Engine Development Design Margins Briefing Charts

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ENGINE DEVELOPMENT DESIGN MARGINS BRIEFING CHARTS

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CHART 1 comments: ENGINE DEVELOPMENT DESIGN MARGINS

BACKGROUND: New engines experience durability problems after entering service. The most prevalent and costly is the hot section, particularly the high-pressure turbine. . The origin of durability problems can be traced back to: 1) the basic aero-mechanical design systems, assumptions, and design margins used by the engine designers, 2) the available materials systems, and 3) to a large extent, aggressive marketing in a highly competitive environment that pushes engine components beyond the demonstrated capability of the basic technology available for the hardware designs. Unfortunately the user must operate the engine in the service environment in order to learn the actual hot section life. Development testing in ground based facilities can point to some early durability problems with the hot section hardware but the actual thrust loading and the time at max effort take-off conditions used in service are needed to determine the hot section life.

Several hundred thousand hours of operational service will be required before the demonstrated reliability of a fleet of engines or the design deficiencies of the engine hot section parts can be determined. Also, it may take three to four engine shop visits for heavy maintenance on the gas path hardware to establish cost effective build standards. Spare parts drive the operator's engine maintenance costs but spare parts also makes lots of money for the engine manufacturer during the service life of an engine. Unless competition prevails for follow-on engine buys, there is really no motivation for an OEM to spend internal money to improve parts durability and reduce earnings derived from a lucrative spare parts business.

If the hot section life is below design goals or promised values, the OEM might argue that the engine is being operated beyond its basic design intent. On the other hand, the airframer and the operator will continue to remind the OEM that his engine was selected based on a lot of promises to deliver spec thrust with little impact on engine service life if higher thrust is used intermittently. In the end, a standoff prevails and nothing gets fixed.

This briefing will propose ways to hold competing engine manufacturers more accountable for engine hot section design margins during the entire Engine Development process as well as provide tools to assess the design temperature margins in the hot section parts of Service Engines.

PURPOSE: The purpose of this briefing will be to define a methodology of interaction between the customer and the competing engine manufacturers during the entire engine development process to achieve desired or promised levels of hot section durability. This briefing will also outline the Development of Design Margins for the Hot Section Components which will include the Combustor and the High and Low Pressure Turbines.

Temperature margins are absolutely needed in the hot section during engine development to reduce the durability problems and performance demands that a new engine type will face as it enters revenue service for the commercial operators or operational service for the military.

ENGINE DEVELOPMENT DESIGN MARGINS

PURPOSE

Define a methodology for customer/developer interaction during the entire engine development process to achieve desired hot section durability.

- Component development
- Engine development
- Service assessments

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CHART 2 comments: **SCOPE**

This briefing is divided into two sections.

The first section is a discussion of engine degradation in service, the impact of usage rates (flight hour/cycle) and flight legs, the sensitivity of engine life to thrust derate and thrust uprating. Comments will be made concerning the origin of durability problems.

Real engine data will be shown to illustrate the impact of engine component degradation on service life as well as the impact of take off thrust rating used in service on engine life.

The second section discusses a proposed methodology for Engine Development Design Margins for hot section parts. Some comments will be made on the value of System Engineering during engine development in terms of reducing overall Life Cycle Cost. Customer involvement is a key element of the engine development process as well as accomplishing independent assessments of critical component designs such as the high-pressure turbine. As higher levels of engine performance are being demanded, higher gas temperatures will be used in the engine cycles which puts greater demands on the accuracy of design systems used by the engine designers for the hot section parts as well as the available materials systems. By tracking the demonstrated progress of component and engine development tests, the customer will be in a better position to judge the potential life of a new engine before it selected and enters service.

Working with several engine manufacturers during an engine development encourages more accountability among the viable competitors but represents an additional workload to evaluate the progress of the hardware development to design intent. However, design deficiencies can be discovered during component testing if sufficient test instrumentation are used and the tests are conducted at temperature levels representative of engine conditions. In full scale engine testing, hot section design deficiencies may be more difficult to analyze because of the lack of high temperature instrumentation and sufficient coverage of the critical areas such as the temperature profiles exiting the high-pressure combustor and entering the exit guide vane and the first stage turbine.

SCOPE

BACKGROUND

- ENGINE DEGRADATION IN SERVICE
- IMPACT OF USAGE RATES AND FLIGHT LEGS
- SENSITIVITY TO THRUST DERATE AND THRUST UPRATING
- ORIGIN OF DURABILITY PROBLEMS

ENGINE DEVELOPMENT DESIGN MARGIN METHODOLOGY

- PROPOSED APPROACH FOR HOT SECTION PARTS
- LCC CONSIDERATIONS DUE TO DURABILITY ISSUES
- CUSTOMER INVOLVEMENT IN ENGINE DEVELOPMENT PROCESS
- TRACKING DEMONSTRATED PROGRESS
- COMPETITION ENCOURAGES MORE ACCOUNTABILITY

CHART 3 comments: **Background**

It is important to understand the impact of engine component degradation as well as the flight environment (length of flight leg and flight hours/N1 Cycle) on engine time on wing. Engine degradation is caused in part by the loss to the tip seals and airfoil shapes plus more peaked temperature profiles exiting the high-pressure combustor. Engine time on wing is reduced for various other reasons to include: 1) the more frequent use of lower derate (higher thrust settings), 2) the flight legs result in fewer flight hours/N1 Cycle or 3) the engine is uprated ("Throttle Pushed") to satisfy the operators need for more thrust. Thus, one user of a given engine type may achieve higher time on wing because his average flight legs are longer. On the other hand, another operator may experience less engine time on wing for the same flight legs than another operator for the reason that his average take off derate thrust settings are much lower which requires higher cycle temperatures resulting in accelerated hot section distress. The origin of the problem may also be that the engine hardware is operating at higher gas temperatures to produce the desired take off thrust than the design intent of the hot section. As the loss in gas path seals increase the tip clearances in both the compression system and the turbines forcing the engine cycle run hotter to produce the same engine pressure ratio, the peaked gas temperatures exiting the combustor may exceed the basic materials capability of the hot section parts. Exposure to prolonged high gas temperatures will result in severe oxidation/erosion on the leading edges and trailing edge tips of the first stage high-pressure turbine blade resulting in increased scrape rate or total replacement of the T1 Blade Stage.

When a high bypass turbofan as shown in **CHART 4** reaches 0 °C EGT Margin (defined as the Redline Temperature minus the Measured EGT measured at station 6), the engine must be scheduled for a shop visit to accomplish heavy maintenance to restore the gas path hardware performance. Otherwise, serious over-temperature to the high-pressure spool parts will result and increase the scrape rate of the vane and blades. Thus, engine time on wing is influenced by the average engine derate used in service, the average length of the flight legs, and the flight hours/N1 Cycle. One N1 Cycle is equivalent to engine idle to max RPM and return to idle. Sometimes the number of partial N1 Cycles are counted at intermediate RPM's and divided by four to increase the total accumulated N1Cycles. Counting the partial cycles may be more important to the military user than the commercial operator.

CHART 5 is provided to clarify some of the terminology commonly used in engine maintenance. Logisticians and maintainers use these metrics to forecast spare parts and workload requirements as well as judging the quality of the maintenance work performed. The concept of computing the SVR using the Shop Visit Factor (SVF) times the ERR was developed to estimate the time between heavy maintenance. Dividing TBO into the overall refurbishment cost of an engine provides the maintenance cost per engine flying hour. SVF is a function of 1 minus the fraction of engines removed by management decision or Returned to Wing for no fault and/or minor repairs. Each user has a different RTW percentage based on the condition of engine hardware in service and the prevailing maintenance policy at the military unit or commercial operator.

BACKGROUND

UNDERSTANDING THE ISSUES:

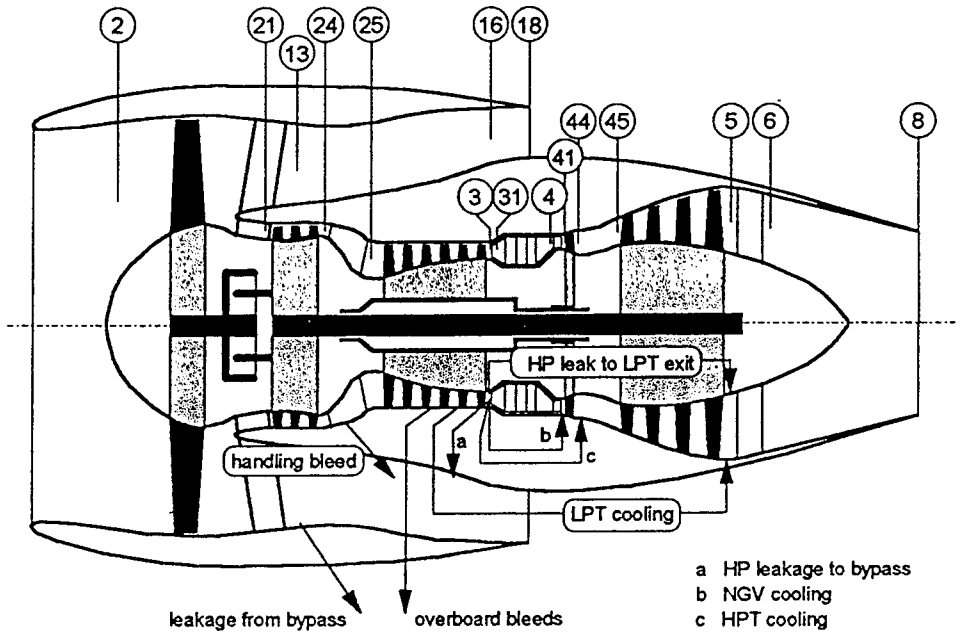
DEGRADATION

MISSION ENVIRONMENT

ENGINE DURABILITY

ORIGIN OF THE PROBLEM

UNMIXED FLOW TURBOFAN



TERMINOLOGY CLARIFICATION

ERR = SER + UER

<u>Where:</u>

ERR (Engine Removal Rate)

SER (Scheduled Engine Removal)

UER (Unscheduled Engine Removal)

SVR = SVF X ERR

SVF (Shop Visit Factor)

SVF = 1 - %RTW/100

RTW (Return to Wing)

ERR, SER, UER AND SVR are reported as 12 mo. rolling averages for the Entire Fleet. Dimensional Units (EVENTS/1000 EFH) (i.e. 0.09/1000 EFH)

<u>Categories</u>

SER's: Forced Inspections, Convenience or Planned Removals

UER's: Ground or Flight Anomaly, FOD, Hardware Failure, Leaks, etc.

TBO = Ave. Total Accumulated Cycles X Fleet Ave. EFH's/Cycle

SVR = 1000/TBO TBO refers to a Heavy Maintenance Action

CHART 6 comments: **Engine Degradation in Service**

New engines as well as refurbished engines are built to established standards by the maintainers. The level of EGT Margin for a new engine delivery is set by the customer whether the military or commercial operators. The engine will not be accepted unless the EGT Margin is above an established level of EGT Margin. During the refurbishment cycle, new parts are installed to restore the gas path performance but the engines can not be economically restored to New Engine condition. The important point of this chart is that the new or rebuilt engine goes through an Initialization and Wear in Period that decreases the available EGT Margin. Initialization occurs during the first 50 to 100 N1 Cycles whereas the Wear in Period is generally completed after 300 to 400 N1 Cycles depending upon the levels of engine derate used by the operator. During Steady State, a New Engine or a refurbished engine will provide the same rate of EGT Margin degradation as shown on the graph. If the base material capability of the T1 blades is marginal at the end of the service life, the rapid deterioration of the T1 blade surfaces will occur causing the more Rapid Deterioration Rate of the EGT Margin.

