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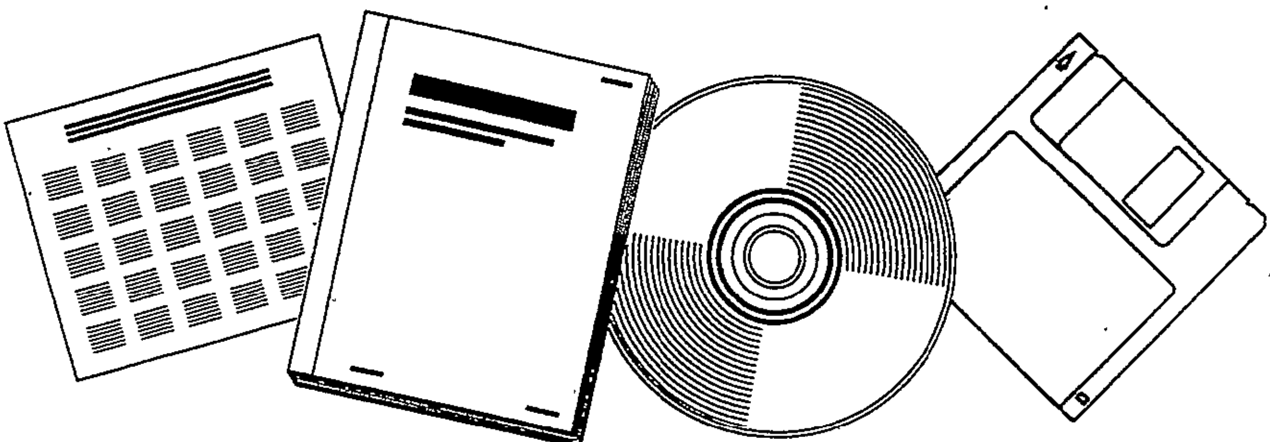

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by

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GENERAL ELECTRIC COMPANY

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16. Abstract
In this program, the cost/benefits of eight advanced materials technologies were evaluated for two aircraft missions. The overall study was based on a time frame of commercial engine use of the advanced material technologies by 1985. The material technologies evaluated were eutectic turbine blades, titanium aluminide components, ceramic vanes, shrouds and combustor liners, tungsten composite FeCrAly blades, γ' oxide dispersion strengthened (ODS) alloy blades, and no-coat ODS alloy combustor liners. They were evaluated in two conventional take-off and landing missions, one Transcontinental and one Intercontinental.

For the specific commercial engine/aircraft mission evaluated in this study, the results indicated substantial economic benefit for ceramic turbine shrouds and vanes, eventhough ceramic turbine vanes, in particular, have a higher risk and uncertainties of introduction. Titanium aluminide LPT blades and static components showed relatively good payoff and ceramic combustor liners showed marginal benefit. All advanced material HPT blades provided a substantial improvement in engine performance, but relatively small economic benefit due to the projected high costs of these components. Major cost reductions in the manufacturing process for the advanced blade materials are necessary to take full advantage of the improved engine performance provided by these materials. This is true specifically for the particular commercial engine/aircraft missions evaluated in this study. A no-coat ODS alloy combustor liner showed no overall economic benefit when applied to the advanced engines of this study.

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FOREWORD

The work described herein was conducted by the Aircraft Engine Group, General Electric Company under NASA Contract NAS3-20074. The NASA Project Manager was Neal T. Saunders, NASA-Lewis Research Center. The General Electric Program Manager was E.N. Bamberger. This report was prepared by Robert V. Hillery and Richard P. Johnston. A similar NASA sponsored cost/benefit study was conducted in parallel to this study by Pratt & Whitney Aircraft Division of United Aircraft Corporation. Readers are referred to report number NASA-CR-135107 for a summary of the results of that independent study.

1.0 SUMMARY

In this program, the cost/benefits of eight advanced materials technologies were evaluated for two aircraft missions. The overall study was based on a time frame of commerical engine use of the advanced material technologies by 1985. The material technologies evaluated were eutectic turbine blades, titanium aluminide components, ceramic vanes, shrouds and combustor liners, tungsten composite FeCrAly blades, γ' oxide dispersion strengthened (ODS) alloy blades, and no-coat ODS alloy combustor liners. They were evaluated in two commercial transport conventional take-off and landing missions, one Transcontinental and one Intercontinental. Methodologies for selecting technical goals, costs and chances of success were developed and these data were utilized in the engine design analysis and cost benefit studies.

For the specific commercial engine/aircraft mission evaluated in this study, the results indicated substantial economic benefit for ceramic turbine shrouds and vanes, even though ceramic turbine vanes, in particular, have a higher risk and greater uncertainties of introduction. Titanium aluminide LPT blades and static components showed relatively good payoff and ceramic combustor liners showed marginal benefit. All HPT blade materials (DS eutectics, W-FeCrAlY composite and γ' ODS alloy) provided a substantial improvement in engine performance, but relatively small economic benefit due to the projected high costs of these components. Major cost reductions in the manufacturing process for these materials are necessary to take full advantage of the improved engine performance provided by these advanced technologies. Based on General Electric's assessment of blade manufacturing costs, only the DS eutectic blade technology is likely to provide an overall economic benefit to future engine systems of the type evaluated. A no-coat ODS alloy combustor liner showed no overall economic benefit when applied to the advanced engines of this study.

Overall ranking of the advanced materials technologies considering economic benefit, development costs and probability of success were:

1. Ceramic shrouds
2. Ceramic vanes
3. Titanium aluminide components
4. D.S. eulectic stage 1 HPT blades
5. Ceramic combustor liners
6. D.S. eutectic stage 2 HPT blades
7. W-FeCrAlY composite blades
8. γ' ODS alloy blades
9. No coat ODS alloy combustor liners

A "Corporate Ranking" of the advanced materials technologies, assessed independent of the criteria used in the study, grouped the materials technologies as follows in order of decreasing benefit:

- DS eutectic HPT blades and ceramic shrouds
- Titanium aluminides, ceramic combustor liners, no coat ODS alloy combustor liners, and ceramic vanes
- W-FeCrAlY and γ' ODS HPT blades

2.0 INTRODUCTION

Advanced materials technologies have historically made substantial contributions to the evolution of aircraft gas turbine engines and must continue to meet the demands for engines with improved performance and cost effectiveness. NASA and industry have recognized the need to investigate and evaluate advanced component and materials technologies for the improved commercial transport engines of the 1980's. Concern for the efficient use of petroleum resources has resulted in an amphasis on applications that will make the engines more energy efficient while at the same time satisfying the environmental considerations of being clean and quiet. Finally, and most important in this study, the economic impact on the airlines resulting from these increased energy and environmental constraints has been recognized and taken into account.

To help fulfill these needs in the area of materials technology, NASA and industry have a cooperative effort -- the MATE (Materials for Advanced Turbine Engine) Program -- to accelerate the introduction of new materials technologies into advanced aircraft turbine engines. Part of this overall program has involved a periodic assessment of the costs and potential benefits of selected materials technology advances when applied in turbofan powered commercial transports. The results of these studies provide input which helps in the selection of technologies to be developed in the MATE effort. The study program summarized in this report has established costs and benefits for several advanced materials technologies as applied to specific components of a turbofan engine for a CTOL commercial aircraft sized for Transcontinental and Intercontinental missions.

Materials technologies selected for this study included high temperature turbine blades, vanes and shrouds, combustor liners and low pressure turbine blades and structural components. The methodology used to assess benefits, costs and risks of the advanced materials technologies as applied to future CTOL propulsion systems is described in the report. The overall study was based on a time frame of commercial engine use of the advanced materials technologies by 1985.

Results generated under this program are expected to aid the selection and subsequent development of those materials technologies which offer the greatest potential for use in future aircraft turbine engines. It should be recognized, however, that the ranking of the materials technologies as defined by this study does not represent the sole basis for engineering development and engine application. Other significant factors, which require engineering judgment and may play a major role in ultimate program selection and technology development, are not necessarily included in this cost/benefit study.

2.1 STUDY APPROACH

The approach used in the materials technology cost/benefit study is shown schematically in Figure 1 and a brief description of the overall process follows. The methodology and rationale used in the selection of the materials technologies, costs and probability of success is described in some detail in Section 2.2 of this report.

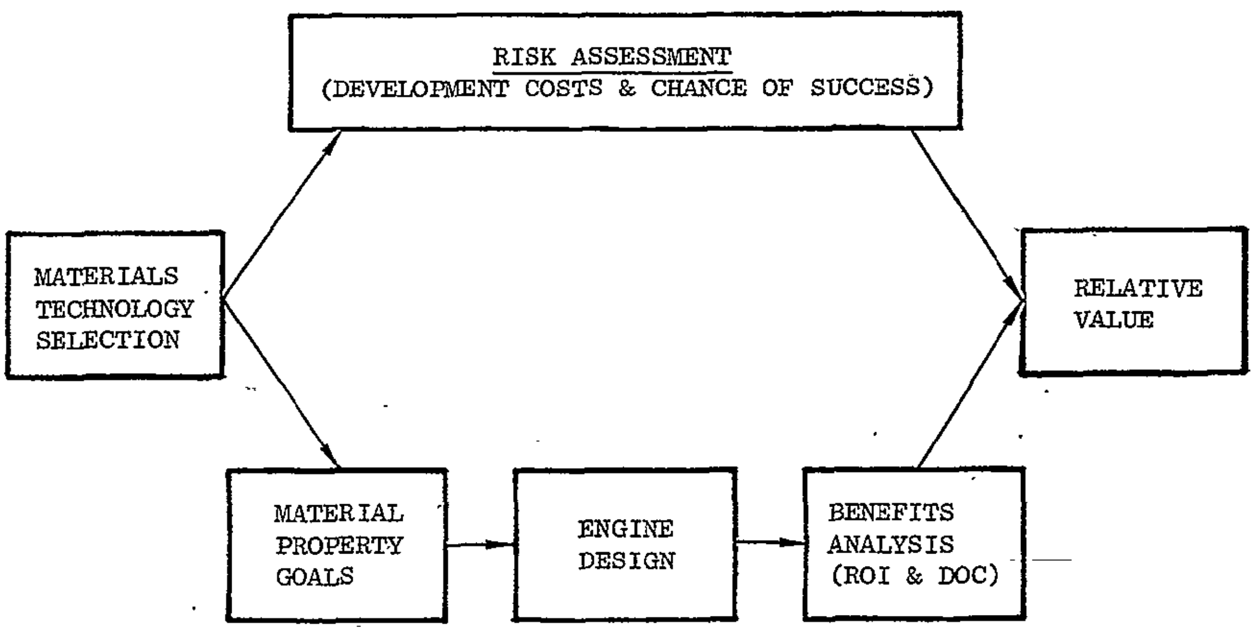

Figure 1. Schematic Representation of Cost/Benefit Study Approach

First, material property goals were established for each of the selected technologies. Technology development costs* and the risk (chance of success) were estimated for each material technology. The overall effects of the advanced technology in terms of changes in engine performance, weight, initial costs and maintenance costs were determined and incorporated in the benefit analysis. In the benefit analysis, the effect of the advanced material technology on the engine-airplane system was assessed using previously developed economic trade factors pertinent to the aircraft-economic system. The results of the benefit analysis are expressed in terms of change (Δ) in the Return on Investment (ROI), and Direct Operating Costs (DOC) and Fuel Consumption (W_f) due to substitution of the new materials where:

Direct Operating Cost (DOC) includes crew pay, fuel, oil, hull insurance, maintenance labor (including burden) and materials as well as the amortized costs of the aircraft and engines. The 1967 ATA formulae[1] is used except that depreciation is taken over a 15 year period rather than 12, and 20% engine spares are assumed rather than the 40% used in the ATA formula. Also, engine maintenance and materials costs were taken at rates obtained from General Electric experience, which differs from those used in the ATA formula.

Return on Investment (ROI) is the annual return on the initial investment, considering direct operating costs, indirect operating costs, revenue, taxes and profits. A positive (+) value in ΔROI and a negative (-) value in ΔDOC are favorable results in any given technology. Additionally, the effect of materials substitutions on the weight of fuel used (ΔW_f) for which negative values are desired, were evaluated.

The ΔROI factor was incorporated in the Relative Value which is a NASA cost/benefit merit factor described by the following equation:

$$\text{"Relative Value"} = \frac{\text{Benefit } (\Delta\text{ROI})^{**}}{\text{Development Cost}} \times \text{Probability of Success}$$

Relative value as used herein represents one way of assessing a material technology benefit and ranking the potential benefits of several material technologies on the same relative basis. It should not be construed to represent the sole, or necessarily the prime basis for selecting material technologies for engineering development and engine applications. Other significant factors, such as engineering judgment for a particular engine application and an assessment of the overall merits of a particular material technology for a range of engine applications could play a major role in the final ranking and selection of the technologies.

In the determination of relative value used throughout the cost/benefit study, it was assumed that all property goals established for a given material would be met. To determine the impact of possible deviations from full property goals, "sensitivity" analyses were conducted to show the effect on the economic benefit of the technologies. The results of these sensitivity analyses for the several materials technologies are described in detail in Section 6 of this report.

*Development costs, finished parts, costs, and economic benefits are expressed in 1977 dollars.

**The ΔROI is quoted in this report as "points" rather than %, such that (1) point ROI is the difference between 15 and 16% ROI, for example.

2.2 METHODOLOGY FOR SELECTION OF GOALS, COSTS, AVAILABILITY, AND PROBABILITY OF SUCCESS.

Advanced materials technologies selected for this study are shown in Table I. These were selected for the specific applications shown in consultation with the NASA Program Manager. The materials technologies were chosen because of their anticipated potential benefits in the engine/aircraft application with particular emphasis on their potential effects in reducing engine fuel consumption. The methodology used to determine technical material goals, finished part cost goals, projected year of availability and probability of success for each of these is outlined in the following sections, and illustrated by a typical example in Table II.

TABLE I

ADVANCED MATERIALS TECHNOLOGIES SELECTED FOR COST/BENEFIT STUDY

- Titanium Aluminide LPT Blades and Structural Components
- No Coat ODS Alloy Sheet Shingled Combustor
- Ceramic Turbine Shrouds
- Ceramic Turbine Vanes
- Ceramic Shingled Combustor
- Advanced High Pressure Turbine Blades
  - Gamma Prime ODS Alloy
  - W-FeCrAlY Composite
  - Directionally Solidified Eutectic Alloy

2.2.1 Material Property Goals

The technical material goals were divided into "critical" goals and "others".

The critical goals were those that must be met (or very closely approached) to make the materials technology (alloy/process) worthwhile. If any of the critical goals were not met, the alloy/process development would not be pursued further. The "others" goals are ones that are also important, but failure to meet one of these would not necessarily result in elimination of the alloy/process from consideration. For example, the maximum density goals for the eutectic turbine blade alloy used herein could probably be exceeded by a few percent if the rupture strength goal were achieved.

TABLE II

EXAMPLE OF MATERIAL TECHNOLOGY GOALS FOR TURBINE BLADE ALLOY DESIGNATED Ni69XB IN 1966

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Rupture Strength: ≥ IN 100	75
• Hot Corrosion Resistance: ≥ U700	60
• 871C (1600F) Tensile R of A: ≥ 15%	60
OTHERS	
• Density: ≤ 8.17 g/cm^3 (.295 lb/in^3)	
• Stability: No deleterious phases affecting long time rupture life	75
• Oxidation Resistance: ≥ IN 100	

FINISHED PART COST GOAL

≤ Cast U700

PROJECTED DEVELOPMENT COSTS

Total	$ 775,000
1967 & Beyond	$ 700,000

PROJECTED YEAR OF AVAILABILITY 1969

PROBABILITY OF SUCCESS

%	
Material	60
Design	90
Manufacturing	80
OVERALL PROBABILITY OF SUCCESS	**60%**

The technical goals for each alloy/process were determined by General Electric materials experts and where possible compared to currently used materials. Mechanical, physical and environmental properties were considered in determining the goals, as well as component life. After extensive discussions, a probability of success was determined for each critical goal and for the "others" as a group. The lowest probability of success number then became the "material" input to determine the overall probability of success -- discussed in Section 2.2.5.

The technical goals are illustrated in Table II, which shows the 1966 goals for Ni69XB* in the current program's cost benefit study format. As shown, the most difficult technical goals for Ni69XB turbine airfoils were hot corrosion resistance and 871C (1600F) tensile ductility and these might have been given a 60% probability of success. Both of these "critical" goals were met in the Rene' 80 alloy that was developed to meet these goals.

2.2.2 Finished Part Cost Goal

The finished part cost goals were determined by General Electric materials and manufacturing specialists and took into consideration the projected cost of the material and/or process involved to produce the finished part -- including all finishing processes such as machining, coating, etc. The part cost goals were based on the costs estimated for the 250th engine set time period and are therefore production (not development) costs for the generic part. In all cases the cost goals were related to costs of materials used in current General Electric engines.

As the study progressed, however, specific design configurations were developed for each component and, as a result, the input to the manufacturing cost analysis became more precise. With this maturing material/design and manufacturing knowledge, projected estimates of specific part costs were refined. It was more appropriate, therefore, to use these specific part costs in this study rather than the generic part cost goal listed in Section 3 of this report. Both "goal" and "specific" part costs are listed for the several materials technologies in Tables IV-XI (Section 3).

It should also be emphasized that, in cases where the projected part cost radically affected the overall cost benefit analysis for a given component, the sensitivity analysis (Section 6) was used to demonstrate this effect. Hence, although the specific and goal part costs were not necessarily the same for any given component, the sensitivity analysis allowed an assessment as to what was actually required to provide an economically viable technology for the mission evaluated.

*Since 1966, GE advanced alloy goals have had designations which described their probable base (Ni = nickel), year of availability (69 = 1969), and application (B = blade).

2.2.3 Projected Development Costs

This included the costs projected for:

1) Laboratory development (developing the material/process, scaling up to production size heats, etc., and developing the required design data).

2) Manufacturing process development (demonstration of adaptability of standard joining and fabrication and coating technology - process development of new and novel processes if necessary to accommodate unique characteristic of a material).

3) Transition to Manufacturing (establish new processes or process modifications in the manufacturing area to achieve engineering requirements).

4) Trial hardware (work with outside vendors to produce trial hardware for property evaluation and component tests).

5) Engine test (proof of concept, evaluation in test engine).

These costs did not include production hardware or facilities (occasionally costs for tooling were included).

The costs are those projected for the total development and are separated into (1) total costs for the entire program; and (2) the costs for 1977 and beyond. (The latter figure was used in the relative value calculations in the Benefit Analysis -- Section 5.0.)

2.2.4 Projected Year of Availability

This is the year in which small quantities of parts made from the material/process are projected to be ready for reliable procurement to a specification, at reasonable cost, and with sufficient design data curves available to assure reliable property predictions.

2.2.5 Overall Probability of Success

This was the most difficult of the goals to develop due to the high degree of subjectivity involved. For this determination many discussions were held with the materials, design and manufacturing experts and each expert estimated the probability of success in meeting the goals. (The materials chance of success has previously been discussed -- Section 2.2.1.) The design expert considered what breakthrough (if any) had to occur in his field to make viable the material/process to be developed. For example, the no coat ODS alloy should be comparatively easy to incorporate into an existing shingled combustor design and was therefore given a 90% chance of success, while ceramic vanes would be a difficult and complex design problem and therefore assigned only a 40% chance of success.

The manufacturing expert considered the problems involved in meeting (or closely approaching) the finished part cost goal before the manufacturing probability of success was determined.

The lowest probability of success number of the three (material, design, or manufacturing) then became the "overall probability of success" for each of the materials/processes to be studied. For example, the "lowest" and therefore "overall" probability of success for DS eutectic turbine blades was determined to be 60% -- which was the manufacturing probability of success, compared to 70% for material (alloy) and 80% for design. This was due to the difficulty of meeting the part cost goal in manufacturing, which was believed more difficult than developing an alloy to meet the technical goals, or incorporating the material into a viable design.

The overall probability along with the development costs (1977 and beyond) for the six materials/processes to be studied are listed in Table III. Those numbers will be used with the Return on Investment (ΔROI) value (as determined later in the Benefits Analysis section), to determine the "Relative Value" of the materials/processes being studied.

TABLE III

DEVELOPMENTAL COSTS AND PROBABILITIES OF SUCCESS FOR TECHNOLOGIES STUDIED

Material Technology	Estimated Development Costs ($)	Probabilities Of Success
Titanium Aluminide LPT Blades	3,000,000	40%
No Coat ODS Alloy Shingled Combustor	1,500,000	50%
Ceramic Turbine Shrouds	3,500,000	60%
Ceramic Turbine Vanes	6,000,000	40%
Ceramic Shingled Combustor	3,000,000	50%
Gamma Prime ODS Turbine Blades	4,000,000	20%
W-Reinforced FeCrAlY Turbine Blades	8,000,000	20%
D.S. Eutectic Turbine Blade	8,000,000	60%

3.0 STATUS AND GOALS FOR THE MATERIALS/PROCESSES EVALUATED

The eight materials technologies/component combinations evaluated in the cost/benefit study are at various stages of development in the industry as a whole and as General Electric programs. In this section, the status of the technologies will be discussed briefly and related to the goals established for each.

Goals were based upon historical development experience and projections of current state-of-the-art for each materials technology. The goals listed are those upon which the study was performed and represent the most optimistic projections. "Sensitivity", or less optimistic, goals were also considered to show the impact on the economic benefit of deviations from full property goals and these are discussed for each technology in Section 6 of this report.

3.1 TITANIUM ALUMINIDE LPT BLADES

Titanium aluminides, based on the intermetallic compounds Ti_3Al or TiAl, offer the potential for high strength to weight ratio materials suitable for low pressure turbine blades and certain hot section structural components now fabricated in Inconel 718 or similar Ni-base alloys. LPT blades were evaluated as the original application for titanium aluminides in this study but, as will be discussed, stator parts were also considered. The major problem to be overcome with this class of materials is the inherent poor ductility at low to intermediate temperatures, and the industry as a whole is addressing this shortcoming. General Electric is evaluating several experimental compositions as well as performing forging and rolling studies on selected compositions.

