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STUDY OF TURBOFAN ENGINES DESIGNED FOR LOW ENERGY CONSUMPTION

by

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16. Abstract

A study of subsonic transport turbofan engine design and technology features which have promise of improving aircraft energy consumption is described. Task I addressed the selection and evaluation of features for the CF6 family of engines in current aircraft, and growth models of these aircraft. Task II involved cycle studies and the evaluation of technology features for advanced technology turbofans, consistent with initial service in 1985. Task III pursued the refined analysis of a specific design of an advanced technology turbofan engine selected as the result of Task II studies. In all of the above, the impact upon aircraft economics, as well as energy consumption, was evaluated. Task IV summarized recommendations for technology developments which would be necessary to achieve the improvements in energy consumption identified in this study.

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SECTION I

SUMMARY

The overall objective of this study was to identify and evaluate subsonic-transport, turbofan-engine, design and technology features for low energy consumption; thereby assisting in the guidance of future technology work directed toward improved aircraft energy consumption.

A. Current Engines

Task I of this study analyzed features for reduced energy consumption on the CF6 family of engines. A series of features for which significant technology development would be required were identified as well as more straightforward design improvements, some of which have already been included in General Electric plans for the CF6 engines.

The specific features considered were as follows:

1. Technology Dependent Features; 2 to 3% sfc Potential
   - Improved fan aerodynamics
   - Composite fan blades and frame
   - New compressor casing coating*
   - Self-acting seals in midsump
   - Advanced, directionally solidified, turbine-blade material
   - High pressure turbine (HPT) clearance control*
   - Ceramic HPT shrouds*
2. Mixed-Flow, Composite Nacelle; 3 to 3-1/2% Installed Cruise sfc Potential (Results generated under Douglas/GE contract to NASA-Langley)
3. Other Design Improvements; 1-1/2 to 2% sfc Potential
   - Improved, turbine-blade material
   - Improved HPT shroud design and material*

*Will contribute to improved performance retention.

Although the total installed sfc improvement indicated is 7 to 8%, each item must be judged on its own merits considering payoff, technical risk and cost.

- Low expansion, compressor-casing material
- Cycle trimming

The long-duct, mixed-flow, composite, nacelle design showed the largest potential for improved cruise sfc, approximately 3 to 3-1/2%. Other features totaled about 4% sfc potential, with varying degrees of difficulty involved.

The potential advantage of each feature (from a fuel-usage standpoint) was determined, and the impact upon aircraft economics estimated for DC-10 type aircraft. All the features showed potential for reducing fuel usage, but the impact upon aircraft economics varied. In many cases there was an adverse impact upon aircraft economics for fixed-payload aircraft, particularly on a retrofit basis. However, most of the features provided an improvement in aircraft economics when evaluated on a growth-aircraft basis, where the aircraft could take advantage of an engine sfc or weight improvement by means of a fuselage stretch to increase design payload.

B. Advanced Engines

Task II of this study involved the investigation of cycle parameters and design features for new turbofan engines. An initial service date of 1985 was specified in order to delineate the level of technology to be considered. Advanced technology aircraft, designed for transcontinental and intercontinental ranges with a cruise Mach No. of 0.80, were used for this evaluation. As a result of the study of cycle variations, the parameters listed below were selected for the preliminary design of a specific engine in Task III. The selection was made assuming advanced engine technology, compatible with the year the engine is scheduled to enter service, and involved a balance of energy consumption, aircraft economics, and growth potential.

• Turbine inlet temperature	1427° C (2600° F) at takeoff/ 1327° C (2420° F) at max. cruise
• Cycle pressure ratio	38:1 at altitude design point
• Fan pressure ratio	1.7 at altitude design point
• Bypass ratio	7
• Exhaust	Mixed

Advanced technology and design features evaluated from an energy consumption and aircraft economics standpoint in Task II are summarized below:

- Component aerodynamics improvement for compact engine design
- Composite fan blades and frame
- Clearance control; core compressor and turbine
- Ceramics for hot, static, flowpath parts
- Advanced turbine-blade materials and cooling
- Long-duct, mixed exhaust
- Integrated composite nacelle
- Low noise features

Each of the items listed offered an advantage in energy consumption and, with the exception of very advanced turbine-blade materials and cooling, aircraft economics. The features with payoff were then incorporated in the Task III design, while some of the more speculative items (such as ceramic turbine vanes) were reserved for later growth of the engine.

Task III of this study involved the refined analysis, or preliminary design, of the advanced engine selected. In order to illustrate the magnitude of improvement achievable with a new engine incorporating advanced technology, comparisons were made with the CF6-50C engine; believed to be a good representative of a current high bypass engine in terms of technology and performance. An improvement in installed sfc (including nacelle drag) of just over 10% was estimated in this study. It must be emphasized that this included the effect of advanced technology in terms of component performance, cooling, and materials technology. A reduction in installed weight of 12% below tht of the CF6-50C, scaled to the same take-off thrust, was indicated. Since the advanced-engine ratings were set to provide relatively higher cruise thrust than the CF6-50C, the weight reduction was 24% when compared at the same cruise thrust. It was also estimated that the advanced engine, plus nacelle, would have a production cost comparable to a scaled CF6-50C at the same point in the production run. The design selected involved a relatively small number of parts, which should contribute to low maintenance costs. The cycle and design parameters were selected so that 20 to 25% growth could be obtained in later versions of the engine.

The effect of the above installed-engine improvements was estimated for the advanced reference aircraft utilized in the study. Approximately 13% improvement in fuel usage, and 6% improvement in DOC (Direct Operating Cost) were obtained for the transcontinental trijet. The corresponding numbers for the intercontinental quadjet were 17% and 10%, respectively.

The advanced Task III engine was projected to meet the noise goal of FAR Part 36 (1969) minus 10 EPNdB for the reference aircraft (defined in Task II) utilizing the advanced, noise-reduction technology identified in this study. The proposed 1981 EPA (Environmental Protection Agency) emissions requirements will require advanced combustor-technology features, but no acceptable approach to meeting the Nitrogen Oxides (NO_x) requirement has yet been identified.

Task IV of the study addressed the technology required to achieve the improvements in energy consumption identified. In order to achieve these improvements in energy consumption, and improvements in aircraft economics, technology advancement in all areas of the propulsion system is required.