CHART 7 comments: **Flight Leg Impact of TBA**

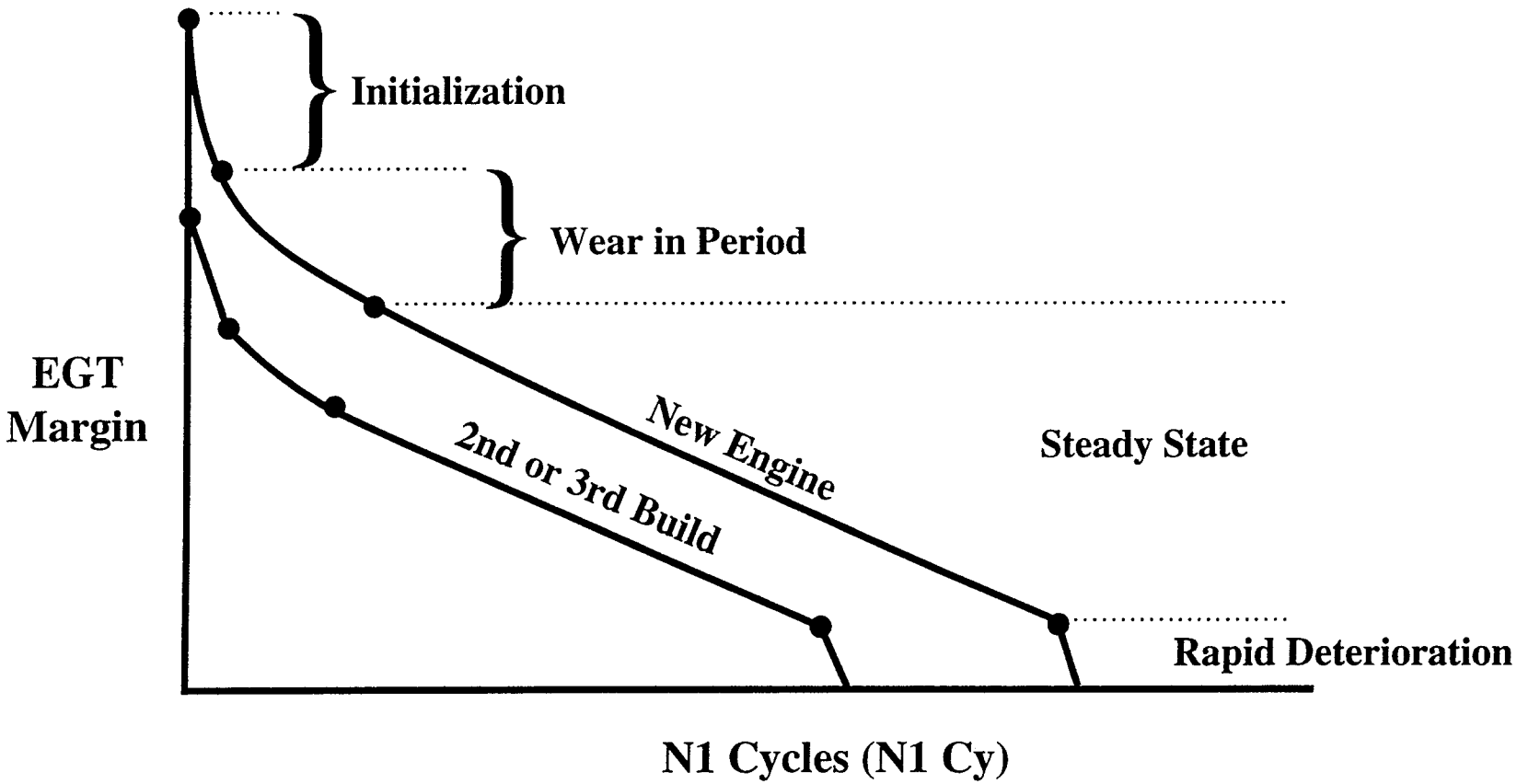
This chart provides an insight of the impact of the flight leg on engine time on wing. Commercial operators will fly long and short haul routes but accumulate only 1 N1 Cycle/Takeoff and Landing. Whereas, a military transport will fly a 3.5 hour mission and accumulate 0.7 to 3.5 EFH/N1 Cycle. High power usage accelerates engine deterioration. N1 Cycles count the number of events the Hot Section is exposed to high gas temperatures.

There is a misconception that may create some confusion, if not some targeted disinformation. The Time on Wing of an engine that experiences fewer N1 Cycle may be misleading. For instance, the build standard for a 9000 EFH engine may be around 1000 N1 Cycles if the engine flies a 9 hour flight leg. Whereas the build standard for an engine flying a 1.9 hr flight leg will be around 5000 N1 Cycles to achieve 8400 EFH's of service. The ERR of the long haul engines are generally lower than the short haul engines. The disinformation is created when certain marketing imply that the long ERR can be achieved for the short haul application. To avoid any confusion, the average flight leg should be provided for the Engine Removal Rates (ERR) that are stated to sort out the long haul engine durability from the short haul engine durability. Compare like flight leg ERR's between various engine offerings. Adjust the ERR for the same flight leg before judging the time on wing of competing engine offerings.

CHART 8 comments: **EGT MARGIN SENSITIVITY**

This chart indicates the sensitivity of EGT Margin loss for a 1% loss in engine component efficiency for a high pressure ratio engine cycle with a Bypass Ratio of 6. The most notable is the High Pressure Turbine (HPT) with a –26.2 °C. loss in EGT Margin. Depending on the level of distress from oxidation and erosion, the loss in component efficiency could be greater than 1.5 to 2%. See Comments:

ENGINE DEGRADATION IN SERVICE



FLIGHT LEG IMPACT ON TBO

- VARIOUS OPERATORS FLY A MIX OF FLIGHT LEGS AND ROTATE AIRCRAFT AND ENGINES TO MAXIMIZE TIME ON WING

- COMMERCIAL AIRLINES FLY CERTAIN AIRCRAFT TYPES ON SHORT AND LONG HAUL ROUTES AT 1 CYCLE/TAKE OFF AND LANDING

- MILITARY TRANSPORTS WILL FLY 3.5 EFH/MISSION BUT AVERAGE CYCLE RATES WILL RANGE FROM FROM 0.7 TO 3.5 EFH/CYCLE

- A 1000 CYCLE Build on a 9 HR Flt Leg will achieve 9,000 EFH before a Heavy Maintenance. A 5,000 CYCLE Build flying a 1.9 HR FLT LEG will achieve around 8,400 EFH before EGT Margin forces engine off wing.

- High Power Usage accelerates Engine Deterioration. N1 CYCLES count events the Hot Section is exposed to high gas temperatures.

- "High time engines" generally experience fewer N1 CYCLES. Beware of Low ERR Marketing unless Ave. Flight Leg is provided. Short Haul durability may be considerably worse than a competitor's engine.

EGT MARGIN SENSITIVITY
(1% LOSS IN ENGINE COMPONENT EFFICIENCY)

COMPONENT	MARGIN LOSS	COMMENTS:
• FAN	- 3.2 °C.	< 1/2 % ON A FULL RUN ENGINE
• LPC	- 2.4	< 1/2% UNLESS HIGH EROSION
• HPC	- 10	> 1 to 1.5%, HIGHER WITH SEVERE BLADE EROSION
• HPT	- 26.2	> 1.5 to 2%, DEPENDING ON DISTRESS (OXIDATION AND EROSION)
• LPT	- 16.5	< 0.1 to 0.2%, UNLESS SEVERE LOSS OF TIP SEALS

COMMENTS: CORE FLOW DECREASE FROM GAS PATH DETERIORATION WILL INCREASE MARGIN LOSS FOR THE HPC AND HPT.

CHART 9 comments: **CAUSES OF DETERIORATION**

This chart is self-explanatory and identifies the causes for component deterioration for the compression system, the combustor and the high and low-pressure turbines.

CHART 10 comments: **NON-ATTRIBUTION STATEMENT**

Real engine data are used to describe the impact of flight leg, EFH/N1 Cycle and thrust derate on engine time on wing and illustrate the need for development design margins. Since these analyses were performed, the T1 Blades for the military and commercial versions of the same engine type have been replaced with upgraded cooling designs and coating systems. The upgraded blade design should be independently reviewed at a future date (circa 1998-9) in order to determine the demonstrated improvements in operational service.

CHART 11 comments: **IMPACT OF INCREASED THRUST RATING (Same Hardware)**

This chart shows the accelerated rate of EGT Margin reduction when a 37.5K take off thrust rated engine (Baseline Engine) is operated at a 40K (Throttle Push) take off thrust rating. The fleet of 37.5K commercial engines uses an average derate of 19 to 21% based on a 40K thrust rating whereas the 40K rated engine used an average derate of 5 to 8%. The rate of deterioration increased from 10 °C/1000 N1 Cycles to a range from 14-16 °C/1000 N1 Cycles. The Initialization and Wear in Period for the engines operating at the higher thrust rating occurred at lower levels of N1 Cycles than the 37.5K rated engines.

CHART 12 comments: **PW2037 EGT MARGIN DEGRADATION**

This chart shows the degradation rate of several first time out commercial engines with an average flight leg of 2.7 EFH. All engines were removed for T1 blade distress or performance deterioration. The wear in drop was around 19-23 °C at 700 N1 Cycles. The rate of EGT Margin Degradation for four engines ranged from 10.5 °C/1000 N1 Cycles over a range from 700 to 2000 N1 Cycles to 9.2 °C/1000 N1 Cycles from 2000 to 3200 N1 Cycles. It was notable that rapid degradation (33-55 °C/1000 N1 Cycles) occurred around 8-10 °C EGT Margin remaining. Microstructure analyses indicated leading edge metal temperatures exceeded 2150 to 2250 °F. Severe loss of material on the blade tips due to oxidation and erosion was noted from the blade mid span back to the trailing edge. Loss of material on the blade tips reached 70 to 80 mils on most blades at the trailing edge.

CAUSES OF DETERIORATION

- COMPRESSION SYSTEM
  - LOSS OF TIP SEALS AND AIRFOILTIP EROSION
  - SECONDARY FLOW LEAKAGE (MINOR)

- COMBUSTOR
  - PEAKED EXIT TEMPERATURE PROFILE
  - OXIDATION AND EROSION IN THE DOME REGION
  - WARPAGE OF EXIT GUIDE VANES

- HIGH AND LOW PRESSURE TURBINES
  - INCREASED TIP CLEARANCE
    - COATING SYSTEM LOSS AND OXIDATION/EROSION
    - DEEP TIP SCHROUD RUB AND/OR LOSS OF CERAMIC MATERIAL
    - SOME BLADE WARPAGE OR UNTWIST

- GRADUAL LOSS IN CORE AIRFLOW AND PRESSURE RATIO

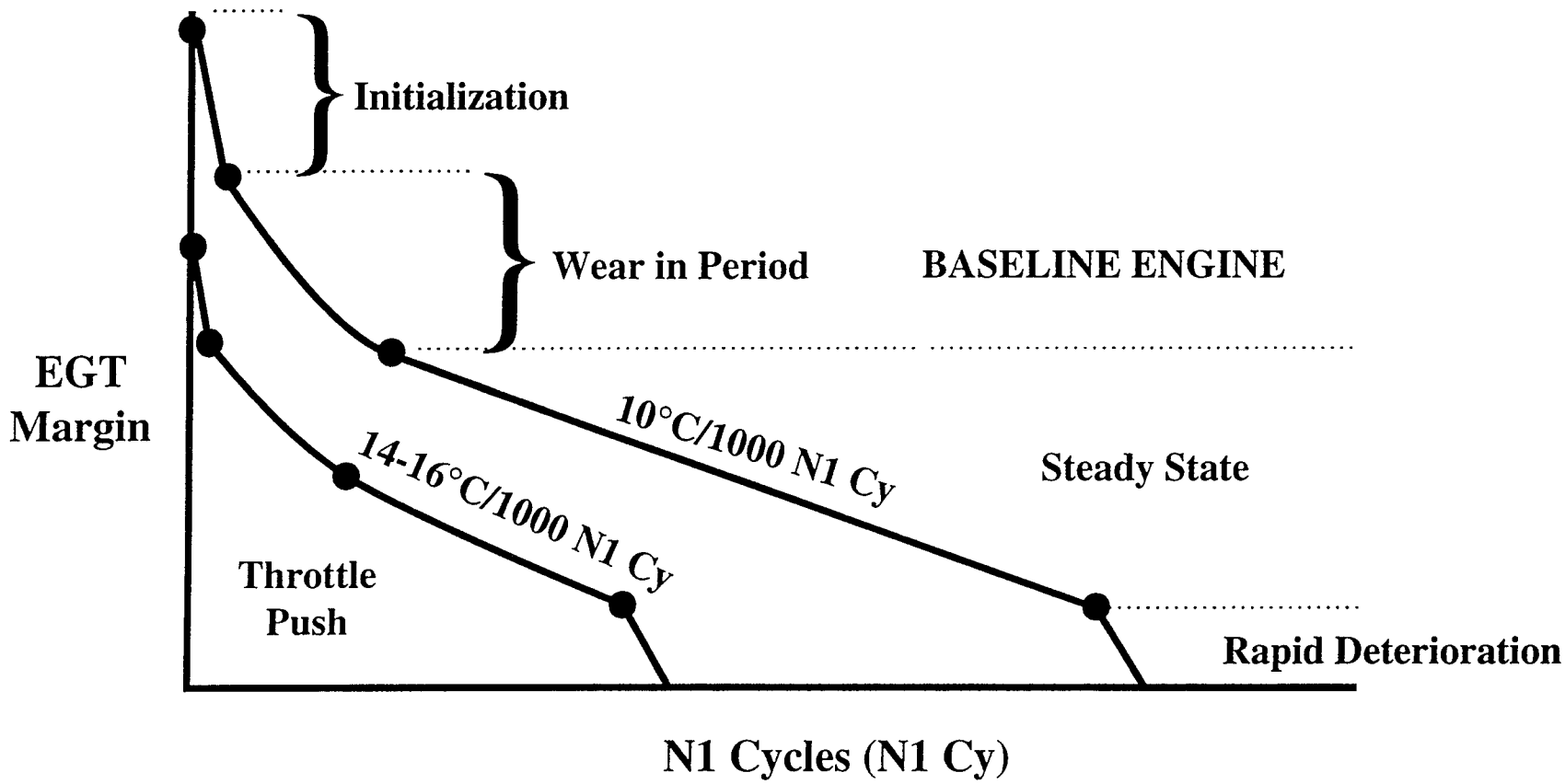
NON-ATTRIBUTION STATEMENT

ENGINE COMPONENT PERFORMANCE DETERIORATION AND THRUST UPRATE CAN SERIOUSLY IMPACT HOT SECTION DURABILITY

COMMERCIAL AND MILITARY DATA ARE PRESENTED FOR THE SAME ENGINE TYPE WITH DIFFERENT TAKE OFF THRUST RATINGS AND CYCLE USAGE RATES. OPERATING ENVIRONMENT ALSO A FACTOR.