The goals for titanium aluminide LPT blades are presented in Table IV and it is clear that the low temperature ductility is the limiting item in achieving these goals. Programs to achieve these goals are, however, receiving appropriate emphasis in the aircraft engine industry and a material with these or similar properties should be attainable.

3.2 NO COAT ODS SHEET SHINGLED COMBUSTOR

An oxide dispersion strengthened (ODS) sheet alloy was evaluated in the study for its potential payoff as a combustor shingle material. This material would have excellent oxidation and corrosion resistance together with good rupture strength and thermal fatigue resistance. Property goals used in this study are presented in Table V.

It is believed that the rupture strength, formability and environmental resistance requirements of these goals could be achieved utilizing the recently developed ODS FeCrAl alloy, MA956. General Electric is evaluating this material for possible use in advanced engines although there are no programs in place to evaluate this type of material specifically in a shingled combustor design. General Electric does, however, have a significant commitment to ODS alloys in general and the development and evaluation of an ODS sheet alloy by the materials vendors is being encouraged with ultimate use seen possible in several components, including combustors.

TABLE IV

TITANIUM ALUMINIDE LPT BLADES

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Creep and Tensile Strength Superior to U700 on a Strength/Density Basis	75
• Impact Resistance to FOD Comparable to Current Turbine Blade and Vane Alloys	40
• Tensile Ductility 3% Elong., 5% Red. Area Over Temperature Range R.T. to 1600F	40
• Density <4.0 g/cc (.145 lb/in^3)	90
OTHERS	
• Low and High Cycle Fatigue Strength Superior on a Strength/Density Basis to U700 or Comparable Blade Alloys	
• Stability - No significant Loss of Fatigue, Ductility or Impact Resistance Due to Phase Changes	50
• Corrosion/Oxidation Resistance Equivalent to Commonly Used Coated or Uncoated LPT Blade Alloys	

FINISHED PART COST GOAL	SPECIFIC PART COST USED
≤ Cast U700 or X40	1.1 X Conventionally Cast Ni-Base LPT Blades, or 0.8 X Inconel Sheet

PROJECTED DEVELOPMENTS COSTS

Total	\$3,000,000
1077 & Beyond	\$3,000,000

PROJECTED YEAR OF AVAILABILITY 1982

PROBABILITY OF SUCCESS

%		
	Material	40
	Design	80
	Manufacturing	70
	OVERALL PROBABILITY OF SUCCESS	40%

TABLE V

NO COAT ODS SHEET SHINGLED COMBUSTOR

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Oxidation Resistance ≥ HS188	90
• Hot Corrosion Resistance ≥ Coated MarM509	90
• Rupture Strength ≥ 62 MPa/1037°C/100 hr (9 ksi/1900°F/100 hr.) (Long. & Transv.)	90
• Thermal Fatigue Resistance ≥ HS188	50
• Formability ≥ HS188 Sheet	80
OTHERS	
• Erosion Resistance > HS188	50
• Rupture Ductility = HS188	

FINISHED PART COST GOAL	SPECIFIC PART COST USED
1X Cast MarM509 Shingled Combustor	2.05 X HS188 Sheet Liner
1.2 X HS188 Sheet Shingled Combustor	

PROJECTED DEVELOPMENT COSTS

Total	$1,500,000
1977 & Beyond	$1,500,000

PROJECTED YEAR OF AVAILABILITY 1980

PROBABILITY OF SUCCESS

%		
	Material	50
	Design	90
	Manufacturing	75
	OVERALL PROBABILITY OF SUCCESS	50%

3.3 CERAMIC TURBINE SHROUDS

The use of ceramics for turbine shrouds would allow higher temperatures, longer life and reduced cost compared to currently used turbine shroud materials such as transpiration cooled wire mesh (Poroloy). Developments over the past several years at the General Electric Corporate Research and Development Center have resulted in a promising ceramic turbine shroud material which should provide the necessary impact resistance and rub tolerance.

Impact and rub tests, including cold rotor rub tests, have given very promising results and General Electric is actively pursuing further development and evaluation programs aimed at the introduction of ceramic shrouds into General Electric engines. The goals for ceramic shrouds as used in the present study are presented in Table VI.

3.4 CERAMIC TURBINE VANES

As with ceramic shrouds, the use of ceramic turbine vanes would allow higher temperatures, longer life and reduced cost compared to currently used vane materials.

Turbine vanes in current jet engines are either cast cobalt base alloys such as X-40 or Mar-M509, which are comparatively inexpensive but have low incipient melting points [1288C (2350F)] or are wrought oxide dispersion strengthened (ODS) nickel base alloys (DSNiCr, MA 754), which are expensive but have the necessary high melting points [1371C (2500F)] needed in advanced engines. The possible use of ceramics would combine the lower costs of the cast alloys with the high (probably much higher) melting points of the ODS alloys. The goals for the ceramic vanes are shown in Table VII. Current work at the General Electric Corporate Research and Development Center (CR & DC) has been very promising, with silicon carbide emerging as a prime candidate. The eventual use of ceramic vanes in jet engines will require close coordination between material development, design and manufacturing to assure success. The low overall 40% probability of success shown in Table VII, resulting from anticipated design problems and the ≤50% probability for materials and manufacturing, attest to the difficulty anticipated in achieving these goals for ceramic vanes.

3.5 CERAMIC SHINGLED COMBUSTOR

The goals for the ceramic shingled combustor as used in the present study are listed in Table VIII. A number of materials, design and manufacturing problems must be overcome before this technology could be realized in practice but the advances being made in ceramics provide some degree of optimism.

3.6 GAMMA PRIME ODS TURBINE BLADES

As indicated with reference to other technologies, oxide dispersion strengthened (ODS) alloys are being developed and evaluated for several applications in General Electric engines. As a class, these alloys have very high melting points and excellent resistance to creep and offer good potential for nozzle vanes and possibly several other high temperature applications. A "gamma prime" (γ') ODS alloy has been developed by the International Nickel Company which is reported to have excellent high temperature capabilities due to a com-

TABLE VI

CERAMIC TURBINE SHROUDS

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Thermal Shock Resistance with Surface Simulating Engine Gas Erosion: 1593C to 538C to 1593C (2900°F to 1000°F to 2900°F). ≥ 1500 Cycles (No Macro Cracking)	70
• No Scabbing on Blade Rub	60
• Ballistic Impact Resistance: Absorb 4.1J (3 ft.-lb) at Surface Exposed to Gas Path Without: 1. Loss of Pieces Larger Than .32 cm (.125 in.) 2. Cracking Through the Entire Section on Subsequent Application of 100 Thermal Cycles	70
OTHERS	
• Maximum Surface Temperature: 1593°C (2900°F)	
• Maximum Bulk Temperature: 1482°C (2700°F)	
• Dimensional Stability: ≤ 0.2% Change After Above Thermal Shock Test	75
• Gas Erosion (In a High Velocity, Cycled Burner Rig): Will Not Exceed .38 mm (.015 in.) of Surface Recission in 3000 Cycles	

FINISHED PART COST GOAL	SPECIFIC PART COST USED
≤ 50% of Poroloy	0.87 X Poroloy

PROJECTED DEVELOPMENT COSTS

Total	$3,500,000
1977 & Beyond	$3,300,000

PROJECTED YEAR OF AVAILABILITY 1981

PROBABILITY OF SUCCESS

%		
	Material	60
	Design	75
	Manufacturing	80

OVERALL PROBABILITY OF SUCCESS	60%

TABLE VII

CERAMIC TURBINE VANES

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Maximum Surface Temperature: ≤ 1593°C (2900°F)	75
• Thermal Shock Resistance: ≥ 1500 Cycles, 1593°C to 538°C to 1593°C (2900°F to 1000°F to 2900°F) with .020" diameter cooling holes drilled into a leading edge configuration	40
• Ballistic Impact Resistance: a. Absorb 4.1J (3 ft.-lb.) of Ballistic Energy Without Loss of Pieces Larger than .32 cm (.125 in) b. Absorb 4.1J (3 ft.-lb) of Ballistic Energy Without Cracking Through the Entire Section on Subsequent Application of 100 Thermal Cycles	40
• Modulus of Rupture: ≥ 276 MPa (40 ksi) at 1482°C (2700°F) with a Weibull Modulus ≥ 10	70
OTHERS	
• Dimensional Stability: ≤ 0.2% Change After Above Thermal Shock Test • A Compliant Ceramic Air Seal Material Layer Capability: ≥ 7 MPa (100 psi) pressure drop at 1371°C (2500°F) • Gas Erosion Resistance: ≤ 15 mils Surface Recession at 2400°F Mach 1.0 in 500 hours	50

FINISHED PART COST GOAL	SPECIFIC PART COST USED
≤ 50% of ODS Vane Assemblies	0.67 X ODS Alloy Vane (Stg 1) 1.64 X R125 Alloy Vane (Stg 2)

PROJECTED DEVELOPMENT COSTS

Total	$6,000,000
1977 & Beyond	$6,000,000

PROJECTED YEAR OF AVAILABILITY 1984

PROBABILITY OF SUCCESS

%	
Material	40
Design	40
Manufacturing	50
OVERALL PROBABILITY OF SUCCESS	40%

TABLE VIII

SHINGLED COMBUSTOR - CERAMIC

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Thermal Shock Resistance: 1482°C (2700°F) to 649°C (1200°F) to 1482°C (2700°F) ≥ 5000 1 Hour Cycles	70
• Creep Rate at 1482°C (2700°F): ≤ 10^{-4}/Hr. at $35N/m^2$ (5 ksi)	90
• Environmental Resistance up to 1315°C (2400°F): ≤ .37 mm (.0151 in) Depth of Penetration in 1000 Hrs. with Fuel Initially Condensed on the Surface	50
OTHERS	
• Maximum Surface Temperature: 1482°C (2700°F)	
• Dimensional Stability: ≤ 0.2% Change After Above Thermal Shock Test	60
• Fretting Resistance in Metal Holder to Accommodate ≤ 7 MPa (1000 psi) Stress Vibrations for 10^5 Cycles: Less than .25 mm (.010 in) Depth of Wear at 649°C (1200°F)	

FINISHED PART COST GOAL

≤ Conventionally Cast MarM509 Shingles

SPECIFIC PART COST GOAL

1.15 X HS188 Sheet Liner

PROJECTED DEVELOPMENT COSTS

Total	$3,000,000
1977 & Beyond	$3,000,000

PROJECTED YEAR OF AVAILABILITY 1981

PROBABILITY OF SUCCESS

%		
	Material	50
	Design	50
	Manufacturing	50
	OVERALL PROBABILITY OF SUCCESS	50%

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bination of γ' and oxide dispersion strengthening. Preliminary data suggest, however, that intermediate temperature (~1400F) strength and 1300-1800°F ductility are inadequate for turbine blade applications. In addition, manufacturing operations and related finished part cost goals are a major concern with this type of material for blading. The goals for the γ' ODS turbine blades are given in Table IX.

3.7 W-REINFORCED FeCrAlY BLADES

The goals for a tungsten-reinforced FeCrAlY high pressure turbine blade used in the present study are given in Table X. There are many technical problems expected to be encountered in manufacture of turbine blades with this composite which account for the very low overall probability of success projected. This, together with the limited payoff relative to other advanced blade alloys, has not appeared to warrant the large investment necessary to develop and scale up the material system.

In this study, a coatability requirement was included in the "less critical" goals for this material since it was felt that possible exposure of the tungsten fibers to a high temperature air environment would be detrimental to the overall performance. However, it has been suggested that an environmental resistant coating may not be required. For this reason, the cost of a coating was not considered with respect to DOC, ROI and Relative Value for the W-FeCrAlY material. An estimate of the effect of the added cost of a coating system for this material, were it proven to be necessary, can be obtained from the sensitivity analysis discussed in Section 6 of this report.

3.8 DIRECTIONALLY SOLIDIFIED EUTECTIC TURBINE BLADES

Development of directionally solidified (DS) eutectic turbine blades is an active program at General Electric. The alloy has been designated Ni82XB and is aimed at advanced turbine blades with 165°F greater rupture and creep strength compared to Rene' 125. General Electric is developing DS eutectic alloys on an Air Force Program. The technology goals for Ni82XB used in this study are listed in Table XI.

The finished part cost goal for Ni82XB is $\leq$ 2.5X the finished part cost for a conventionally cast Rene' 80 HPT blade with current General Electric CF6 cooling technology. This projected cost is high primarily due to the very slow directional casting solidification rates found necessary for the NiTaC-13 eutectic alloy developed early in the program. General Electric has now introduced a rapid, automated DS casting process (RAM-DS) which is expected to considerably reduce the cost of Ni82XB. Improved cost factors have been incorporated in the Cost Benefit Study, as will be discussed in the following Sections of this report.

TABLE IX

GAMMA PRIME ODS TURBINE BLADES

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Longitudinal Rupture/Creep 92°C (165°F) > Rene' 125	40
• Longitudinal Tensile Ductility at 704 - 982°C (1300 - 1800°F) > 10%	20
• Longitudinal Thermal Fatigue Resistance = Rene' 125	70
• Transverse Properties = Rene' 125	20
OTHERS	
• Ballistic Impact 649 - 982°C (1200 - 1800°F) > Rene' 125	50
• Bare Oxidation Resistance = Rene' 125	
• Bare Hot Corrosion Resistance = Rene' 125	
• Density < 8.7 g/cc (.315 lb/in^3)	
• Nil Strength 1316°C (2400°F)	

FINISHED PART COST GOAL

≤2.5 X Rene' 80
(Conventionally Cast CF6 Type HPT Blade)

SPECIFIC PART COST USED

2.23 X R125 (Stg. 1)
2.78 X R125 (Stg. 2)

PROJECTED DEVELOPMENT COSTS

Total	$4,000,000
1977 & Beyond	$4,000,000

PROJECTED YEAR OF AVAILABILITY 1982

PROBABILITY OF SUCCESS

%		
	Material	20
	Design	80
	Manufacturing	50
	OVERALL PROBABILITY OF SUCCESS	20%

TABLE X

W-REINFORCED FeCrAlY

TECHNICAL GOALS

CRITICAL	% PROBABILITY OF SUCCESS
• Stress Rupture 92°C (165°F) > Rene' 125	20
• Strain Cycle LCF at 871°C (1600°F) at .5%: > 10 X Rene' 80 in Coated Condition	70
• Process: Meet Finished Part Cost Goal 2.5 X Rene' 80 in a Blade Form that Can be Competitive with Complex Air-Cooled Castings	20
OTHERS	
• Tensile Ductility at 704-982°C (1300-1800°F): ≥ 10% RA	
• Thermal-Cycling Stability: After 5,000 Cycles from 982°C (1800°F) to R.T. Must Retain 90% of 982°C (1800°F) Rupture Strength	
• Hot Corrosion Res.: ≥ Rene' 100: Oxidation Res: ≥ Rene' 80	
• High Temperature Stability: No Unpredictable Losses in Long Time Rupture Life	
• Must be Joinable for Maximum Temperature Up to 1288°C (2350°F)	70
• Nil Strength Temperature: ≥ 1316°C (2400°F)	
• Physical Properties (to 1066°C (1950°F): ± 5% of Rene' 80	
• Density: ≤ 8.58 g/cm^3 (.315 lb/in^3)	
• Youngs Modulus (Parallel to Growth Direction): ≥ 90 X 10^3 MN/m^2 (13 X 10^6 psi) ≤ 110 X 10^3 MN/m^2 (16 X 10^6 psi) at 870°C (1600°F)	
• 649-982°C (1200-1800°F) Impact (After Stressed Exposure):	
- Ballistic: ≥ 2.7J (2 ft-lb) No Cracks, .44 cm (175 in°) Steel Ball, .23 cm (.090 in) Thick Spec.)	
- Charpy V-Notch: ≥ 13.6J (10 ft-lb)	
• Coatable for 12,000 Hours Life to 927°C (1700°F) 5000 Hours to 1037°C (1900°F) 1000 Hours to 1149°C (2100°F) + Recoatable	

TABLE X (Cont.)

TECHNICAL GOALS

FINISHED PART COST GOAL	SPECIFIC PART COST USED
≤ 2.5 X Rene' 80 (Conventionally Cast CF6 Type HPT Blade)	2.2 X R125 (Stg. 1) 2.67 X R125 (Stg. 2)

PROJECTED DEVELOPMENT COSTS

Total	$8,000,000
1977 & Beyond	$6,500,000

PROJECTED YEAR OF AVAILABILITY 1981

PROBABILITY OF SUCCESS

%		
	Material	20
	Design	50
	Manufacturing	40
	OVERALL PROBABILITY OF SUCCESS	20%

TABLE XI

D.S. EUTECTIC TURBINE BLADES

TECHNICAL GOALS*

CRITICAL	% PROBABILITY OF SUCCESS
• Rupture and Creep Strength: 92°C (165°F) > Rene' 125 in Coated Condition	75
• Strain Cycle LCF at 871°C (1600°F) at .5% ≥ 10 X Rene' 125 in Coated Condition	90
• Coatable for 12,000 Hrs. Life to 927°C (1700°F) 5,000 Hrs. to 1038°C (1900°F), 1,000 Hrs. to 1149°C (2100°F) + Recoatability	70
• Process: Meet Finished Part Cost Goal (See Manufacturing Probability of Success ~ 60%)	
OTHERS	
• Tensile Ductility at 704-982°C (1300-1800°F): ≥ 10% RA	
• Thermal Cycling Stability: After 5,000 Cycles from 982°C (1800°F) to R.T. Must Retain 90% of 982°C (1800°F) Rupture Strength	
• Hot Corrosion Res: ≥ Rene' 125 Oxidation Res: ≥ Rene' 80	
• High Temperature Stability: No Unpredictable Losses in Long Time Rupture Life	
• Must be Joinable for Max. Temperature Up to 1288°C (2350°F)	70
• Nil Strength Temperature: ≥ 1216°C (2400°F)	
• Physical Properties (to 1066°C (1950°F)): ± 5% of Rene' 80	
• Density: ≤ 8.72 g/cm^3 (.315/lb/in^3)	
• Youngs Modulus (Parallel to Growth Direction): ≥ 90 X 10^3 MN/m^2 (13 X 10^6 psi) ≤ 110 X 10^3 MN/m^2 (16 X 10^6 psi) at 870°C (1600°F)	
• 649-982°C (1200-1800°F) Impact (After Stressed Exposure):	
- Ballistic: ≥ 2.7J (2 ft-lb) No Cracks, .44 cm (175"D). Steel Ball, .23 cm (.090") Thick Spec.)	
- Charpy V-Notch: ≥ 13.6J (10 ft-lb).	

*Mechanical Properties - Longitudinal Direction

TABLE XI (Cont.)

TECHNICAL GOALS

FINISHED PART COST GOAL	SPECIFIC PART COST USED
≤ 2.5 X Rene' 80 (Conventionally Cast CF6 Type HPT Blade)	1.74 X R125 (Stg. 1) 1.8 X R125 (Stg. 2)

PROJECTED DEVELOPMENT COSTS

Total	$8,000,000
1977 & Beyond	$6,500,000

PROJECTED YEAR OF AVAILABILITY 1982

PROBABILITY OF SUCCESS

%		
	Material	70
	Design	80
	Manufacturing	60
	OVERALL PROBABILITY OF SUCCESS	60%

4.0 ADVANCED ENGINE DESIGN

Prior to performing the cost benefit and sensitivity analyses, the various aspects of the design of the advanced engine and the materials technology selection had to be defined and integrated. This Section describes those factors and discusses the design evaluation procedure utilized in this program.

4.1 ENGINE DEFINITION

Two engine designs were selected for this study consistent with the technology expected to be available for a 1985 certification period. Both designs were configured for CTOL commercial airline aircraft, with one sized for a Transcontinental mission and the other for an Intercontinental mission. The general engine configuration is described in Table XII.

The fan design and engine cycle are similar to those employed for the baseline engine in the NASA USTEDLEC studies[2] and the core engine employs a compressor design very similar to that evolved under the NASA AMAC study[3]. A representative cross-section of the engine is shown in Figure 2.

4.2 MATERIALS SELECTION

The five general technology areas listed in Table XIII were selected for study in this program. These were selected based on prior work on advanced systems and preliminary materials goals. As the study proceeded, however, it was necessary to make some revisions and additions to the originally selected technologies and to the baseline technologies used for comparison. These changes, shown in Table XIV, were made to permit more realistic design configurations as new data became available throughout the period of the study. The reason for each addition is briefly described below:

- Rene' 125 was added as a baseline technology material becuase it was believed that the strength characteristics of MA754 may have been marginal for the Stage 2 HPT vane.

- An 8% higher density W-reinforced FeCrAlY composite blade was evaluated because the increased temperature capability resulting from the higher tungsten fraction made the blade more attractive from a cost-effectiveness standpoint. The relatively low tip speed of the high pressure turbine made it possible to carry the increased weight of these blades.

- Titanium aluminide static structural components were added when it became evident that large weight savings, together with moderate cost savings, might be possible with this material. Further, it was felt that applications for this type of material might be more readily developed for static, rather than high speed rotating components.