SECTION II

INTRODUCTION

NASA initiated studies of advanced, subsonic-transport-system technologies in 1970 as part of the Advanced Transport Technologies program (ATT). References 1 and 2 report the results of studies carried out by General Electric under contract to NASA-Lewis. In these studies, the emphasis was placed on cruise at high subsonic speeds, to take advantage of supercritical aircraft-wing technology, and on lower noise. An advanced technology engine from the ATT studies was used in one portion of the Douglas/General Electric acoustic composite nacelle study reported in Reference 3.

After completion of the ATT contract effort, General Electric continued in-house studies with the emphasis placed on engine technology to improve aircraft economics thru improved installed sfc, weight, and cost. Engines under study at that time were used as the basis for two NASA studies directed at evaluating the benefits of composites and advanced materials (References 4 and 5). The benefits analysis approach utilized in Task II of the study covered in this report was an extension of that used in Reference 5.

As a result of the recently escalated concern for diminishing petroleum-based energy supplies, NASA sponsored the study reported herein. The title initials of the contract, "Study of Turbofan Engines Designed for Low Energy Consumption," were employed to produce the acronym "STEDLEC" referred to in various portions of this report for identification.

The purpose of the study was to identify and evaluate subsonic-transport, turbofan design and technology features for reduced energy consumption. The study consisted of the following tasks:

- Task I - Low energy consumption features for the CF6 family of high bypass turbofan engines.
- Task II - Low energy consumption cycle and technology features for a new engine with 1985 introduction into service.
- Task III - Refined analysis for a selected 1985 turbofan design.
- Task IV - Technology recommendations.

The advanced engines, involved in Tasks II and III, were evolutionary from the original ATT studies and follow-on General Electric in-house studies. The emphasis was placed upon reduced energy consumption, and the cruise Mach number set at 0.8 for a balance between energy consumption and acceptability to the airlines. General Electric also continued to place emphasis on achieving a substantial improvement in aircraft economics, since it was felt this would be mandatory to justify development of a new engine. The technology level for the advanced engine was established, in accordance with the contract Statement-of-Work, to be consistent with introduction into service in 1985. The study was also structured to place about 25% of the contract effort on the identification and evaluation of features for improved energy consumption applicable to the CF6 engine family (Task I).

SECTION III

TASK I - CURRENT ENGINES

A. Approach

The CF6 family of engines was selected for Task I because it is the General Electric commercial-transport engine family now in service and production. Various versions of the engine are expected to remain in production for many years, thereby providing the opportunity to incorporate features for reduced energy consumption.

The opportunities for reduced energy consumption of CF6-6 and CF6-50 engines were surveyed first. A selection of the most promising features was made and categorized as to whether technology-dependent, or more straightforward design changes. The effects of the specific design features on engine characteristics (including sfc, weight, first cost and replacement costs) were then estimated. Suitability for retrofit in existing engines was also assessed.

An evaluation procedure was selected for the CF6-6 engine in a DC-10-10 type airplane, and the CF6-50 engine in a DC-10-30 type airplane. The effects of the various design features were then evaluated in terms of fuel usage and aircraft economics on a new engine basis, with and without aircraft redesign to increase payload, and on a retrofit basis.

B. CF6 Engine Description

The major cycle and design characteristics of the two CF6 engine models are summarized in Table I. A cross section of the CF6-6 engine is shown in Figure 1, and the CF6-50 engine in Figure 2. The engines utilize the same fan. The higher thrust CF6-50 engine was evolved from the CF6-6 engine by adding two booster stages to increase the core engine flow. In addition, two stages were removed at the rear of the compressor, and the flow path in the combustor and high pressure turbine was modified for the higher volume flow. One less fan turbine stage was required because of the lower bypass ratio of the CF6-50.

C. Features for Improved Energy Consumption

The features selected for evaluation in this study are listed in Tables II and III. The applicability to the two CF6 models, and the suitability for retrofit, are shown on the right side of each table. These features were divided into those requiring technology development (Table II) and into the more straightforward design improvements (Table III). Table IV presents a list of features considered, but not pursued. This does not necessarily mean that the features in Table IV were not practical, but only that it was necessary to limit the study to those with the greatest promise for reduced energy consumption.

Table I. CF6 Engine Description.

	CF6-6D	CF6-50C*
Thrust, N (lb)	178,000 (40,000)	227,000 (51,000)
Fan Diameter - m (in.)	2.195 (86.4)	2.195 (86.4)
Bypass Ratio	5.8	4.2
Fan Pressure Ratio - Cruise	1.67	1.71
Overall Pressure Ratio - Cruise	28	31
Turbine Temperature - Takeoff Hot Day	1316° C (2400° F) Class	1316° C (2400° F) Class
Staging - Fan	1	1
- Boosters	1	3
- HP Compressor	16	14
- HP Turbine	2	2
- LP Turbine	5	4
Installation	Separate Flow Short Duct	Separate Flow Short Duct
Reversers	2	2

*Initially rated at 218,000 N (49,000 lb). Growth model quoted at 240,000 N (54,000 lb)