REAL ENGINE DATA ARE USED TO ILLUSTRATE THE NEED FOR DEVELOPMENT DESIGN MARGINS

IMPACT OF INCREASED THRUST RATING
(SAME HARDWARE)



PW2037 EGT MARGIN DEGRADATION

FIRST TIME OUT COMMERCIAL ENGINES

10-12% AVE. DERATE (37.5K Fn RATED T.O.)

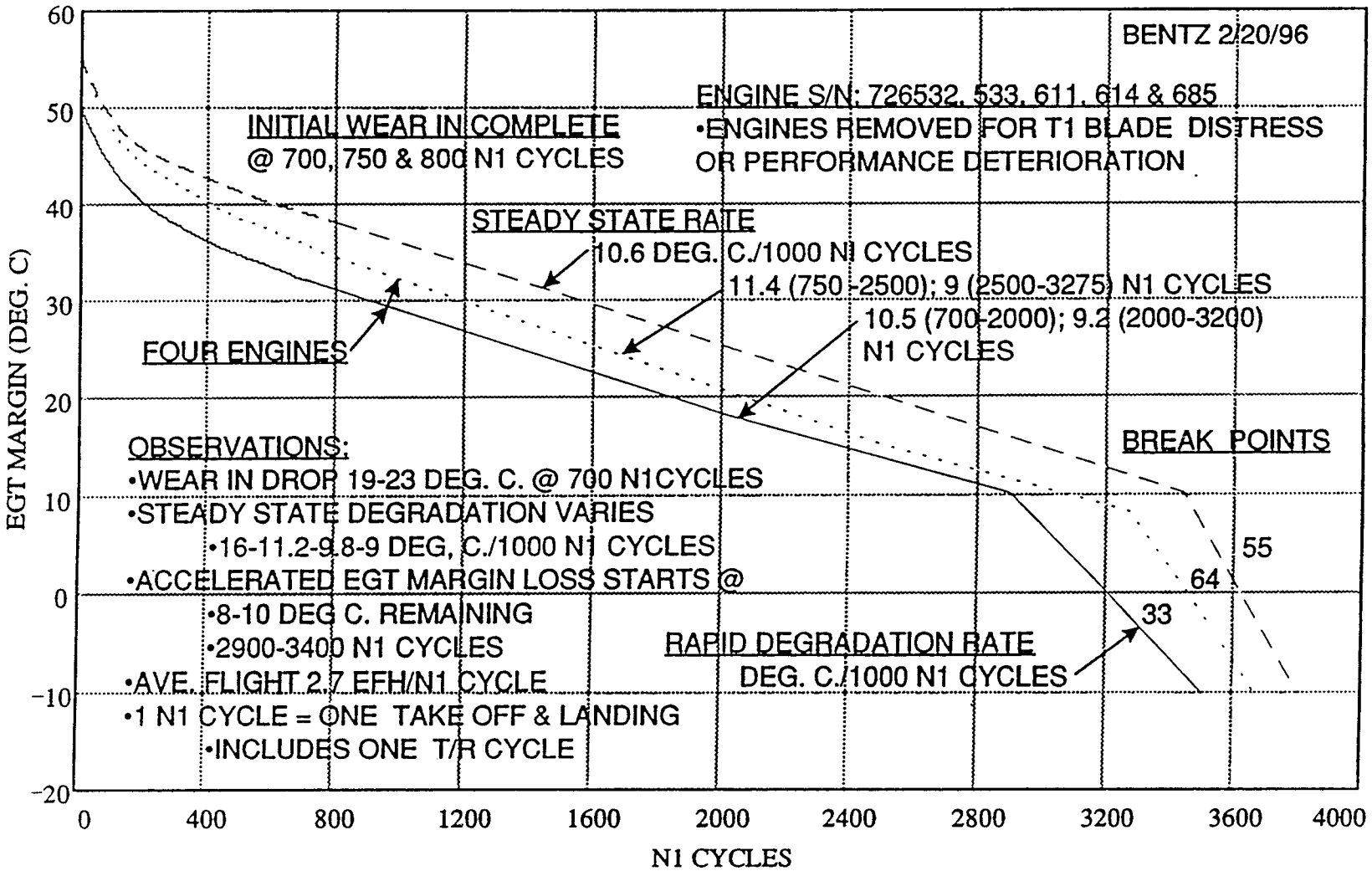


CHART 13 comments: **DURABILITY OF NEW BLADE DESIGN** (Same engine type)

This chart shows the impact that a 10% lower engine derate (higher thrust usage) has on the engine time on wing before T1 Blade distress drive the engine off the wing. The other difference was that Operator B had more Max Effort climb outs during take off. As noted, both Operators have an average flight leg of 1.94 hours however Operator B has lower Time on Wing and lower N1 Cycles before a heavy maintenance is required. The T1 blades on these engines had TBC coverage on the blade platforms only.

CHART 14 comments: **F117 DO-1 EGT MARGIN DEGRADATION**

An independent assessment of seven engines was accomplished circa early CY96. It was determined from the data analyzed that the average engine degraded at a rate of 14.5 °C/1000 N1 Cycles. The average take off derate was 12 to 16%. Max take off thrust was used during 7 to 20% of the take off conditions for the engines analyzed. Rapid degradation also occurred at around 8 to 10 °C EGT Margin remaining. **CHART 15 (COMMENTS ON ENGINE USAGE)** discusses some of the noted trends in the engine data analyzed for "average" type engine usage. The LIGHT DUTY and HARD-PRESSED Engine Usage curves were estimated based on all the data analyzed.

CHART 16 comments: **DEGRADATION CHARACTERISTICS COMPARED**

This chart summarizes the differences in several engine metrics between the commercial and military versions of the same engine type. The military version is operated at a 40K take off rating while the commercial version is rated at 37.5K thrust. The average engine usage is shown on the lower portion of the chart. The commercial version has only one max take off per month whereas max take off thrust is used in the range from 7 to 20% on some of the military engines. The other notable difference is the partial N1 Cycles per flight. The commercial engine has one partial cycle during each landing when the thrust reverser is deployed. The military engine can have 4 to 8 partial N1 Cycles on the ground and/or in flight performing one or more of the mission scenarios.

CHART 17 comments: **INDEPENDENT ASSESSMENT**

This chart summarizes the impact of uprating a commercial engine for military use, which is usually a more severe environment due to the various mission profiles, and austere operating conditions. "Throttle Push" is usually needed to offset increased drag and airframe weight or to provide for increased payload and range. However, "Throttle Push" with reduce engine life as noted by the analyses performed in the foregoing charts. The OEM must upgrade the hot section parts to provide acceptable engine time on wing and manage the durability problems of fielded engines.

DURABILITY OF NEW BLADE DESIGN

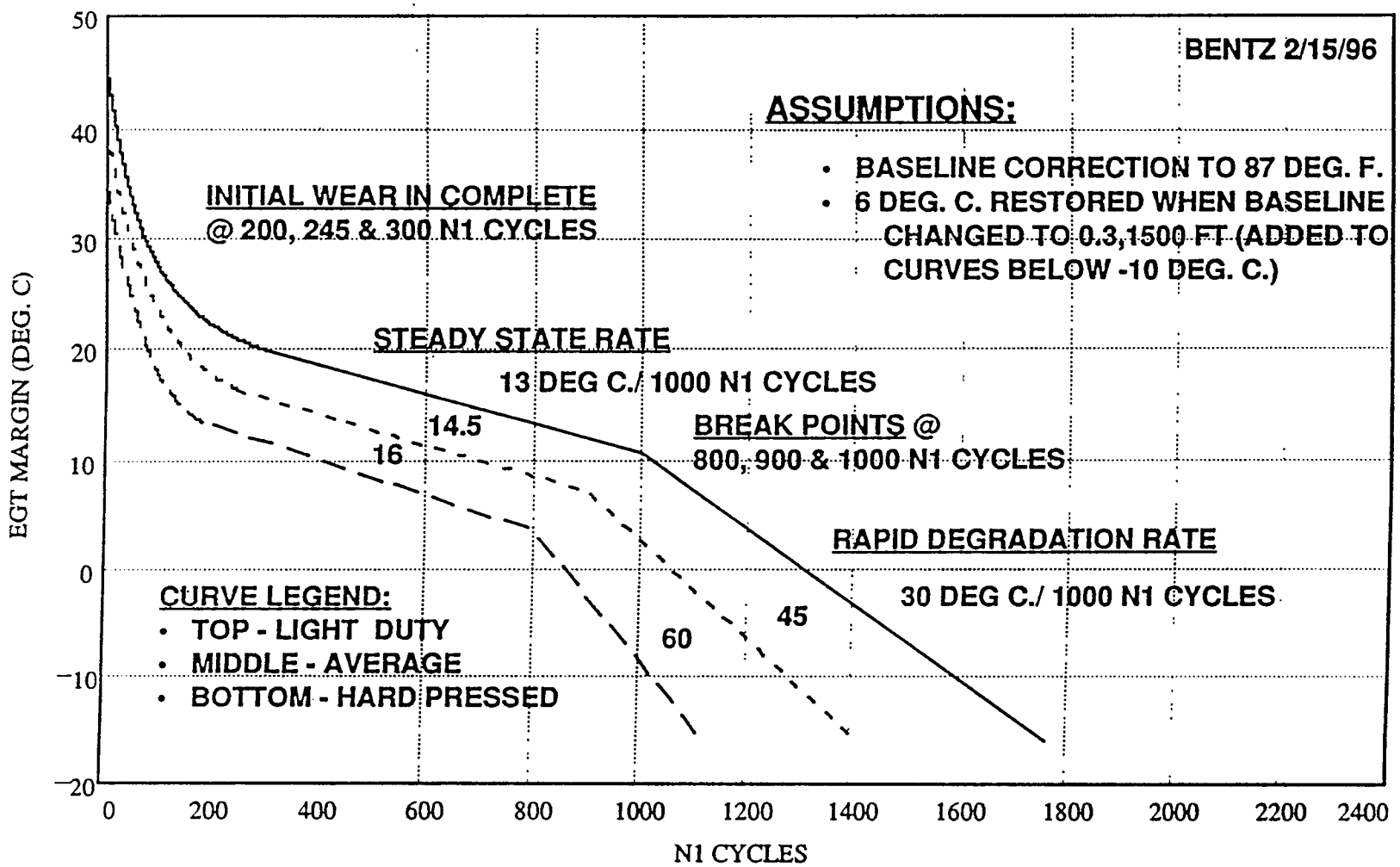
(SAME ENGINE TYPE)

METRICS	OPERATOR A	OPERATOR B
EFH'S	8200 - 9200	6100 - 6400
DERATE	19 to 21%	9 to 11%
CLIMB OUT	Throttle Retard for Noise Abatment	Many Max. Effort
AVE. FLT LEG	1.94 Hours	1.94 Hours
N1 CYCLES BEFORE HEAVY MAINTENANCE	4200 - 4700+	3100 - 3300
T. O. CONDITIONS	Some High Alt. Moderate Temp. Clean Runways	Many High Alt. Hotter Climate More Debris

F117 DO-1 EGT MARGIN DEGRADATION

P9, 10, 13, 14, 16, 17 & 18 ENGINES THRU JAN 96

(INDEPENDENT DATA ASSESSMENT)



COMMENTS ON ENGINE USAGE
F117 DO-1 EGT MARGIN DEGRADATION

TYPE USAGE	% AVE DERATE	% @ MAX T.O. Fn
LIGHT DUTY	16 TO 20	1 TO 7
AVERAGE	12 TO 16	7 TO 20
HARD PRESSED	8 TO 12	20 TO 35

NOTED TRENDS IN GO-81 DATA: (LAST 10 PTS. AVERAGED EACH REPORTING PERIOD):

- SOME ENGINES OPERATED AT HIGHER % @ MAX T.O. Fn DURING WEAR IN PERIOD WITH NO IMMEDIATE IMPACT ON EGT MARGIN DROP.
- LOWER AVERAGE DERATE INCREASES WEAR IN DROP AND STEADY STATE DEGRADATION RATES.
- AFTER 300 N1 CYCLES, THE AVERAGE % @ MAX T.O. POWER INCREASED AROUND 4 TO 8% WHILE THE AVERAGE DERATE REMAINED FAIRLY CONSTANT. AVERAGE DERATE ON SOME ENGINES INCREASED BY 3 TO 5%.