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TABLE XII

Engine Description

•	Single Stage Fan with Booster	
•	Fan Pressure Ratio (Max. Cruise)	1.7
•	Overall Pressure Ratio	45.1
•	Turbine Rotor Inlet Temperature (Take-Off)	2600°F
•	Highly Loaded Low Pressure Turbine	
•	Mixed Flow Exhaust	
•	Installation Configured for Low Noise	

T_4 = 2600°F (SLS), CYCLE P/P = 38:1 (MAX CLIMB)

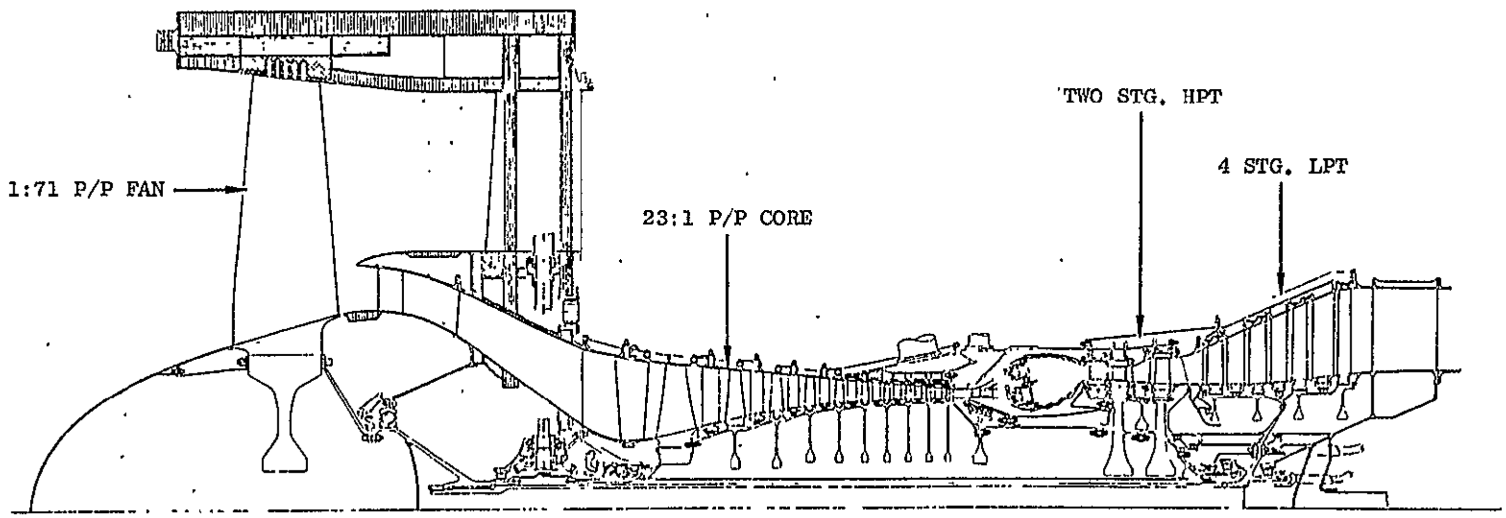

Figure 2. Baseline Engine Layout

TABLE XIII

MATERIALS TECHNOLOGY SELECTIONS

Technology Selections	Baseline Technology
• Ceramic Turbine Shrouds	Poroloy
• Ceramic Turbine Vanes	MA754
• Combustor Liner	HS188
– No Coat ODS	
– Ceramic	
• Advanced Turbine Blading	Rene' 125
– W-Wire Reinforced FeCrAlY	
– Eutectics	
– γ' ODS	
• Ti-Aluminides (LPT Blades)	Inconels

TABLE XIV

REVISIONS TO MATERIALS TECHNOLOGY SELECTIONS

Technology Selections	Baseline Technology
• Ceramic Turbine Vanes	MA754 Rene' 125*
• Advanced Turbine Blading	
- W-Wire Reinforced FeCrAlY	Rene' 125
A) Rene' 125 Equivalent Density	
*B) +8% Increased Density	
• Ti-Aluminides (LPT Blades) (Static Parts)*	Inconels

*Revisions

4.3 DESIGN EVALUATION PROCEDURE

The design evaluation procedure was formulated to ascertain the effects of substituting the advanced materials for the current technology materials in the respective engine parts.. These effects fall into the following categories:

- Cycle and performance effects (SFC, efficiencies, etc.)
- Design and weight
- Initial parts cost differences
- Maintenance cost differences (labor and replacement part cost)

Procedures used for each of the above design evaluation categories are outlined in the following sections. They follow quite closely those procedures used in the USTEDLEC study referenced earlier [2] and the same aircraft missions were used as a basis for evaluation. A typical design evaluation procedure is summarized in Table XV.

4.3.1 Cycle and Performance Effects

Most of the overall operating characteristics of the baseline engine were maintained as the advanced technology parts were substituted. These constant parameters included the following major characteristics:

- Thrust
- Turbine Inlet Temperature
- Overall Pressure Ratio
- Fan Pressure Ratio

To maintain a balanced cycle as the engine component efficiencies or cooling flows varied with substitution of the advanced part changes in core size and bypass ratio were required. In addition to the primary cycle/efficiency changes, small weight and cost differences for the engine resulted from changes in core size when the engine was used to maintain the thrust needed for the mission requirements. These secondary changes were added to the primary weight and cost changes resulting from the direct parts replacement.

Substitution of advanced technology materials for the baseline materials had varying effects on the engine cycle and performance. For some substitutions, such as the substitution of titanium aluminides for nickel base alloy parts, there were no cycle or performance effects since only weight and cost were affected. On the other hand, substitution of ceramic for metal vanes had a considerable effect on cycle and performance. The large differences in cooling air needs resulted in significant differences in core engine weight, combustor exit temperature and bypass ratio.

TABLE XV

TURBINE BLADE DESIGN EVALUATION PROCEDURE

Base Engine	1985 Advanced Technology Engine with Best Current Blade Materials.
Life Requirement	Consistent with Typical GE CF6 Commercial Mission and Use. All Blades Designed to Same Life.
Blade Temperature Limits	Bulk Blade Metal Temperatures Set to Meet Above Life Requirements with Imposed Operating Stresses.
Blade Relative Gas Temperatures	Average Pitchline Cycle Temperatures Adjusted for Margin (Tolerances, Transients, Deterioration) With Vector Velocity Effects Incorporated. Pitchline Temperatures Include Radial Profile Effects.
Cooling Flows	Determined by Calculating Cooling Flow Needed for Assumed Cooling Technology Effectiveness to Attain Bulk Temperature Limits.
Material Density Effects	Constant Stress Designs Used in Supporting Structures Along with Part Weights Calculated.
Core Scaling Effect	Core Size Scaled with Required Cooling Flow to Keep Constant Thrust Engine with Given Fan Size. Weight and Cost Scaled Accordingly.

Changes in cooling flows were determined using the design technique of setting life requirements, estimating the gas path environment taking into account engine temperature margins and radial profiles, and setting an allowable bulk metal temperature for the estimated stress levels. The amount of cooling air was then related to the cooling effectiveness required to achieve the allowable bulk metal temperature with a cost effective cooling scheme.

The changes in cooling flow impacted the cycle in several ways. Component efficiency was affected by the quantity and placement of cooling flows within the turbine. The size of the core was also affected together with some variation in cycle station to station gas path temperatures. Bypass ratios were also affected by changes in the balanced cycle core size.

4.3.2 Design and Weight

Baseline engine weight and design were established using baseline material technology parts. Advanced technology parts were then designed and substituted directly and any weight differences calculated.

Generally, it was possible to make substitutions without the necessity for configuration changes in the substituted parts, but some advanced materials technologies required very different configurations. An example of this is shown in Figure 3. When substituting ceramic for metal in the HPT vanes and shrouds, a new concept in design configuration is required.

Another example of a less complex configuration change was that required when comparing a shingle liner combustor with a more conventional sheet metal combustor. Figure 4 is a schematic drawing of how the shingles are arranged and utilized in a combustor.

In one case, there was a small configuration change required due to the different manufacturing requirements of the advanced materials being substituted. The two different HPT blade cooling schemes, one employed for a castable alloy, such as Rene' 125 or Ni82XB, and the other required for a material such as the Tungsten composite material and the γ' ODS alloys, are shown in Figure 5. To avoid penetration of areas of dense tungsten wire concentrations, film cooling hole locations were limited to the leading edge and one downstream location on the airfoil. Therefore, an improved internal cooling scheme, such as the impingement tube employed in this case, was necessary to keep overall cooling effectiveness high. Fabrication limitations for the wrought γ' ODS alloy also necessitated a less complex internal cooling passage configuration and the utilization of an internal impingement cooling system.

In each case, however, consistent design and life requirements were applied where possible so that the overall engine life and reliability would not be affected by the materials substitutions. The minor unavoidable variations in these requirements that did arise were due to configuration or basic material capability differences.

As each advanced material technology part was substituted for the baseline part, any effects due to weight differences were calculated. These effects resulted from changes in the weight of the part itself or in associated hardware. For example, when the denser Tungsten composite alloy blades were substituted, not only was the blade itself heavier, but the HPT disks necessarily became heavier.

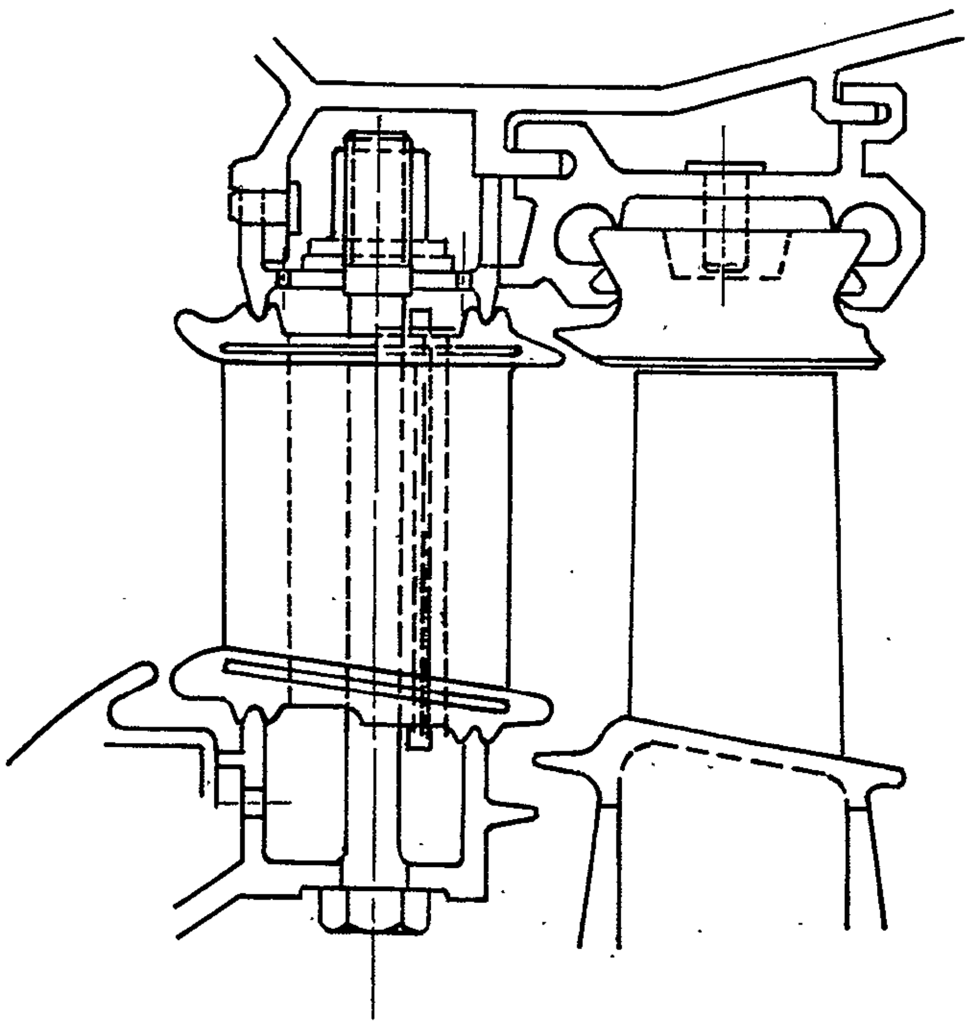

Figure 3. Ceramic Vane and Shroud Design

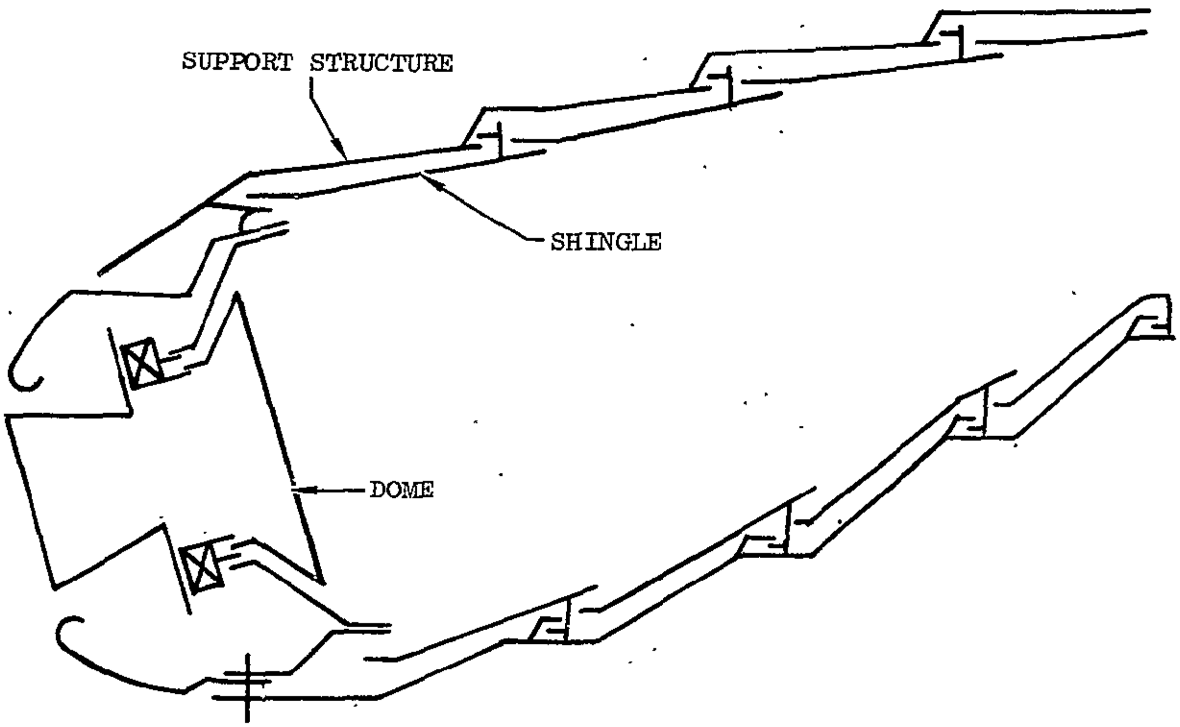

Figure 4. Shingled Combustor Design

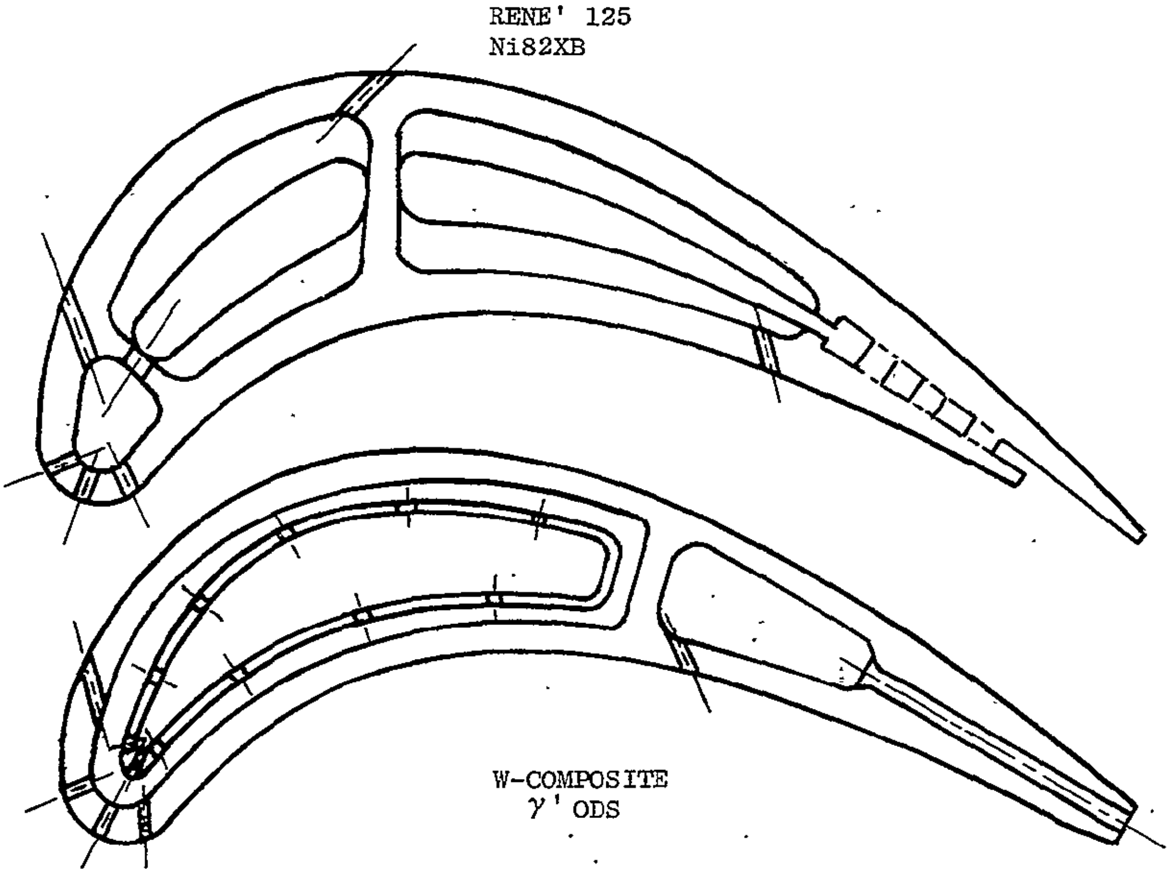

Figure 5. Cooling Schemes for High Pressure Turbine Blades

Most of the re-design with the new materials was, however, quite straight forward with new property goals (Tables IV-XI) applied and consistent design requirements applied to both the current and the advanced material part.

4.3.3 Initial Parts Costs Differences

For the baseline engine parts, estimates of the initial parts costs were made based upon input from Advanced Manufacturing personnel in conjunction with Manufacturing people now associated with similar production parts. The impact on price was scaled to the engine sizes required for the two missions evaluated in this study.

Costs for the advanced technology parts were derived by the same advanced manufacturing personnel but with input from Materials Laboratory personnel familiar with the specific materials and technologies. By doing this, it was felt a more consistent costing mechanism resulted, especially since the manufacturing steps that were common to similar parts often resulted in the largest portion of a part cost. Naturally, the cost estimates for the advanced material parts are less firm than the current material part costs. For this reason, sensitivity studies were conducted on many of these costs to evaluate the effect of deviations from the nominal cost estimates. These results are presented in Section 6 of this report. The advanced material goals given in Section 3 contain part cost goals and these were incorporated into the cost estimates as appropriate, although in many cases estimated costs did differ from the goal. The sensitivity studies point out those material technologies where actual parts cost radically affects the overall benefit attributed to that material.

4.3.4 Maintenance Cost Differences

Maintenance costs were assumed to consist of two major items, namely parts costs and labor. Rather than try to develop a labor cost for each part based on engine teardown time, the labor costs portion was attributed to each part on the basis of a fixed ratio of the cost of the replacement part. For this study, the maintenance cost total was assumed to consist of a 70% part cost and a 30% labor cost. Therefore, once a part cost was developed, the labor portion of the maintenance costs for that part was 3/7 of that figure.

As previously noted, constant design lives and design criteria were applied wherever possible to all component substitutions. Therefore, for most parts, the only large differences in maintenance costs between current and advanced technology materials were due to replacement part cost differences. However, for some of the components, replacement factor* estimates between the baseline technology and the advanced technology were quite different and this significantly impacted maintenance costs.

Mature General Electric CF6 fleet estimates were used as a guide for the initial estimates of the replacement factors, particularly for the baseline technology parts. Then, any inherent advantages or disadvantages of the advanced technology materials were included in the replacement factor estimate. For example, the expected superior low cycle fatigue properties of the advanced HPT blade alloys resulted in a 20% reduction in the replacement factor estimates. Table XVI shows the estimated replacement factors used for this study.

*The value of the replacement factor indicates how often that particular part would be replaced over an assumed engine life of 36,000 hours on the average.

TABLE XVI

EXPECTED PART REPLACEMENT FACTORS FOR 36,000 HOURS ENGINE LIFE

Part	Baseline Technology	Advanced Technology
HPT Shroud	9.1 (Poroloy)	4.5 (Ceramic)
HPT Stage 1 Vane	5.0 (MA754)	4.0 (Ceramic)
HPT Stage 2 Vane	6.1 (Rene' 125)	4.0 (Ceramic)
Combustor Liner	2.8 (HS188)	0.8 (Ceramic)
		2.0 (No Coat ODS Alloy)
HPT Stage 1 Blade	2.3 (Rene' 125)	1.9 (All Advanced Alloys)
HPT Stage 2 Blade	2.1 (Rene' 125)	1.7 (All Advanced Alloys)
LPT Blades	0.3 (Ni-Base Alloy)	0.3 (Titanium Aluminides)
Static Hot Parts	0.05 → 0.4 (Ni-Base Alloy)	0.05 → 0.4 (Titanium Aluminides)

Since the replacement factors are only estimates and subject to some error, particularly for the new technologies, sensitivity studies were conducted for those parts where the replacement factor appeared to be critical. The results of the sensitivity studies are presented in Section 6.

4.4 SUMMARY OF ENGINE DESIGN STUDIES

Differences in weight, part cost, maintenance cost and performance effects resulting from substitution of advanced materials for baseline materials were determined in the design evaluation. These data were required to perform the benefit analysis portion of this study to determine the effects on the total airframe/engine system.

