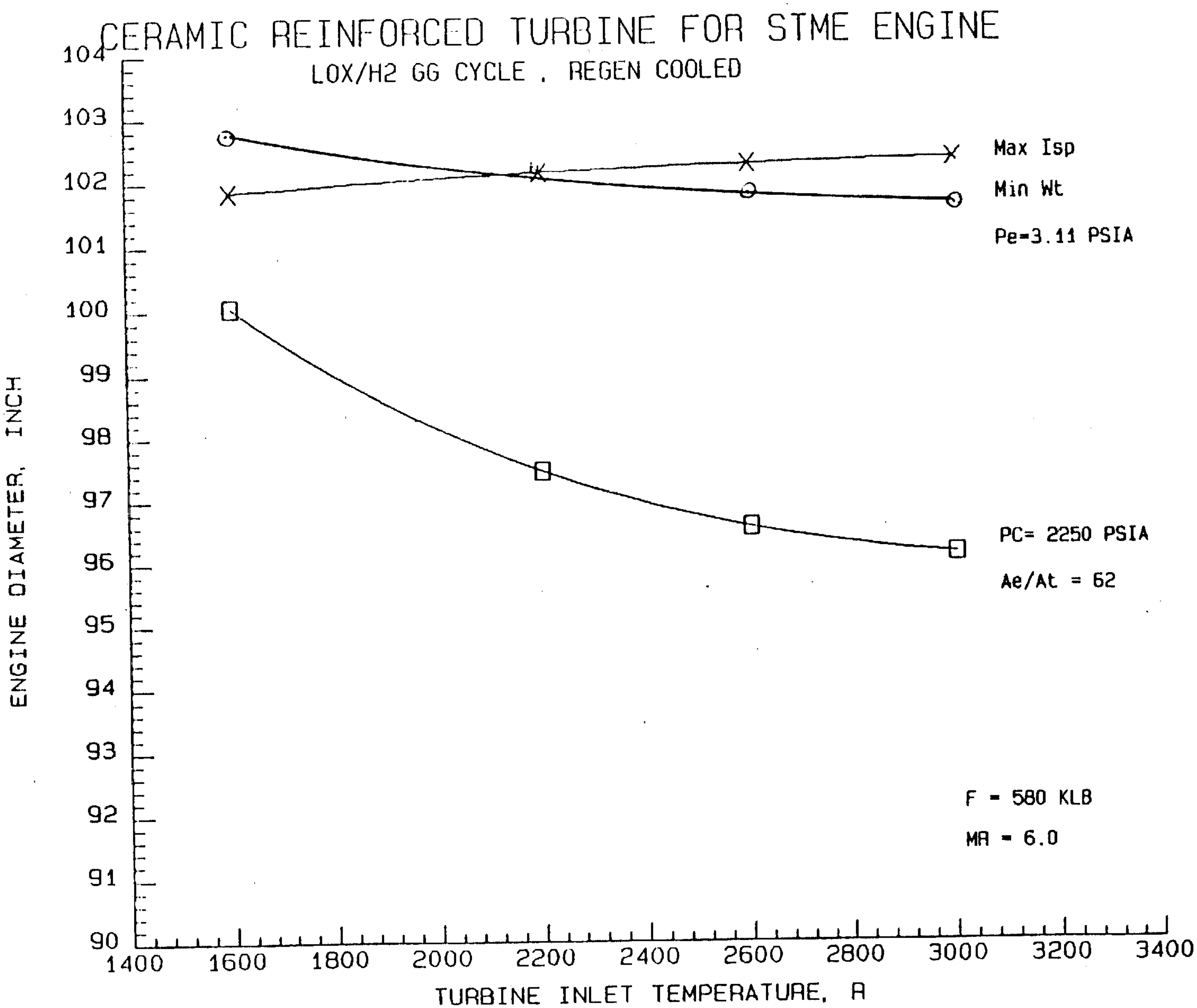

CERAMIC REINFORCED TURBINE FOR STME ENGINE
LOX/H2 GG CYCLE , REGEN COOLED
Max Isp
Min Wt
Pe=3.11 PSIA
PC= 2250 PSIA
Ae/At = 62
F = 580 KLB
MR = 6.0
ENGINE DIAMETER, INCH
90
91
92
93
94
95
96
97
98
99
100
101
102
103
104
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200
3400
TURBINE INLET TEMPERATURE, R

Figure 13

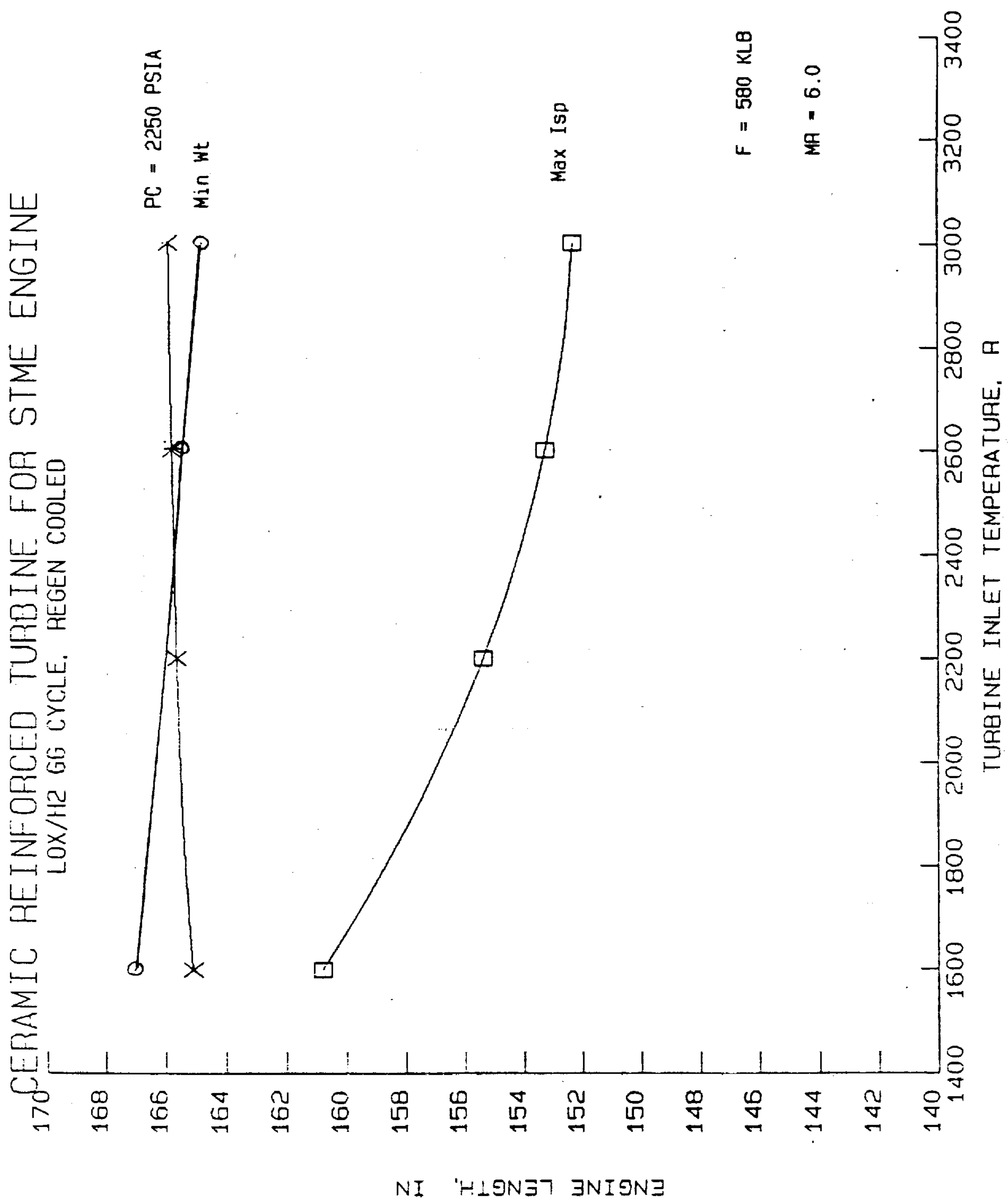
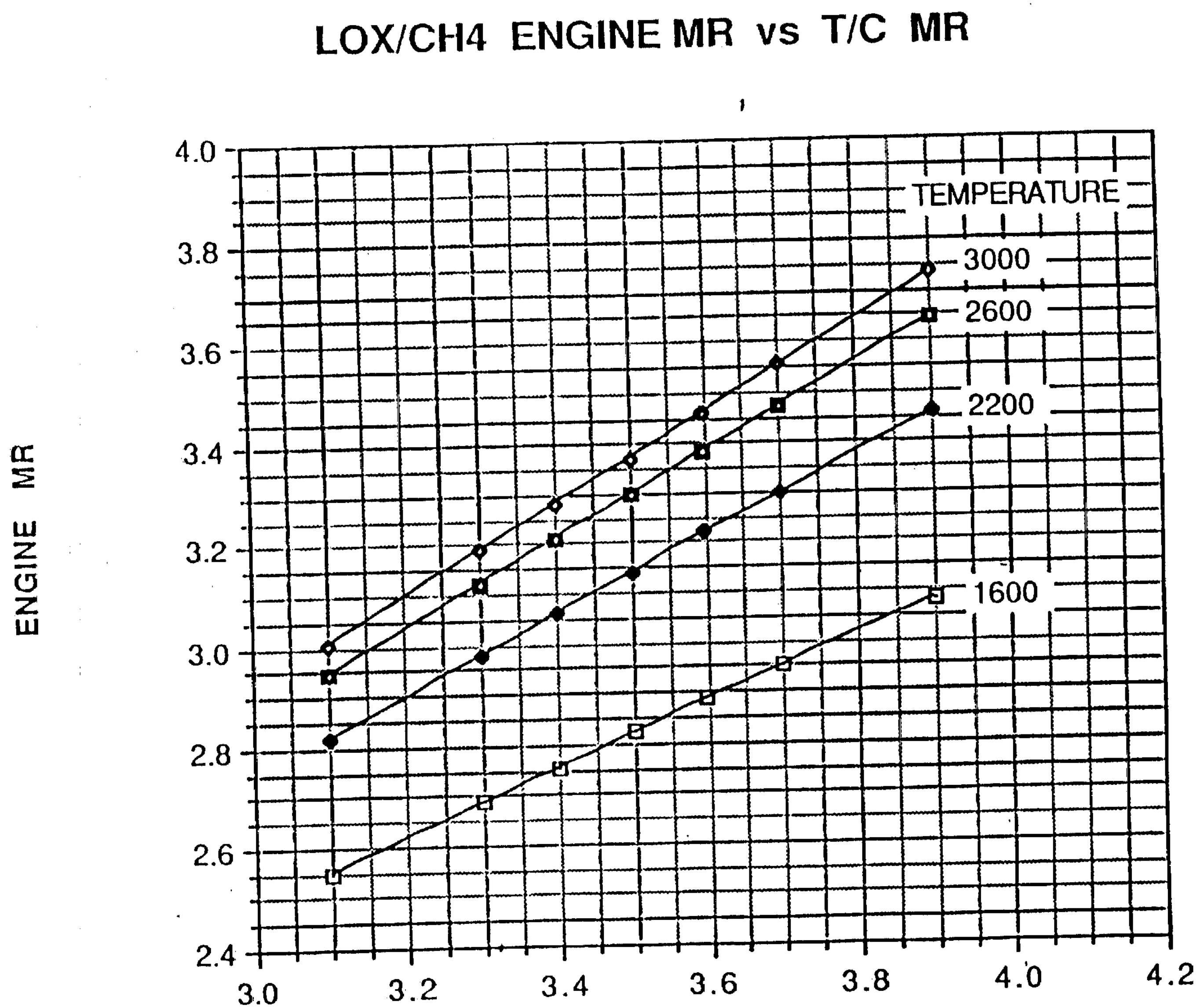

CERAMIC REINFORCED TURBINE FOR STME ENGINE
LOX/H2 GG CYCLE, REGEN COOLED
PC = 2250 PSIA
Min Wt
Max Isp
F = 580 KLB
MR = 6.0
ENGINE LENGTH, IN
TURBINE INLET TEMPERATURE, R
170
168
166
164
162
160
158
156
154
152
150
148
146
144
142
140
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200
3400

Figure 14


LOX/CH4 ENGINE MR vs T/C MR
TEMPERATURE
3000
2600
2200
1600
ENGINE MR
THRUST CHAMBER MR
4.0
3.8
3.6
3.4
3.2
3.0
2.8
2.6
2.4
3.0
3.2
3.4
3.6
3.8
4.0
4.2
Figure 15

LOX/CH4 THEORETICAL PERFORMANCE(ODE)