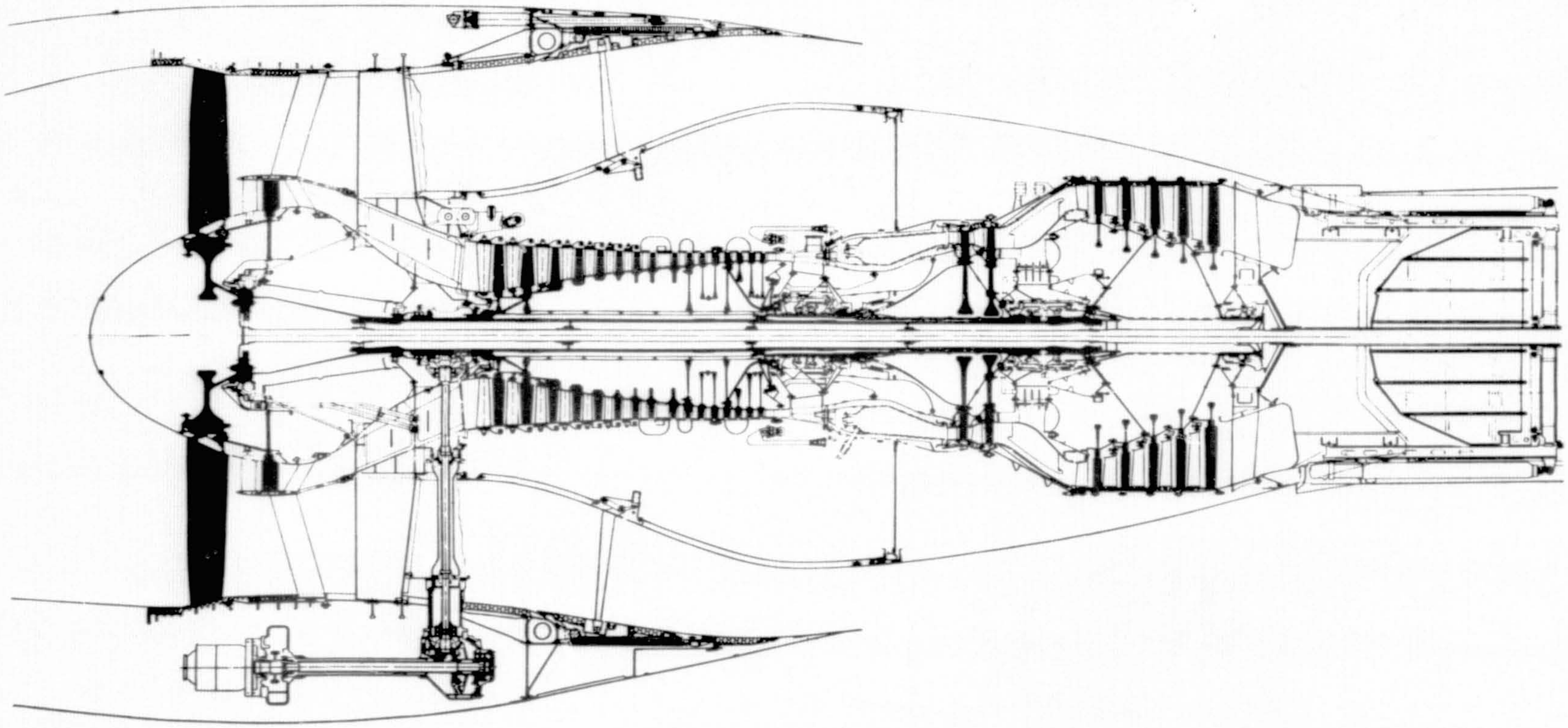

Figure 1. The General Electric CF6-6 Engine Cross Section.

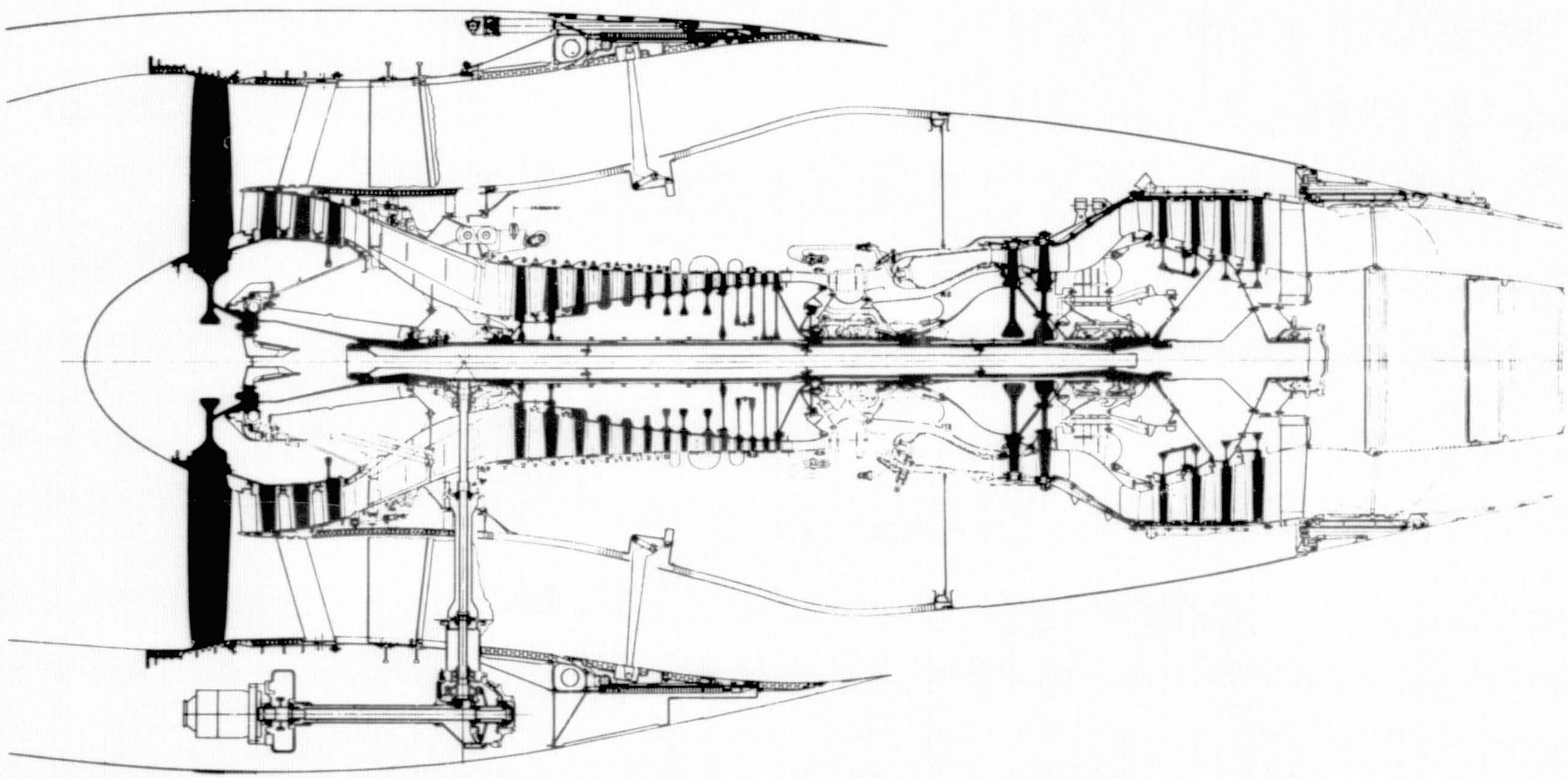

Figure 2. The General Electric CF6-50 Engine Cross Section.