Results of the engine design evaluation are shown in Table XVII in which data for both the Transcontinental mission [~ 89,000N (20,000 pounds thrust) engine] and the Intercontinental mission [~ 93,400N (21,000 pounds thrust) engine] are presented. Differences between the two occur as a result of the inherently more expensive engine required for the Intercontinental mission rather than as a result of differences in efficiencies.

Generally, the higher temperature capabilities of the blade, vane and shroud materials were reflected in lower SFC and lower weight. However, for some of the materials, especially the HPT blade materials, these favorable effects were offset to some degree by the much higher initial costs and the higher maintenance costs (higher maintenance costs reflecting primarily the high cost of the replacement part). The combustor liner and titanium aluminide materials substitutions did not produce any SFC changes and the major impact of these substitutions was to cause differences in weight, initial cost, maintenance cost and TOGW.

The various design effects were combined and evaluated in the cost benefit portion of the study discussed in the following Section.

TABLE XVII

DESIGN AND ECONOMIC EVALUATION RESULTS

MISSION SIZE TRANSCONTINENTAL Δ's/INTERCONTINENTAL Δ's

			Design Factors			
Component	Adv. Tech.	Base Tech.	Δ SFC %	Δ Price $1000	Δ Maint. $/Hr.	Δ Wt. Lbs.
Stg 1 HPT Shrouds	Ceramic	Poroloy	-.18/-.18	-4.1/-5.1	-1.83/-2.19	-15/-16
Stg. 2 HPT Shrouds	Ceramic	Poroloy	-.19/-.19	-.4/-.4	-1.83/-2.19	+9/+10
Stg 1 HPT Vanes	Ceramic	MA754	-.43/-.43	-13.5/-16.7	-3.48/-4.14	-6/-6
Stg 2 HPT Vanes	Ceramic	R125	-1.04/-1.04	-9.1/-11.2	+.29/+.34	-91/-98
Combustor Liner	Ceramic	HS188	0/0	+11.5/+14.2	-1.24/-1.47	+8/+8
	No Coat ODS	HS188	0/0	+22.4/+27.7	+1.30/+1.55	+18/+19
LPT Blades	TI-A1	Inconel	0/0	+1.3/+1.6	+.04/+.04	-32/-34
Static Parts	TI-A1	Inconel	0/0	-18.6/-23.1	-.18/-.20	-75/-79

TABLE XVII (Cont.)

MISSION SIZE TRANSCONTINENTAL Δ's/INTERCONTINENTAL Δ's

Component	Adv. Tech.	Base Tech.	Design Factors ΔSFC %	ΔPrice $1000	ΔMaint. $/Hr.	ΔWt. Lbs.
Stage 1 HPT Blade	NI82XB	R125	-.33/-.33	+8.9/+11.1	+1.02/+1.21	-46/-48
	γ' ODS	R125	-.33/-.32	+21.2/+26.2	+1.98/+2.35	-46/-48
	W-Comp. (.305)	R125	-.35/-.35	+19.7/+24.4	+1.93/+2.30	-50/-53
	W-Comp. (.33)	R125	-.39/-.39	+18.8/+23.3	+1.93/+2.30	-50/-53
Stage 2 HPT Blade	N182XB	R125	-.19/-.19	+13.3/+16.4	+.82/+.97	-17/-18
	γ' ODS	R125	-.19/-.19	+33.2/+41.1	+2.21/+2.63	-17/-18
	W-Comp. (.305)	R125	-.22/-.22	+30.7/+38.1	+2.07/+2.47	-19/-20
	W-Comp. (.33)	R125	-.25/-.25	+30.3/+37.5	+2.07/+2.47	-13/-14

5.0 BENEFITS ANALYSIS

5.1 MISSION AND AIRCRAFT DEFINITION

Two aircraft missions, Transcontinental and Intercontinental, were evaluated in the benefits analysis study. The aircraft, design range, payload and aerodynamic performance are briefly described in Table XVIII. The construction of the aircraft is consistent with the technology anticipated for a 1985 time period and significant portions of the airframe make use of advanced structural composites. Aircraft characteristics also include supercritical wing aerodynamics and high aspect ratios.

The evaluation procedure was very similar to that employed in a previous cost benefit study performed by General Electric for NASA[4]. The Direct Operating Costs (DOC) calculations have been updated to reflect 1977 dollars, a 35 ¢ and 45 ¢ per gallon fuel charge (Transcontinental and Intercontinental rates respectively) and generally agree with the 1976 Boeing update of the ATA formulae. Indirect Operating Costs (IOC) were determined using the 1970 Lockheed Georgia Report[5]. Return on Investment (ROI) was calculated using the DOC's and IOC's as determined above, a 48% tax rate, and discounting the resulting cash flow.

Design change effects on the aircraft system size and economics were determined using the trade factor derivatives shown in Table XIX. The derivatives are based on an average mission range and a 55% load factor as defined in the table.

A general description of the cost benefit evaluation procedure is presented in Table XX. The evaluation was carried out maintaining aircraft payload and range constant, while allowing the gross weight to vary in response to changes in the engine's characteristics as materials substitutions were introduced.

As each substitution was performed, the effect of that change was determined in terms of design effects which were then translated into aircraft system economic and size effects. The engine weight and cost were then adjusted based on the required thrust of the resized aircraft using the engine and installation price and weight scaling exponents shown in Table XX. Utilization of engine to aircraft system derivatives allowed both economic (ΔDOC, ΔROI, ΔROI, ΔW_f) and size effects (TOGW) to be calculated as each advanced materials technology substitution was made. It should be pointed out that the aircraft/engine system evaluated in the above procedure was an attempt to describe a representative airframe/engine system that could be available in the 1985 time period. It was not intended to be an assessment of the airframe capability but was rather a vehicle required to determine the trade factor derivatives needed for the cost benefit study.

5.2 ADVANCED MATERIALS BENEFITS

All the design effects resulting from the advanced materials technologies were entered in a benefit analysis procedure that determined the change in benefit from the baseline material technology configuration. The benefits were expressed in terms of ΔDOC, ΔROI, and ΔW_f. The trade factor derivatives described above were employed to convert the design effects into the system economic effects.

TABLE XVIII

AIRCRAFT DEFINITION SUMMARY

Baseline Aircraft Description	Transcon Trijet	Intercon Quadjet
• Technology Level	Supercritical wing; structural composites	
• Design Range (n.mi)	3,000	5,500
• Cruise Altitude (ft)	35,000	35,000
• Cruise Mach Number	.80	.80
• TOGW (lbs)	223,000	320,000
• Number of PAX	200	200
• Design Payload (lbs)	41,000	43,000
• SLS T/O Fn/Engine (lbs)	20,000	21,000
• Wing Aspect Ratio	12	12
• Cruise L/D Avg.	17	18

TABLE XIX

MISSION TRADE FACTORS

	Transcon Trijet				Intercon Quadjet			
Average Mission Range (n.mi.)	700				3000			
Load Factor (%)	55				55			
Fuel Costs (¢/gal)	35				45			
	Δ's -- %				Δ's -- %			
	DOC	ROI	TOGW	W_f	DOC	ROI	TOGW	W_f
1% SFC	.42	-.12	.47	1.09	.71	-.25	.87	1.44
100 lb Engine/Nacelle wt	.17	-.06	.32	.26	.22	-.09	.39	.31
$1/Flt. hr - Maint. Cost	.17	-.03	--	--	.19	-.04	--	--
$1000 Engine/Nacelle Initial Price	.006	-.004	--	--	.005	-.003	--	--
$1000 Engine Initial Price with Replacement Parts	.010	-.004	--	--	.010	-.004	--	--

NOTE: Benefits Arise when

DOC - TOGW -
ROI + W_f -

TABLE XX

EVALUATION PROCEDURE

- Baseline Aircraft
  - 3000 n.mi./200 PAX Trijet (Transcon)
  - 5500 n.mi./200 PAX Quadjet (Intercon)
- Constant Payload and Range, Variable Gross Weight
- Mission Trade Factors Developed for Various Engine Changes
- Baseline Engine 1.71 P/P Fan, Mixed Flow with Advanced Technology Compatible with Late 1980's Service Entry
- Effect of Change in Installed Engine Characteristics Determined for Each Engine Variation Studied
- Effects of Engine Price Related to Production Costs - 1977 $
- Individual Parts Replacement Rates Considered in Engine Maintenance Costs
- Engine Scaled to Thrust Required by Baseline Aircraft

Engine Scaling Exponent	Weight - 1.25	Price - .55
Installation Scaling Exponent	Weight - 1.1	Price - .80

Results of the materials substitutions on ΔDOC, ΔROI, and ΔW_f for the Intercontinental mission are presented schematically in Figures 6-9. The quantitative values for each factor and both missions are listed in Table XXI. In comparing the results for the two missions, it must be remembered that the increased effect of fuel cost on the Intercontinental mission is reflected in fuel conserving materials substitutions showing more favorably in the analysis.

Generally, the substitution of ceramic components had a very favorable economic impact except in the case of the ceramic combustor liner where the benefit was marginal. Similarly, the substitution of titamium aluminides for the baseline technology material showed an overall benefit; this was not a particularly large benefit for any one component evaluated but became significant when the benefits from several parts were accumulated. Surprisingly, the benefit of titanium aluminides was not as great for LPT blades as it was for the static components. This is a direct result of the higher cost of the titanium aluminide LPT blade offsetting much of the weight advantage. The payoff might be larger with some other LPT configuration such as that employed in a high speed geared engine.

Within the scope of this specific engine/aircraft mission, substitution of the high temperature blade alloys for the baseline material was not generally economically attractive except for a small benefit resulting from utilization of the DS eutectic alloy. This situation resulted from the estimated high cost of the finished blades more than offsetting the economic gain resulting from reduced fuel consumption. For these alloys to be more generally attractive in a commercial environment, initial cost would have to be reduced from that now projected. On the other hand, in an aircraft system where fuel consumption and weight are much more important, such as a military fighter, these advanced HPT alloys will show a significant payoff even with the high initial cost.

An overall ranking of each of the advanced materials in terms of the "relative value" (defined in Section 2) is given in Figure 10. The starred items were not ranked strictly in accordance with the definition because their ΔROI was negative. However, they were ranked in terms of ΔW_f saved and increasing ΔROI penalty.

5.3 PRESENT WORTH EVALUATION

As part of the cost benefit study, a present worth evaluation of each of the advanced technology material substitutions was conducted. The main assumptions made in this evaluation were:

- Development costs spent during 1977
- Advanced technology utilization in fleet begins in 1985
- Fleet build-up, stabilization, and decline lasts 22 years
- Constant utilization time per aircraft per year in the fleet

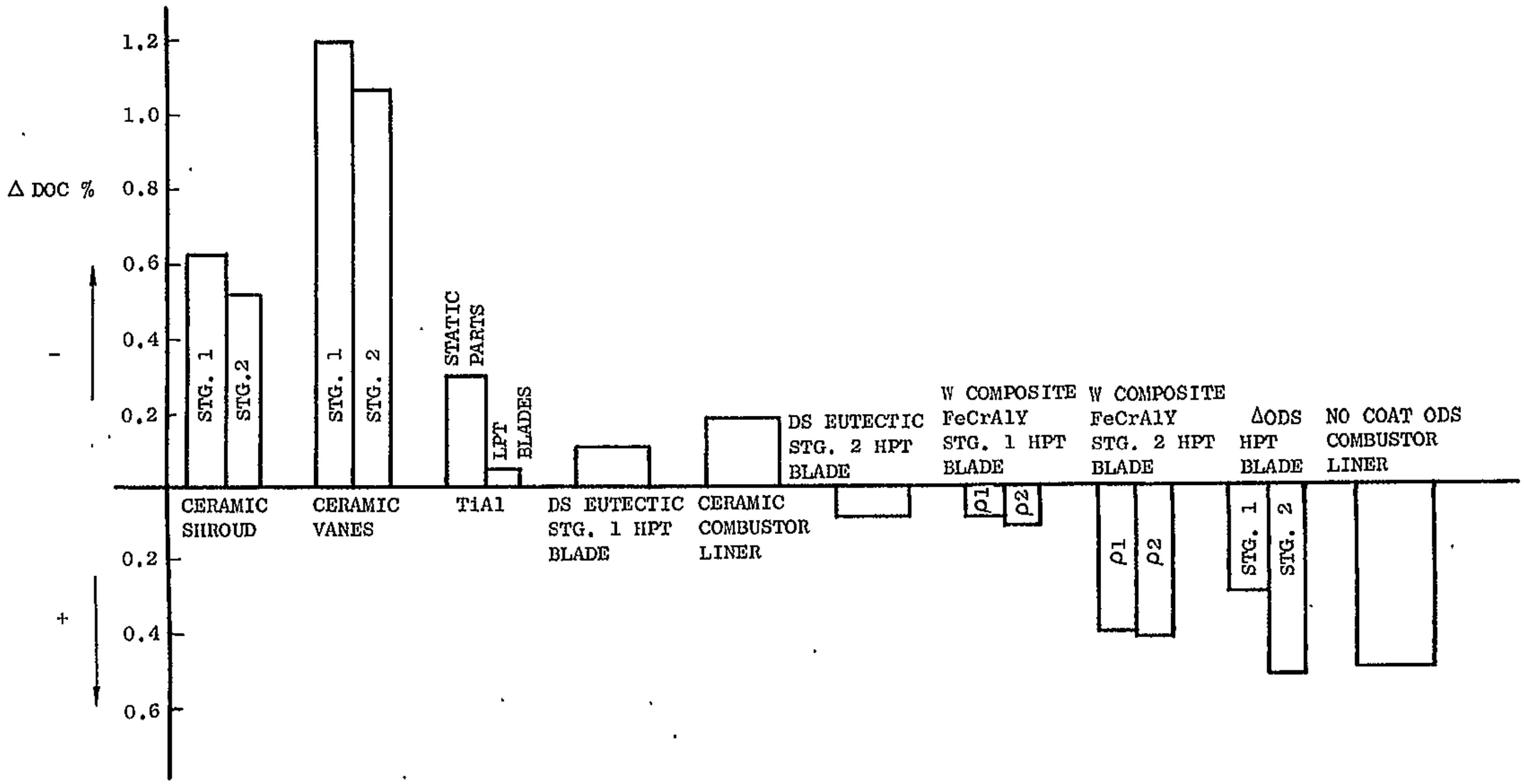

Figure 6. Effect of Advanced Materials Substitution on ΔDOC For Intercontinental Mission

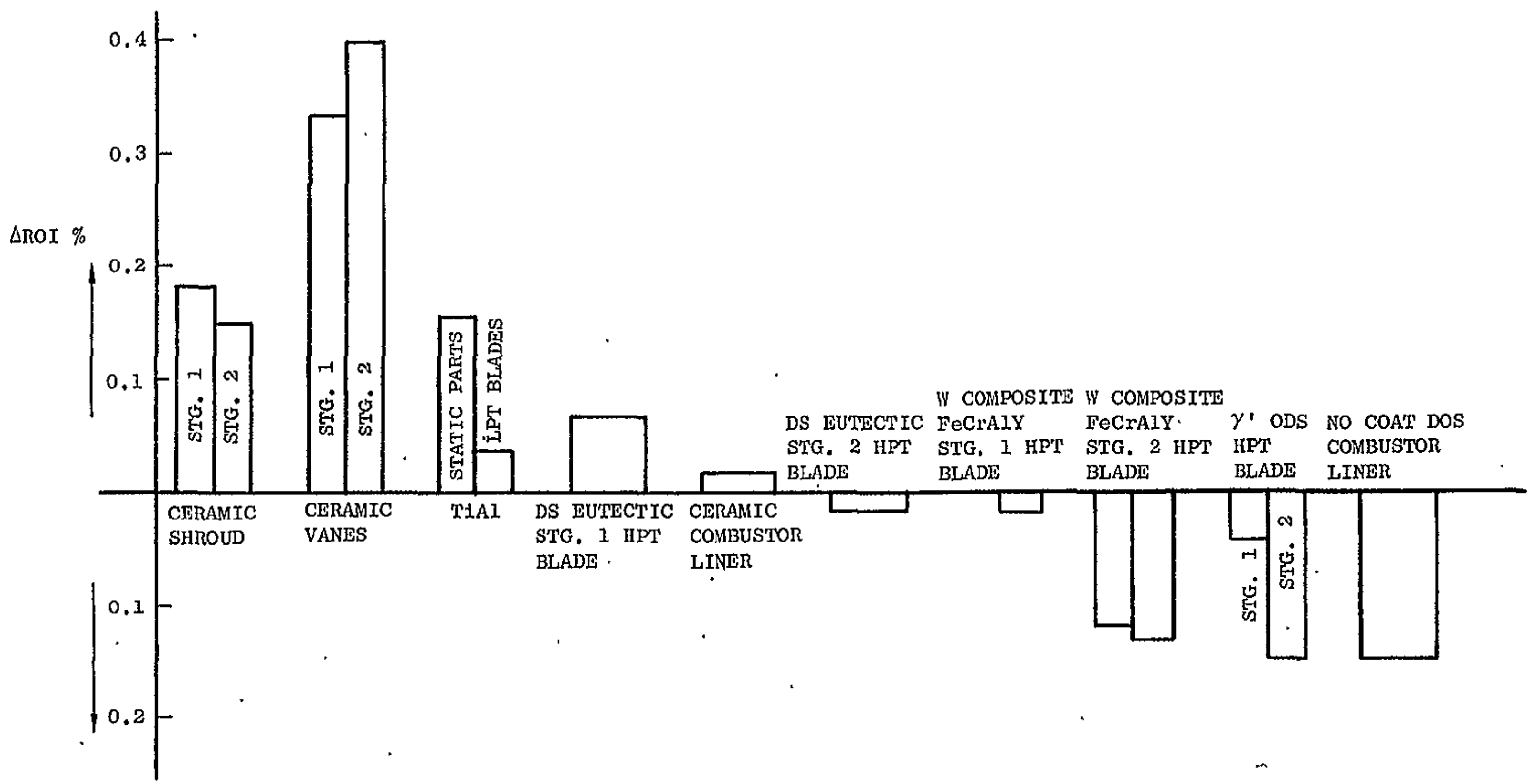

Figure 7. Effect of Advanced Materials Substitution on ΔROI For Intercontinental Mission

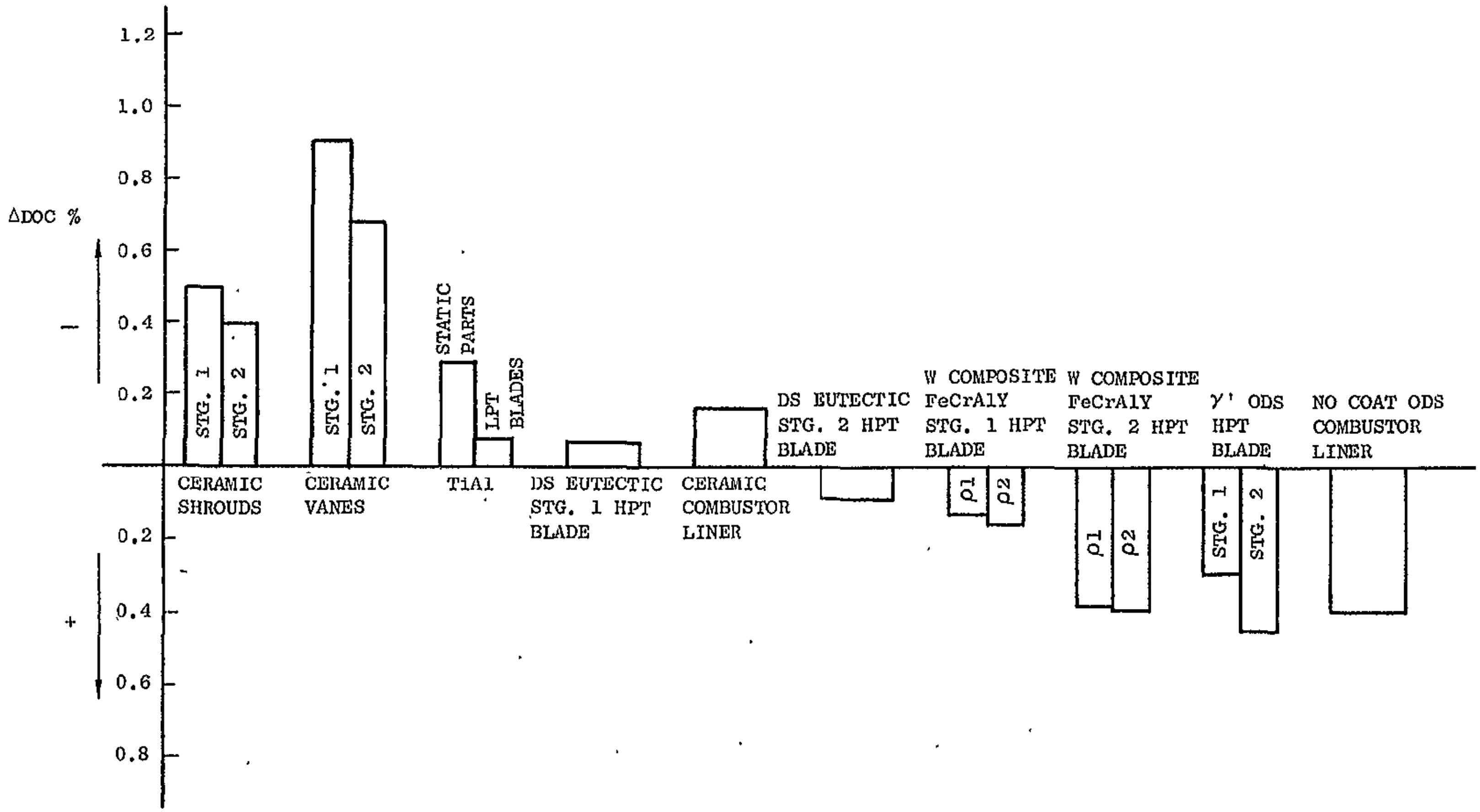

Figure 9. Effect of Advanced Materials Substitutions on ΔDOC For Transcontinental Mission

TABLE XXI

ADVANCED MATERIALS ECONOMIC BENEFITS

			Mission Results					
			Transcontinental Δ's			Intercontinental Δ's		
Component	Adv. Tech.	Base Tech.	ΔW_f	ΔDOC	ΔROI	ΔW_f	ΔDOC	ΔROI
Stg 1 HPT Shrouds	Ceramic	Poroloy	-.23	-.46	+.10	-.31	-.62	+.17
Stg 2 HPT Shrouds	Ceramic	Poroloy	-.18	-.39	+.07	-.24	-.54	+.13
Stg 1 HPT Vanes	Ceramic	MA754	-.49	-.88	+.21	-.64	-1.19	+.33
Stg 2 HPT Vanes	Ceramic	R125	-1.37	-.68	+.22	-1.8	-1.02	+.39
Combustor Liner	Ceramic	HS188	+.02	-.14	-.01	+.03	-.19	+.01
	No Coat ODS	HS188	+.05	+.38	-.13	+.06	+.47	-.16
LPT Blades	TI-A1	Inconel	-.08	-.04	+.01	-.1	-.06	+.02
Static Parts	TI-A1	Inconel	-.19	-.26	+.12	-.25	-.33	+.15

TABLE XXI (Cont.)