Pc=3253 psia,Hf=-19.42 kcal/mole,Area Ratio=57

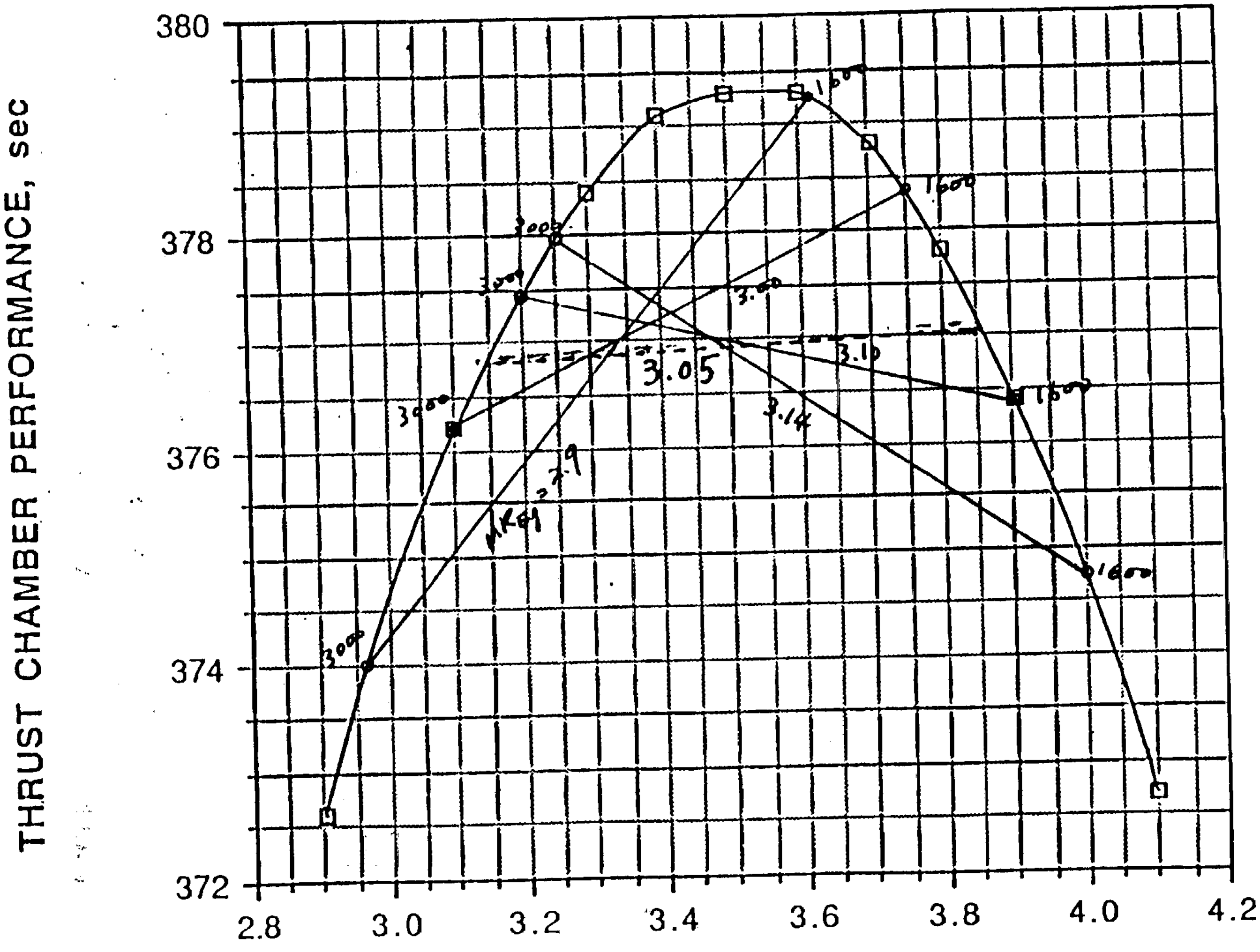
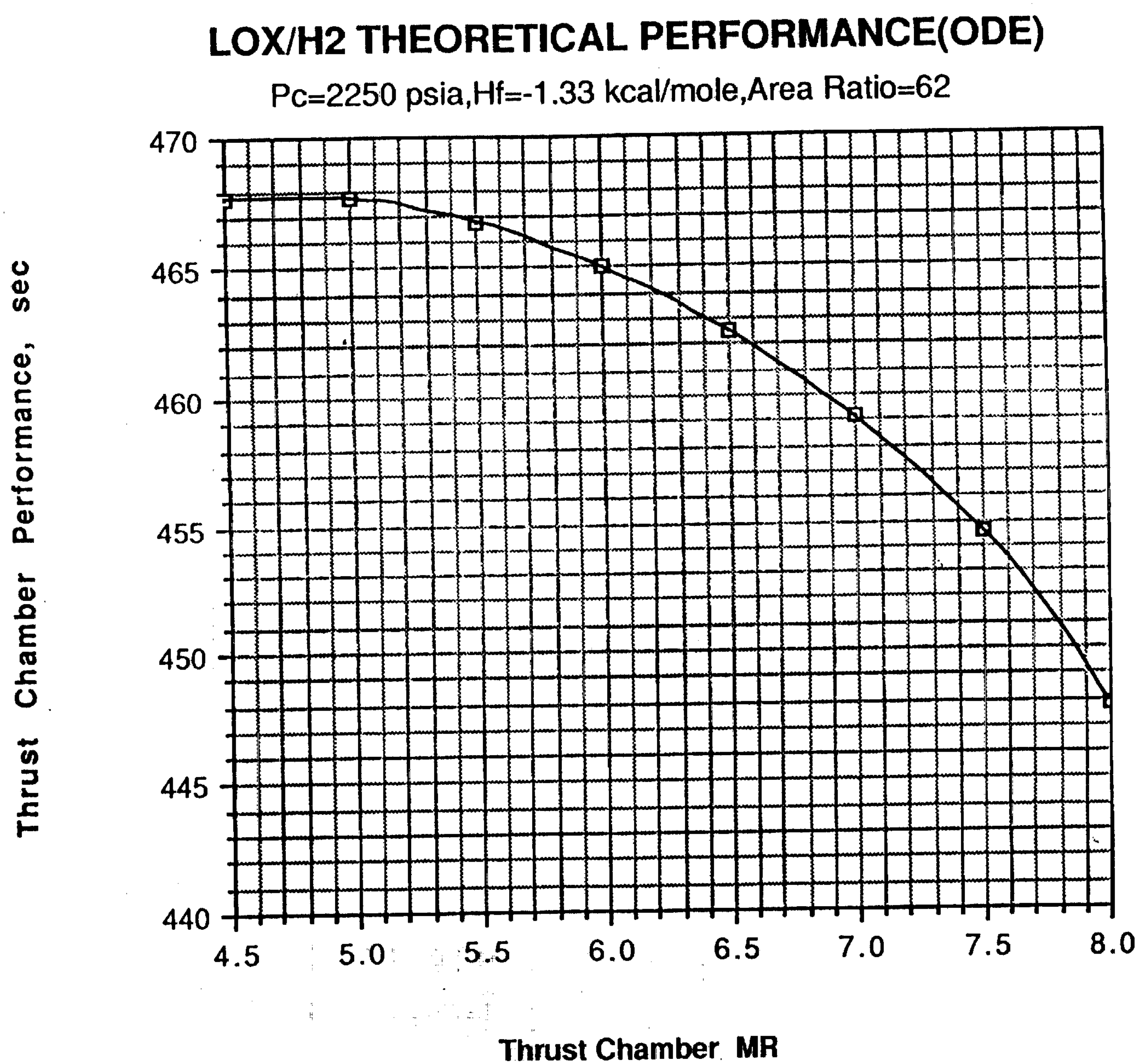


Figure 16



LOX/H2 THEORETICAL PERFORMANCE(ODE)
Pc=2250 psia,Hf=-1.33 kcal/mole,Area Ratio=62
470
465
460
455
450
445
440
4.5
5.0
5.5
6.0
6.5
7.0
7.5
8.0
Thrust Chamber Performance, sec
Thrust Chamber MR

Figure 17

APPENDIX C

CONCEPTUAL DESIGN

APPENDIX C1

ENGINE/COMPONENT SELECTION

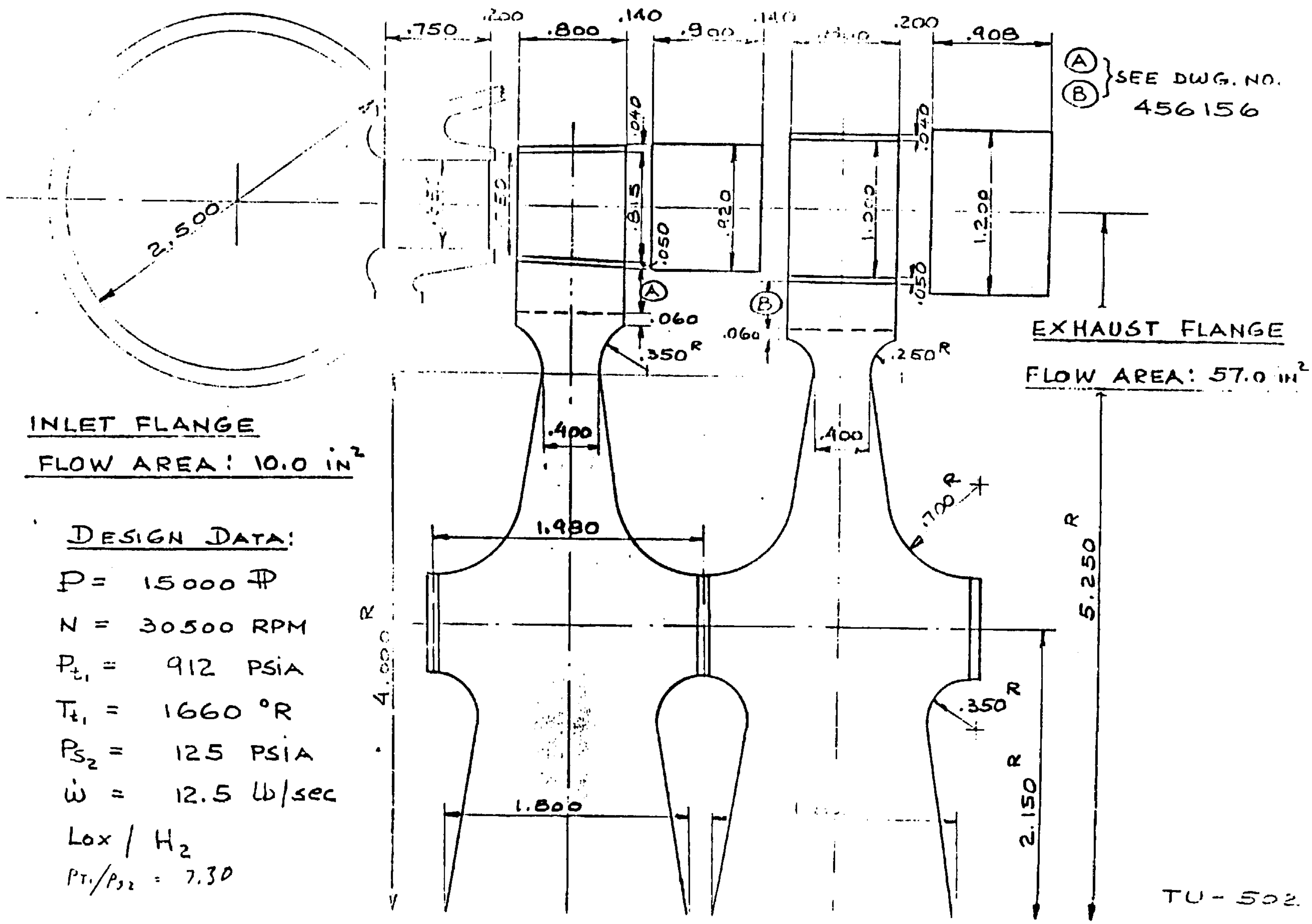
<u>COMPARISON OF ALS FUEL TURBINE WITH MARK 29 FUEL TURBINE</u>

TURBINE		ALS STME FUEL	J2S MARK 29 FUEL
1. DESIGN PARAMETERS - TOTAL-TO-STATIC:			
PRESSURE RATIO	-	6.65	7.30
ISENTROPIC VELOCITY RATIO	-	0.177	0.167
EFFICIENCY	-	0.602	0.610
NOZZLE OUTLET MACH NUMBER	-	1.56	1.60
ROTOR INLET RELATIVE MACH NO.	-	1.27	1.30
LOADING COEF. - OVERALL	-	9.94	10.88
LOADING COEF. - FIRST ROTOR	-	7.02	6.81
LOADING COEF. - SECOND ROTOR	-	2.92	4.06
FLOW COEF. - AVERAGE	-	1.11	1.27
2. GEOMETRY PARAMETERS:			
MEAN DIAMETER	INCH	14.07	10.50
EFFECTIVE AREA	SQUARE INCH	4.63	3.00
NUMBER OF ROTORS	-	2	2
STAGING	-	VELOCITY COMPOUNDED	VELOCITY COMPOUNDED
NOZ. HEIGHT/MEAN DIAM.	-	0.040	0.062
LAST ROTOR HEIGHT/MEAN DIAM.	-	0.085	0.095