DISCLAIMER:

- THE ABOVE RANGES FOR % AVE DERATE AND % @ MAX T.O. Fn WERE DERIVED FROM ALL DATA THRU JAN 96 AND REFERRED TO AS THE AVERAGE ENGINE. THE LIGHT DUTY AND HARD PRESSED TRENDS WERE ESTIMATED USING LOWER AND HIGHER DEGRADATION RATES DURING ENGINE WEAR IN AND STEADY STATE OPERATION.

DEGRADATION CHARACTERISTICS COMPARED
F117 DO-1 VERSUS COMMERCIAL PW2037

ITEM	F117 DO-1	PW2037
WEAR IN DROP (DEG. C.)	24 - 32	19 - 23
ACCEPTANCE EGT MARGIN (DEG. C.)	45 - 55 (ENG)	55 (A/C)
START STEADY STATE RATE (DEG. C.)	13 - 21	32 - 36
@ N1 CYCLES	200 - 300	700 - 800
DEGRADATION RATE (DEG. C./1000 N1 CYCLES)	13 - 16	9 - 11.4
START RAPID DEGRADATION (DEG. C.)	8 - 10	8 - 10
@ N1 CYCLES	800 - 1000	2900 - 3400
AVERAGE ENGINE USAGE		
% DERATE @ TAKE OFF	12 - 16	19 - 21 (A)
% @ MAX T.O. THRUST	7 - 20 (B)	1
EST. PARTIAL N1 CYC PER FLT/GRND OPER.	4 - 8 (C)	1

LEGEND: (A) DERATE BASED ON F117 T.O. RATING
(B) TIME AT HIGH POWER ACCELERATES TURBINE BLADE DISTRESS
(C) NOT ALL PARTIAL CYCLES SHOULD BE COUNTED AS A MAJOR CYCLE

INDEPENDENT ASSESSMENT

- FREQUENT HIGH POWER TAKE OFF AND CLIMB OUT ACCELERATES HOT SECTION DISTRESS
- THRUST UPRATING DEMANDS INHERENT GROWTH MARGIN IN HOT SECTION COMPONENT PARTS ESPECIALLY THE HIGH PRESSURE TURBINE
- HIGHER PROFILE FACTOR AND REDUCED BLADE COOLING PERFORMANCE LEADS TO HIGHER BLADE METAL TEMPERATURES
- MICROSTRUCTURE ANALYSES VERIFIED ANALYTICAL PREDICTIONS, BLADE SURFACE TEMP. EXCEEDED DESIGN INTENT BY +150 TO +200°F
- "THROTTLE PUSH" TO OFFSET INCREASED DRAG AND AIRFRAME WEIGHT OR PROVIDE INCREASED PAYLOAD/RANGE WILL IMPACT ENGINE LIFE
- THE OEM MUST RESPOND BY UPGRADING THE HOT SECTION DESIGN AND MANAGE THE CURRENT ENGINE DURABILITY PROBLEMS

CHART 18 comments: **A CRITICAL OBSERVATION**

This chart reiterates the fact that new engines will experience durability after entering service. The most prevalent and costly to fix are the HPC, Combustor and the HPT Blades. The origin of engine durability problems is outlined again for review. Something can be done to reverse the trends but will take informed and dedicated Systems Engineering and the disciplined use of **DEVELOPMENT DESIGN MARGINS.**

CHART 19 comments: **DEVELOPMENT DESIGN MARGINS**

This chart summarizes the various topics to be covered in this section of the briefing.

CHART 20 comments: **DESIGN MARGINS APPROACH**

This chart covers the basic approach of establishing and tracking Development Design Margins. It outlines a Systems Engineering Approach that requires analytical work to be accomplished independent of the engine manufacturer. Preliminary estimates of T1 Blade leading edge temperatures are made independent of the engine designer based on assumed state of the art component characteristics. Design deficiencies are determined from component and engine testing. The customer must determine the source of the deficiencies that were determined through component or engine testing. These deficiencies may be a direct result of the design system or the component assumption used by the engine designer. The customer must also perform independent microstructure analyses of hot section parts to verify the max temperature exposure. The engine designer should be made aware that the customer will be conducting independent analyses of the hot section designs. Algorithms and design templates will be provided to accomplish the independent analyses.

CHARTS 21 and 22 comments: **LCC CONSIDERATIONS and ROI BENEFITS**

These two charts attempt to show the LCC impact of Hot Section problems and the benefits of performing dedicated Systems Engineering during an engine development. If an engine hot section were to provide only half the goal life for a fleet of 400 engines and effective System Engineering were applied to successfully achieve the goal life, then the ROI in maintenance cost reductions is estimated to be 1000.

CHART 23 comments: **MORE AFFORDABLE APPROACH**

This chart outlines a Systems Engineering Approach for use by the government and a more affordable approach for commercial operators. Specialized consulting could also be considered to accomplish some or all of the analyses.

A CRITICAL OBSERVATION

- New Engines Experience Durability Problems after Entering Service
- Most Prevalent and Costly to Fix: HPC, Combustor and HPT Blades
- Origin of the Problem
  - Basic Aero-Mechanical Design Systems, Assumptions and Design Margins
  - Available Seals and High Temperature Materials Systems
  - Development Testing Does Not Simulate Actual Service Environment
  - Aggressive Marketing in a Highly Competitive Environment
  - Evolving Mission Usage Changes (Cycles Rates and Flight Legs)
  - Voluntary Uprate Beyond Demonstrated Hot Section Durability Limits
  - Engine Choice Based on Criteria other than Sound Engineering Data
  - User Needs a New Aircraft to Replace an Aging Fleet
  - Airframer/OEM Advocacy, Keep Design Teams Busy and Production Steady
- Durability Short Falls Revealed during First Run Time in Service

DEVELOPMENT DESIGN MARGINS

PROPOSED APPROACH

LCC CONSIDERATIONS

MARGINS TEMPLATE

CUSTOMER INVOLVEMENT

TRACKING PROGRESS

COMPETITION AND ACCOUNTABILITY

DESIGN MARGINS APPROACH

Calculate Engine Cycle Performance for Candidate Engine Configurations

- Request Cycle Pressure Ratios, Bypass Ratio, Max. Turbine Inlet Temperature
- Use Design Point Component Efficiencies, Pressure Drops, Core Airflow at Sea Level Static and Cruise Conditions

Establish Demonstrated Performance Baseline for all Engine Components

- Determine Combustor Pattern and Profile Factors from Low & High Pressure Rig Tests. (Were Test Rigs Full Scale and when were tests conducted?)
- Obtain design assumptions for Overall Blade Cooling Performance (Were these levels demonstrated in both a Component Rig and Engine Test?)
- Request or Estimate Stage Cooling Flows for the Exit Guide Vane and the High Pressure Turbine (Use Increased Levels of Cooling as Required.)

Participate in Preliminary and Final Hot Section Design Reviews

- Compare independent results with Hot Section Designers Interface Control Document and the Blade Leading Edge Metal Temperature Requirements
- Evaluate the Results of Rig and Engine Tests. Sufficient Test Time at Max. Gas Temperature is Important. <u>Obtain Blades for Microstructure Analysis.</u>

Encourage OEM to Update Hardware Design, Assumptions and Practices

- Continue to Track Development Work to Correct Indentified Deficiencies

Request all Design Assumptions for Proposed Engine Design and Results of Engine Development Testing (Data should not be guarded by OEM.)

LCC CONSIDERATIONS

(ASSUME 25 YEARS SERVICE LIFE)

ASSUME THE FOLLOWING CATEGORIES IN LIFE CYCLE COST:

- DEVELOPMENT PROGRAM: $5 TO 10B (Estimate) OR NO COST IF CFE
- ENGINE COST: $8 TO 20M EACH PLUS INITIAL SPARES PROVISIONING
- MAINTENANCE COST: $1.5 TO 2M FOR EACH HEAVY MAINTENANCE WITH 8 TO 15 REQUIRED BUILDS DEPENDING ON DURABILITY AND LIFE LIMITS
- FUEL BURNED: FUEL PRICES WILL ESCALATE, $0.90 TO 1.50/GAL.

ASSUME THE FOLLOWING BREAKOUT OF TOTAL LIFE CYCLE COSTS:

- ENGINE ACQUISITION: 20 TO 25%
- MAINTENANCE COST: 25 TO 35%
- FUEL BURNED: 40 TO 60%

IMPACT STATEMENT: If the HOT SECTION provides only HALF the GOAL LIFE, then the MAINTENANCE COST will <u>increase</u> by $5B (Est.) for a Fleet of 400 Engines. Military pays CATALOG PRICE for COMMERCIAL PARTS whereas Commercial Customers demand and get prorated CREDITS for all WARRANTED PARTS and EXCESS FUEL BURNED.

ROI BENEFITS

IF 40 M/Y'S OF SYSTEM ENGINEERING WERE DEDICATED TO TRACKING ENGINE DEVELOPMENT TO INSURE THAT DURABILITY GOALS WERE ACHIEVED BEFORE DOWNSELECT, THE COST WOULD BE AROUND $5M. (Assumed $125K/MY)

THE ROI IN MAINTENANCE COST REDUCTIONS = 1000

FUEL BURNED INCREASES AS GAS PATH COMPONENTS DEGRADE, ESPECIALLY PERFORMANCE RETENTION IN THE HPC AND HPT.

A 5% INCREASE IN FUEL BURNED WOULD COST AROUND $0.6 B. THE ROI WOULD INCREASE IN THE RANGE FROM 1120 TO 1200.

THE ROI FOR IMPROVED PERFORMANCE RETENTION = 120 TO 200

MORE AFFORDABLE APPROACH

System Engineering Requires Partime and Full Time Specialists in each of the Component Areas before Engine Down Select

- Program Manager for each Engine Manufacturer
- Performance Analysts
- Dedicated Development Engineers for each Engine
- Materials and Structures Specialists
- Test Engineers

A More Affordable Approach is Possible for Commercial Operators

- Establish a Performance and Durability Analyst Position for Current and New Engines in the Propulsion Maintenance Facility
- The Program Manager for each Engine in Service, New Engines or Upgrades could request a technical assessment on Durability Issues.
- Empower the Analyst to obtain the necessary design information from the Engine Manufacturer(s) for the internal analyses.
- Explain that Future Business Opportunities would be contingent upon their full cooperation and commitment to fix deficiencies

Specialized Consulting could also be considered as needed

CHART 24 comments: **DESIGN MARGINS TEMPLATE**

This chart summarizes the four major elements of establishing and tracking Development Design Margins. They are: 1) Establish and track engine component performance parameters, 2) Establish State of the Art Values for the Performance Coefficients that affect blade and vane metal temperatures (Phi, CPF, PF, Coatings Systems, etc.), 3) Calculate the Vane and T1 Blade Metal Temperature Requirement (use Tbmr equation), and 4) Calculate metal temperature sensitivity for a range of Pattern and Profile Factors (PF) and the Overall Cooling Performance (Phi). Note: CPF is Combustor Pattern Factor used for the vane while PF is the Profile Factor used in the calculation for the T1 Blade. It is important to use degraded values for gas path component efficiencies, Pressure Ratio's and Core Airflow at the end of the service interval when calculating the gas path temperatures and the Calculated Blade Surface Temperature on the leading edge of the blade. It is also important to compare independent calculations with the design intent of the engine designer. As Component Rig and Engine Test Data are available, compare with estimates and design intent. Urge designer to correct any deficiencies and determine when the next design iteration will be tested. Obtain vanes and blades for microstructure analyses. The materials will indicate the level of high temperature exposure and eliminate some of the smoke and mirrors!

CHART 25 comments: **TRACK COMPONENT PERFORMANCE**

This chart attempts to show critical parameters of the gas path that must be tracked during the development process. The differences between the Desired and the Demonstrated values for component efficiencies and Pressure Ratio can cause a lot of error in the design temperatures of the hot section parts. Unfortunately the turbine blade design is a long lead item and any changes to the cooling design and blade castings are difficult to correct in a timely manner. Temperature Margins are important early in an engine development program to protect the design life of the hot section. Estimated values component efficiency reduction are provided for use in cycle calculations.