Component	Adv. Tech.	Base Tech.	Mission Results: Transcontinental Δ's			Intercontinental Δ's		
			ΔW_f	ΔDOC	ΔROI	ΔW_f	ΔDOC	ΔROI
Stage 1 HPT Blade	DS Eutectic	R125	-.47	-.03	+.01	-.62	-.12	+.06
	γ' ODS	R125	-.47	+.20	-.06	-.62	+.17	-.04
	W-Comp. (.305)	R125	-.51	+.16	-.05	-.68	+.11	-.02
	W-Comp. (.33)	R125	-.55	+.14	-.05	-.73	+.08	0
Stage 2 HPT Blade	DS Eutectic	R125	-.25	+.09	-.04	-.33	+.06	-.02
	γ' ODS	R125	-.25	+.44	-.15	-.33	+.49	-.16
	W-Comp. (.305)	R125	-.29	+.39	-.13	-.38	+.42	-.14
	W-Comp. (.33)	R125	-.31	+.38	-.13	-.40	+.41	-.13

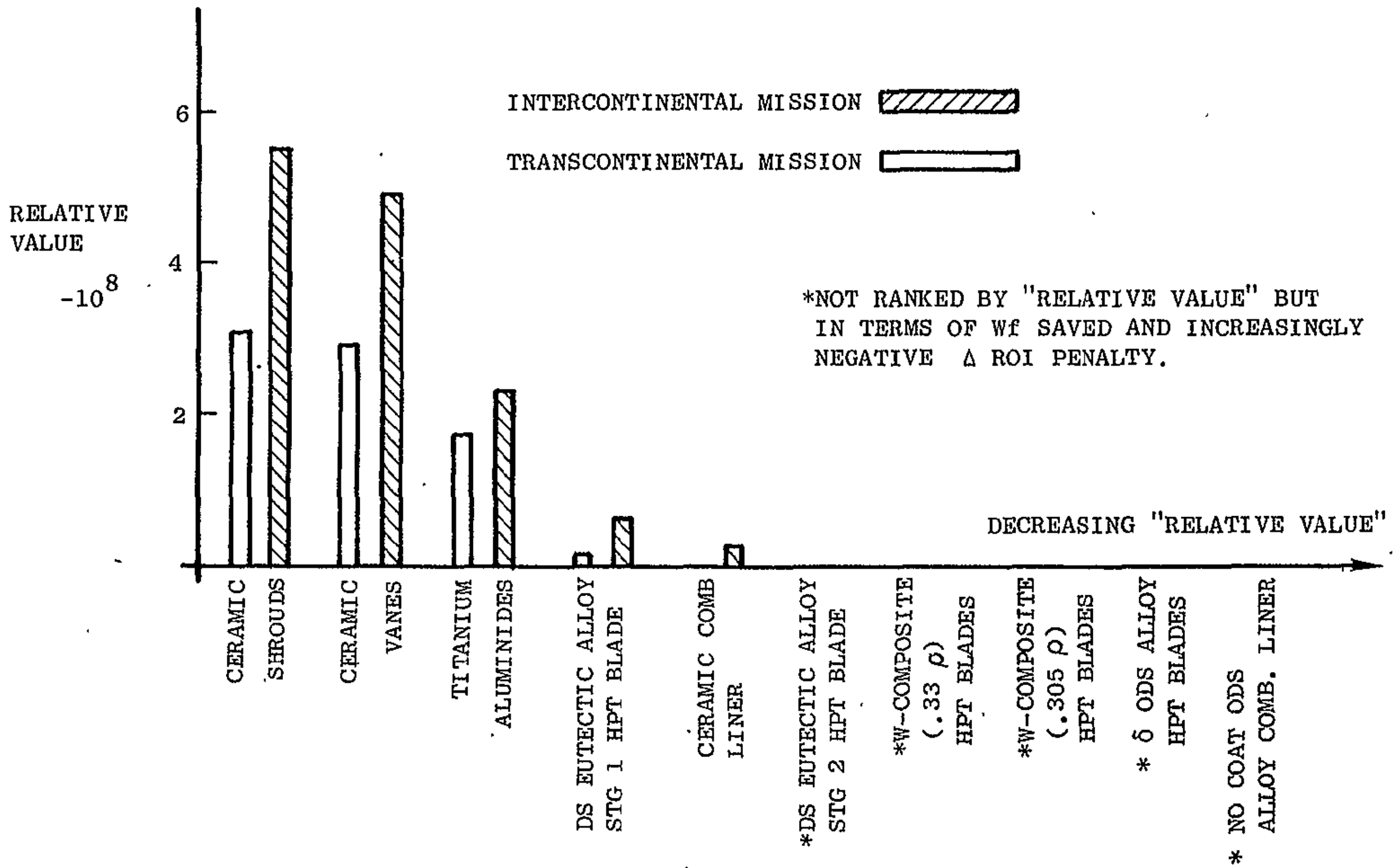

Figure 10 "Relative Value" Ranking of Materials Technologies

The actual savings per year were calculated by multiplying the Direct Operating Cost per hour by the negative ΔDOC values generated in the benefit analyses. These savings per hour were then multiplied by the number of aircraft in the fleet for that year and the utilization rate for the type of aircraft being considered. The values used for the assumptions and a bar chart illustrating graphically the build-up and decline of the fleet size as a function of year are shown in Figure 11.

Using the savings per year values calculated, the present worth (1977) of the savings was determined using the discounted present worth factor for the period of time from 1977 to the year in which the savings were accrued. The results of these calculations for each component studied are shown in Figure 12 for both the Transcontinental and Intercontinental missions. As a class, the ceramic components had the largest present worth benefits. The titanium aluminides had the next largest present worth with the Ni82XB stage 1 HPT blade material showing a marginal benefit.

Tables XXII through XXIV summarize the results of the cost benefit study and compare the various figures of merit that have been used. In Table XXII, a comparison of "present worth" to development cost is presented for each class of materials determined to have positive "relative value". The ratios vary considerably, ranging from > 1:1 for the advanced HPT blade. Although there are some reversals in ranking, it can be seen from Table XXIII that both the "relative value" and "present worth" figures of merit are relatively consistent. Finally, Table XXIV shows the relative value, the ΔROI, the development cost and the probability of success for the five technologies with positive relative values to summarize the major findings of the cost benefit analysis.

	TRANSCONTINENTAL	INTERCONTINENTAL
DOC - $/FLT HR	1303	1833
UTILIZATION - HRS/YR-ACFT	3410	4040
INVESTMENT YEAR	1977	1977
DISCOUNT RATE - %	14.6	14.6

PRESENT WORTH = THEN YEAR SAVINGS · PRESENT WORTH FACTOR

FLEET SIZE AND BUILD-UP RATE

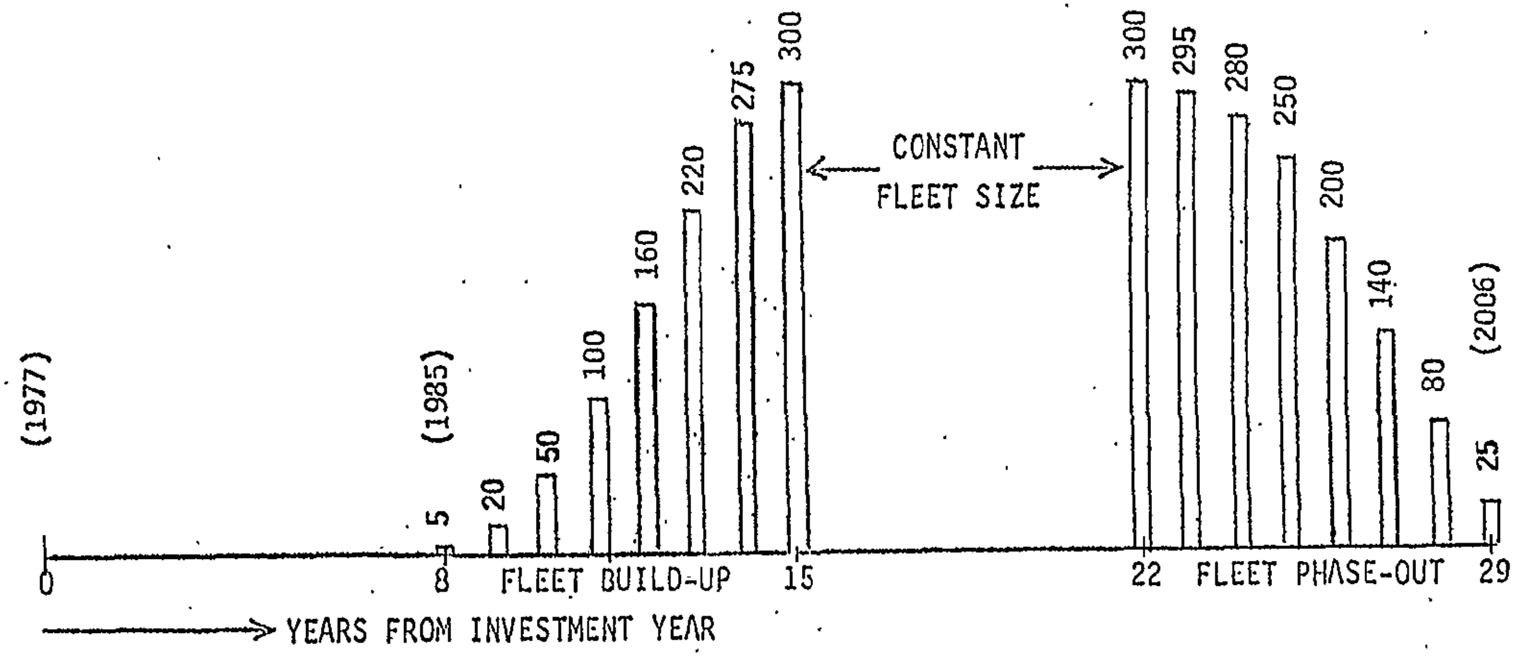

Figure 11. Present Worth Evaluation Procedure

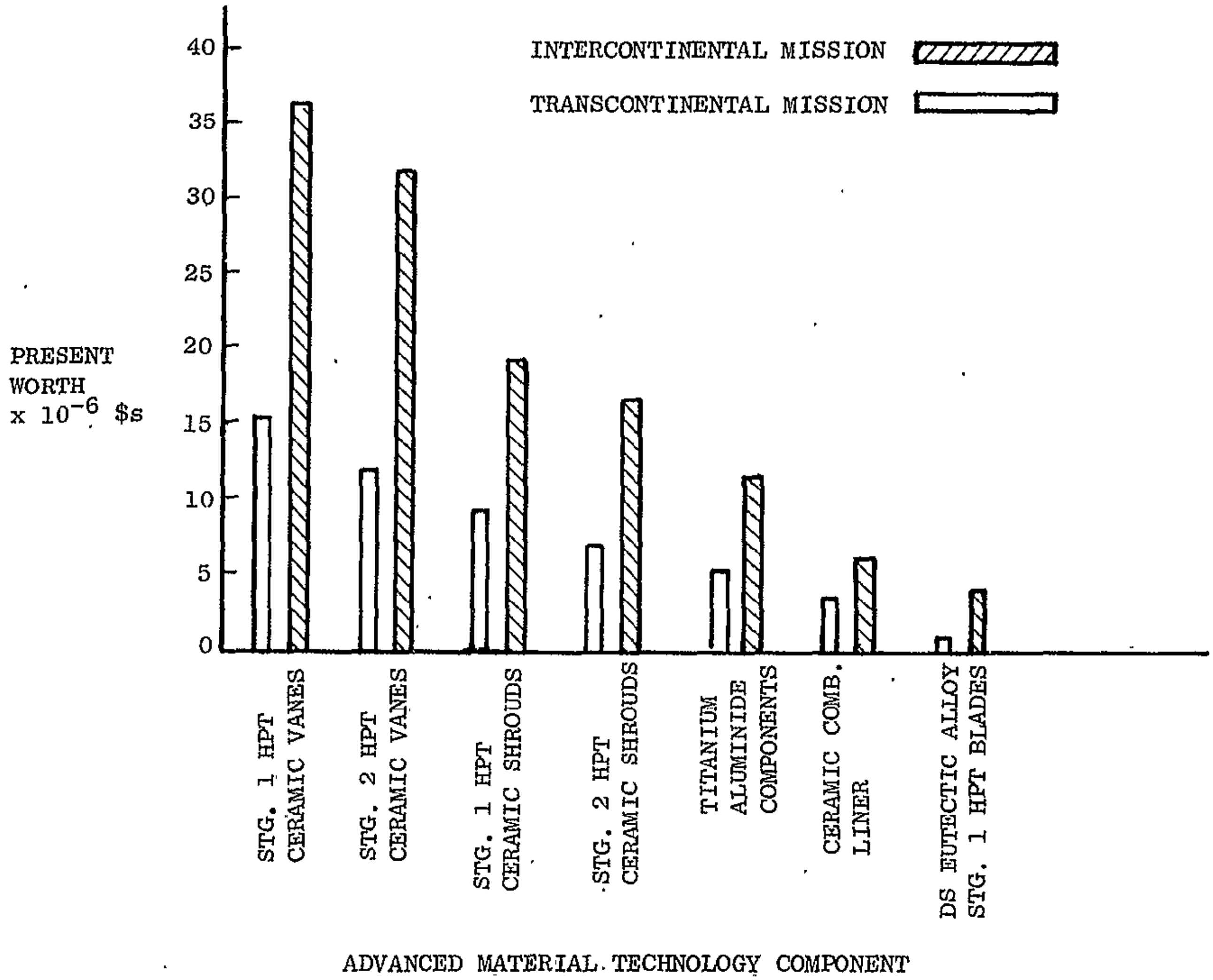

Figure 12 Present Worth Evaluation Results

TABLE XXII

PRESENT WORTH EVALUATION RESULTS

Technology	Development Cost X 10^{-6} - $	Present Worth X 10^{-6} - $ Transcon.	Intercon.
Ceramic HPT Shrouds	3.3	15.0	34.2
Ceramic HPT Vanes	6.0	27.6	66.6
TI-AL Parts	3.0	4.9	11.3
Ceramic Combustor Liner	3.0	2.5	5.7
DS Eutectic Stg 1 HPT Blade	6.5	.6	3.6

TABLE XXIII

COMPARISON OF RELATIVE VALUE AND PRESENT WORTH FOR MATERIALS TECHNOLOGIES (TRANSCONTINENTAL MISSION)

Technology	"Relative Value" Rank	Present Worth Rank
Ceramic HPT Shrouds	1	2
Ceramic HPT Vanes	2	1
TI-AL Static and Rotating Parts	3	3
Ceramic Combustor Liner	-	4
DS Eutectic HPT Blades	4	5

TABLE XXIV

ADVANCED TECHNOLOGY RESULTS SUMMARY TRANSCONTINENTAL RESULTS/INTERCONTINENTAL RESULTS

Advanced Technologies	Relative Value x 10^{+8}	ΔROI	Development \$ x 10^{-6}	Probability of Success %
Ceramic HPT Shrouds	3.1/5.5	+.17/+.30	3.3	60
Ceramic HPT Vanes	2.9/4.9	+.43/+.72	6.0	40
Ti-Al Static Parts and LPT Blades	1.7/2.3	+.13/+.17	3.0	40
DS Eutectic Stg 1 HPT Blades	.1/.5	+.01/+.05	6.5	60
Ceramic Combustor Liner	-/.2	-/+.01	3.0	50

6.0 SENSITIVITY ANALYSIS

In the cost benefit studies, nominal values of the design parameters associated with each advanced material technology were utilized to develop the system economics figures of merit. Failure to attain the desired properties or cost goals, or any deviation from the nominal values used in the analysis, would clearly result in a change in the benefits accrued. It was determined in the study that certain parameters for each technology were more critical than others and that relatively minor changes in these parameters could have a profound effect on the eventual system benefit. Sensitivity analysis were performed by varying design values and deriving the change in benefits to determine quantitatively the impact of deviations in critical properties from nominal values. The Transcontinental mission was selected for these sensitivity studies to illustrate the effects, although similar changes would be anticipated for the Intercontinental mission.

6.1 CERAMIC SHROUDS

With the maximum cycle T_4 (2600°F) selected for this engine, it is anticipated that a very significant lowering of ceramic operating temperature capability (on the order of 350°F) would be necessary to affect the results previously obtained. Therefore, the most serious problem would probably be the inability to achieve the expected reliability assumed for the shrouds since the HPT shrouds have traditionally been a high replacement rate part. Figure 13 illustrates the effect on ΔDOC and ΔROI if the reliability (or replacement factor) is not as was assumed in the benefits study. The replacement rate would have to reach a value about 50% higher than the baseline Poroloy shroud before all benefit of the ceramic would be lost. Therefore, the ceramic shroud would appear to be relatively insensitive to property deficiencies which could cause this to become an uneconomical development.

6.2 CERAMIC VANES

The concern in the sensitivity study for the ceramic vanes was essentially the same as that for the shrouds except that, since the vane is exposed directly to the gas path, the reliability and its effect on anticipated benefit are even more critical.

Both Stage 1 and Stage 2 HPT ceramic vanes replacement factors were varied to determine the effect on benefits. In Figure 14, the slope of ΔDOC as a function of parts index is such that it would require more than a doubling of the assumed replacement factor to lose all economic benefit. However, when the ceramic vane is compared to a Rene' 125 vane (for Stage 2), a relatively small upward change in replacement factor for the ceramic vane results in loss of benefit compared to the lower cost Rene' 125 (Figure 15). This indicates that the ultimate benefit to engine operation of a ceramic Stage 2 vane is very dependent on the material and design application being successfully applied in this location to maximize reliability.

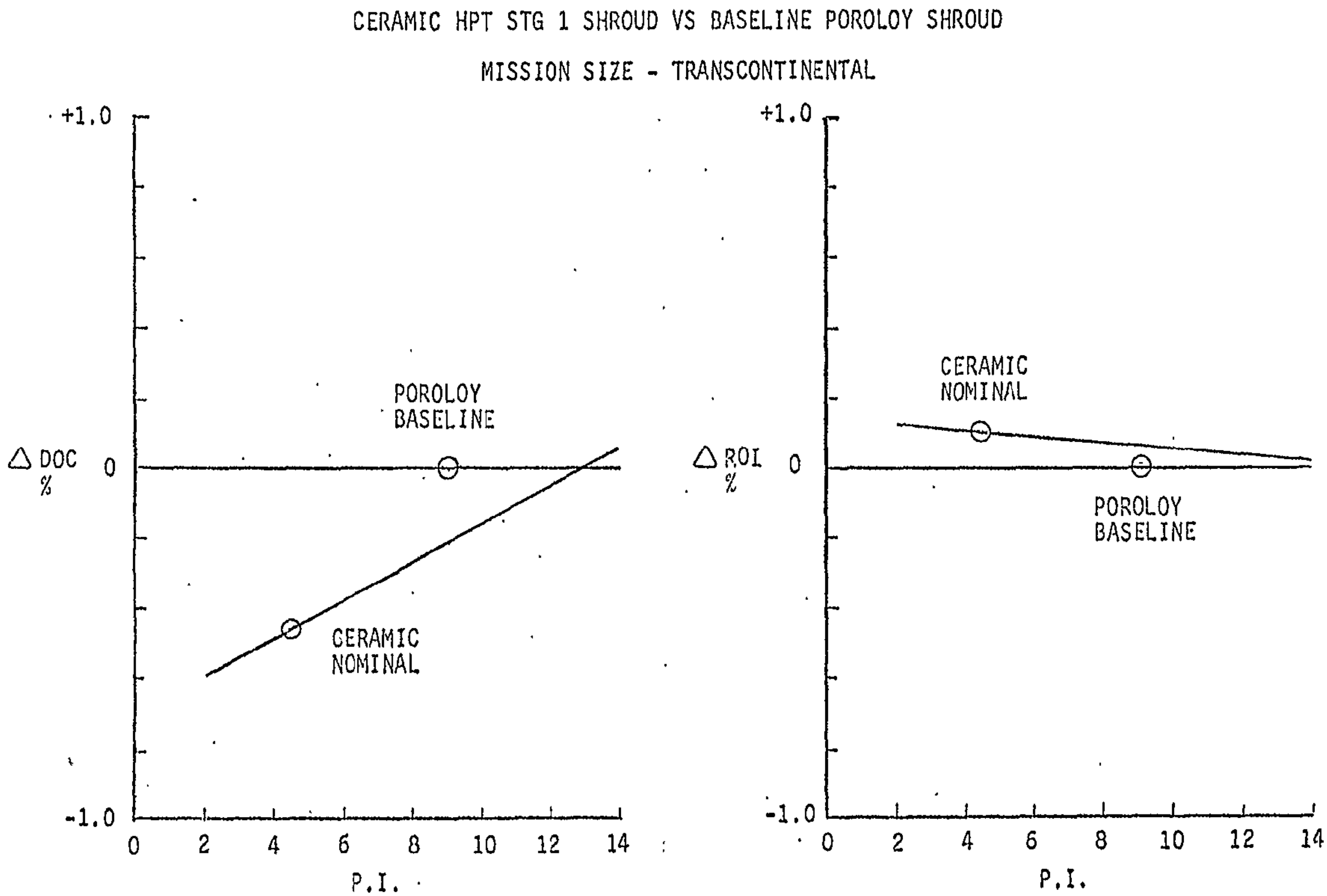

Figure 13. Sensitivity Analysis - Variation in Stg. 1 Shroud Replacement Factor (P.I.)