COMPARISON OF ALS FUEL TURBINE WITH MARK 29 FUEL TURBINE

TURBINE		ALS STME FUEL	J2S MARK 29 FUEL
3. STRUCTURAL PARAMETERS:			
SPEED SQUARED-ANNULUS AREA	(RPM-INCH)E2	28x10E9	31x10E9
MEAN BLADE SPEED	FT/SEC	1397	1412
4. DESIGN OPERATING CONDITIONS:			
OUTPUT POWER	HP	55,200	15,000
SHAFT SPEED	RPM	23,000	30,500
SHAFT TORQUE	FT-LBF	12,605	2583
WORKING FLUID - COMBUSTION PRODUCTS OF		O2/H2	O2/H2
FLOWRATE	LBM/SEC	51.0	12.5
INLET TOTAL TEMPERATURE	R	1600	1660
INLET TOTAL PRESSURE	PSIAT	2239	912
OUTLET STATIC PRESSURE	PSIAS	337	125
OUTLET TEMPERATURE	R	1195	1233
5. OTHER PARAMETERS:			
SPECIFIC WORK	BTU/LBM	765	848

MK 29 - F BLADE PATH AND DISC PROFILE



INLET FLANGE

FLOW AREA: 10.0 in^2

EXHAUST FLANGE

FLOW AREA: 57.0 in^2

DESIGN DATA:

P = 15000 HP

N = 30500 RPM

P_{t_1} = 912 PSIA

T_{t_1} = 1660 °R

P_{S_2} = 125 PSIA

$\dot{w}$ = 12.5 lb/sec

Lox / H_2

P_{T_1}/P_{S_2} = 7.30

APPENDIX C2

FRCMC TURBOPUMP CONCEPTUAL DESIGN

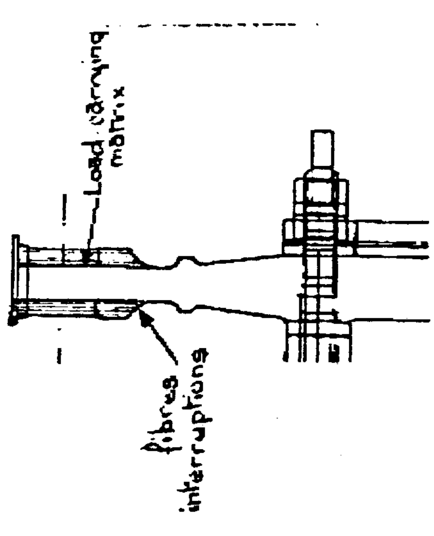


Figure 1

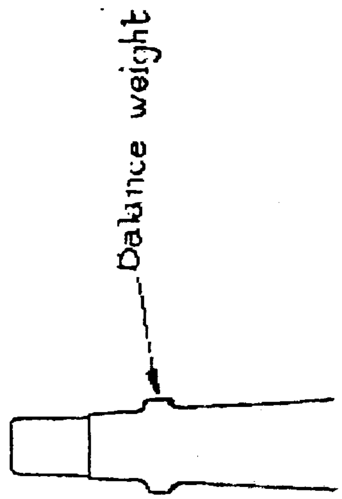


Figure 2

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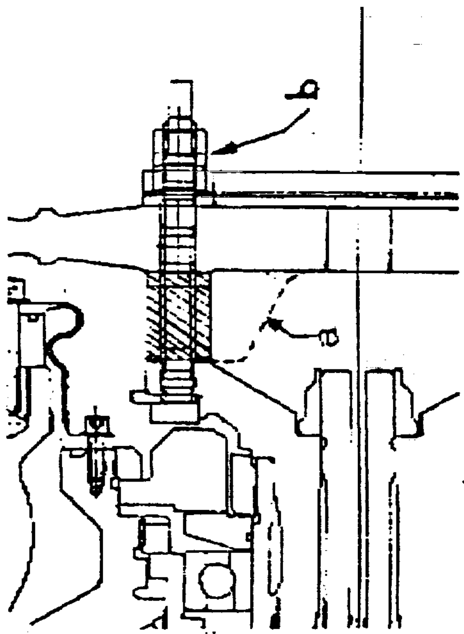


Figure 3

FRCMC WHEEL ATTACHMENT CONCEPT

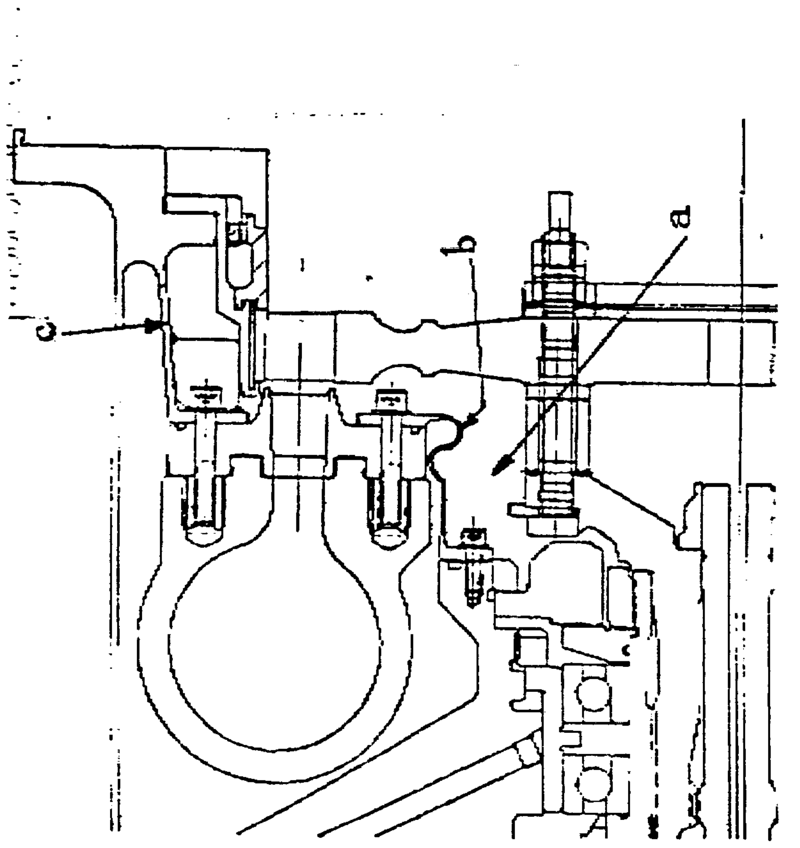
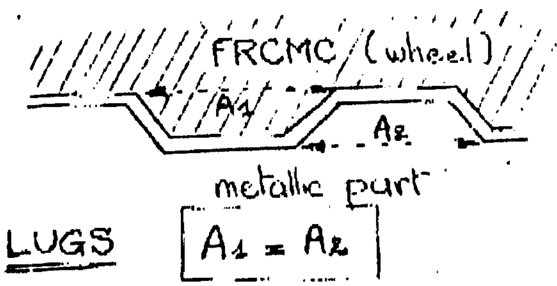
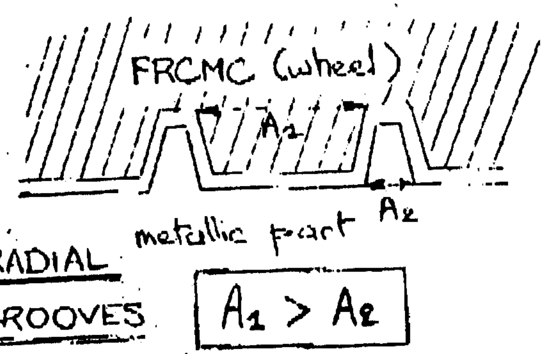


Figure 4



LUGS

load carrying areas being equal in each part, the torque transmission capacity is given by the shear strength of FRCMC, which is negligible compared with metallic alloys.



RADIAL GROOVES

Efforts to be transmitted by the wheel can be dispatched on a much greater area, so that shear level is limited, and the torque is more important than in the previous case.

APPENDIX D

DETAILED DESIGN AND SPECIFICATION

APPENDIX D1

FRCMC BLADE COUPON DESIGN

RELEASED DOCUMENT

-	-	-	-	-	-	-	-	-	-
-	-	-	-	-	-	-	-	2	1
REVISION STATUS OF SHEETS									

-3	CERAMIC COMPOSITE	--	⟨8⟩	--	--
NO.	MATERIAL	SIZE	SPECIFICATION	SH	ZONE

SEE SEPARATE PARTS LIST

UNLESS OTHERWISE SPECIFIED
- DIMENSIONS ARE IN INCHES AND APPLY PRIOR TO FINISH
- DRAWING INTERPRETATIONS PER RF0004-018
- SURFACE TEXTURE
  - SYMBOLS PER ANSI Y14.36
  - REQUIREMENTS PER ANSI B46.1
  - 125/ MACH SURF ROUGHNESS
- TOLERANCES ON
  - ANGLES ±0°30'
  - DECIMALS .XX ± .03 .XXX ± .010

DO NOT SCALE PRINT

CONTR NAS3-25468		
DWN L KADO	DATE 89/11/17	
CHK G F TUTTLE (CC)	DATE 90/01/05	
DSGN G F/ TUTTLE	XXX	
MATL J W BROCKMEYER	XXX	
STRUCT L ORR	XXX	
APVD R McCLELLAND	DATE 90/01/09	

Rockwell International Corporation
Rocketdyne Division
Canoga Park, California

COUPON, BLADE, FRCMC

SIZE	FSCM NO	DRAWING NO
E	02602	7R032077
SCALE 4/1		SHEET 1 OF 2

REVISIONS			
REV	DESCRIPTION	DATE	APPROVED

NOTE: UNLESS OTHERWISE SPECIFIED

1. MACHINE PER RA0103-016.
2. CLEAN PER RA1110-003.
3. ENGRAVE PART NUMBER PER RF0004-044 TYPE V.
4. THE STACKING AXIS IS A STRAIGHT LINE PASSING THRU THE BASIC STACKING POINT OF EACH SECTION
5. THE STACKING POINT OF EACH SECTION MAY VARY .005 INCH IN ANY DIRECTION FROM THE BASIC STACKING AXIS
6. THE AIRFOIL SURFACES SHALL BLEND SMOOTHLY BETWEEN THE DEFINED CONTOUR POINTS AND SECTIONS.
7. THE AIRFOIL SECTIONS ARE SIMILAR TO THE MK29F FIRST STAGE TURBINE BLADE SECTIONS (REF DWG 461390) EXCEPT FOR THE FOLLOWING
   a. SOLID CORE
   b. HUB SECTION CONTOUR INCREASED .010 INCHES
   c. CENTROID OF THE HUB SECTION IS OFFSET .010 INCHES FROM THE STACKING POINT
8. MATERIAL PER RB0115-017 WITH LAMINATIONS NORMAL TO THE X AXIS.