Table II. Features Selected Requiring Technology Development.

Component	Change	Area of Improvement	Applicable to CF6-6	CF6-50	Suitable for Retrofit
Fan Blades	Modified Aerodynamics	Efficiency	X	X	X
Fan Blades	Composites	Weight, Cost, and Safety	X	X	No
Fan Frame	Composites	Weight and Cost	X	X	No
Compressor Casing	New Coating	Clearances, Deterioration	X	X	X
B Sump and CDP Seals	Redesign for Hydrodynamic Seals	Leakage	X	X	No
HPT Blades	Ni76XB Material	Cooling	X	X	Later
HPT Shrouds	NiCrAly Material	Clearances, Deterioration	X	X	X
HPT Casing	Redesign for Clearance Control	Clearances, Deterioration	X	X	No
Nacelle	Long Duct Mixed Flow with Composites	Installed Cruise, sfc, Noise	X	X	?

Table III. Features Selected Requiring Design Improvements.

Component	Change	Area of Improvement	Applicable to		Suitable for Retrofit
			CF6-6	CF6-50	
HPT Blades	R125 Material	Cooling	X	X	X
	Cooling Redesign	Cooling		X	X
HPT Stage 1 Shrouds	Film Cooling	Cooling	X	X	X
	Saw Cut Segments	Clearances	X	X	X
LPT Design	Stage 1 Blade - Incidence Angle	Efficiency		X	X
	Bolt Covers - Interstage Seal Supports	Windage	X	X	X
	Redesign Seal Under Stage 1 Vane	Efficiency		X	?
	Fill Honeycomb Over Tip Shrouds	Efficiency	X	X	X
	Add Seal Tooth to Tip Shrouds	Efficiency	X	X	X
	Blacken Cowl Above LPT	Clearances	X		X
Core Jet Nozzle	Increase Area	sfc at Cruise		X	?
Variable Stator Schedule	Close Stators at Cruise	sfc at Cruise	X		?
Rear Compressor Casing	Low Expansion Material	Clearances	?	X	-50 only

Table IV. Features Considered But Not Pursued.

Component	Change	Possible Area of Improvement	Limiting Factors
Compressor Casing	Heating/Cooling Methods	Clearances	Practical Design Not Identified
Compressor Rotor	R95 Material	Weight	Small Reduction
Combustor	New Swirler Dome	Idle Efficiency	Emissions Dominate Small Savings
Combustor and Control	Lower Quality Fuel Capability		Impact Upon Engine Life and Reliability No Net Saving in Energy
HPT Vanes	MA754 Material	Cooling Life	Cost of Advanced Materials (May Be Needed For Growth)
Turbine Blades	Redesign Blades for Nonconstant Work	Efficiency	Advantage Not Clear For Lightly Loaded Design
Fan Jet Nozzle	Two-Position	Cruise sfc	Major Redesign For Small Gain
Control and Fuel System	Lower Flight Idle Power	Descent Fuel	Questionable Improvement
Parasitic Flow System	Add Shut-Off Valves	Idle sfc	Complicated - Small Saving
Diagnostics	Add System	Identify Deterioration	Improvement Not Identified

D. Evaluation Procedure

The evaluation procedure involved General Electric calculations of the effects of engine changes on DC-10 type aircraft fuel usage and economics. The reference aircraft characteristics are presented in Table V and breakdowns of the DOC's are illustrated in Figure 3, along with the significant assumptions.

Three methods of evaluation were used for; incorporating improvement features, new engines for fixed and growth aircraft, and retrofit into existing engines while the engines are the shop (Table V). Mission trade factors for changes in engine parameters were calculated and the results are tabulated in Table VI, for the DC-10-10, and in Table VII for the DC-10-30. Note the large difference in the trade factors between fixed- and variable-payload aircraft, especially for engine weight changes. The variable-payload factors shown are quoted on a per-seat basis and therefore offer a significantly greater reduction in fuel consumption, or DOC, because of the increased number of seats.

The procedure consisted of first determining the effects of incorporating a given design feature into the engine, then taking the following factors into account.

1) sfc: The direct effect of a component performance improvement was determined at constant thrust (exception was mixed flow). For new engines the secondary effect of reduced cooling, allowed by the lower turbine temperature resulting from component performance improvement, was taken into amount.

2) Weight: Estimated directly.

3) Engine Price Change: For new engines, estimates of manufacturing cost change and the nonrecurring costs of the design change were reflected in an engine price change, using typical pricing methods. Each model of the engine was treated separately for each evaluation approach. For the retrofit cases, an equivalent price change was determined, which reflected the price of a new part relative to the estimated value of the scrapped part (again including nonrecurring costs of the design change).

4) Maintenance Costs: Account was taken for the change in parts price and the estimated replacement rates for the part in question. For the retrofit cases, an estimate of the improvement in maintenance costs associated with the lower turbine temperatures, related to component performance improvement (constant thrust), was included.

Table V. Evaluation Procedure.

Aircraft and Missions	Engine	CF6-6	CF6-50
	Aircraft	DC-10-10	DC-10-30
	Design TOGW	195,000 kg (430,000 lb)	252,000 kg (555,000 lb)
	Design Range	5560 km (3000 nmi)	10,190 km (5500 nmi)
	Fuel Cost	\$71/m^3 (\$.27/gal nmi)	\$106.7/m^3 (\$.40/gal)
	Price Level	1974	1974
	No. of PAX	270	270

Alternate Means of Evaluation

- Aircraft in Service - Retrofit when Engines in the Shop
- New Aircraft - Fixed Payload
- Growth Aircraft - Variable Payload by Fuselage Stretch - Includes Associated Penalties for Longer Fuselage
  - Improvement Quoted on Per-Seat Basis
  - Constant Design Range

Trade Factors Used To Determine Effects of Engine Changes on Fuel Usage and Aircraft Economics

Direct Operating Cost Breakdown (DOC)

DC-10-10 (Domestic)
Range = 1,300 km (700 nmi)
Load Factor = 55%
DOC = 2.33 \$/km (4.31 \$/nmi)
DOC = 0.77 ¢/Seat/km (1.43 ¢/Seat/nmi)
Fuel at 71.0 \$/m^3 (\$0.27/gallon)