CHARTS 26-28 comments: **ESTIMATING T1 BLADE MAXIMUM SURFACE TEMPERATURE**

The next three charts plus the appendix entitled **DERIVATIONS** outline the general equation for estimating the maximum surface temperature on the leading edge of the first stage turbine blade. The **DERIVATIONS** Appendix contains the same derivation for the Vane. These equations are first order approximations of the surface temperature on the leading edge of the vane and blade airfoils.

DESIGN MARGINS TEMPLATE

Establish and Track Engine Component Performance Parameters

- Demonstrated Baseline at each Engine Manufacturer
- Establish Desired Levels for Proposed New Engine Centerline
- Determine Reduction in Efficiency, Pressure Ratio and Core Airflow at the end of a Service Interval

Establish State of the Art Values for Performance Coefficients that Affect Blade and Vane Metal Temperatures (Phi, CPF, PF, Coating Systems, etc.)

- Set Reasonable Levels for Degraded State at end of a Service Interval

Calculate Vane and T1 Blade Metal Temperature Requirement (Tbmr Eq.)

- Use Degraded Engine Component Efficiencies at End of Service Interval
- Compare with Design Intent, Rig and Engine Test Data

Calculate Metal Temperature Sensitivity for a Range of Pattern and Profile Factors and Overall Cooling Performance

- Use Sensitivity Curves to Determine Demonstrated Levels of CPF, PF and Phi using data from highly instrumented Developmental Engine Test
- Microstructure Analyses will provide Actual Metal Temperature Exposure

TRACK COMPONENT PERFORMANCE

Component	Demonstrated Efficiencies	Desired Efficiencies	Degraded State (Estimated)
• Fan	_____%	_____%	-1/2%
• LPC	_____%	_____%	- 1%
• HPC	_____%	_____%	- 1%
• Combustor	_____%	_____%	- 3%
• HPT	_____%	_____%	-1 1/2%
• LPT	_____%	_____%	-1/2%

	Demonstrated	Desired	Degraded State
• Fan	_____PR	_____PR	_____PR
• LPC	_____PR	_____PR	_____PR
• HPC	_____PR	_____PR	_____PR
• Combustor	_____ΔP/P	_____ΔP/P	_____ ΔP/P
• HPT	_____PR	_____PR	_____PR
• LPT	_____PR	_____PR	_____PR
• Corr. Wa	_____Core	_____Core	_____Core

ESTIMATING T1 BLADE MAXIMUM SURFACE TEMPERATURES

The general equation for estimating the maximum surface temperature on the leading edge of the first stage turbine blade is:

Tm = (1 - Phi)/RF[PF(T4-T3) + T41] + Phi(T3 + delta Tp)

Definition of Coefficients:

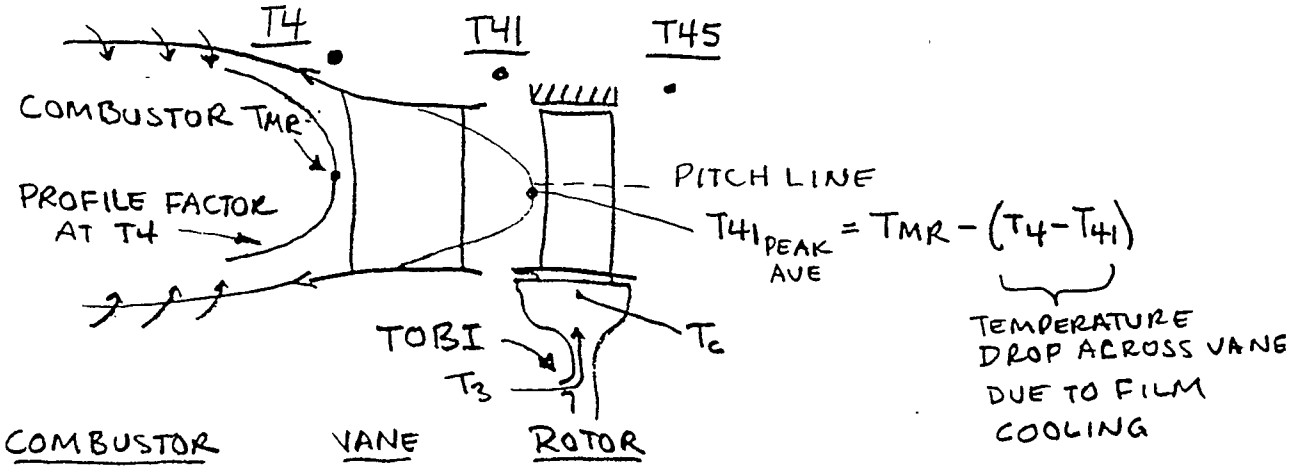
Phi	Overall Blade Cooling Performance used in the blade design
RF	Relative Rotor Factor used to calculate the relative rotor temperature
PF	Combustor Profile Factor used to estimate the maximum mean radial temperature entering the turbine stage
Tp	Delta temperature rise of the cooling air T3 due to disk pumping

Comments:

The above equation is a first order approximation of the surface temperature on the leading edge of the first stage turbine blade at the pitch line.

Tm = Tcbs Calculated Blade Surface Temperature

COMBUSTOR EXIT TEMPERATURE PROFILE



(PF) COMBUSTOR PROFILE FACTOR $= \dfrac{T_{MR} - T_4}{T_4 - T_3}$

(T_{RR}) $T_{RELATIVE\ ROTOR} = \dfrac{T_{41}}{\text{ROTOR FACTOR}}$

(PHI) OVERALL COOLING PERFORMANCE $= \dfrac{T_{RR} - T_M}{T_{RR} - T_C}$

$$T_C = T_3 + \Delta T_{PUMPING}$$

T_{MR} = MEAN RADIAL TEMPERATURE

FROM GAS TURBINE COMBUSTION BY ARTHUR H. LEFEBVRE

Thus, it is highly desirable that rig work on the improvement of temperature traverse quality should be carried out at the maximum engine pressure, since this corresponds to maximum heat-transfer rates to nozzle guide vanes and turbine blades.

Another difficulty in the assessment of pattern factor or temperature traverse quality stems from lack of confidence in the reliability of the experimental data. Because of the time and cost involved, temperature surveys are necessarily based on a limited number of thermocouple readings or gas samplings, and a very small change in the location of a probe can make all the difference between recording and failing to record a "hot spot" in the exit-temperature traverse. Owing to the inherent variability, it is highly desirable to examine data obtained from three or four chambers, all built to the same standard, when defining a typical pattern.

The most important temperature parameters are those that affect the power output of the engine and the life and durability of the hot sections downstream. As far as overall engine performance is concerned, the most important temperature is the turbine inlet temperature T_4, which is the mass-flow-weighted mean of all the exit temperatures recorded for one standard of liner. Since the nozzle guide vanes are fixed relative to the combustor, they must be designed to withstand the maximum temperature found in the traverse. Thus, the parameter of most relevance to the design of nozzle guide vanes is the overall temperature distribution factor, which highlights this maximum temperature. It is normally defined as

$$\text{Pattern factor} = \frac{T_{max} - T_4}{T_4 - T_3} \tag{4.42}$$

where T_{max} = maximum recorded temperature
T_3 = mean inlet air temperature
T_4 = mean exit temperature

The temperatures of most significance relative to the turbine blades are those that constitute the average radial profile. They are obtained by adding together the temperature measurements around each radius of the liner and then dividing by the number of locations at each radius, i.e., by calculating the arithmetic mean at each radius. A typical radial temperature profile is shown in Fig. 4.27. The expression used to describe the radial temperature distribution factor, also known as the *profile factor*, is

$$\text{Profile factor} = \frac{T_{mr} - T_4}{T_4 - T_3} \tag{4.43}$$

where T_{mr} = maximum circumferential mean temperature

The pattern factor and profile factor, as defined above, are best suited for situations in which a perfectly uniform exit-temperature distribution would be considered ideal. However, in modern high-performance engines, which employ extensive air cooling of both nozzle guide vanes and turbine blades, the desired average radial distribution of temperature at the combustor exit plane is far from flat; instead, it usually has a profile that peaks above the midheight of the blade, as

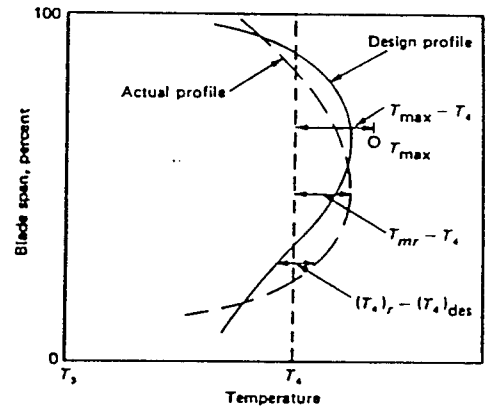


FIG. 4.27 Explanation of terms in exit temperature profile parameters.

illustrated in Fig. 4.27. A parameter that takes the design profile into account is the *turbine profile factor*, defined as

$$\text{Turbine profile factor} = \frac{(T_{4,r} - T_{4,des})_{max}}{T_4 - T_3} \tag{4.44}$$

where $(T_{4,r} - T_{4,des})_{max}$ is the maximum temperature difference between the average temperature at any given radius around the circumference and the design temperature for that same radius.

Statistical methods are sometimes used to describe the temperature distribution at the combustor outlet. Their main application is in combustor development, where they are used both in analyzing the key factors governing the temperature distribution of any given combustor, and in helping to identify the causes of differences in measured temperature distribution among combustors of nominally the same design [48 to 50].

Dilution-Zone Design

At this stage in the design process, the amount of air available for dilution purposes will have been established, using charts of the type illustrated in Fig. 4.28, along with estimates of liner diameter and liner pressure-loss factor. The principal dilution-zone design variables are the number and size of the air-admission holes and the zone length. To ensure a satisfactory temperature profile at the chamber outlet, there must be adequate penetration of the dilution air jets, coupled with the correct number of jets to form sufficient localized mixing regions. Now the penetration of a round jet is a function of its diameter [see Eq. (4.27)]. If the total dilution-hole area is spread over a large number of small holes, penetration will be inadequate, and a hot core will persist through the dilution zone. At the other extreme, the use

CHART 29 comments: **BLADE TEMPERATURE SENSITIVITY TO COMPONENT PERFORMANCE COEFFICIENTS**

This chart is provided to show the range of performance degradation that can be expected for the performance factors shown. It should be pointed out that the levels for the Overall Cooling Performance (Phi) are very optimistic. The New Engine Phi should be in the range of 0.55 to 0.5 rather than 0.75 to 0.7. These later values were used in an advanced study. The importance of using degraded values for the Performance Factors and the Advanced Coating Systems is to insure that the condition of the hot section parts at the end of each service interval are serviceable and have a low scrape rate. Otherwise, the cost to refurbish the high-pressure turbine will be extremely high. The Relative Rotor Factor is a function of the vane and blade stage design and difficult to calculate but must be considered. The engine designer must declare a development pad for the T1 Blade to accommodate unforeseen development problems and for a "Throttle Push" that may be needed to provide more engine thrust later in the program. Uprating the engine should not result in a corresponding decrease in engine time on wing. The OEM should provide a new T1 Blade design for the uprated engine especially if the current blade design is not meeting engine life goals.

CHART 30 comments: **PROPOSED EQUATION FOR ESTIMATING T1 BLADE METAL REQUIREMENTS**

The equation to calculate the Base Metal Requirement (Tbmr) for the T1 Blades contains eight terms that are described under Definition of Terms. The bracketed terms are referred to as the component sensitivity factors that take into consideration the additional temperature margin that has to be accounted for at the end of the service interval in order to avoid any distress on the first stage blades and allow repair. Use of this equation will be demonstrated in **CHART 33**. A more detailed discussion of the rationale for each of the terms in the equation is contained in a report prepared for the Turbine Engine Division of the Air Force Propulsion and Power Laboratory at WPAFB, OH.

CHART 31 and 32 comments: **MIXED FLOW TURBOFAN and SAMPLE GASTURB CALCULATION**

A cross section of the an mixed flow turbofan engine is provided in **CHART 31** to understand the sample cycle calculations made by GasTurb for Windows shown in **CHART 32**. A baseline cycle calculation is made using the proposed component efficiencies, pressure drops and design pressure ratio's at the select design point. GasTurb calculates Tcbs in the Composed Values for different levels of Overall Cooling Performance (Phi). It can be noted that a Phi of 0.5 provides a 243 °F higher Tcbs than a blade design using a Overall Cooling Performance of 0.7. This large increase in blade surface temperature for a small reduction in cooling performance sounds the alarm to verify the overall cooling performance in the engine rather than a component rig test.