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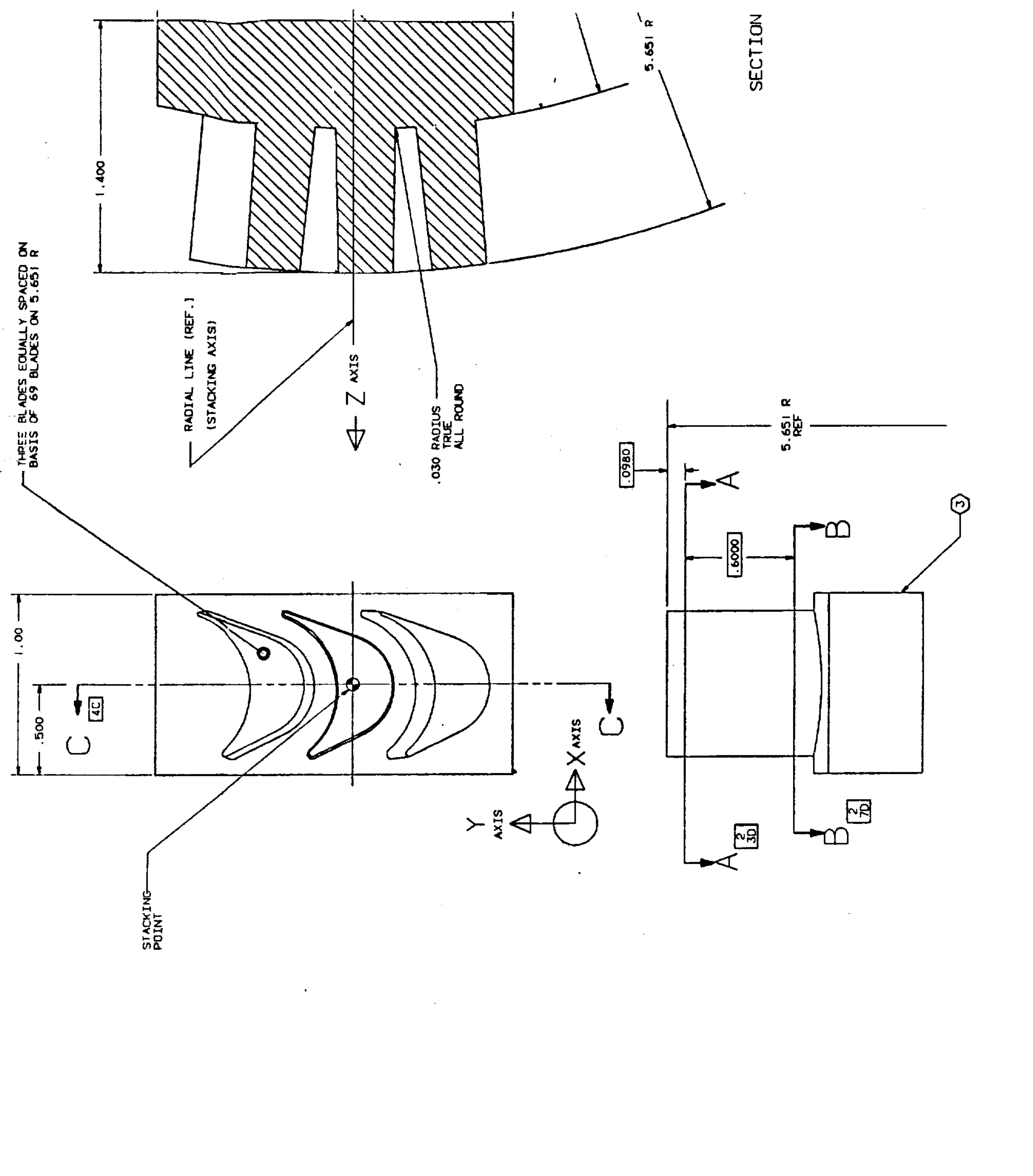
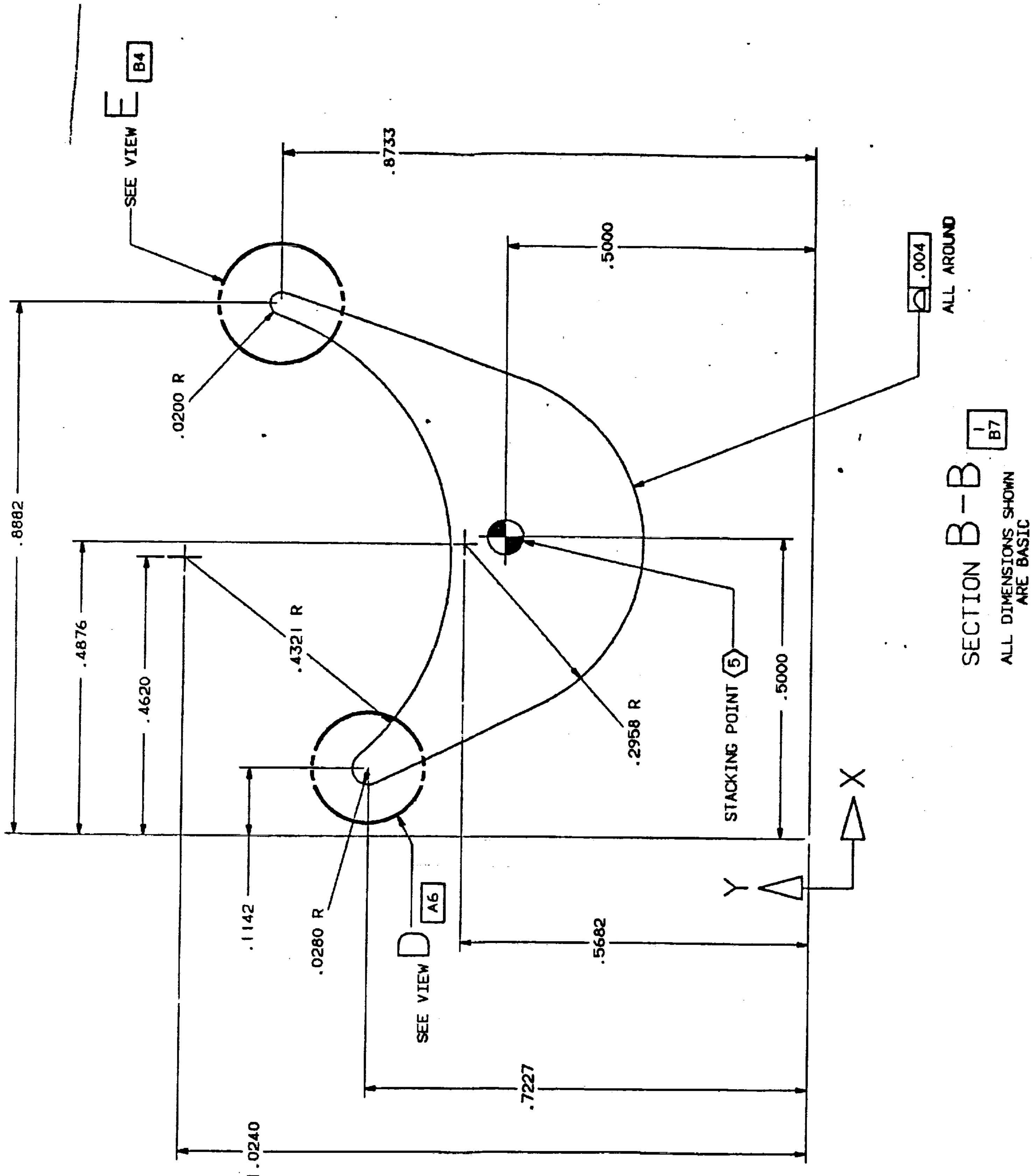
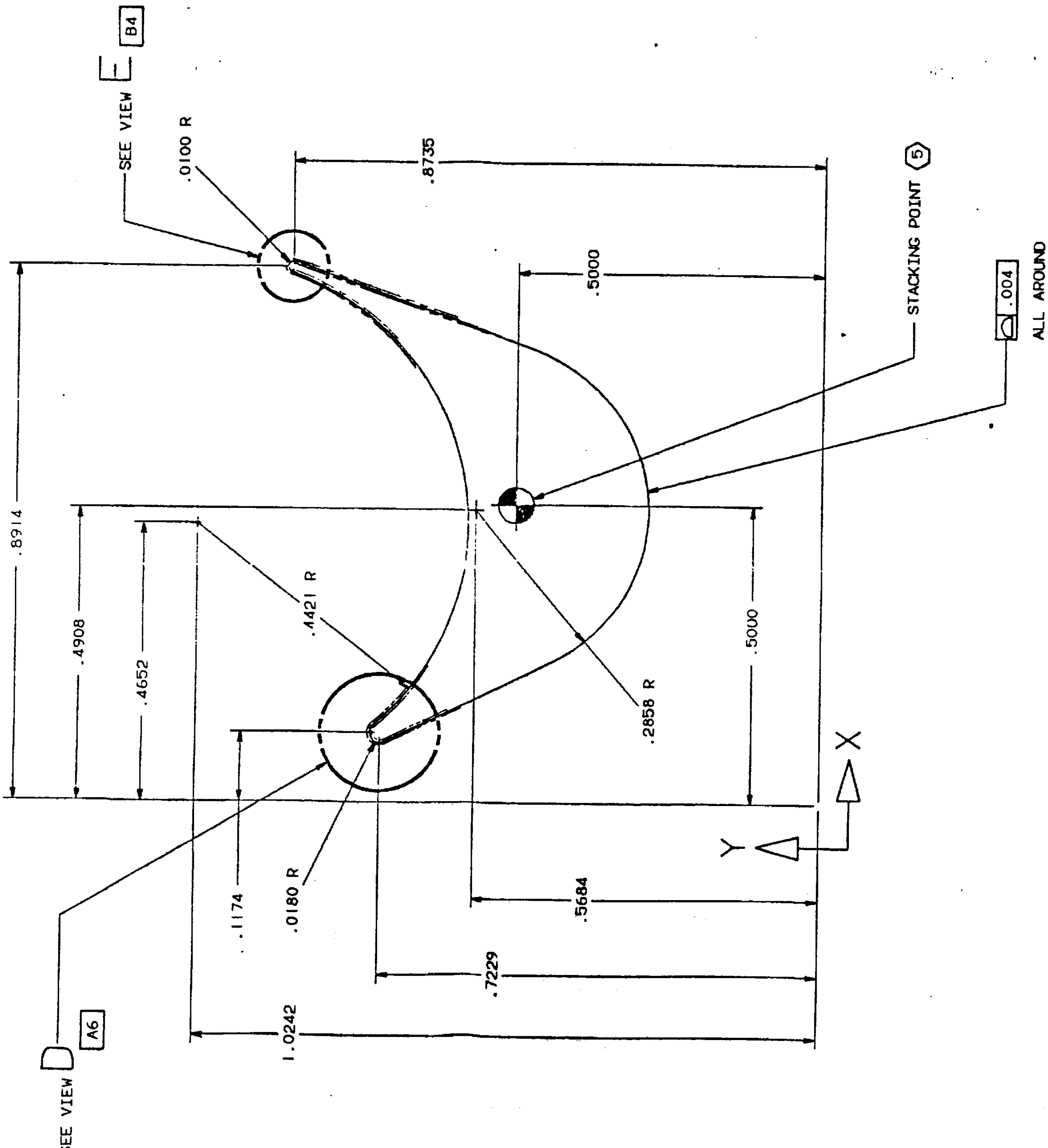

THREE BLADES EQUALLY SPACED ON BASIS OF 69 BLADES ON 5.651 R
RADIAL LINE (REF.) (STACKING AXIS)
Z AXIS
.030 RADIUS TRUE ALL ROUND
1.400
5.651 R
SECTION
5.651 R REF
.0980
.6000
A
B
3
1.00
.500
C
4C
STACKING POINT
X AXIS
Y AXIS
2 3D
2 7D



SEE VIEW E
B4
.8733
.5000
.004
ALL AROUND
.0200 R
.8882
.4876
.4620
.4321 R
.2958 R
STACKING POINT 5
.5000
SECTION B-B
1
B7
ALL DIMENSIONS SHOWN ARE BASIC
Y
X
.1142
.0280 R
SEE VIEW D
A6
.5682
.7227
1.0240



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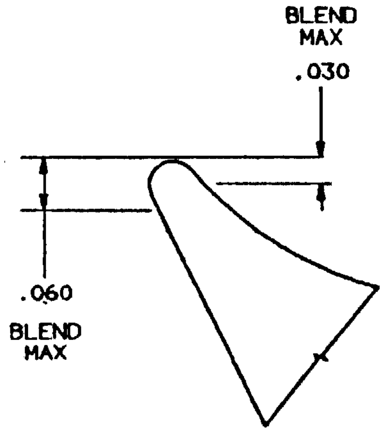
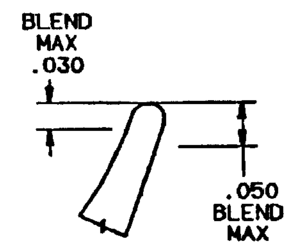


VIEW D F7



VIEW E G1

APPENDIX D2

PRELIMINARY MATERIALS SPECIFICATION

PRELIMINARY MATERIALS SPECIFICATION
for
CARBON FIBER-REINFORCED SILICON CARBIDE (C/SiC)
LIQUID ROCKET ENGINE COMPONENTS

1. SCOPE

1.1 Scope
This specification establishes the architecture, processing, properties and inspection requirements for continuous carbon fiber-reinforced silicon carbide, ceramic matrix composite materials, hereinafter referred to as C/SiC, intended for use in the manufacture of liquid fuel rocket engine turbomachinery components. The specification is limited to components requiring orthotropic properties prepared from fiber architectures as described in Section 3.

1.2 Classification

1.2.1 Type
This material shall be of the type referred to as fiber-reinforced ceramic matrix composites (FRCMC).

1.2.2 Class
This specification refers specifically to the class of FRCMC known as carbon fiber-reinforced silicon carbide (C/SiC) and further refers specifically to C/SiC produced by the chemical vapor infiltration (CVI) of pre-forms produced from continuous carbon fibers.