DC-10-30 (International)
Range = 3,700 km (2,000 nmi)
Load Factor = 55%
DOC = 2.65 \$/km (4.91 \$/nmi)
DOC = 0.92 ¢/Seat/km (1.70 ¢/Seat/nmi)
Fuel at 106.0 \$/m^3 (\$0.40/gallon)

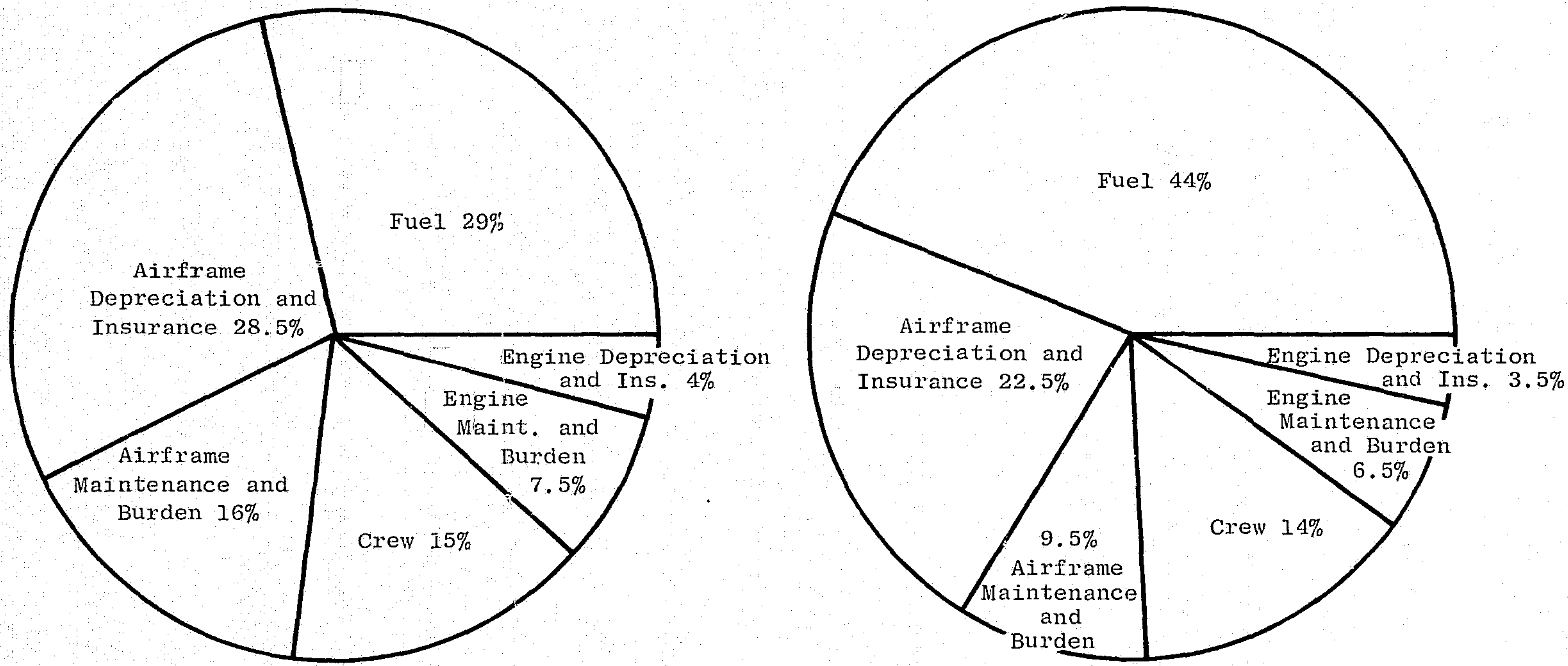

Figure 3. Direct Operating Cost Breakdown, DC-10.

Table VI. Mission Trade Factors, DC-10-10.

DC-10-10 – 1300 km (700 nmi) – 55% Load Factor						
	Effect on Fuel Usage		Effect on DOC		Effect on ROI*	
Engine Change	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload
+1% sfc	+1.1%	+2.0%	+0.30%	+1.2%	-0.047%	-0.35%
+45.4 kg (+ 100 lb) Weight/Engine (~0.8% Installed Engine Weight)	+0.04%	+0.25%	+0.01%	+0.22%	-0.002%	-0.07%
+ $10,000 Initial Price/Engine	-	-	+0.05%	+0.05%	-0.032%	-0.032%
+ $10,000 Replacement Parts Price/Engine (During Life of Engine)	-	-	+0.05%	+0.05%	-0.008%	-0.008%

Fixed Payload – Improvements in Range Approximately Same As Improvements in Fuel Usage

Variable Payload – One Row of Seats Requires 2.8% Improvement in sfc

*% ROI is Difference Between ROI's (11% ROI - 10% ROI = ΔROI of 1%)

Table VII. Mission Trade Factors, DC-10-30.

DC-10-30 – 3700 km (2000 nmi) – 55% Load Factor						
	Effect on Fuel Usage		Effect on DOC		Effect on ROI	
Engine Change	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload
+1% sfc	+1.1%	+2.2%	+0.48%	+1.6%	-0.10%	-0.57%
+45.4 kg (+ 100 lb) Weight/Engine	+0.05%	+0.24%	+0.02%	+0.22%	-0.005%	-0.09%
+ $10,000 Initial Price/Engine	-	-	+0.035%	+0.035%	-0.024%	-0.024%
+ $10,000 Replacement Parts Price/Engine	-	-	+0.035%	+0.035%	-0.008%	-0.008%

Fixed Payload – Improvements in Range Approximately Same as Improvements in Fuel Usage

Variable Payload – One Row of Seats Requires 2.2% Improvement in sfc

The effects of a given design feature on aircraft fuel usage, DOC, and ROI (Return on Investment) were then determined by applying the trade factors, listed in Tables VI and VII, to the estimated changes in the four engine characteristics described above.

E. Effects of Design Features on Engine and Aircraft Characteristics

The results of evaluation of selected engine design features are summarized in Table VIII thru XII. The estimated effects upon engine sfc, aircraft fuel usage, and ROI are shown for each feature. Note that minus is good for changes in sfc, fuel usage, and DOC, while plus is good for ROI. Table VIII lists the estimates for those features which are of a type that could be considered for retrofit in existing CF6-6 or CF6-50 engines. Table IX presents the effects of the more straightforward design improvements on new CF6-6 engines, and Table X lists the effects on new CF6-50 engines. The results are shown for both fixed and growth (variable-payload) aircraft. Table XI lists the effects of technology-dependent features on new CF6-6 engines, and Table XII lists the effects on new CF6-50 engines. Again, both fixed and growth aircraft were considered.