BLADE TEMPERATURE SENSITIVITY TO COMPONENT PERFORMANCE COEFFICIENTS

Performance Factors	Range of Performance	
Overall Cooling Performance	New Engine	0 75 to 0 7
	End of Service Life	0 6 to 0.5
Combustor Profile Factor	New Engine	0.25 to 0.28
	End of Service Life	0.28 to 0.32

Note: End of Service Life refers to End of Service Interval

Advanced Coating Systems	New Engine	200 °F
1st Generation	End of Service Life	100
2nd Generation	New Engine	300 °F
	End of Service Life	100
3rd Generation	New Engine	400 °F
	End of Service Life	100

Note: Current TBC Systems initially provide 50 to 75 °F of protection on the blade depending on thickness, but **due to erosion on the leading edge down to the base material, TBC provides no protection of the base material near the end of the Service Interval and all blades must be scraped due to oxidation.** The assumed 100 °F protection at the end of a Service Interval is within the inspection capability of flight line boroscopes since erosion or spallation will expose the base material on some of the blade leading edges. It is also difficult to maintain uniform thickness of TBC on the blade leading edge within 1 to 2 mils.

Rotor Factor	New Engine	0 °F. (1.15 Baseline)
Development margin	N2 speed changes	+/- 20 °F.
Tm Pad for "Throttle Push"	Need more thrust	+50 °F

Note: The assumed +50 °F. Pad provides for engine growth with components that have achieved their design efficiency and pumping goals.

PROPOSED EQUATION FOR ESTIMATING T1 BLADE METAL REQUIREMENTS

Tbmr = Tcbs - Ttbp + Tgmr + [Tdec + Tltp + Tlcp + Thpf] +/- Trrf

Note. The bracketed terms, [Tdec + Tltp + Tlcp + Thpf], are referred to as the component sensitivity factors.

Definition of Terms:

Tbmr Base Metal Requirement for the T1 Blades.

Tcbs Calculated Blade Surface Temperature using the **baseline** assumptions for the blade design. In this study, Phi=0.7, PF=0.25, RF=1.15 and a delta temperature rise of 156 °F for disk pumping of the cooling air, T3.

Ttbp Thermal Barrier Protection of TBC Systems.

Tgmr Assigned Growth Margin Reserve.

Tdec Degraded Engine Component Efficiencies. Assumes pumping capacity of the Fan, LPC and HPC and the turbine work coefficient have all been achieved.

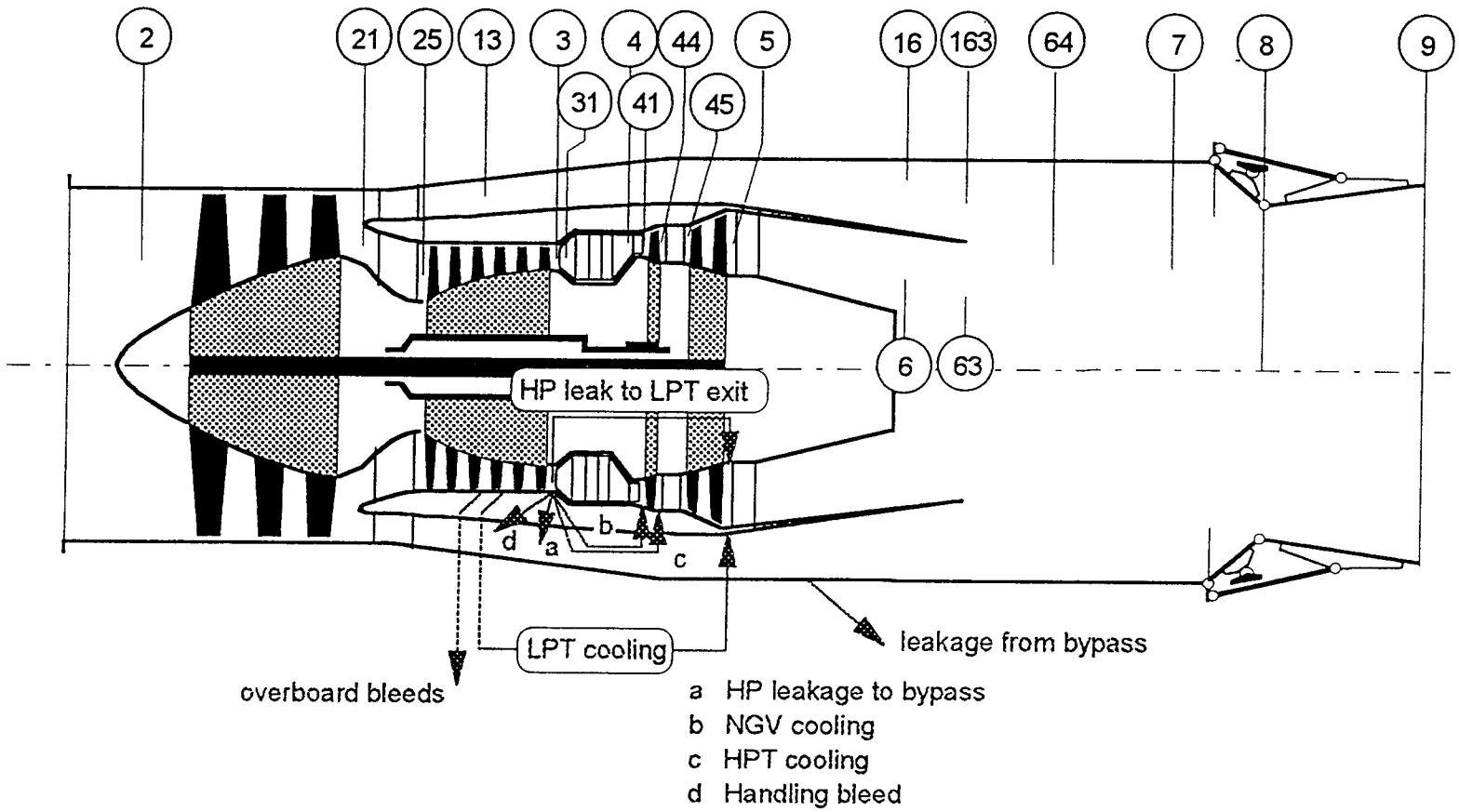
Tltp Loss in Thermal Protection for TBC system due to erosion and/or spallation.

Tlcp Lower Overall Cooling Performance demonstrated than design intent and from deterioration during operational use.

Thpf Higher Combustor Profile Factor than design intent and/or from deterioration during operational use.

Trrf Relative Rotor Factor.

MIXED FLOW TURBOFAN
WITH VARIABLE EXHAUST NOZZLE



SAMPLE GASTURB CALCULATION

File: C:\GTB WIN\LAST_MTF.CYM - modified
Date: Jun2497
Time: 10:06

mixed Turbofan Alt=60000ft / Mn=3.00 ISA

T4=4121.8°R., T3=2460, 2X Cooling, Degraded Comp. Perf., Phi=0.7,0.6,0.5

Station	W	T	P	WRstd
amb		389.97	1.040	
2	156.571	1091.92	30.902	108.035
13	37.005	1402.24	66.008	
21	119.565	1552.69	91.070	33.383
25	119.565	1552.69	91.070	33.383
3	117.174	2459.83	486.311	7.711
4	105.677	4121.80	463.844	9.439
41	115.242	3994.91		10.133
45	120.025	3152.20	134.357	32.365
5	122.416	2677.29	59.291	68.936
6	122.416	2677.29	58.106	
16	37.005	1402.24	61.017	
64	159.422	2402.75	58.453	
8	159.422	2402.75	58.453	86.267

P2/P1 = 0.8088 P4/P3 = 0.9538

Efficiencies:	isentr	polytr	RNI	P/P
Outer LPC	0.7929	0.8124	0.60	2.136
Inner LPC	0.7879	0.8153	0.60	2.947
HP Compressor	0.8832	0.9042	0.98	5.340
HP Turbine	0.8748	0.8565	1.04	3.418
LP Turbine	0.8862	0.8752	0.45	2.266
Mixer	0.9900			
Con-Di Nozzle:				
A9*(Ps9-Pamb)			7.648	

FN	=	8068.83
SFC	=	1.5387
WF Burner	=	3.4488
BPR	=	0.3095
P25/P21	=	1.0000
Core Eff	=	0.6390
Prop Eff	=	0.7866
P3/P2	=	15.737
P16/P6	=	1.05011
A64	=	772.08933
A63	=	689.61064
A163	=	82.47869
XM63	=	0.18263
XM163	=	0.32351
A8	=	257.62282
P6/P5	=	0.98000
P16/P13	=	0.92440
W NGV/W25	=	0.08000
WHcl/W25	=	0.04000
WLcl/W25	=	0.02000
WBLD/W21	=	0.00000
WBLD/W25	=	0.00500
PWX	=	0
A9/A8	=	5.99000
XM9	=	3.20704
CV9	=	0.97388

Composed Values:

Fn/W2	= 51.53469		
0.635*T3+0.0652*T4+0.261*T41+109.2	= 2982.60327	(2523 °F)	ϕ = 0.7
0.513*T3+0.087*T4+0.348*T41+93.6	= 3104.31665	(2644 °F)	0.6
0.391*T3+0.1087*T4+0.435*T41+78	= 3225.61719	(2766 °F)	0.5
P8qamb	= 56.19679		

CHART 33 comments: **SAMPLE CALCULATION OF THE TURBINE BASE METAL REQUIREMENT**

The end of the service interval calculations must be accomplished to determine the temperature increases that must be accounted for when the engine components degrade. A baseline cycle calculation is used to determine the delta increase in the Tdec. Loss in Thermal Protection, Leading Edge Phi and Higher Profile Factor are determined for each of the design points. The examples shown are for the SLS and M3/60K Cruise operating points. It is noted that the estimated Tbmr of 2583 °F at SLS is beyond current materials capabilities. Current materials start to oxidize and erode above 2050 °F and melt at around 2400 °F. The loss in thermal barrier protection is caused by erosion that decreases the TBC thickness and cannot be avoided. Limited exposure to fine particles on the ground or in the air can cause a sudden drop in EGT Margin and a corresponding increase in Turbine Inlet Temperature to maintain the same engine thrust which will destroy the T1 Blades.

CHART 34 and 35 comments: **Tm SENSITIVITY TO COOLING PERFORMANCE and Tm SENSITIVITY TO PROFILE FACTOR (Both at SLS Conditions)**

If Cooling Performance (Phi) decreases from 0.7 to 0.5, Tm will increase by 200 °F at a T4 of 3460 °R. At 4070 °R, Tm would increase approximately 325 °F. A Profile Factor change from 0.25 to 0.35 would result in approximately 36 °F increase in Tm at a T4 of 3460 °R. Overall Cooling Performance provides a significant change to Tm as noted. It is extremely important that advanced high temperature cycles use realistic and attainable levels of Phi. These levels should be based on engine testing rather than component rig or coupon tests as the scale of turbulence, air density and configuration differences will result in a poorly executed cooling system design and negative temperature margins.