2. APPLICABLE DOCUMENTS

2.1 Government Documents
This section is not applicable to this specification.

2.2 Nongovernment Documents
The following documents form a part of this specification to the extent specified in Sections 3, 4, and 5. The effective issue of these documents shall be that issue incorporated herein or that referenced specifically by change letter.

APPENDICES

1. T300 carbon fiber, Amoco publication, Appendix A
2. SEP 2D and Novoltex Material Properties, Appendix B
3. Properties of C/SiC, DuPont publication, Appendix C

3. REQUIREMENTS

3.1 Fiber Architecture

3.1.1 Weave

The fiber lay-up shall consist of individual, 2-dimensional, plain woven layers of carbon fabric prepared from fibers as described in section 3.2.

3.1.2 Lay-Up

Successive 2-dimensional layers shall be rotated in 30° + or - 5° increments about a central vertical axis throughout the entire thickness of the lay-up in order to create a structure with essentially isotropic in-plane properties that when finally infiltrated can be described and analyzed through the use of orthotropic materials properties.

3.1.3 Loading

In-plane fiber loading shall be 40 + or - 2 volume %.

3.2 Fiber and Tow Description

3.2.1 Filament Description

Individual filaments shall be Amoco T300 or equivalent polyacrylonitrile (PAN) carbon fiber filaments with properties as specified in Appendix A for T300 fibers.

3.2.2 Tow Count

Fibers shall consist of 1000 (1K) filaments per tow.

3.2.3 Twist and Surface Treatment

Fiber twist and surface treatment shall be vendor specified.

3.2.4 Fiber Surface Post-Treatment

Fiber surface post-treatment for interfacial control is vendor proprietary and shall be controlled in conformance with their standard practices as required to achieve the desired properties.

3.3 Porosity

3.3.1 Allowable Porosity

Apparent porosity of the final part shall be less than or equal to 13 volume % as determined by final inspection.

3.3.2 Distribution of Porosity

Porosity shall be uniformly distributed both in-plane and through-thickness such that there are no indications of delamination.

3.4 Homogeneity

3.4.1 Properties

The completed material shall be essentially homogeneous throughout such that the macroscopic properties data are typical for any location throughout the part with minimal reduction in properties at leading or trailing edges, bores or fillets.

3.4.2 Layer Spacing

Individual fiber layers shall be uniformly spaced with no apparent delaminations.

3.4.3 Intralayer Uniformity

Within layers, fibers shall be uniformly spaced, allowing uniform infiltration with no evidence of segregated porosity or delaminations.

3.4.4 Porosity Distribution

Porosity shall be uniformly distributed and of uniform size with no areas of segregated porosity indicative of delaminations.

3.4.5 Density Variation

Matrix infiltration shall be uniform both in-plane and through-thickness such that density variations will not exceed + or - 5% from the nominal.

3.5 General Requirements

3.5.1 Density

Density shall be consistent with the porosity, fiber loading and materials requirements and shall not deviate more than 5% from the nominal value.

3.5.2 Surface Finish

Surface finish of the machined material shall be TBD or better as determined by inspection.

3.5.3 Surface Treatment

Surface treatment shall consist of a final infiltration and deposition step with the SiC matrix to prevent the occurrence of exposed carbon fibers.

3.5.4 Chemistry

Matrix chemistry shall be consistent with that of a chemical vapor infiltrated, SiC matrix and is subject to verification by x-ray crystallographic means.

4. QUALITY ASSURANCE PROVISIONS

4.1 In-process Inspection

4.1.1 Responsibility
In-process inspection shall be performed by the vendor.

4.1.2 Fabric
Fabric shall be inspected to insure conformance to fiber type and weave architecture requirements.

4.1.3 Lay-Up
Lay-up or pre-form shall be inspected to insure conformance to architecture requirements.

4.1.4 X-Ray Inspection
Material shall be inspected both visually and by x-ray following each intermediate infiltration step to verify conformance to density, porosity and homogeneity requirements.

4.2 Finished Materials Inspection

4.2.1 Responsibility
Finished materials inspection shall be performed by the vendor and is subject to confirmation by Rocketdyne.

4.2.3 Visual Inspection
Material shall be inspected visually for the appearance of any flaws, cracks, chips, excessively segregated porosity, exposed fibers or delaminations that shall be cause for rejection.

4.2.4 Geometry
Material shall be measured to insure conformance to geometry as called out on the drawing or as otherwise specified.

4.2.5 Weight
Material shall be weighed to assure conformance to average density and porosity requirements.

4.2.6 X-Ray Inspection
Material shall be inspected by x-ray to insure conformance to porosity and density uniformity requirements.

4.2.7 Ultrasound Inspection
(Optional) Material shall be inspected ultrasonically to verify density distribution.

5. PREPARATION FOR DELIVERY

5.1 Packaging
This section is not applicable to this specification.

5.2 Labeling
No markings shall be applied directly to the material; identification shall be applied to the respective container.

6. NOTES
This section is not applicable to this specification.

7. APPROVED SOURCE
SEP, represented in the U. S. by DuPont, is the only currently approved source.

8. NUMBERING
R-B-01-15-xxx

9. APPENDICES
A, B, and C attached

Amoco Performance Products, Inc.
380 Grove Street
Ridgefield, CT 06877

THORNEL® OXIDATION RESISTANT CARBON FIBERS

TYPICAL PROPERTIES

	THORNEL® T-40R*		THORNEL® T-300R		
	3K	12K	3K	6K	12K
TENSILE MODULUS (MPSI)	40	40	35	36	33
TENSILE STRENGTH (KPSI)	500	500	500	500	500
STRAIN (%)	1.1	1.1	1.6	1.5	1.5
DENSITY G/CM^3	1.77	1.79	1.78	1.78	1.77
YIELD G/M	.18	.75	.19	.40	.81
OXIDATION RESISTANCE					
- % WT. LOSS AFTER 1,000 HOURS @ 600°F, 4 ATMOSPHERE	.4	.43	10.8	9.6	10.4

*T-40R IS DOMESTIC - CARBONIZED IN GREENVILLE, SC FROM PRECURSOR PRODUCED AT THE SAME LOCATION

F-7080/REV. 1
9/86
817A

PRODUCT INFORMATION

TECHNICAL INFORMATION

THORNEL® Carbon Fiber T-300 1K

1. Description

THORNEL Carbon Fiber T-300 1K is a continuous length, high-strength, high-modulus fiber consisting of 1,000 filaments in a one-ply construction. The fiber surface has been treated to increase the interlaminar shear strength in a resin matrix composite.

2. Typical Properties and Characteristics

Property	U.S. Customary Units	Value	S.I. Units	Value
Tensile Strength	$lb/in^2 \times 10^3$	500	GPa	3.45
Tensile Modulus	$lb/in^2 \times 10^6$	33.5	GPa	231
Density	lb/in^3	0.063	Mg/m^3	1.76
Filament Diameter	m	7.0	mm	7.0
Elongation at Break	%	1.4	%	1.4
Elastic Recovery	%	100	%	100
Carbon Assay	%	92	%	92
Surface Area	m^2/g	0.45	m^2/g	0.45
Longitudinal Thermal Conductivity	BTU-ft/hr(ft²)(°F)	5	W/m K	8.5
Electrical Resistivity	$Ohm\text{-}cm \times 10^{-4}$	18	mohm-m	18
Longitudinal CTE at 70 °F (21 °C)	PPM/°F	−0.3	PPM/K	−0.5

3. Typical Strand Properties

Property	U.S. Customary Units	Value	S.I. Units	Value
Yield	yd/lb	7500	m/g	15.2
Denier	g/9000m	595	g/9000m	595
*Twist	tpi	0.6	tpm	25
Filaments/Strand	—	1000	—	1000
Fiber Area in Yarn Cross Section	$in^2 \times 10^{-5}$	5.8	mm^2	0.037

* Also available untwisted

THORNEL is a registered trademark of Amoco Performance Products, Inc., U.S.A.

F-4964 Rev. 9

PRODUCT INFORMATION

TECHNICAL INFORMATION

THORNEL® Carbon Fiber T-300 3K

1. Description

THORNEL Carbon Fiber T-300 3K is a continuous length, high-strength, high-modulus fiber consisting of 3,000 filaments in a one-ply construction. The fiber surface has been treated to increase the interlaminar shear strength in a resin matrix composite.

2. Typical Properties and Characteristics

Property	U.S. Customary Units	Value	S.I. Units	Value
Tensile Strength	$lb/in^2 \times 10^3$	530	GPa	3.65
Tensile Modulus	$lb/in^2 \times 10^6$	33.5	GPa	231
Density	lb/in^3	0.064	Mg/m^3	1.76
Filament Diameter	μ	7	μm	7
Elongation at Break	%	1.4	%	1.4
Elastic Recovery	%	100	%	100
Carbon Assay	%	92	%	92
Surface Area	m^2/g	0.45	m^2/g	0.45
Longitudinal Thermal Conductivity	BTU-ft/hr(ft²)(°F)	5	W/m K	8.5
Electrical Resistivity	$Ohm\text{-}cm \times 10^{-4}$	18	μohm-m	18
Longitudinal CTE at 70 °F (21 °C)	PPM/°F	−0.3	PPM/K	−0.5

3. Typical Strand Properties

Property	U.S. Customary Units	Value	S.I. Units	Value
Yield	yd/lb	2510	m/g	5.05
Denier	g/9000m	1780	g/9000m	1780
*Twist	tpi	0	tpm	0
Filaments/Strand	—	3000	—	3000
Fiber Area in Yarn Cross Section	$in^2 \times 10^{-5}$	17.5	mm^2	0.113

* Also available with a twist of 0.4 tpi (16 tpm)

F-4311 Rev. 9

PRODUCT INFORMATION

Thornel

TECHNICAL INFORMATION

THORNEL® Carbon Fiber T-300 6K

1. Description

THORNEL Carbon Fiber T-300 6K is a continuous length, high-strength, high-modulus fiber consisting of 6,000 filaments in a one-ply construction. The fiber surface has been treated to increase the interlaminar shear strength in a resin matrix composite.

2. Typical Properties and Characteristics

Property	U.S. Customary Units	Value	S.I. Units	Value
Tensile Strength	$lb/in^2 \times 10^3$	530	GPa	3.65
Tensile Modulus	$lb/in^2 \times 10^6$	33.5	GPa	231
Density	lb/in^3	0.064	Mg/m^3	1.76
Filament Diameter	μ	7	μm	7
Elongation at Break	%	1.4	%	1.4
Elastic Recovery	%	100	%	100
Carbon Assay	%	92	%	92
Surface Area	m^2/g	0.45	m^2/g	0.45
Longitudinal Thermal Conductivity	BTU-ft/hr(ft^2)(°F)	5	W/m K	8.5
Electrical Resistivity	Ohm-cm $\times 10^{-4}$	18	μohm-m	18
Longitudinal CTE at 70°F (21°C)	PPM/°F	−0.3	PPM/K	−0.5

3. Typical Strand Properties

Property	U.S. Customary Units	Value	S.I. Units	Value
Yield	yd/lb	1255	m/g	2.53
Denier	g/9000m	3565	g/9000m	3565
*Twist	tpi	0	tpm	0
Filaments/Strand	—	6000	—	6000
Fiber Area in Yarn Cross Section	$in^2 \times 10^{-5}$	35	mm^2	0.226

* Also available with a twist of 0.4 tpi (16 tpm)

F-5409 Rev. 9

PRODUCT INFORMATION

TECHNICAL INFORMATION

THORNEL® Carbon Fiber T-300 12K

1. Description

THORNEL Carbon Fiber T-300 12K is a continuous length, high-strength, high-modulus fiber consisting of 12,000 filaments in a one-ply construction. The fiber surface has been treated to increase the interlaminar shear strength in a resin matrix composite.