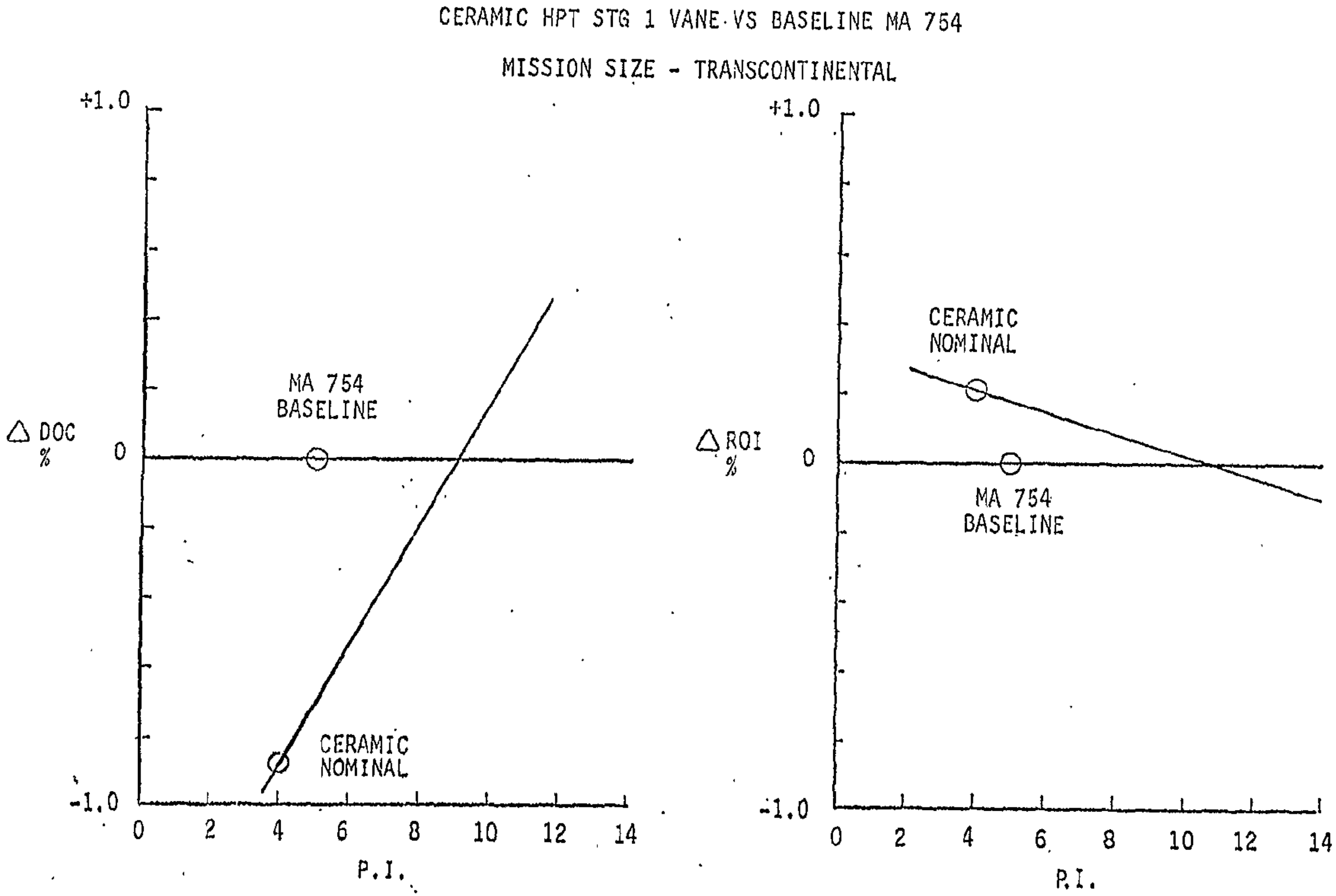

Figure 14. Sensitivity Analysis - Variation in Stg. 1 Vane Replacement Factor (P.I.)

CERAMIC HPT STG 2 VANE VS BASELINE R125

MISSION SIZE - TRANSCONTINENTAL

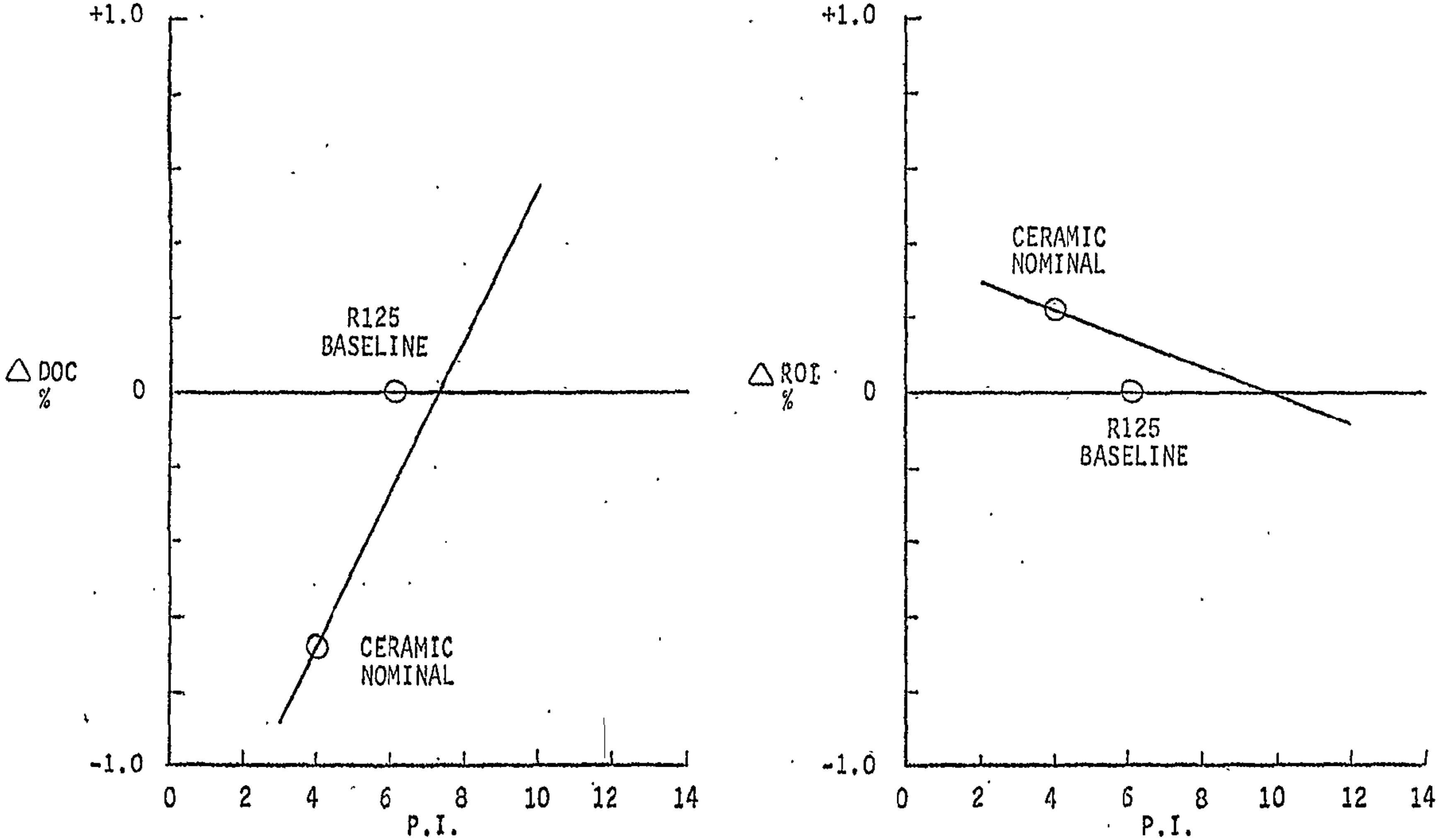

Figure 15. Sensitivity Analysis - Variation in Stg. 2 Vane Replacement Factor (P.I.)

6.3 COMBUSTOR LINERS

Figure 16 shows a small benefit for the ceramic liner application if the replacement factor assumed is actually achieved, as well as pointing out the extreme sensitivity to change in this factor. Therefore, as evaluated, without taking advantage of any second order benefits such as an improved combustor gas temperature profile, any benefit is very dependent on achieving very long life and maintenance free service.

In Figure 17, a possible second order benefit of both the No-Coat ODS alloy and the ceramic liner is explored. This is based on the assumption that smaller amounts of cooling air needed for the advanced shingle designs will result in a better radial profile thereby reducing peak gas temperatures experienced by downstream turbine components. If this were true, the nominal benefits developed in the cost benefit study could be increased by the amount shown using 7% cooling (ceramic) and 13% cooling (ODS) versus the nominal 20% cooling (HS188). This would not change the overall rating of the ODS alloy liner, but would result in the ceramic liner showing an ROI advantage, even in the Transcontinental mission.

6.4 HPT ADVANCED BLADE ALLOYS

At the outset of this study, the cost goals and initial technical goals for each of the advanced HPT blade alloys were identical which, if carried throughout the program, would have resulted in essentially identical relative values for each of the technologies. (Since the cost of development and the probability of success for each technology differ, the relative values would have differed but a thorough assessment of each material would not have been made.) As the study proceeded, therefore, and as blade designs using each material were finalized, the cost goals were refined to the best cost estimates (specific costs) based on General Electric knowledge and experience. Similarly, the technical goals were modified to reflect the optimum capabilities of the individual materials; for example, the higher temperature rupture capability of the W-FeCrAlY composite was used in the study. It was these technical capabilities and specific costs that were ultimately used to determine the Relative Values, rather than the originally established goals.

Although raw material costs for each of the advanced blade technologies were high, the labor costs (casting, machining, drilling, joining, finishing) accounted for the majority of the estimated overall finished blade cost. The alloying elements in the DS eutectic alloy made the raw material cost for this material considerably higher than that for either the W-FeCrAlY or the γ' ODS materials. However, the subsequent amount of labor involved in fabricating the finished blade was estimated to be substantially higher for the wrought γ' ODS alloy and the W-FeCrAlY composite than for the cast DS eutectic. The electro-discharge machining (EDM) of both the external shape and internal cooling passages in the wrought ODS alloy was assessed as a time consuming, and therefore expensive, process. This high labor cost resulted in the γ' ODS material being the most expensive of the three advanced blade materials. Similarly, the lay up and finishing of the W-FeCrAlY composite structure was estimated as a time consuming and expensive process based on General Electric experience with similar processing.

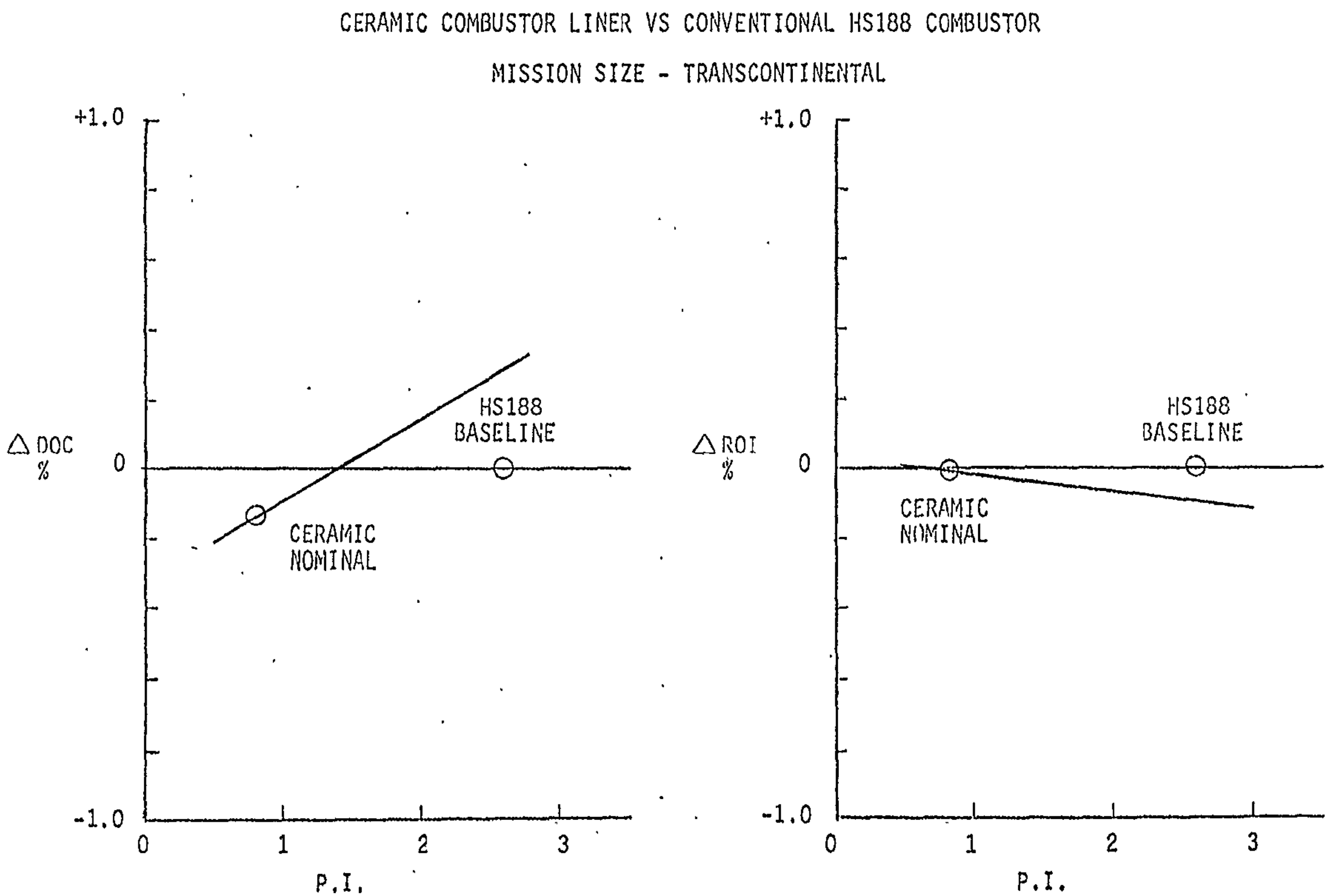

Figure 16. Sensitivity Analysis - Variation in Combustor Replacement Factor (P.I.)

REDUCED COOLING CERAMIC/NO COAT ODS LINER VS BASELINE PTF

MISSION SIZE - TRANSCONTINENTAL

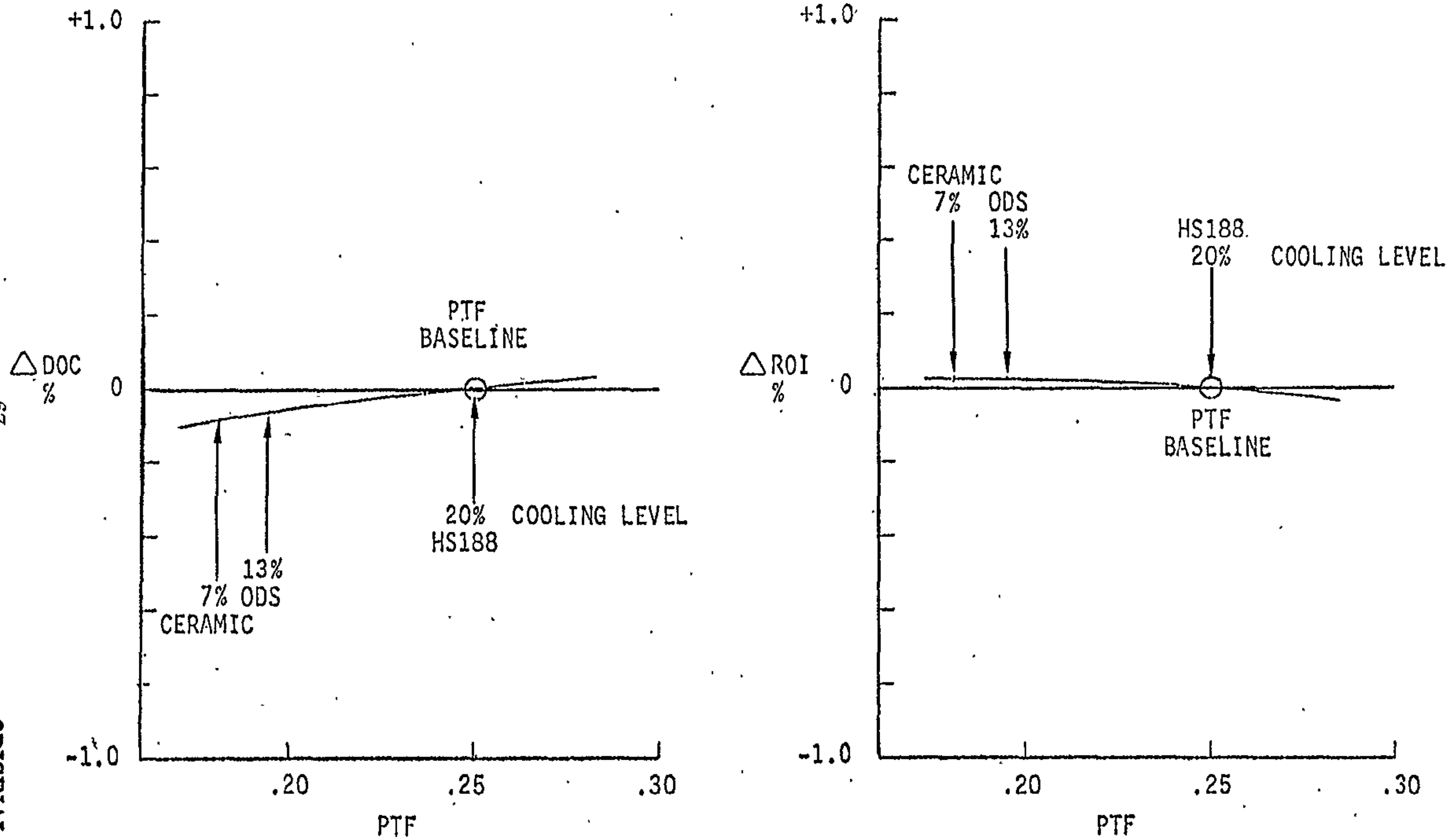

Figure 17. Sensitivity Analysis - Variation in Combustor Pattern Factor (PTF)

Obviously each of the cost estimates necessarily reflected to some extent the General Electric manufacturing experience with the technologies, and this experience is significantly greater with casting than with either of the other technologies. However, the costs do reflect the best comparative estimates by manufacturing specialists working with advanced manufacturing processes at General Electric. Furthermore, the sensitivity analysis performed for these three technologies (as with all other materials technologies in this study) allows an estimation of the effect of alternative cost figures as they became available or are better defined as a result of increased experience.

The major obstacle, therefore, to obtaining maximum economic benefits from the class of materials studied for advanced HPT blade applications is the high initial part costs. Therefore, results of the sensitivity analysis in which the finished part price of the stage 1 and 2 HPT blades in the Tungsten FeCrAlY composite material was varied are shown in Figures 18 and 19. A cost ratio based on the baseline material (R125) was used as a variable. Results for both the R125 equivalent density material ($\rho = .305$) and the heavier material ($\rho = .33$) are very close and indicate that, for the stage 1 blade, a DOC and ROI benefit would only occur when the finished part is less than about 1.8 times the cost of a comparable R125 blade. Similarly, the break-even point for the Stage 2 HPT blade in this material is approximately 1.5 times the cost of a R125 blade. The nominal cost estimates used in this program were higher than these break-even points and indicate that progress is needed in reducing costs before this material technology could be a practical alternative for a commercial engine application.

Several sensitivity parameters were investigated for the Ni82XB advanced technology material. The results are specifically for the eutectic alloy but are qualitatively applicable to all the other advanced blade alloy materials.

The effect of varying the turbine inlet temperature and the blade cooling effectiveness are shown in Figure 20. The nominal turbine inlet temperature used in the benefits study was 2600°F and this was allowed to vary up to 3000°F in the sensitivity analysis. Although the temperature limit for the lower cost no insert design for the R125 baseline material is about 2850°F, this limit did not apply to the DS eutectic with the goal properties defined in Section 3. The sensitivity study indicates that for T_4 temperatures up to at least 3000°F, the lower cost cast cooling system is more cost effective than the more complex two insert system for the Ni82XB alloy.

Figure 21 illustrates the effect of varying the replacement rate of the Stage 1 and 2 HPT Ni82XB blades. As can be seen by the close grouping of the R125 and Ni82XB replacement factors (ρ.I.), a very small upward change in this factor over the value assumed in the benefits study would eliminate the cost effectiveness.

Since the application of DS eutectic to the stage 2 HPT blade did not show any benefits in terms of a positive relative value, the turbine inlet temperature was varied to 3000°F to see if a payoff would occur at a higher T_4 than the baseline 2600°F. The sensitivity analysis results shown in Figure 22 indicate that these higher T_4 temperatures would not produce any overall ΔDOC or ΔROI benefit for Ni82XB as a stage 2 HPT blade in the commercial mission used in this program.