F. Discussion of Results

1. Retrofit Features

The items deemed suitable for retrofit included both design-improvement items and technology-dependent items which are described below. The results for the retrofit cases, listed in Table VIII, differ from those for all-new engines because the evaluation procedure was selected as being appropriate for the retrofit situation. In general, the magnitude of the retrofit improvements, for a given design change, were less than those for new engines in fixed aircraft. Many of the fuel-saving, retrofit features show no advantage in aircraft economics (when evaluated on an ROI basis) since the initial cost to the airlines was weighted heavily in the ROI procedure. On this basis, the cases with significant economic benefit were R125 blades (CF6-50 only), the redesigned HPT shroud, a new compressor casing coating, and the NiCrAly HPT shroud lining.

2. Design Improvements

Design improvement items are those which involved changes to the engine design, and appropriate proof testing, but did not require significant technology development effort to decide whether the change should be undertaken. However, the cost of making the change, and the payoff which was expected, were significant factors in making a decision for their use. Note that certain of the features listed are currently planned for future models of the CF6 engine.

1) R125 Blades: The substitution of R125 blade material in both HPT stages in place of R80 material now used.

Table VIII. Evaluation Summary: Retrofit Possibilities.

Design Change	Δ sfc - %		Δ Fuel Usage - %		Δ DOC - %		Δ ROI - %	
	CF6-6	CF6-50	CF6-6	CF6-50	CF6-6	CF6-50	CF6-6	CF6-50
R125 HPT Blades	-0.2	-0.6	-0.22	-0.69	+0.19	-0.21	-0.11	+0.01
HPT Shroud*	-0.3	-0.25	-0.34	-0.27	-0.12	-0.09	+0.01	+0.015
LPT Changes	-0.4	-0.5	-0.41	-0.54	+0.02	-0.12	-0.065	-0.02
Core Jet Nozzle Area Change		-0.2		-0.22		-0.06		-0.005
Variable Stator Schedule	-0.25		-0.27		0		-0.03	
Low Expansion Compressor Casing		-0.15		-0.19		-0.07		+0.005
Improved Fan Aerodynamics	-0.25	-0.25	-0.27	-0.28	+0.09	-0.03	-0.09	-0.04
New Compressor Casing Coating*	-0.3	-0.2	-0.31	-0.21	-0.22	-0.11	+0.03	+0.02
NiCrAly HPT Shrouds*	-0.2	-0.15	-0.20	-0.16	-0.08	-0.07	+0.01	+0.01

*Includes Reduced Deterioration Effect

Table IX. Evaluation Summary; Design Improvements (CF6-6, New Engines).

Design Change	Δ sfc %	Δ Fuel Usage - %		Δ DOC - %		Δ ROI - %	
		Fixed Payload	Variable Payload	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload
R125 HPT Blades	-0.35	-0.36	-0.65	+0.27	-0.02	-0.10	0
HPT Shrouds*	-0.4	-0.44	-0.79	-0.02	-0.38	-0.005	+0.11
LPT Changes	-0.5	-0.60	-1.08	-0.10	-0.59	0	+0.16
Variable Stator Schedule	-0.25	-0.27	-0.49	-0.05	-0.27	0	+0.07

*Includes Reduced Deterioration Effect

Table X. Evaluation Summary; Design Improvements (CF6-50, New Engines).

Design Change	Δ sfc %	Δ Fuel Usage - %		Δ DOC - %		Δ ROI - %	
		Fixed Payload	Variable Payload	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload
R125 HPT Blades	-0.9	-1.05	-2.03	-0.28	-1.35	+0.035	+0.47
HPT Shroud*	-0.3	-0.34	-0.66	-0.09	-0.44	+0.015	+0.16
LPT Changes	-0.55	-0.64	-1.25	-0.18	-0.84	+0.02	+0.29
Core Jet Nozzle Area Change	-0.2	-0.22	-0.44	-0.09	-0.28	+0.015	+0.08
Low Expansion Compressor Casing	-0.25	-0.28	-0.55	-0.11	-0.40	+0.02	+0.14

*Includes Reduced Deterioration Effect

Table XI. Evaluation Summary; Technology-Dependent Features (CF6-6, New Engines).

Design Change	Δ sfc %	Δ Fuel Usage - %		Δ DOC - %		Δ ROI - %	
		Fixed Payload	Variable Payload	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload
Improved Fan Aerodynamics	-0.25	-0.27	-0.49	-0.05	-0.27	-0.005	+0.07
Composite Blades							
-0.3% Efficiency	+0.4	+0.32	+0.04	-0.31	-0.56	+0.11	+0.21
+1.0% Efficiency	-0.4	-0.55	-1.53	-0.55	-1.54	+0.15	+0.48
Composite Frame	0	-0.14	-0.86	-0.16	-0.88	+0.08	+0.32
New Compressor Casing Coating*	-0.4	-0.46	-0.82	-0.11	-0.48	+0.01	+0.14
Hydrodynamic Seals	-0.75	-0.82	-1.47	+0.38	-0.29	-0.065	+0.16
Ni76 HPT Blades	-0.45	-0.48	-0.86	+0.72	+0.33	-0.24	-0.11
NiCrAly HPT Shrouds*	-0.2	-0.24	-0.43	-0.05	-0.25	0	+0.07
HPT Clearance Control*	-0.3	-0.33	-0.59	-0.02	-0.29	-0.015	+0.07
Mixed-Flow, Composite Nacelle	-3.0	-3.3	-5.9	-0.46	-3.1	-0.13	+0.77

*Includes Reduced Deterioration Effect

Table XII. Evaluation Summary; Technology-Dependent Features (CF6-50, New Engines).