2. Typical Properties and Characteristics

Property	U.S. Customary Units—Value		S.I. Units — Value	
Tensile Strength	lb/in² x 10^3	530	GPa	3.65
Tensile Modulus	lb/in² x 10^6	33.5	GPa	231
Density	lb/in³	0.064	Mg/m³	1.76
Filament Diameter	μ	7	μm	7
Elongation at Break	%	1.4	%	1.4
Elastic Recovery	%	100	%	100
Carbon Assay	%	92	%	92
Surface Area	m²/g	0.45	m²/g	0.45
Longitudinal Thermal Conductivity	BTU-ft/hr(ft²)(°F)	5	W/m K	8.5
Electrical Resistivity	Ohm-cm x 10^{-4}	18	μohm-m	18
Longitudinal CTE at 70 °F (21 °C)	PPM/°F	−0.3	PPM/K	−0.5

3. Typical Strand Properties

Property	U.S. Customary Units—Value		S.I. Units — Value	
Yield	yd/lb	627	m/g	1.26
Denier	g/9000m	7130	g/9000m	7130
Twist	tpi	0	tpm	0
Filaments/Strand	—	12000	—	12000
Fiber Area in Yarn Cross Section	in² x 10^{-5}	70	mm²	0.452

F-7013 Rev. 3

APPENDIX E

SUB-COMPONENT FABRICATION DETAILS

APPENDIX E1

SEP COUPON DESIGN

Appendix : Blades dimensions (mm)

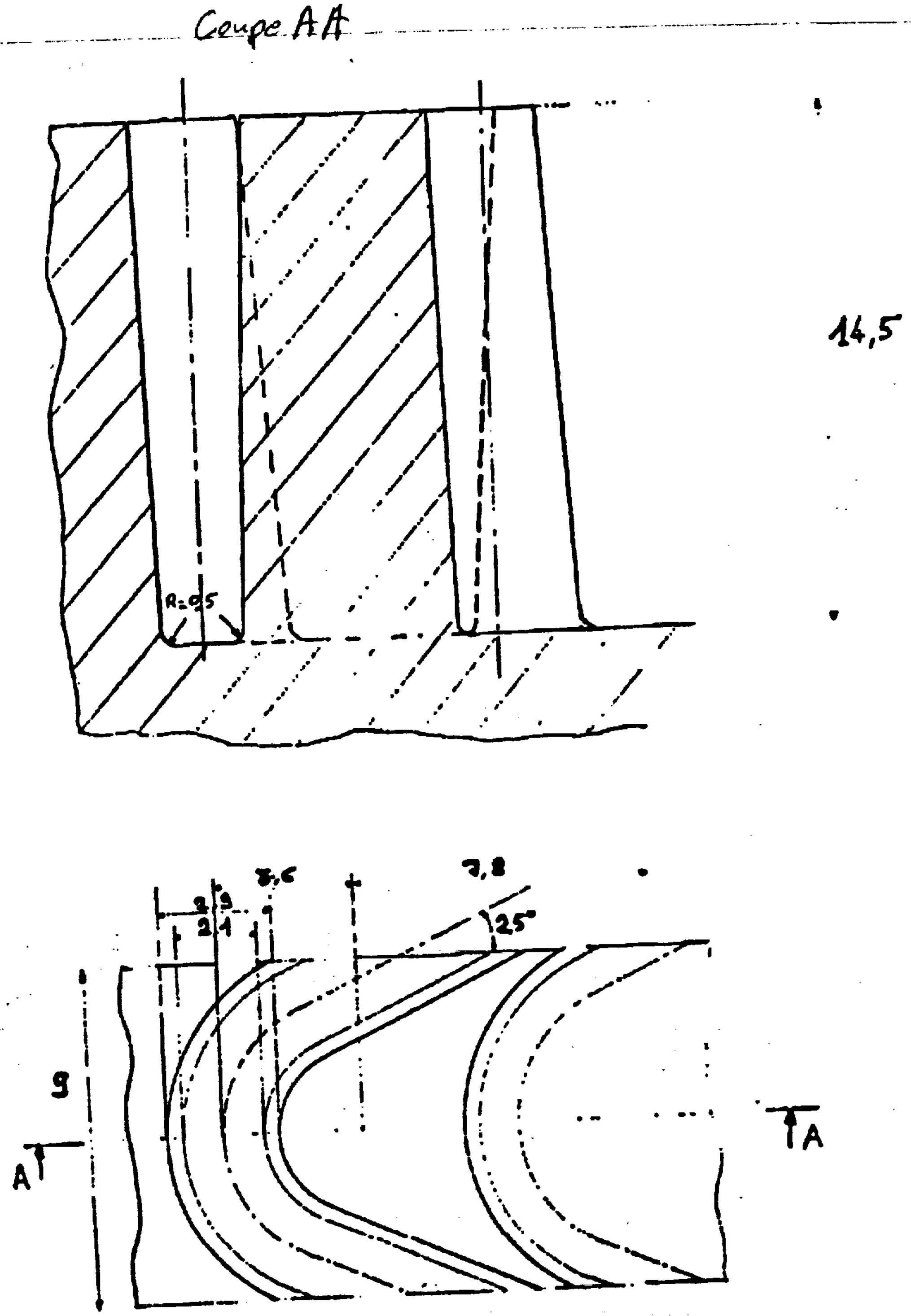


FIG. 1

B4 -> B4

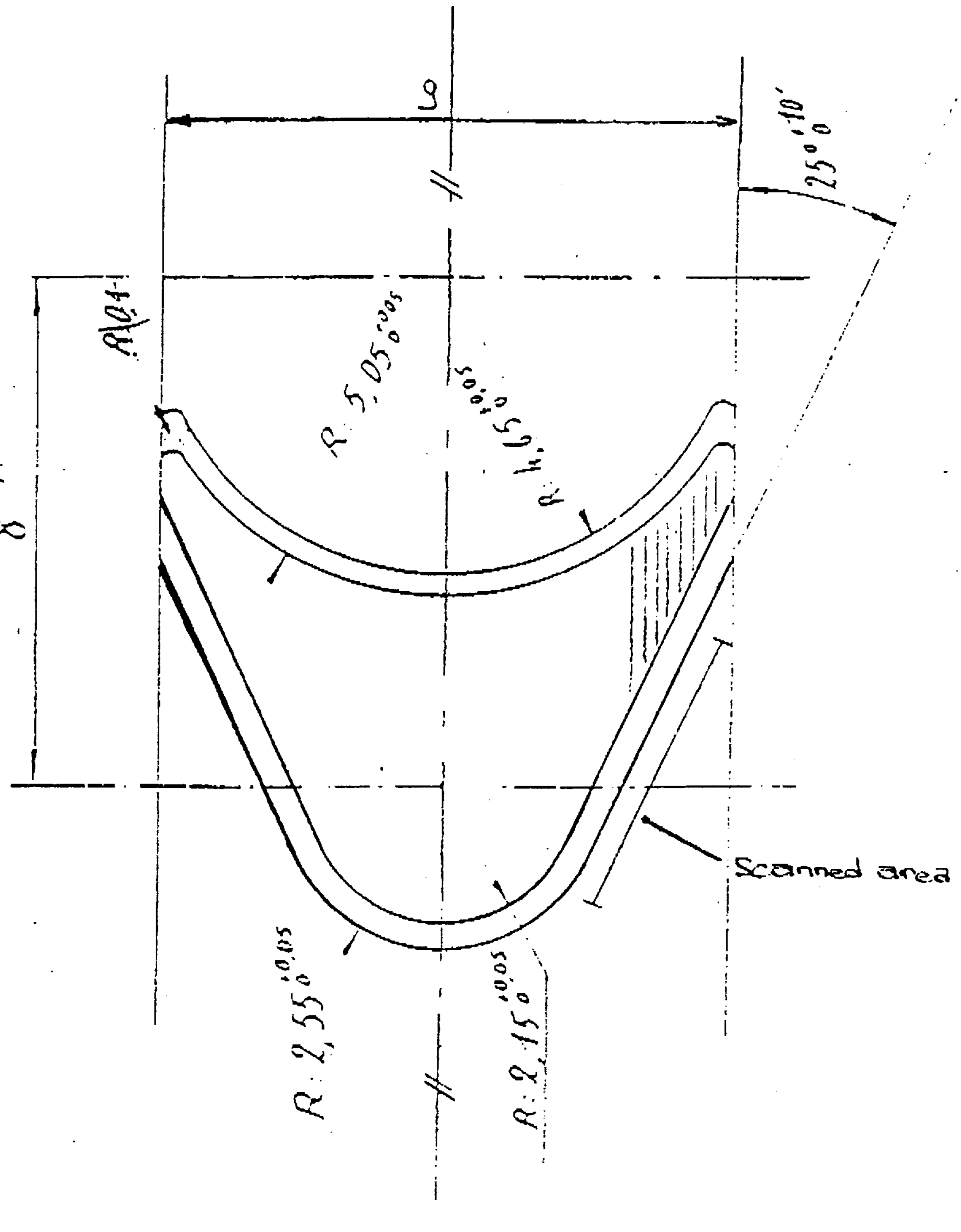


Figure 2

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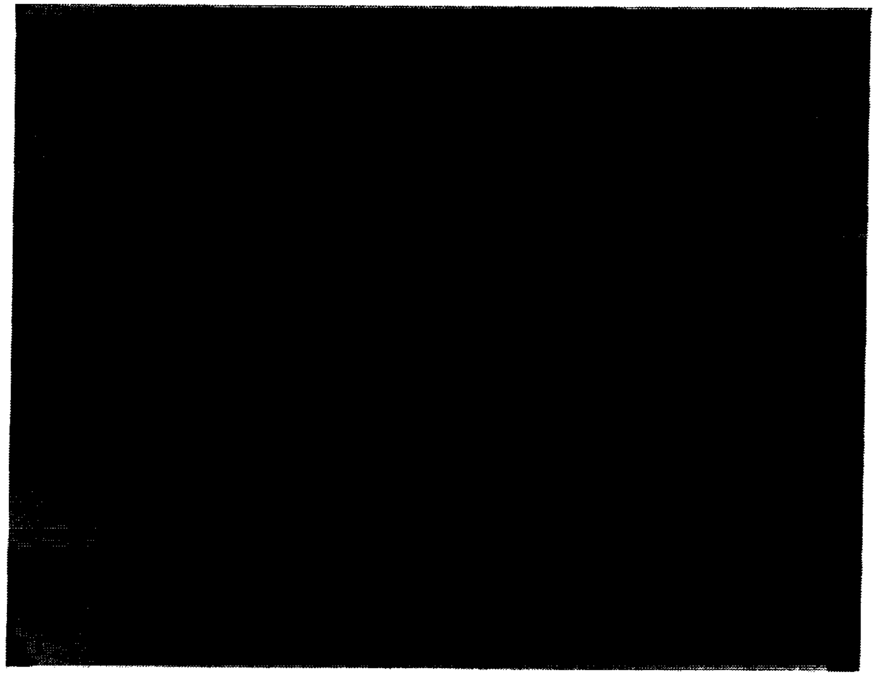
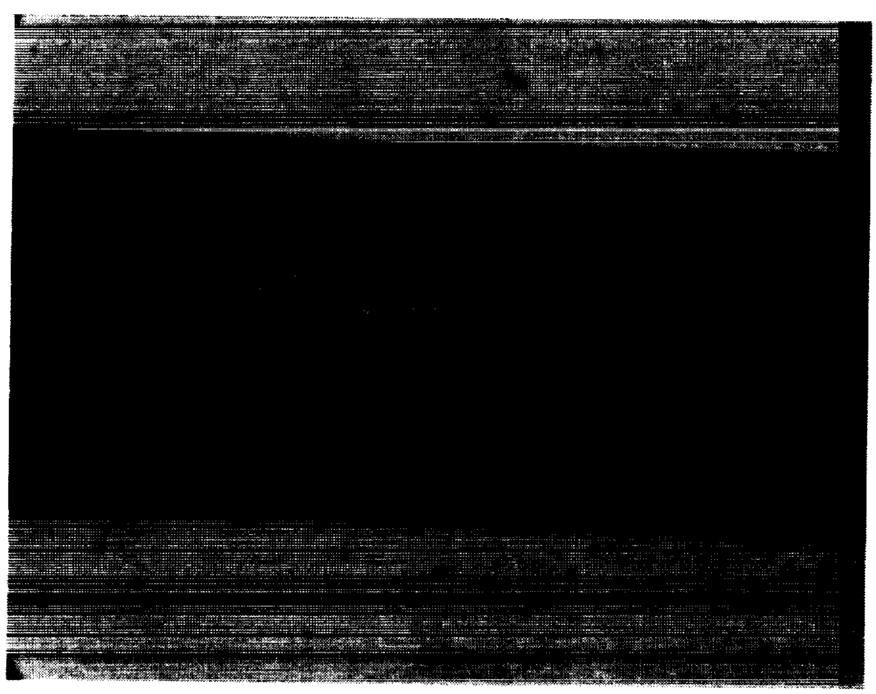
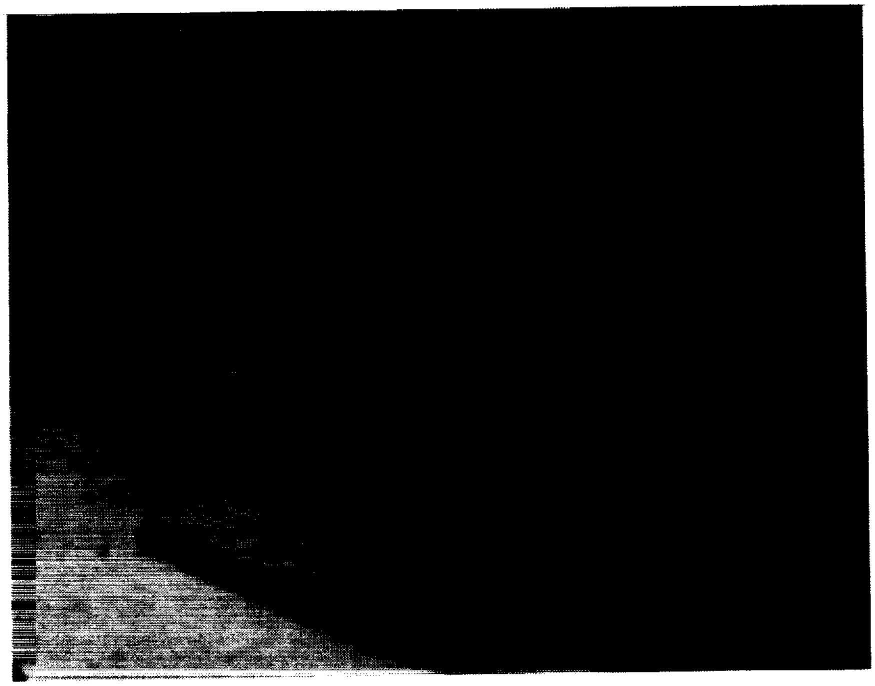
APPENDIX E2