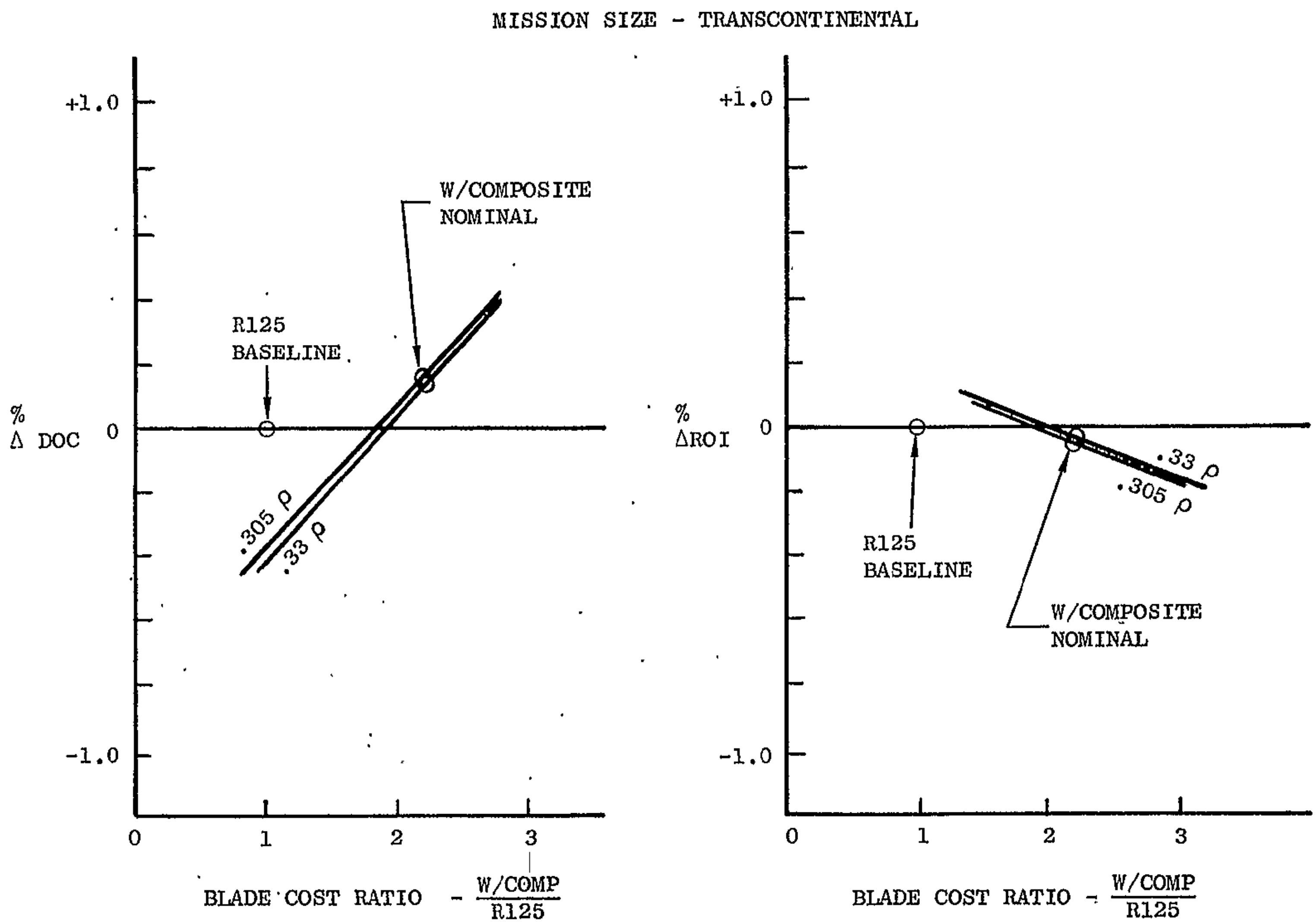

Figure 18 Sensitivity Analysis - Variation in Cost Ratio: W Composite/R125, Stg 1 Blades

MISSION SIZE - TRANSCONTINENTAL

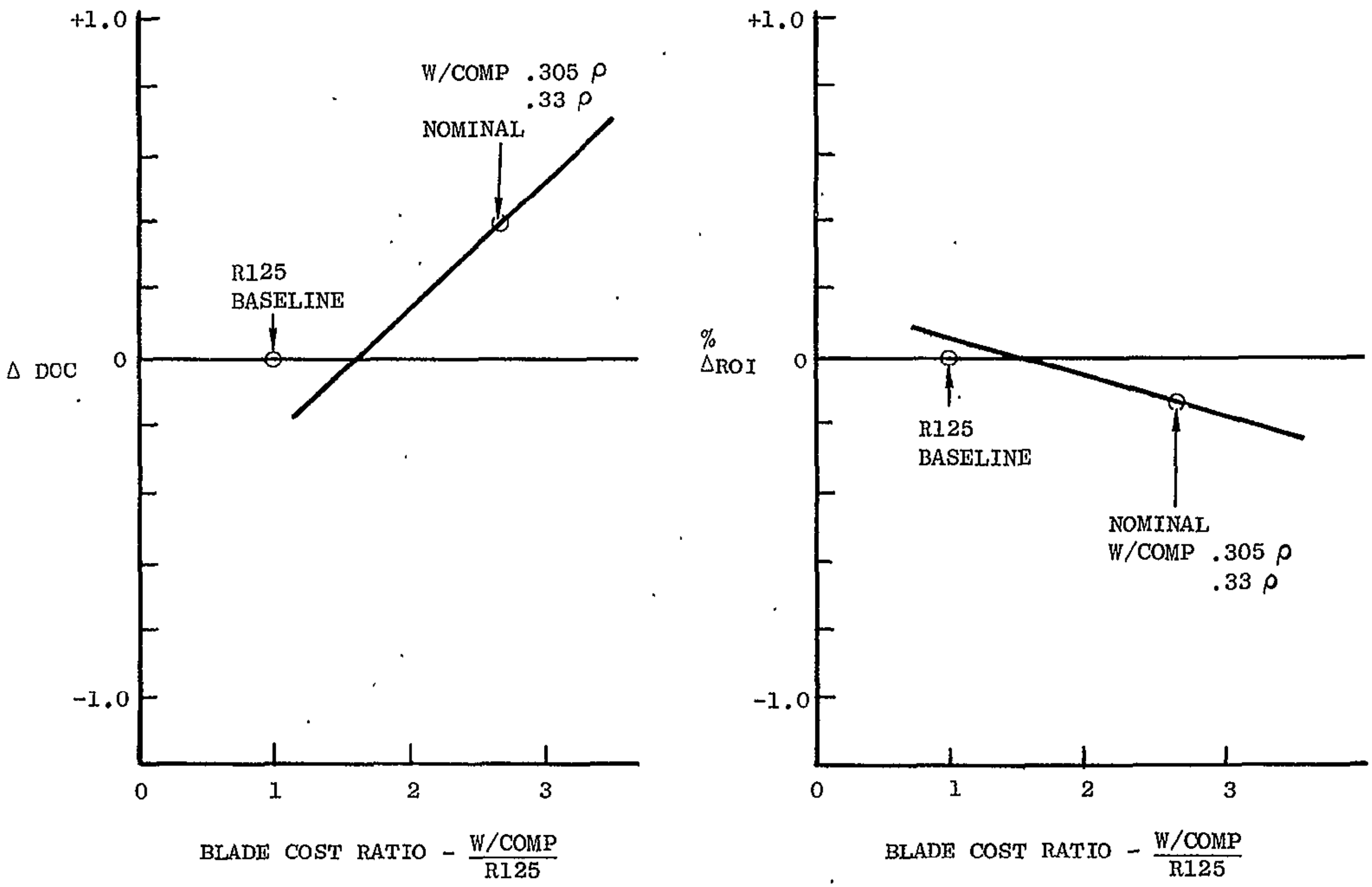

Figure 19. Sensitivity Analysis - Variation in Cost Ratio: W Composite/R125, Stg 2 Blades

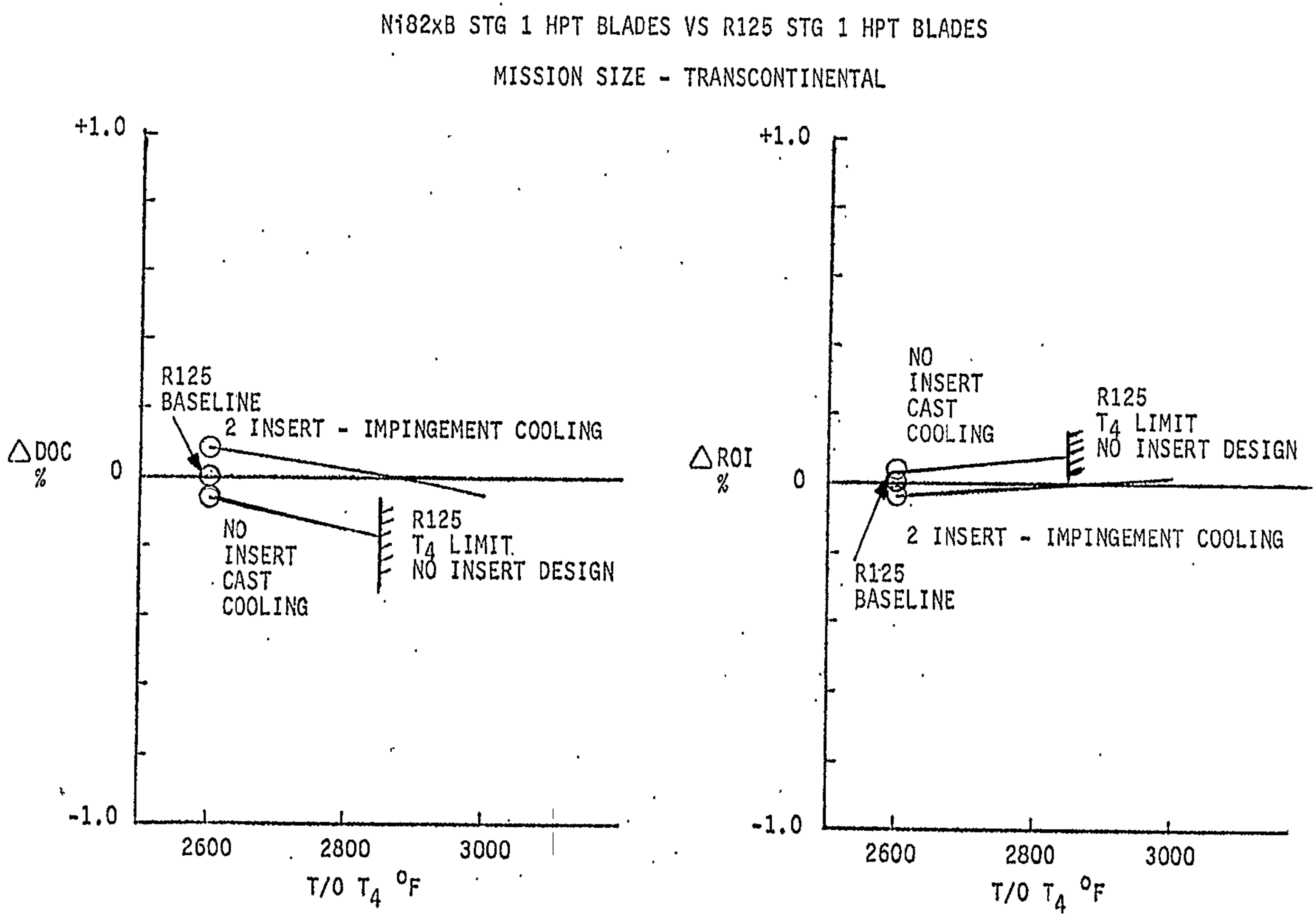

Figure 20. Sensitivity Analysis - Variation in Turbine Inlet Temperature and Cooling Effectiveness

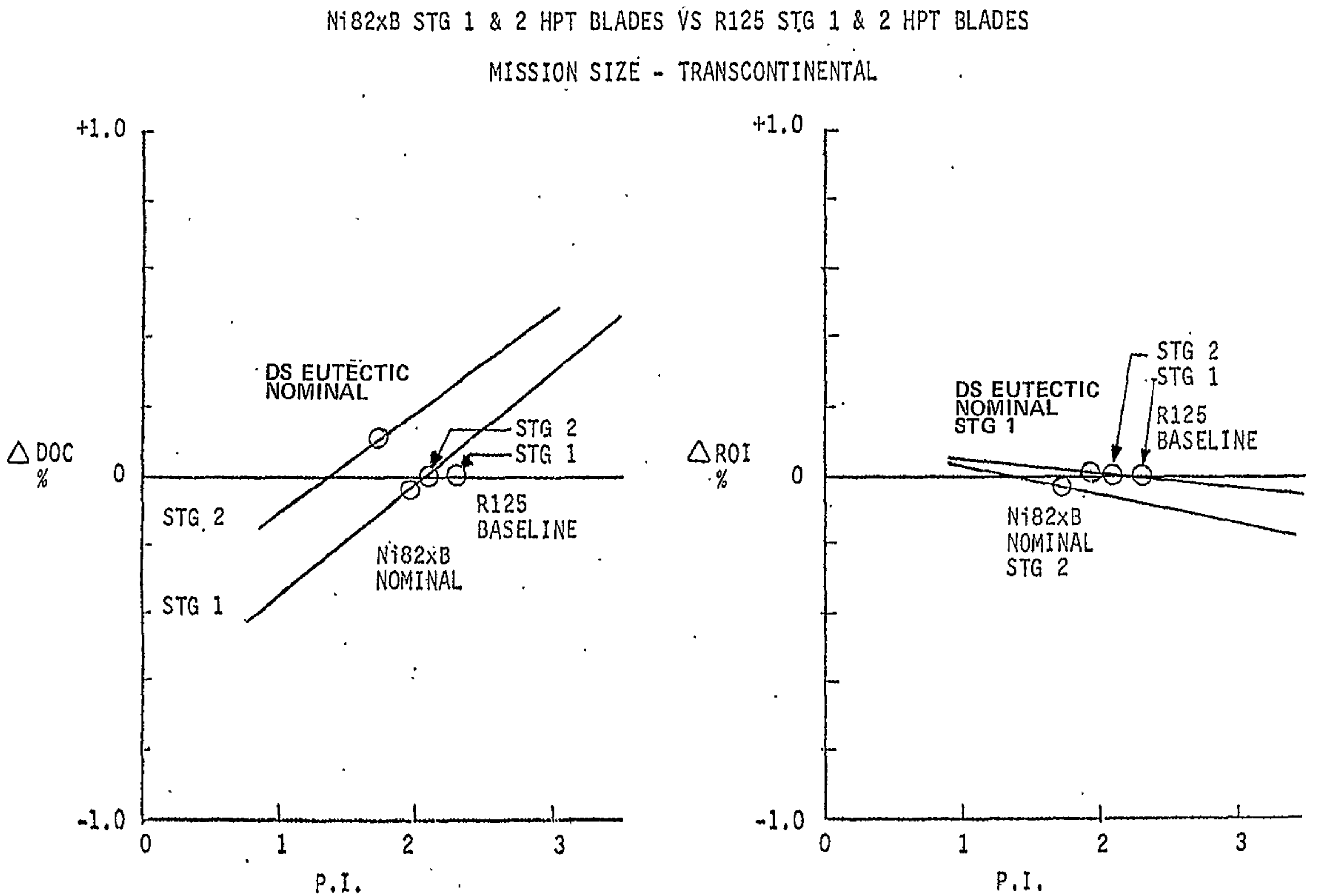

Figure 21. Sensitivity Analysis - Variation in Stg 1 and Stg 2 Blade Replacement Factor (P.I.)

Ni82xB STG 2 HPT BLADES VS R125 STG 2 HPT BLADES

MISSION SIZE - TRANSCONTINENTAL

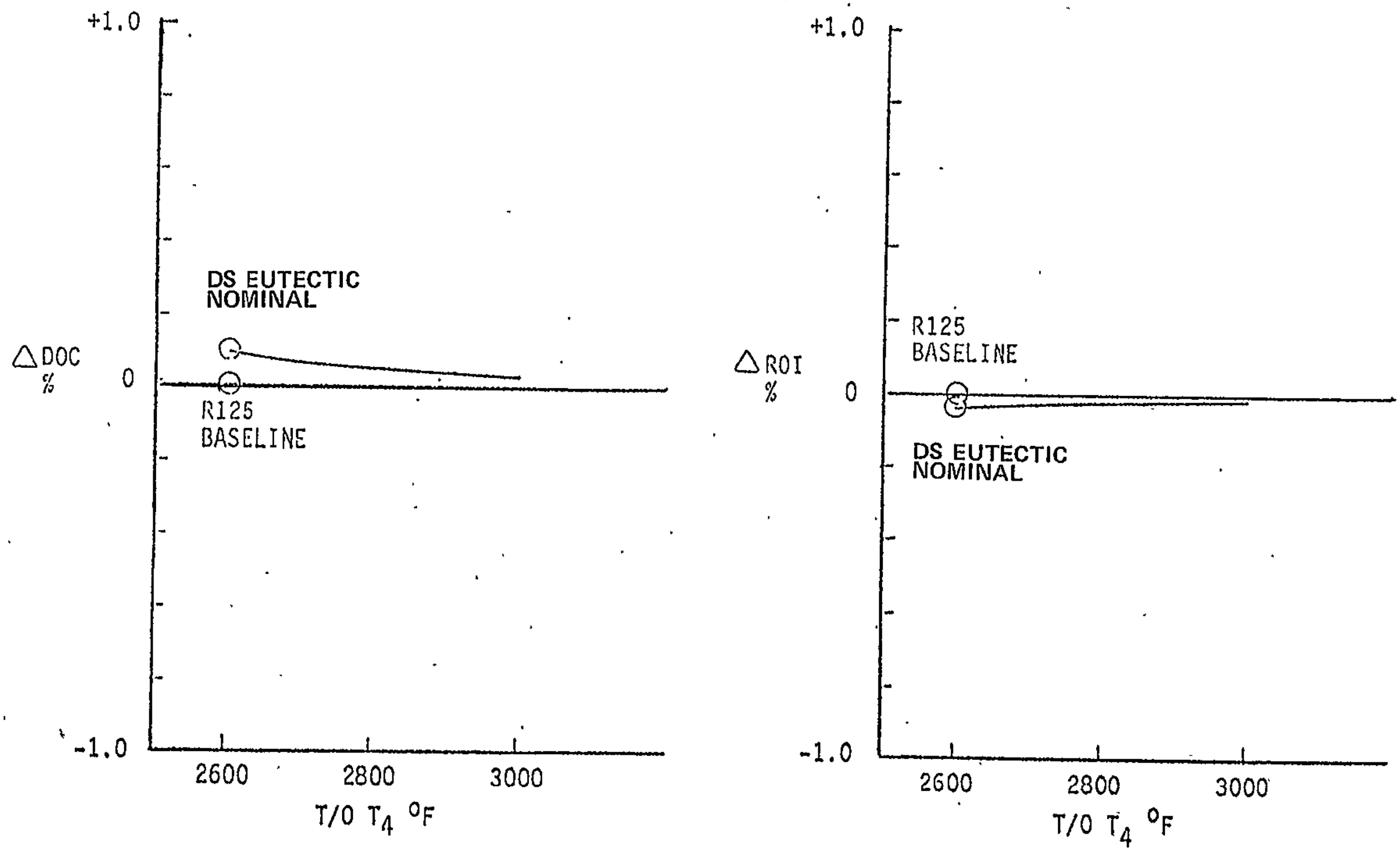

Figure 22. Sensitivity Analysis - Variation in Turbine Inlet Temperature, Stg 2 Blades

Finally, for the cast DS eutectic alloy, the effect of variations in casting cost was evaluated in the sensitivity analysis (Figure 23). If the casting cost ratio (i.e. cost relative to that of R125) is 3 or less for the stage 1 HPT blade and 1.3 or less for the stage 2 blade, the advanced alloy is cost effective. Again, these ratios are applicable only to the cycle/mission conditions as defined in this study and will vary significantly with changes in application.

In conclusion, the benefits realized through the use of the advanced technology HPT blade materials were very sensitive to changes in cost, replacement factors and cooling effectiveness and relatively insensitive to the range of T_4's that could reasonably be expected in the 1985 period. This indicates that for future commercial applications of the type modeled in this study, a successful program will require achieving or surpassing all critical property and cost goals. For other applications where SFC, weight, fuel consumption (W_f) and take off gross weight (TOGW) are of greater importance, such as in a military fighters, bombers and helicopters, these advanced turbine alloys, particularly Ni82XB, would be expected to produce significant benefits without any change in the property and cost goals used in this study.

6.5 TITANIUM ALUMINIDE LPT BLADES AND STATIC PARTS

One of the more uncertain aspects of the behavior of titanium aluminides is their reliability, which relates directly to the rate at which a given part must be replaced. The effect of varying the replacement factor (P.I.) of titanium aluminide LPT blades, nominally set at the replacement factor currently experienced in commercial service of existing blades, is shown in Figure 24. The steep slope of this figure indicates that a very small increase in replacement factor (on the order of 30% would eliminate the benefits for the titanium aluminides in this specific direct drive LPT application. Similarly, as shown in Figure 25, a relatively small increase in parts cost (~ 20%) eliminates the benefits for titanium aluminide LPT blades compared to the current nickel-base alloy blades. Clearly this is a small margin and points out the sensitivity of this technology to variations from nominal parameters and the need to achieve the goals established.

Finally, an evaluation of the effect of change in parts cost for a large static titanium aluminide component (an exhaust mixer) was performed. The sensitivity analysis results (Figure 26) indicate that an increase in cost (versus nominal) of about 50% would be required before all economic benefits would be eliminated. The benefits for the static titanium aluminide components are, therefore, not very sensitive to a modest increase in cost.