Design Change	Δ sfc %	Δ Fuel Usage - %		Δ DOC - %		Δ ROI - %	
		Fixed Payload	Variable Payload	Fixed Payload	Variable Payload	Fixed Payload	Variable Payload
Improved Fan Aerodynamics	-0.25	-0.28	-0.55	-0.11	-0.40	+0.015	+0.13
Composite Blades							
-0.3% Efficiency	+0.4	+0.30	+0.16	-0.15	-0.29	+0.08	+0.14
+1.0% Efficiency	-0.4	-0.60	-1.58	-0.53	-1.59	+0.16	+0.59
Composite Frame	0	-0.17	-0.82	-0.19	-0.88	+0.10	+0.38
New Compressor Casing Coating*	-0.3	-0.31	-0.61	-0.13	-0.45	+0.025	+0.16
Hydrodynamic Seals	-0.5	-0.50	-1.09	+0.10	+0.48	-0.02	+0.22
Ni76 HPT Blades (versus R125)	-0.55	-0.62	-1.20	+0.13	-0.50	-0.09	+0.17
NiCrAly Shrouds*	-0.2	-0.19	-0.37	-0.08	-0.27	+0.015	+0.09
HPT Clearance Control*	-0.4	-0.45	-0.87	-0.16	-0.62	-0.025	+0.21
Mixed-Flow, Composite Nacelle	-3.5%	-3.9	-7.6	-1.47	-5.5	+0.21	+1.85

*Includes Reduced Deterioration Effect

2) HPT Shrouds: Design changes to the shrouds to improve cooling and reduce mechanical distortion in service.

3) LPT Changes: A series of design changes to improve LPT efficiency. They were grouped because they involve the same engine component and all, or a portion, might be accomplished at the same time.

4) Cycle Trimming: The variable stator schedule and core jet nozzle changes were directed at improving cruise sfc, but also resulted in cycle operation changes at other conditions.

5) Low-Expansion, Compressor Casing: A new rear casing material to allow closer steady-state running clearances. It tended to be between the design-improvement and technology-dependant categories because the material characteristics were not completely defined.

The design improvement features for new CF6-6 engines did not show a payoff from an ROI standpoint for fixed aircraft, as presented in Table IX. On a growth aircraft basis, however, the economic situation was better; but the advantages were less than those available for the CF6-50 engine. The reason was the smaller production run expected for the CF6-6 model of the engine, which increased the impact of the nonrecurring costs.

The design improvement features for new CF6-50 engines all showed an economic advantage for fixed aircraft, as presented in Table X. On a growth-aircraft basis, the advantages were quite large and, for that reason are already under consideration for growth versions of the CF6-50 engine.

3. Technology-Dependent Features

The technology-dependent items for the CF6-6 varied in their payoff, as listed in Table XI. On a fixed-aircraft basis, there was an economic penalty in many cases. On a variable-payload basis, however, all features (with the exception of Ni76 blades) showed a payoff.

For the CF6-50 engine the magnitudes of advantages achieved tended to be larger, as presented in Table XII. All features, except the hydrodynamic seals and Ni76 blades showed an economic payoff on a fixed-aircraft basis. Significant advantage was shown for all features on a growth-aircraft basis.

The features, listed in Tables XI and XII, require various degrees of technology development. The following are comments on each category of feature considered.

1) Fan Efficiency: the CF6 fan has been pushed to tip speeds and specific flows higher than its original design point. The CF6-50 fan corrected tip speed is greater than 427 m/sec (1400 fps) at altitude flight conditions, about 4% higher than its design value. It was estimated that a redesign of the outer portion of the fan blade could provide at least a 0.5% improvement in fan bypass stream efficiency in the operating range of interest. The redesign would involve a new blade shape with increased camber in the tip section.

2) Composites: The use of composites can improve fuel usage thru a saving in weight. A reduction of 8% of the total engine weight was estimated for a redesign of the CF6 engine using composites in the fan blade and frame (almost one-half of the estimated reduction is in the frame). The effect of a composite fan blade design on sfc was uncertain at that time and the feature was, therefore, evaluated with both a small loss and an improvement in fan bypass stream efficiency. The composite frame was expected to have no effect upon sfc. There was, however, a significant manufacturing cost saving projected for the use of composites in both the fan blades and frame. It must be noted that the feasibility of composite fan blades, from the bird-strike standpoint, has not been clearly established. Future design changes necessary to provide bird-ingestion capability may affect the evaluation of benefits for composite blades.

3) Compressor Clearance Control: One approach considered for compressor clearance control was the utilization of a low expansion material, such as INCO 903, in the casing. An improvement of 0.4% efficiency was estimated because of the closer steady-state clearances that could be obtained with the better transient thermal match of the compressor casing and rotor.

4) Self-Acting Seals: A redesign of the CF6 midsump to incorporate self-acting (or hydrodynamic) seals in place of labrinth seals was evaluated to provide a reduction in the high pressure leakage of 0.5%. In the case of the main compressor discharge pressure seal, the current speed of 228.6 m/sec (750 ft/sec) will require an advance in the state-of-the-art for hydrodynamic seals.

5) Advanced Turbine Blade Material: An improved, directionally solidified Ni base alloy (GE designation Ni76XB) has been identified with 24° C (75° F) higher metal temperature capability than the best currently available alloy (René 125). This allowed a reduction in cooling flow of 0.6% for the same engine rating. Note that use of R125 material is listed under the design improvement category, the present CF6 material is René 80.

6) Turbine Clearance Control: In the high pressure turbine, a shroud coating material with increased erosion resistance (NiCrAly) was utilized and estimated to allow 0.2% better turbine efficiency, on the average, including reduced deterioration. It was also believed possible to redesign the HPT case to improve thermal matching. A potential improvement of 0.3 to 0.6% efficiency was estimated, but a specific design to achieve the improvement was not carried out in this study due to the magnitude of effort required.

7) Long-Duct, Mixed-Flow, Composite Nacelle: The concept of mixing the exhaust of a turbofan engine has the potential for improving sfc, since a single exhaust jet at a uniform velocity has a higher propulsive efficiency than two jets at different velocities (the situation in a normal separate-flow cycle). A low loss mixer design with high mixing effectiveness (η_{mix}) is required to achieve the sfc improvement. Estimates made in the study were based, in part, upon scale-model testing of mixed exhaust systems.

The characteristics of a new nacelle for the CF6-50 engine were determined in a recent Douglas/General Electric study conducted under contract to NASA-Langley (Reference 3). The results are summarized in Table XIII and were used in the STEDLEC study. Estimates for the factors affecting fuel usage and economics were made for the CF6-6 engine on a consistent basis. Note that no significant change was required to either model of the CF6 engine to incorporate a new nacelle, although the new nacelle itself represents a major change to the propulsion system installation.