CHART 36 comments: **CUSTOMER INVOLVEMENT**

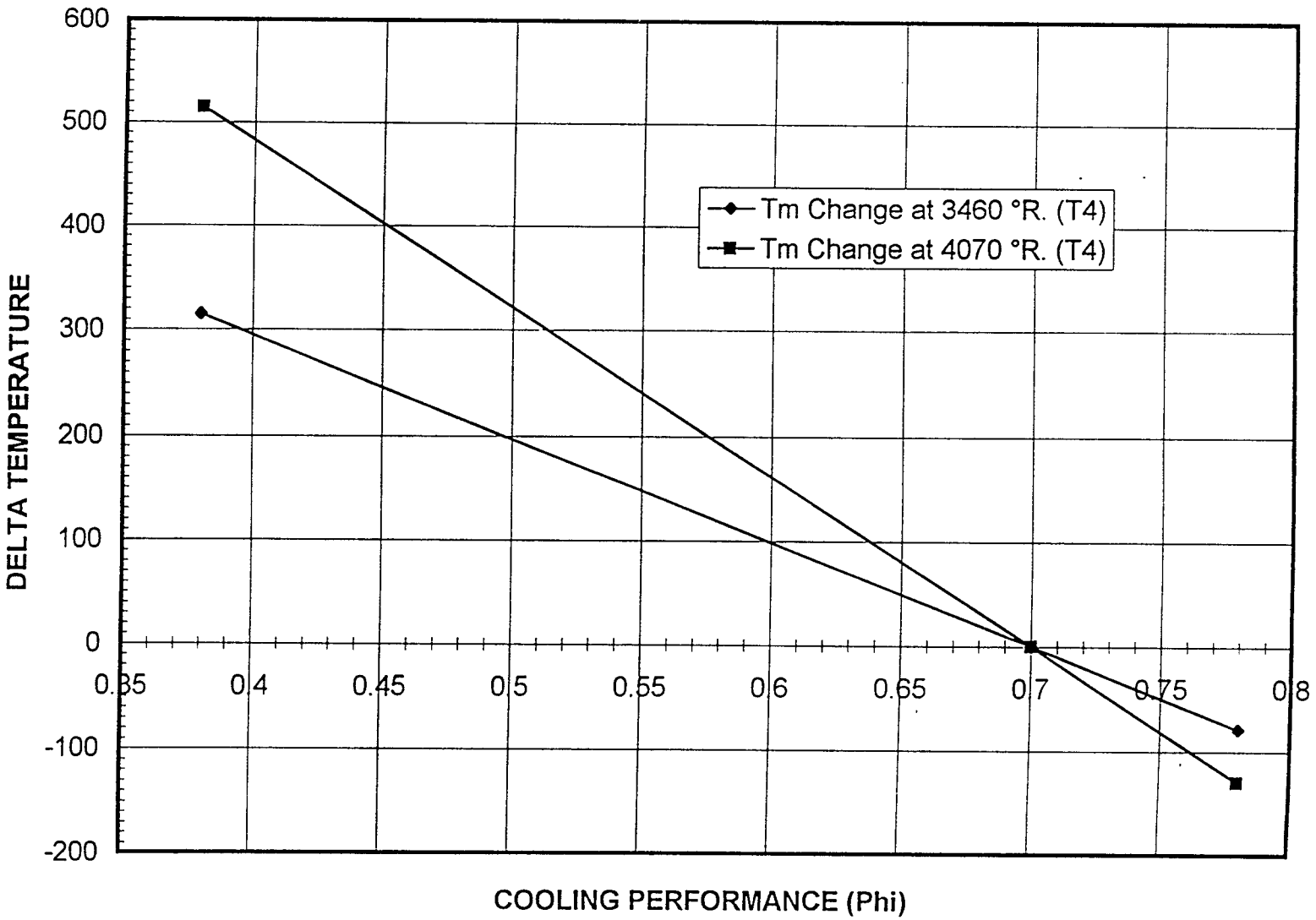
This chart is self-explanatory. Get involved. Obtain responsible design information and development data from all tests. Insure design stability of the gas path components. Nothing is more frustrating for a turbine designer than to have the design temperature increased with no available temperature margins or time for a complete redesign of the blade and cooling schemes. The other important point is to participate in conceptual, preliminary and final design reviews. Do your homework and come to the meeting prepared with good questions. Expect to get slow rolled or isolated! Microstructure analyses of distressed hardware will determine who is right!

SAMPLE CALCULATION OF THE TURBINE BASE METAL REQUIREMENT

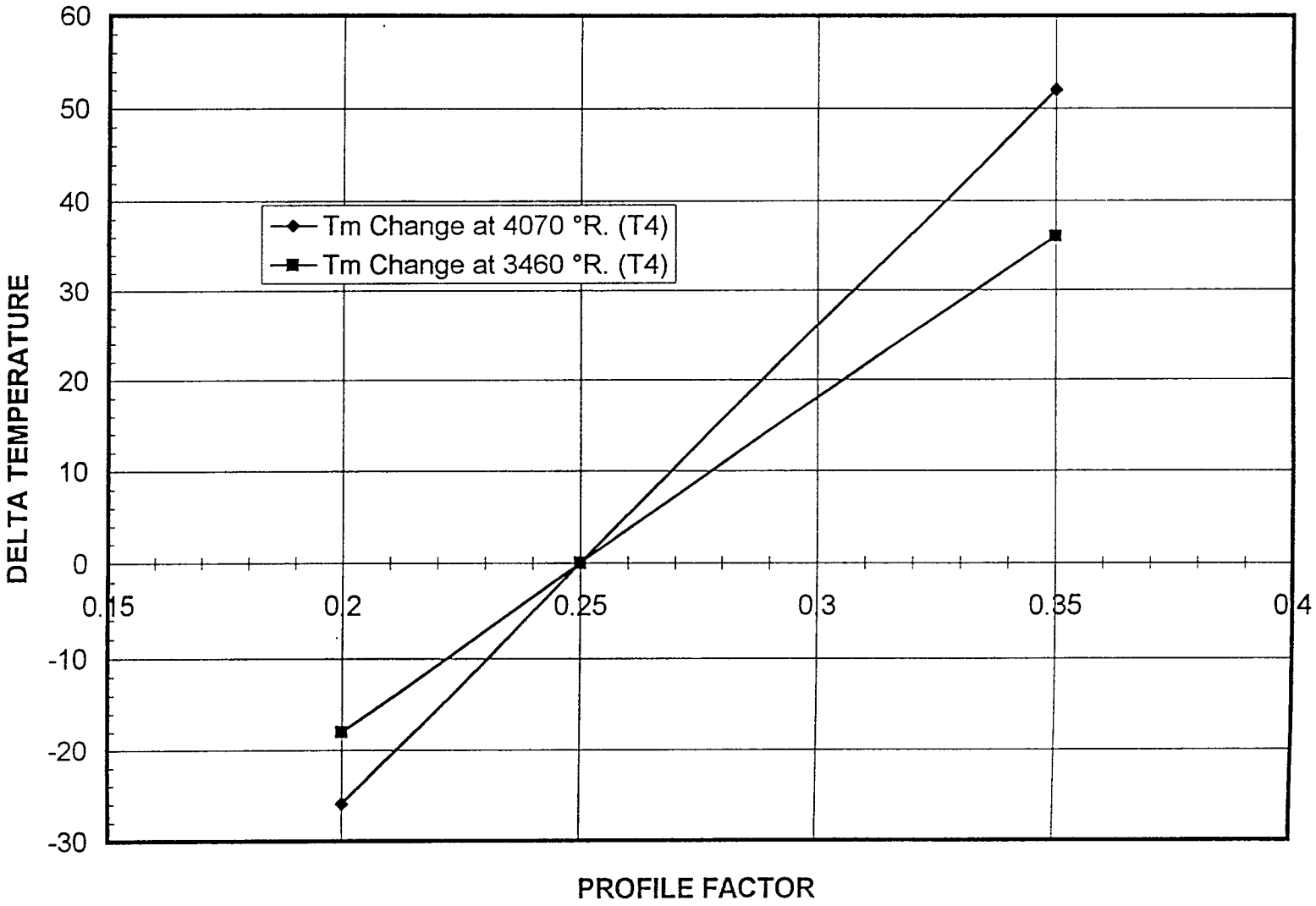
Factor (°F.)	Comments/Assumptions	SLS 50K Fn	M3/60K Cruise
Tcbs	T4=4240 °R. SLS	2285 °F.	
	T4=4122 °R. M3/60K		2523 °F.
	(no efficiency degradation)		
Ttbp	2nd Generation TBC	-300 °F.	-300 °F.
Tgmr	Fn Growth and Reserve	+50 °F.	+50 °F.
Tdec	Combined Efficiency Loss	+31 °F	0
Tltp	Loss in Thermal Protection during Service Interval	+200 °F.	+200 °F.
Tlcp	Leading Edge Phi=0.55 (End of Service Interval)	+279°F (310-31)	+182 °F. (182-0)
Thpf	Higher PF=0.32 (End of Service Interval)	+38 °F.	+30 °F.
Trrf	Estimated RF=1.13 to 1.17	+/- 20 °F.	+/- 20 °F.
Tbmr	Base Metal Requirement (Estimated)	2583 +/- 20 °F. at SLS	2685 +/- 20 °F at M3/60K Cruise

Comments: A seasoned engine designer will argue that the above base metal requirements for the T1 Blades are too one sided. The analysis should have considered a random Monte Carlo distribution in selecting the performance degradations for the various "Component Sensitivity Factors".

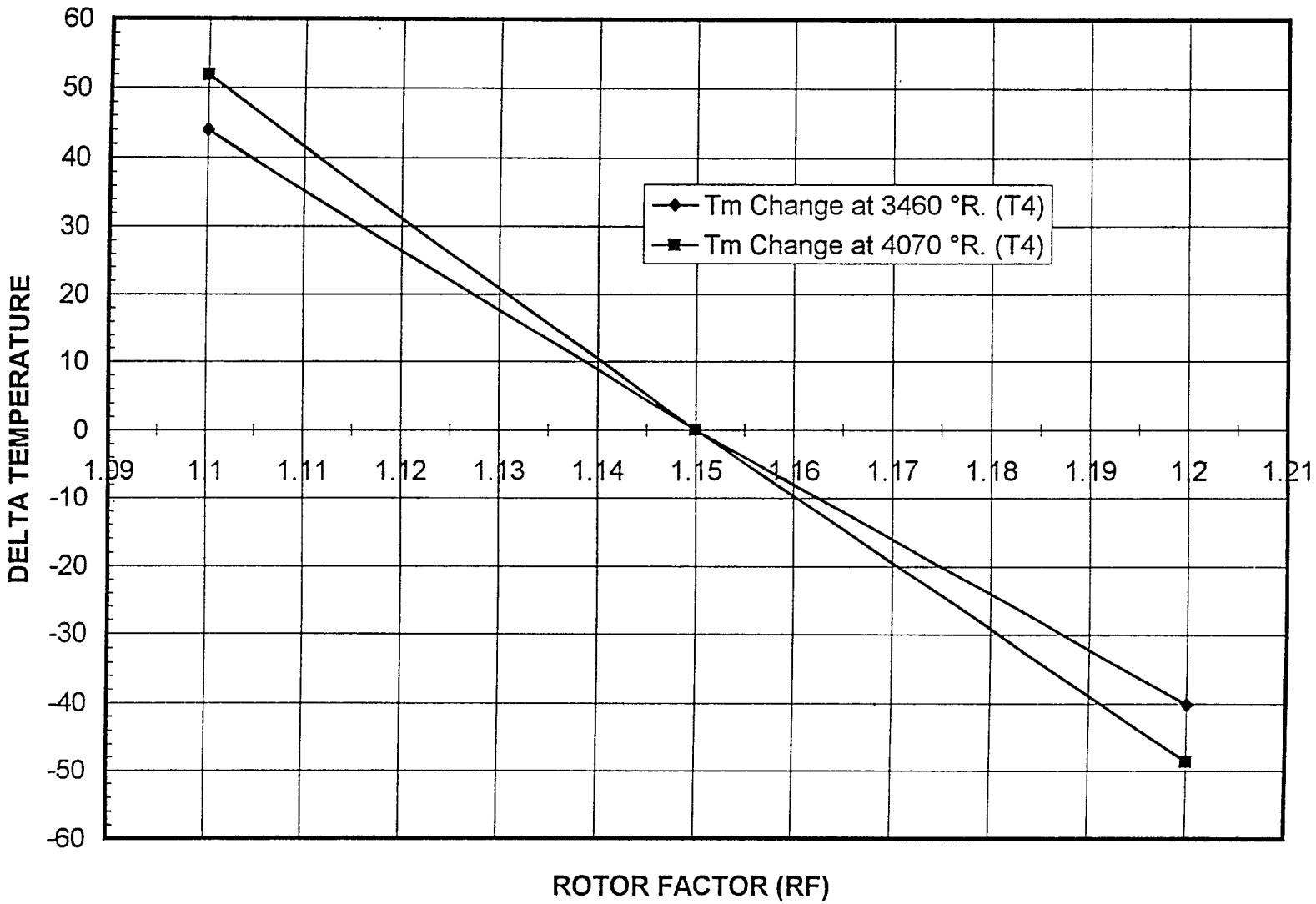
Tm SENSITIVITY TO COOLING PERFORMANCE
SEA LEVEL STATIC CONDITIONS



Tm SENSITIVITY TO PROFILE FACTOR
SEA LEVEL STATIC CONDITIONS



Tm SENSITIVITY TO ROTOR FACTOR (RF)
SEA LEVEL STATIC CONDITIONS



CUSTOMER INVOLVEMENT

Obtaining Responsible Design Information and Development Data will be a Major Challenge

- Don't be lulled into accepting Optimistic Design Assumptions in an Aggressive Advanced Engine Design. Hot Section Life will Suffer.
- Purpose to interact with the Component Designers and Development Engineers in side meetings rather than Top Level Marketing Information
- Current Operational Engines and Advanced Development Engine Hardware will provide many insights if not a lot of concerns!

Insure Design Stability of the Gas Path Components

- Vanes and Blades will run hotter than designed if the Compressor, Combustor and the Overall Cooling Performance goals fall short of the design intent.
- Hot Section Parts require long lead design and development intervals
- 1/4 to 1/2X Blades will Cost the User lots of money. Warrany Guarantees are purposely structured to protect the OEM yet make money on spare parts.

Participate in Conceptual, Preliminary and Final Design Reviews

- Do your homework. Come prepared to asked informed questions based on independent analyses. Expect to get Slow Rolled or Isolated!
- Don't be alarmed if your analyses are not in agreement with the designers.
- Microstructure analyses of distressed hardware will determine who is right!