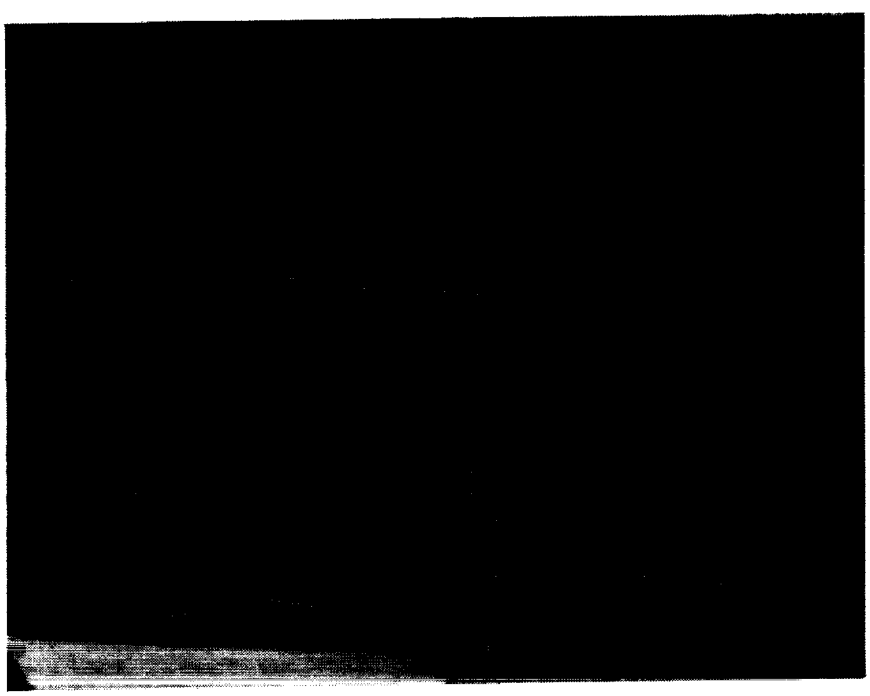
FRCMC COUPON PHOTOMICROGRAPHS





COUPON NO. 1





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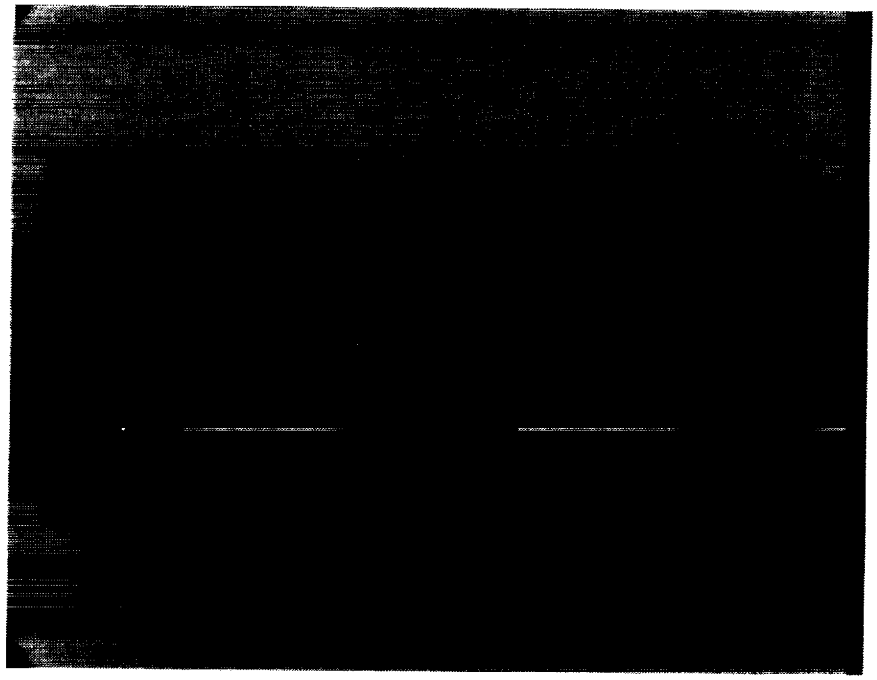

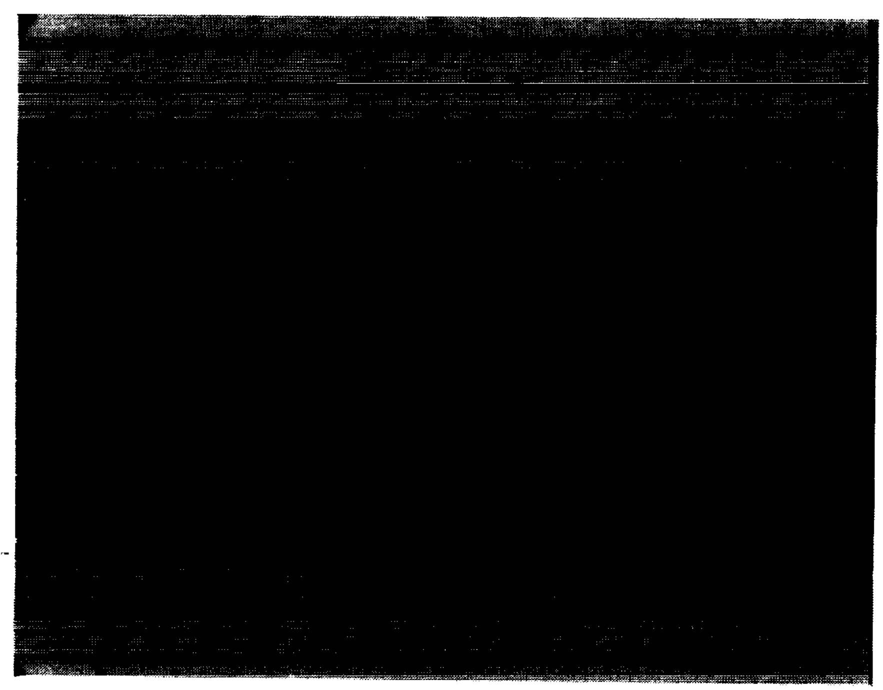

ORIGINAL PAGE
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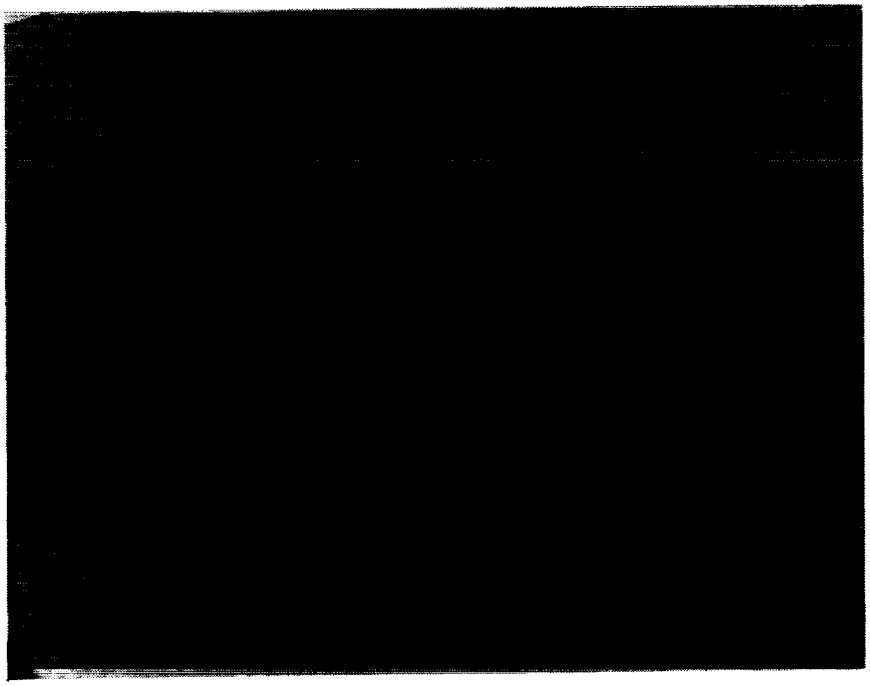
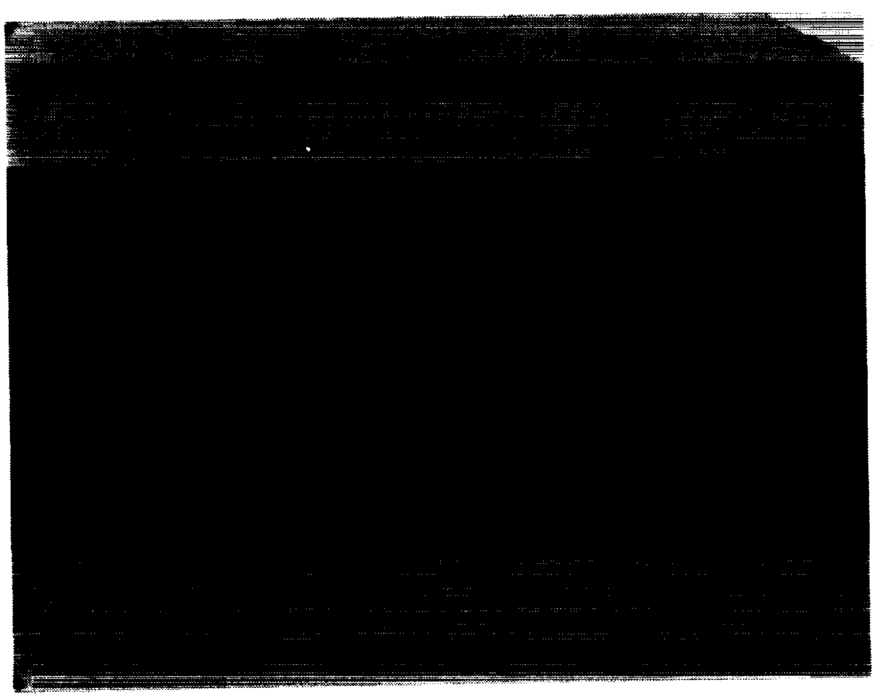
COUPON NO. 4





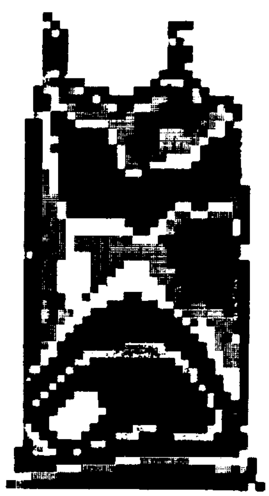
COUPON NO. 3(Retained by Rocketdyne)