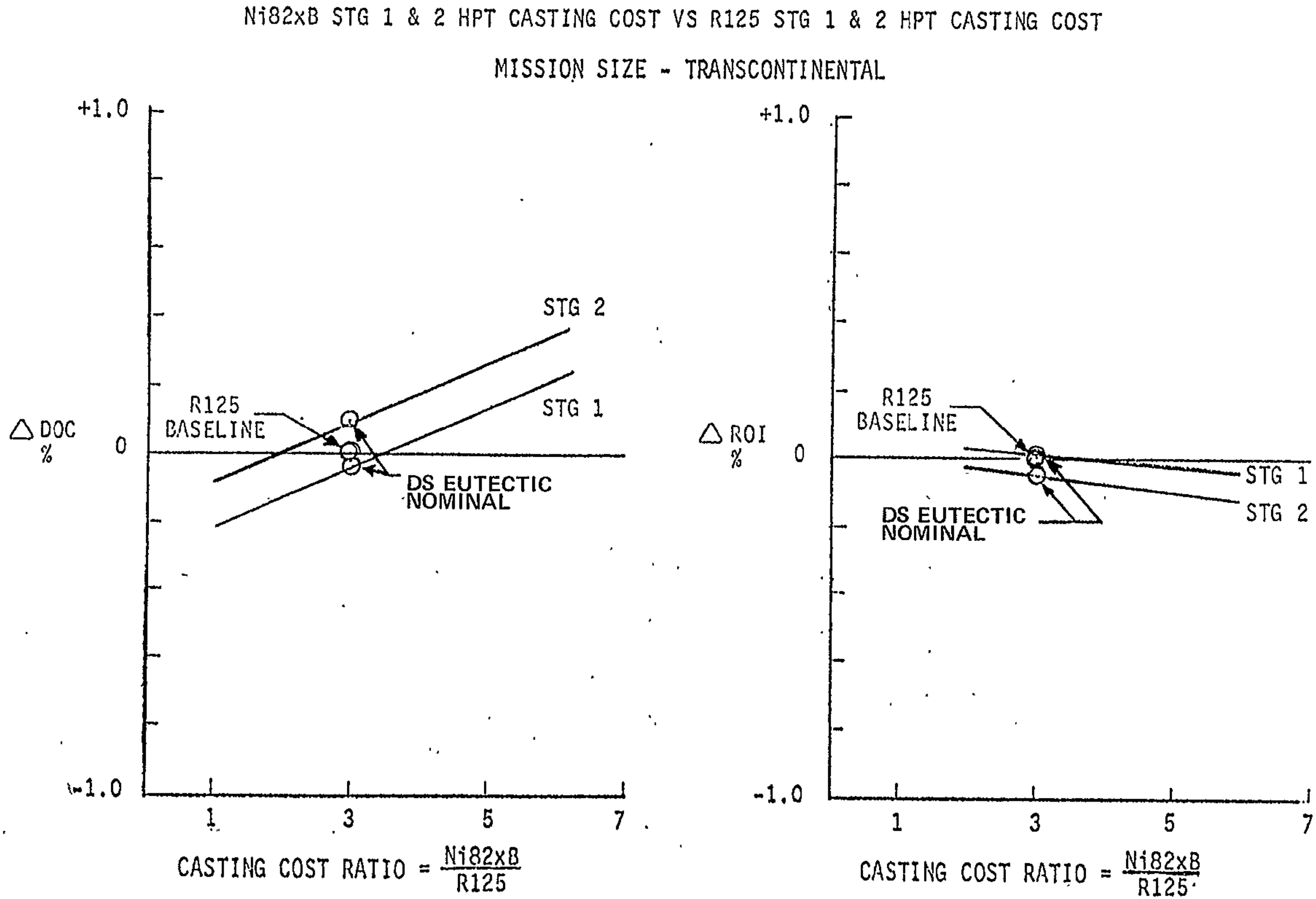

Figure 23. Sensitivity Analysis - Variation in Casting Cost, Stg 1 and Stg 2 Blades

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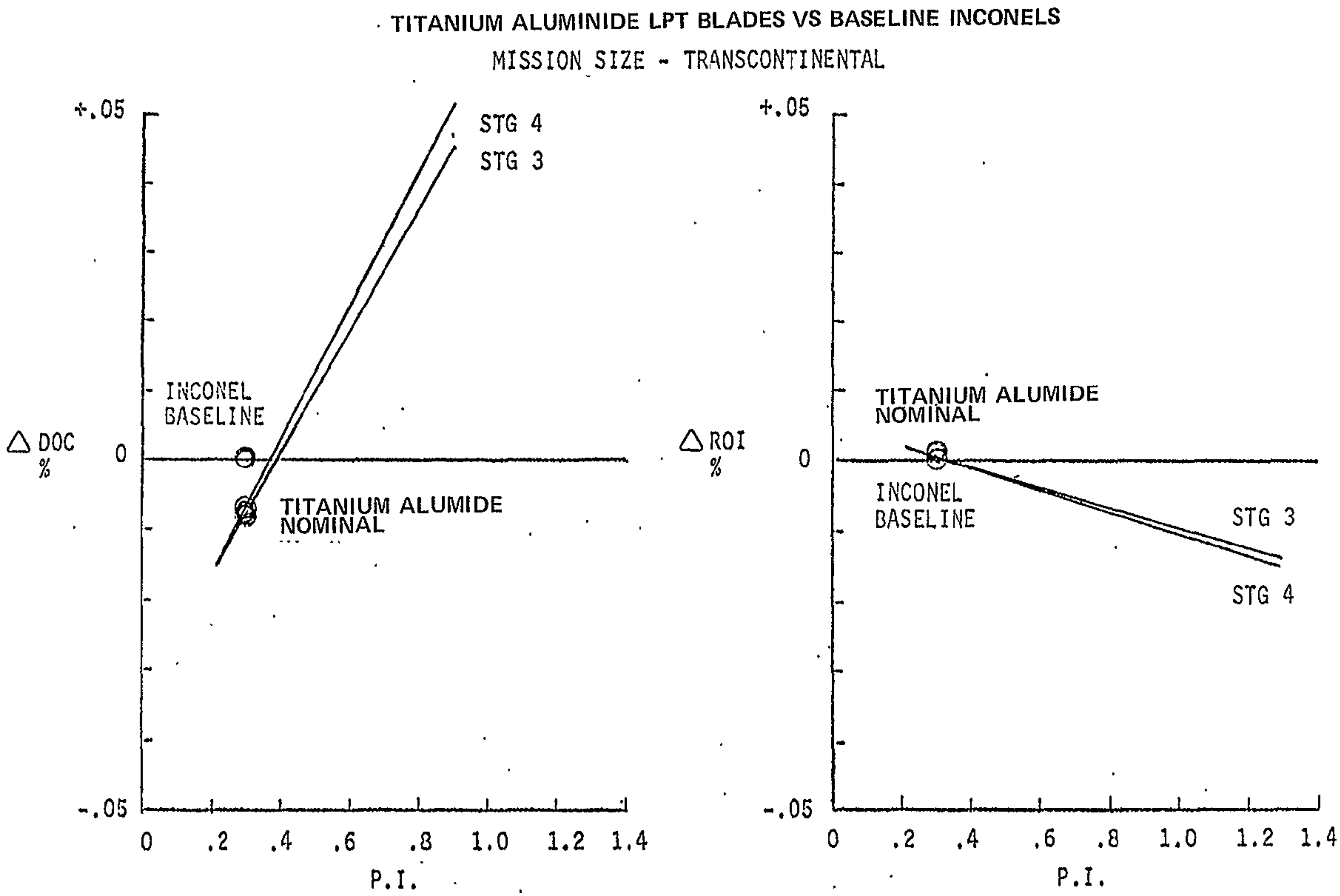

Figure 24. Sensitivity Analysis - Variation in LPT Blade Replacement Factor (P.I.)

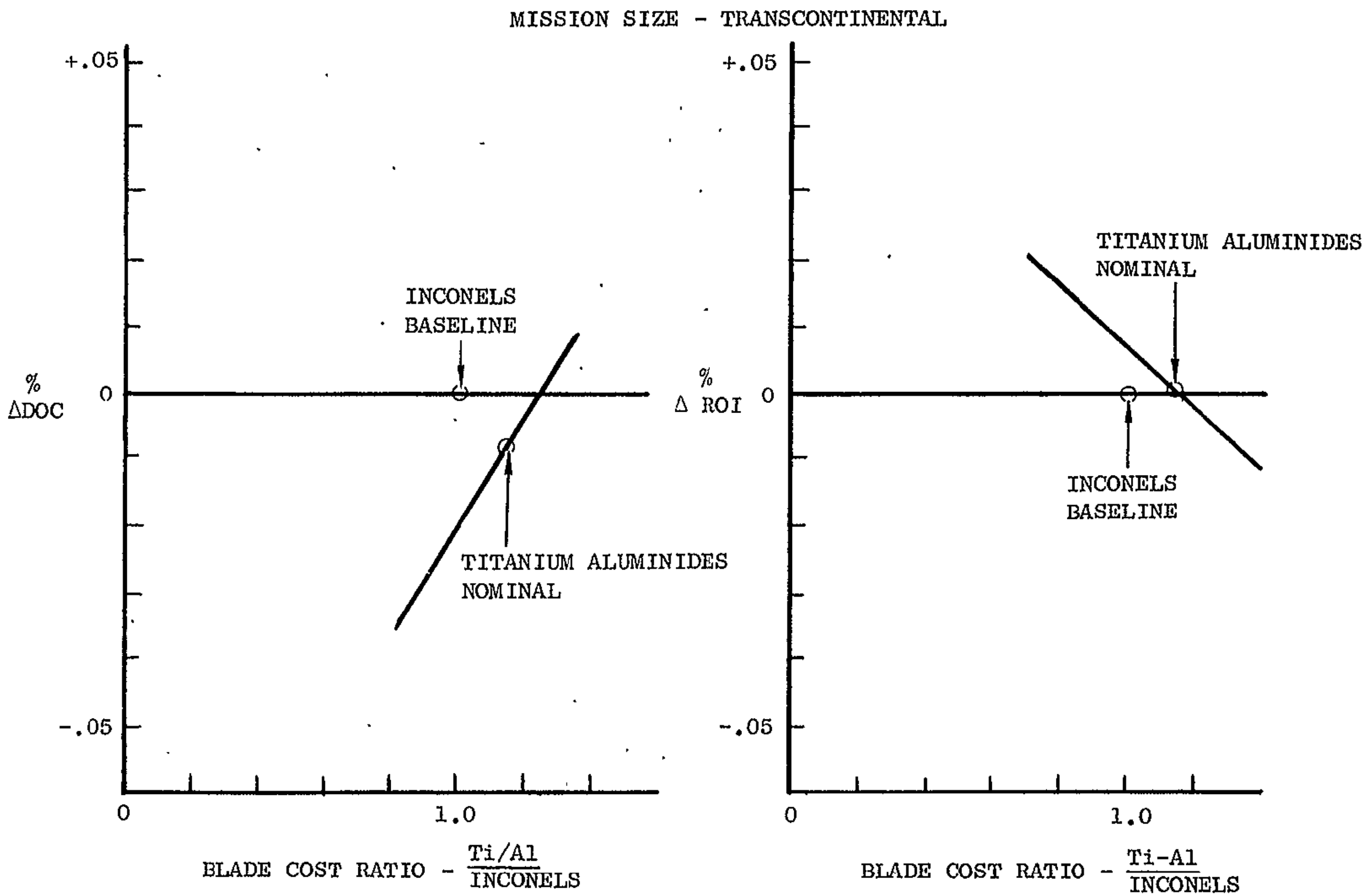

Figure 25 Sensitivity Analysis - Variation in Cost Ratio, LPT Blades

MISSION SIZE - TRANSCONTINENTAL

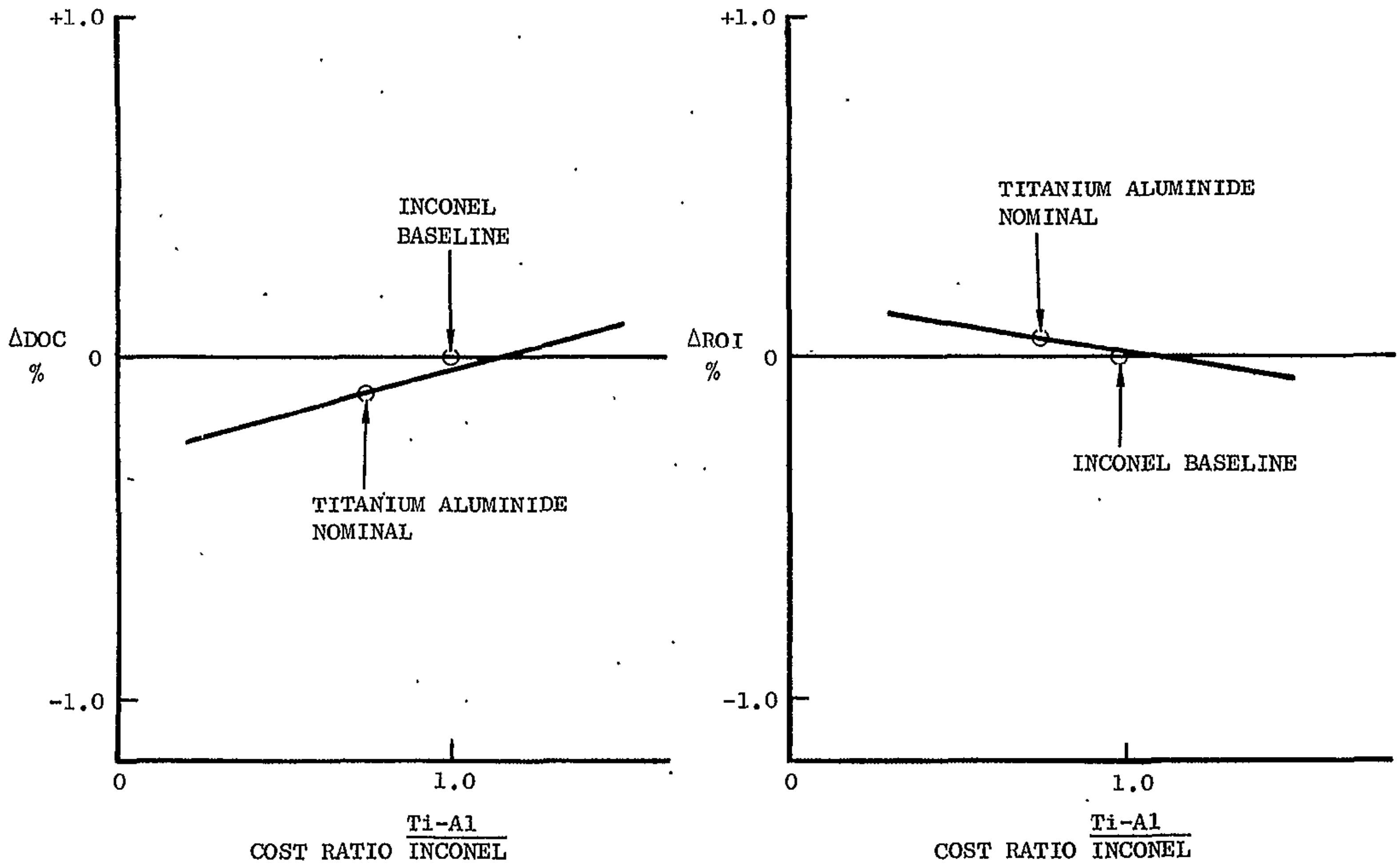

Figure 26. Sensitivity Analysis - Variation in Cost Ratio - Exhaust Mixer

7.0 RANKING OF ADVANCED MATERIALS DEVELOPMENTS

In this study, ROI was the primary criterion in judging the merit of advanced materials technologies. This in turn was used to determine the "Relative Value" parameter defined by NASA for jedging the value of applying development effort to the advanced materials. Relative value is defined by the equation:

$$\text{Relative Value} = \frac{\Delta \text{ROI}}{\text{Development Cost}} \text{ X Probability of Success}$$

The resulting values of ROI and Relative Value are summarized in Table XXV.

Based exclusively on these numbers, a Relative Value ranking was determined as a primary result of this study and has been presented graphically in Figure 10 of Section 5. This is strictly a numerically derived ranking based on the best available input and is <u>pertinent only to the particular mission(s) and engine/aircraft configurations defined in this program</u>.

A number of additional factors, however, might be considered in a more general ranking of advanced materials technologies and the emphasis would probably be significantly different to that determined in the present study. The ranking might be changed as a function of the type(s) of engine to be emphasized (military versus commercial, for example) and the size and mission requirements of the aircraft to which they are applied. In addition certain materials may have greater benefits in situations other than those considered in this study. For example, the advantage of an advanced turbine blade material will tend to be greater in the growth of an existing engine than it is in an all new engine where components can be sized as required. This explains to a great degree the relatively low ranking of advanced turbine blade materials in this particular study since an essentially all new engine was used as the model.

Based on these factors and taking into account the large range of General Electric's Aircraft Engine Group product line comprising applications to several systems (fighter, bomber, helicopter, marine, industrial as well as commercial transport aircraft), as well as an engineering judgment of the projected business needs for the 1985 time period, a "Corporate Ranking" of advanced materials technologies was derived for comparison with the Relative Value ranking defined by the study. This ranking is shown in Table XXVI, with the study ranking listed for direct comparison.

The "Corporate Ranking" basically resulted in three groups of materials technologies in terms of the emphasis deemed appropriate. The first group, comprised ceramic shrouds and Ni82XB HPT blades. The second group of four technologies (titanium aluminides, no-coat ODS alloy and ceramic combustor liners, ceramic vanes) could not realistically be separated from each other in the ranking. Finally, the W-FeCrAlY composite blade and the γ' ODS alloy blade, are rated lowest in the Corporate Ranking. The Corporate Ranking reflects the consolidated viewpoint of General Electric design and materials engineers and, although not specifically arranged to do so, clearly reflects to some degree the intensity with which each of the technologies are being pursued in present development programs.

TABLE XXV

ADVANCED TECHNOLOGY RESULTS SUMMARY TRANSCONTINENTAL RESULTS/INTERCONTINENTAL RESULTS

Advanced Technologies	Δ ROI	Relative Value x 10^{+8}
Ceramic HPT Shrouds	+.17/+.30	3.1/5.5
Ceramic HPT Vanes	+.43/+.72	2.9/4.9
Ti-Al Static Parts and LPT Blades	+.13/+.17	1.7/2.3
DS Eutectic Stg 1 HPT Blades	+.01/+.05	.1/.5
Ceramic Combustor Liner	-/+.01	-/.2

TABLE XXVI

CORPORATE VS. STUDY RANKING OF ADVANCED MATERIALS TECHNOLOGIES

Cost/Benefit Study Relative Value	General Electric Corporate Ranking
Ceramic Shrouds	Ceramic Shrouds
Ceramic Vanes	DS Eutectic HPT Blades
Titanium Aluminides	
DS Eutectic Stg 1 Blade	Titanium Aluminides
Ceramic Combustor Liner	Ceramic Combustor Liner
	No Coat ODS Alloy Combustor Liner
	Ceramic Vanes
DS Eutectic Stg 2 Blade	
W-FeCrAlY HPT Blade	W-FeCrAlY HPT Blade
γ' ODS Alloy HPT Blade	γ' ODS Alloy HPT Blade
No Coat ODS Alloy Combustor Liner	

8.0 CONCLUSIONS

Based on the cost/benefit study conducted, the advanced materials/technologies ranked as follows in order of decreasing benefits:

- Ceramic Shrouds
- Ceramic Vanes
- Titanium Aluminide Components
- DS Eutectic Stage 1 HPT Blade
- Ceramic Combustor Liner
- DS Eutectic Stage 2 HPT Blade
- W-FeCrAlY Composite Blades
- γ' ODS Alloy Blades
- No Coat ODS Alloy Combustor Liner

The "Corporate Ranking", made independent of the study, grouped the technologies as follows in order of decreasing emphasis:

- DS Eutectic HPT Blades and Ceramic Shrouds
- Titanium Aluminides, Ceramic Vanes, No Coat ODS and Ceramic Combustor Liners
- W-FeCrAlY and γ' ODS HPT Blades

The good showing of ceramics, in spite of higher risks, was due primarily to final part costs which were lower than or equivalent to those of the currently used materials, along with their significant advantage in engine performance. The higher costs anticipated for the DS Eutectic Turbine Blades compared to current turbine blades resulted in a marginal economic benefit. Reducing the anticipated costs, greatly increased the benefits of eutectic turbine blades and in all cases engine performance was greatly improved by the use of eutectic turbine blades. Thus, future efforts should be directed towards reducing the processing costs for this class of materials to achieve the better engine performance without an undue economic penalty. This effort would be even more necessary for the W-FeCrAlY and γ' ODS turbine blade alloy materials because of their intrinsically higher cost.

The weight reduction obtained through the use of titanium aluminides was attractive, although the larger cost benefit for this class of materials was in large structural components rather than in the LPT blades.

ABBREVIATIONS USED IN TEXT

AMAC	Advanced Multistage Axial Compressor
USTEDLEC	Unconventional Study of Turbofan Engines Designed for Low Energy Consumption
ATA	Air Transport Association
MATE	Materials for Advanced Turbine Engines
ROI	Return On Investment
DOC	Direct Operating Cost
TOGW	Take Off Gross Weight
W_f	Weight of Fuel Burned
Δ	Change in a Value
CTOL	Conventional Take Off and Landing
HPT	High Pressure Turbine
LPT	Low Pressure Turbine
P.I.	Parts Index (Replacement Factor)
F_n	Thrust in Pounds
CR&DC	Corporate Research and Development Center (General Electric)
ODS	Oxide Dispersion Strengthened
DS	Directionally Solidified
SLTO	Sea Level Take Off
PAX	Passengers

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