CHART 37 and 38 comments: **TRACKING PROGRESS and COMPETITION AND ACCOUNTABILITY**

Independent analyses will create a greater appreciation and understanding of the accomplishments of the engine manufacturer and the areas for improvement. Continue to remind the competitors that Positive Design Margins for the Hot Section Parts are absolutely essential. Always question the Design Practices and Procedures plus the engine design assumption when distressed hot section parts result after an engine test. Use the Cooling Performance and Profile Factor Sensitivity curves prepared for the engine design to debate whether the correct Phi and PF were used in the blade design and/or were the engine component efficiencies lower than design requirements. The benefits and liabilities of TBC Systems must be continually evaluated for affordability and reparability.

Competition is key in having the option to procure the most durable engine type that meets the performance requirements of the airframe. Independent analyses will encourage more accountability from the engine manufacturers in the competition. Continue a relentless pursuit. Operational and Maintenance Budgets will benefit.

CHART 39 comments: **SUMMARY**

In conclusion, become more involved in all phases of the engine design and development process. Use the tools suggested in this briefing. Conduct independent analyses of all component designs. Compare "Hard Pressed Hardware" test results with the design intent. Understand and assess the technology baselines of all components at each engine manufacturer. Evaluate the Design Systems and Best Practices at each engine company. Microstructure analyses of hot section parts are essential to pass judgment. Make sure that the parts that are analyzed have been exposed to sufficient hot time at high power engine design temperatures.

Engine Design Margins are required to successfully field an engine with acceptable durability, survive component degradation and low derate operation in service. Design Margins will allow some thrust uprate as needed to satisfy system requirements.

Always balance risk with reasonable levels of conservatism.

Pursue excellence in Engine Development. Operational and Maintenance budgets will benefit.

TRACKING PROGRESS

Independent analyses will create a greater appreciation and understanding of accomplishments and areas for improvement.

- Let OEM's know that independent analyses will be conducted
- Keep a "Pearl Harblor File" of Design Intent and Demonstrated Capability
- Monitor Redesign work and Development Tests. Re-evaluate results.

Continue to remind competitive engine manufacturers that Positive Design Margins for the Hot Section Parts are absolutely essential

- Achievable levels of Combustor Pattern and Profile Factors and Overall Cooling Performance are encouraged in the Hot Section Design
- Question the Design Practices and Procedures when distress is noted on Hot Section Parts. Were the engine design conditions under-estimated?
- Use the Cooling Performance and Profile Factor Sensitivity Curves to debate whether these design factors are below design intent or the gas path airflow and component efficiencies are below gas path design requirements.

The Benefits and Liabilities of Thermal Barrier Coating Systems and Blade Casting Technologies must be continually evaluated for affordability and repairability. 2 to 3X Blades are desirable. 1X Blades are a "Cash Cow".

COMPETITION AND ACCOUNTABILITY

Competition is key in having the option to procure the most durable engine type that meets the performance requirements of the airframe.

Competition will allow the customer to have access to data to conduct independent analyses and remain involved in the development process.

The drawdown in Defense spending will stretchout the development of "Cutting Edge Technologies". New engines or deriavatives may have to depend on component improvements for commercial engines.

New high temperature materials are needed to design high performance engines. These programs are conducted at company expense over an extended period of time. Progress is slow and guarded.

Independent analyses of Hot Section Design Development Margins will encourage more accountabiltiy from the engine manufacturers in a competiton. Continue a relentless pursuit. O & M Budgets will benefit.

SUMMARY

- Become more involved in all phases of the Engine Design and Development Process
  - Conduct independent analyses of all component designs
  - Compare "Hard Pressed Hardware" test results to design intent
  - Understand and assess the technology baselines of all engine components

- Evaluate the Design Systems and Best Practices of each Manufacturer
  - Microstructure analyses of Hot Section Parts are essental to pass judgement
  - Sufficient Hot Time at High Engine Design Temperatures are required
  - Total engine run test time only is not a good measure of durability in service

- Engine Design Margins are required to successfully field an engine with acceptable durability, survive component degradation and low derate operation in service and provide adequate run times after thrust uprate.

- Balance Risk with Reasonable Levels of Conservatism

- Pursue Excellence in Engine Development. O & M Budgets will benefit.

DERIVATIONS

T1 Blade Leading Edge

Exit Guide Vane Leading Edge

VANE LEADING EDGE SURFACE TEMPERATURE

BENTZ CE
30 JUL 97

1/2

DERIVATION:

COMBUSTOR PATTERN FACTOR = $\dfrac{T_{MAX} - T_4}{T_4 - T_3}$ (1)

SEE p162 & 163 OF REFERENCED REPORT

$$T_{MAX} = CPF\,(T_4 - T_3) + T_4 \quad (2)$$

OVERALL COOLING PERFORMANCE(ϕ) = $\dfrac{T_{MAX} - T_M}{T_{MAX} - T_C}$

$T_C = T_3$ $\quad T_M = T_{CVS}$ CALCULATED VANE SURFACE TEMP. ON LEADING EDGE

$$\phi\,(T_{MAX} - T_3) = T_{MAX} - T_{CVS} \quad (3)$$

$$T_{CVS} = T_{MAX} - \phi\,(T_{MAX} - T_3)$$

$$T_{CVS} = (1-\phi)\,T_{MAX} + \phi\,T_3 \quad (4)$$

$$= (1-\phi)\left[(1+CPF)\,T_4 - CPF\,T_3\right] + \phi\,T_3$$

$$= (1-\phi)(1+CPF)\,T_4 - (1-\phi)\,CPF\,T_3 + \phi\,T_3$$

$$T_{CVS} = (1-\phi)(1+CPF)\,T_4 - \left[(1-\phi)\,CPF - \phi\right] T_3$$

OR $\quad T_{CVS} = (1-\phi)(1+CPF)\,T_4 + \left[\phi - (1-\phi)\,CPF\right] T_3 \quad (5)$

CALCULATED VANE SURFACE TEMPERATURE ON THE LEADING EDGE.

EQS (2) & (4) CAN BE USED TO ACHIEVE THE SAME RESULTS.

BENTI CE
12 MAY 97

T METAL EQUATION DERIVATION

$$PF = \frac{T_{MR} - T_4}{T_4 - T_3} \quad (1)$$

PF = COMBUSTOR PROFILE FACTOR

$$T_{MR} = PF(T_4 - T_3) + T_4 = (PF+1)T_4 - PF\,T_3$$

$$T_{41\,PEAK\,AVE} = T_{MR} - (T_4 - T_{41}) \quad (2)$$

PA

TEMP DROP ACROSS VANE DUE TO COOLING AIR ADDITION

$$T_{41\,PA} = (PF+1)T_4 - PF\,T_3 - (T_4 - T_{41})$$

$$= PF\,T_4 + T_4 - PF\,T_3 - T_4 + T_{41}$$

$$T_{41\,PA} = PF(T_4 - T_3) + T_{41} \quad (3)$$

$$T_{RR} = \frac{T_{41\,PA}}{RF} \quad (4)$$

RF : COMPUTED RELATIVE ROTOR FACTOR BASED ON TURBINE DESIGN GEOMETRY AND ROTOR SPEED

$$T_{RR} = \frac{1}{RF}\left[PF(T_4 - T_3) + T_{41}\right] \quad (5)$$

$$\phi = \frac{T_{RR} - T_M}{T_{RR} - T_c} \quad (6)$$

ϕ = BLADE COOLING PERFORMANCE

$$\phi(T_{RR} - T_c) = T_{RR} - T_M$$

$$T_M = T_{RR} - \phi(T_{RR} - T_c)$$

$$T_M = T_{RR}(1-\phi) + \phi\,T_c$$

$$T_M = \frac{1}{RF}\left[PF(T_4 - T_3) + T_{41}\right](1-\phi) + \phi\,T_c \quad (7)$$

$T_c = T_{COOL}$

$$T_c = T_3 + \Delta T_p \quad (8)$$

ΔT_p – TEMP RISE FROM PUMPING IN TURBINE DISC

$$T_M = \frac{1}{RF}\left[PF(T_4 - T_3) + T_{41}\right](1-\phi) + \phi(T_3 + \Delta T_p) \quad (9)$$

T METAL EQUATION DERIVATION (Cont'd)

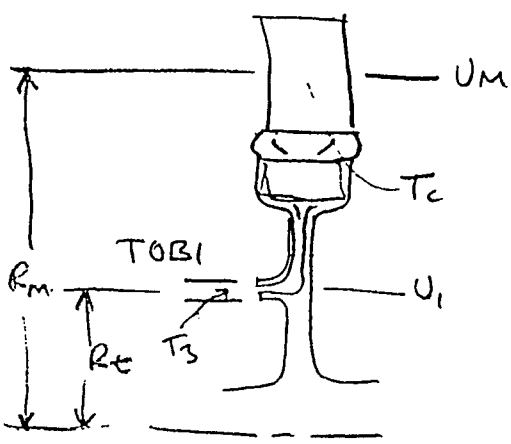
HIGH PRESSURE SPOOL: T_c PUMPING TEMP RISE (ΔT_p)

INITIAL DESIGN: $N_2 = 21{,}335$ RPM

$U_{TIP\ COMP} = 1631$ '/sec

ASSUME: $U_{T\ HPT} \cong 2000$'/sec

$R_T = 0.895'$ $R_M = 0.875'$ $U_M = 1940$'/sec



$U_1 \cong 0.46\ U_M$

$U_1 = 893$ '/sec

ASSUME $\gamma = 1.3266$ FOR $T_3 = 2077\ °R$

$\frac{R_t}{R_m} \simeq 0.46$

$T_c = T_3 + \Delta T_{pumping}$

$$\Delta T_p = \frac{\gamma - 1}{\gamma} \cdot \frac{1}{R g} \left[\frac{U_m^2}{2} - U_1^2 \right] \quad (10)$$

$$= \frac{0.3266}{1.3266} \cdot \frac{1}{(53.3)\ 32.2} \left[\frac{1940^2}{2} - 893^2 \right]$$

$$= \frac{0.3266}{2.277 \times 10^3} \left[\frac{3.764 \times 10^6}{2} - 0.797 \times 10^6 \right]$$

1.085×10^6

$$= 0.1556 \times 10^3$$

$$\underline{\Delta T_p = 156\ °F}$$

T METAL EQUATION DERIVATION (Cont'd)

IF $\phi = 0.5$ (0.7) BASELINE

$PF = 0.25$

$RF = 1.15$ $\Delta T_p = 156$

$$T_M = \frac{(1-\phi)}{RF}\left[PF(T_4 - T_3) + T_{41}\right] + \phi(T_3 + \Delta T_p)$$

$$= \frac{0.5}{1.15}\left[0.25(T_4 - T_3) + T_{41}\right] + 0.5(T_3 + 156)$$

$$= 0.1087\,T_4 - 0.1087\,T_3 + 0.435\,T_{41} + 0.5\,T_3 + 78$$

$$T_M = 0.3913\,T_3 + 0.1087\,T_4 + 0.435\,T_{41} + 78 \qquad (11)$$

IF $\phi = 0.6$

$$T_M = \frac{0.4}{1.15}\left[0.25(T_4 - T_3) + T_{41}\right] + 0.6(T_3 + 156)$$

$$= 0.08695\,T_4 + 0.513\,T_3 + 0.348\,T_{41} + 93.6$$

$$T_M = 0.513\,T_3 + 0.087\,T_4 + 0.348\,T_{41} + 93.6 \qquad (12)$$

IF $\phi = 0.7$

$$T_M = \frac{0.3}{1.15}\left[0.25(T_4 - T_3) + T_{41}\right] + 0.7(T_3 + 156)$$

0.261

$$= 0.0652\,(T_4 - T_3) + 0.261\,T_{41} + 0.7\,T_3 + 109.2$$

$$T_M = 0.635\,T_3 + 0.0652\,T_4 + 0.261\,T_{41} + 109.2 \qquad (13)$$

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13. ABSTRACT ***(Maximum 200 words)***

This workshop has six categories. (1) In the opening sessions we are attempting to provide an overview of the engine technology, applications, and perceived needs from the programmatic point of view. (2) In the second session more details of seals and secondary flows are presented. (3) In the third session some of the tools and techniques are examined. (4) In the fourth session engine externals, the region between the case and the nacelle are examined in more detail. (5) Some sealing needs as applied to High Speed Research (HSR) are presented in a limited exclusive rights session. (6) A short course on engine design development margins by Charles Bentz rounds out the workshop. Through these sessions the efforts at NASA LeRC, contractors and universities are applied to seals/secondary flow problems in a continued effort to expand the U.S. aerospace technology and marketshare. A second volume containing related HSR discussions is available to selected U.S. citizens.

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