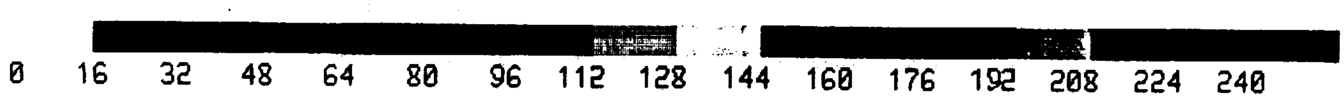
APPENDIX E3

CAT SCAN RESULTS

NAME> N1 N3 N4 2.25MHZ 10MIL
JM1358.PIC
-43 -42 -41 -40 -39 -38 -37 -36 -35 -34 -33 -32 -31 -30 -29 -28

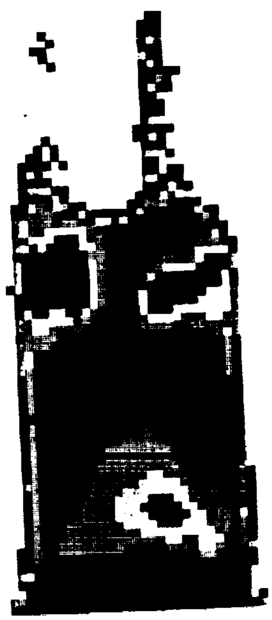
NAME> N1   2.25MHZ   10MIL   JM00N1.PIC

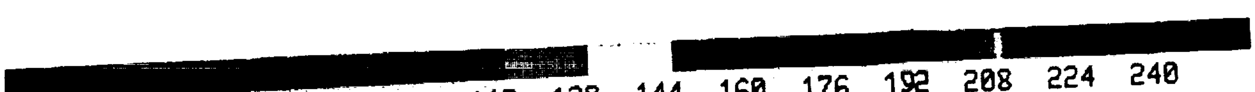




JM00N3.PIC

NAME> N3 2.25MHZ 10MIL





0 16 32 48 64 80 96 112 128 144 160 176 192 208 224 240

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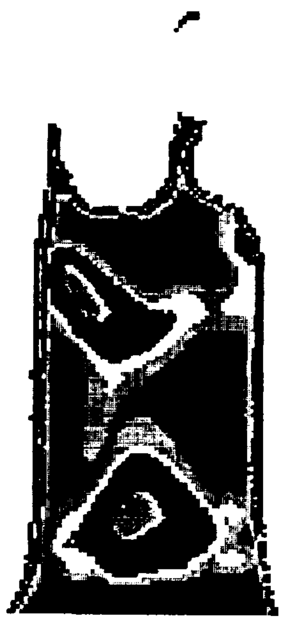

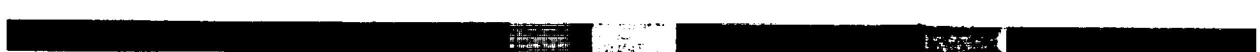

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APPENDIX F

PHASE II PROGRAM

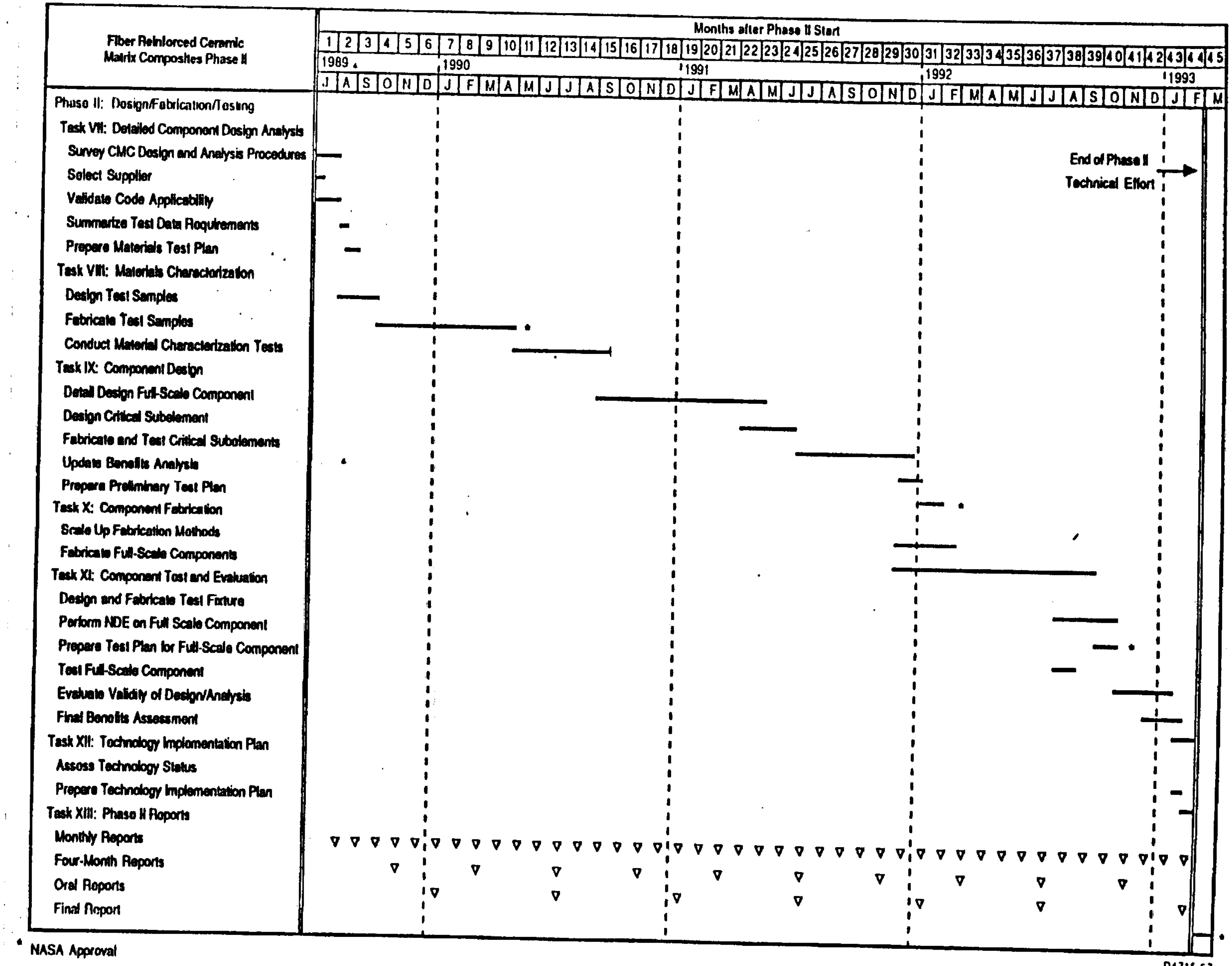

Figure 1-3. Phase II Schedule—Fiber-Reinforced Ceramic Matrix Composites for Earth-to-Orbit Rocket Engine Turbines

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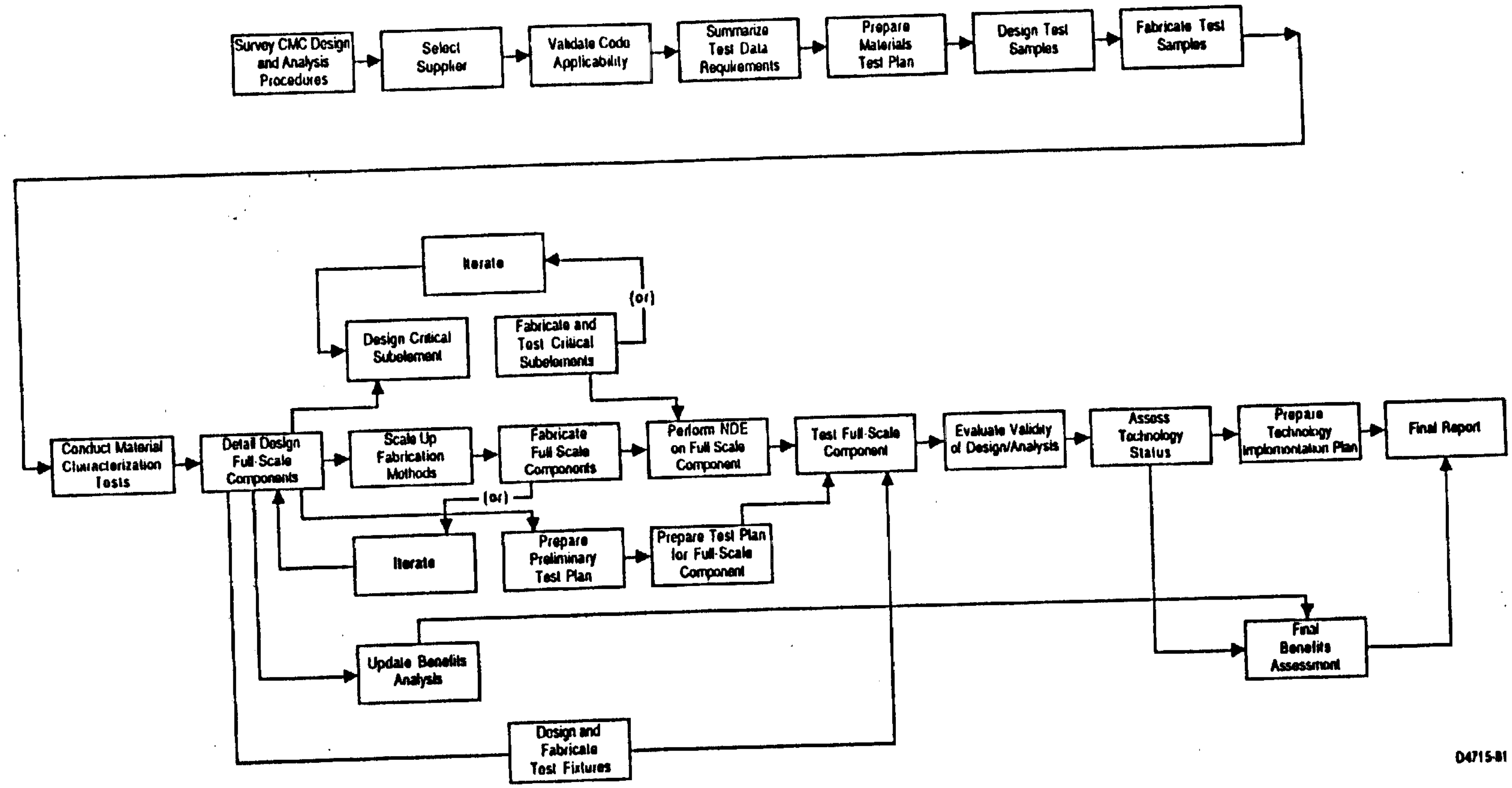

Figure 1-4. Phase II Flow Diagram—Fiber-Reinforced Ceramic Matrix Composites for Earth-to-Orbit Rocket Engine Turbines

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REFERENCES

1. Chandler, W. T., "Materials for Advanced Rocket Engine Turbopump Turbine Blades", NASA-LeRC Contract NAS3-23536, Final Report, November, 1983.

2. Tellier, J. G., "High Temperature Turbine Study", Air Force Rocket Propulsion Laboratory Contract F04622-81-C-0053, Final Report, August, 1986.

3. Petrasek, D. W. and Stephens, J. R., "Fiber Reinforced Superalloys for Rocket Engines", AGARD Conference Proceedings No. 449, October, 1988.

4. Misra, A. K., "Thermodynamic Analysis of Chemical Stability of Ceramic Materials in Hydrogen Containing Atmospheres at High Temperatures", NASA Contractor Report 4271, January, 1990.

5. "NASA Probes Ceramics for the 21st Century", Ceramic Industry, April, 1988.

6. Cawley, J. D., "Joining of Ceramic-Matrix Composites", Ceramic Bulletin, Vo. 68, No. 9, 1989.

7. Halada, J.; Hargis, M. and Nagy, P., "Composite Ceramic Component Propulsion Opportunities", Air Force Wright Aeronautical Laboratories Contract F33615-85-C-2552, Final Report, March, 1988.

8. Couts, J. S., "Engineering Design with ANSYS Software", Ceramic Bulletin, Vol. 68, No. 12, 1989.

9. Nemeth, N. N.; Manderscheid, J. M. and Gyekenyesi, J. P., "Designing Ceramic Components with the CARES Computer Program", Ceramic Bulletin, Vol. 68, No. 12, 1989.

10. Carpenter, H. W., "Ceramic Turbine Elements", NASA-MSFC Contract NAS8-35327, Final Report, June, 1984.

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16. Abstract
FRCMC are emerging materials systems that offer potential for use in liquid rocket engines. Advantages of these materials in rocket engine turbomachinery include: performance gain due to higher turbine inlet temperature, reduced launch costs, reduced maintenance with associated cost benefits, and reduced weight. This program was initiated to assess the state of FRCMC development and to propose a plan for their implementation into liquid rocket engine turbomachinery.

A complete range of FRCMC materials was investigated relative to their development status and feasibility for use in the hot gas path of earth-to-orbit rocket engine turbomachinery. Of the candidate systems, carbon fiber-reinforced silicon carbide (C/SiC) offers the greatest near-term potential. Critical hot gas path components were identified, and the first stage inlet nozzle and turbine rotor of the fuel turbopump for the liquid oxygen/hydrogen Space Transportation Main Engine (STME) were selected for conceptual design and analysis. The critical issues associated with the use of FRCMC were identified. Turbine blades were designed, analyzed and fabricated. The "Technology Development Plan", completed as Task V of this program, provides a course of action for resolution of these issues.

17. Key Words (Suggested by Author(s))
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