The primary advantage of the mixed-flow cycle was an estimated 3 to 3-1/2% improvement (for the CF6-6 and CF6-50 respectively at 80% cruise power setting) in installed sfc, a portion of which was the lower drag of the long-duct nacelle. There was also an estimated reduction in noise. With the use of composites in the cold section of the nacelle, there was no weight penalty for adding the long duct. The net effect was an improvement in fuel usage ranging from 3.3%, for the CF6-6 in a fixed-payload aircraft, to 7.6% for the CF6-50 in a variable-payload aircraft.

The alternate nacelle approaches for the CF6-50 engine are illustrated in Figure 4, again using data from Reference 3. Compared to the current design, it was estimated that the mixed-flow nacelle with partial composites could be designed for the same nacelle weight. Also shown are the relative weights for a new long-duct, metal nacelle; and a nacelle making maximum use of composites. A separate-flow nacelle, redesigned in metal or composites (without a turbine reverser), is also possible and was estimated to provide the improvements presented in the lower-right portion of Figure 4.

Note the estimated benefits presented in Table XIII, and Figure 4, assumed advancements in noise suppression and nacelle design technology and, for that reason, Task I results did not represent a direct mixed- versus separate-flow comparison. This question is, however, addressed in Task II on a consistent technology basis.

Table XIII. Advantages of New Mixed-Flow Nacelle for CF6-50 Versus Current Nacelle.

Installed sfc Improvement	3-1/2% at Normal Cruise	
Installed Thrust	+5% at Max. Climb and Max. Cruise +1% at Takeoff	Constant Turbine Temperature
Noise Reduction	4 EPNdB at Takeoff 2 to 3-1/2 EPNdB at Approach	With Noise Technology
Reverse Thrust (Nominal Static Value)	42% Current Nacelle with Core Reverser 18% Separate Flow Without Core Reverser 30-35% Mixed Flow with Aerodynamic Spoiler Effect	
Nacelle Weight	Composite Version - No Change 181 kg (+400 lb) for New Metal Design	

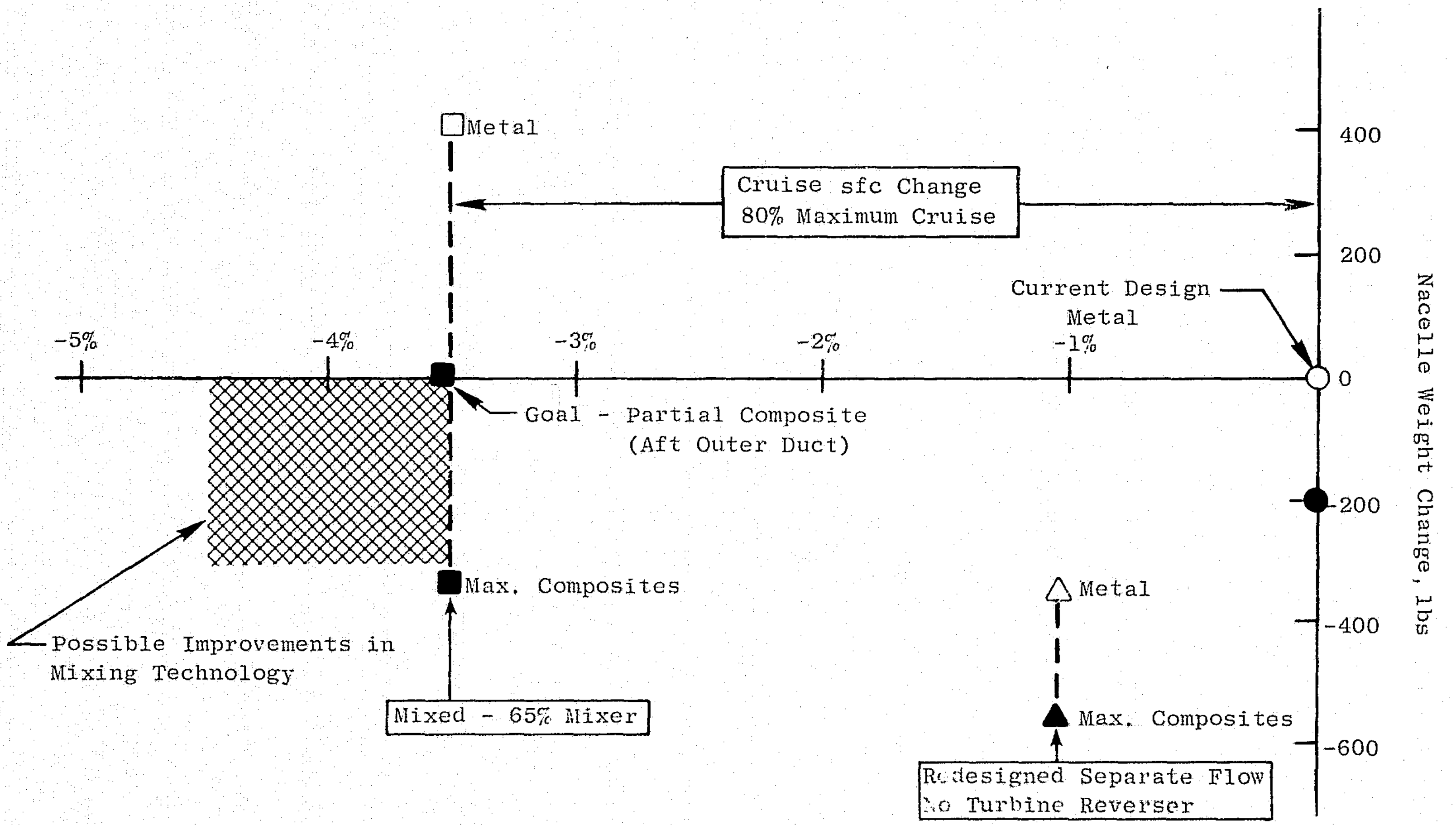

Figure 4. CF6-50 Nacelle Alternatives; Installed sfc and Weight Summary.

G. Summary of Results - Task I

A series of design changes for the CF6 family of engines, many requiring technology development, were evaluated in terms of their potential for reducing energy consumption and impact upon aircraft economics. The engine cost implication of the various changes, as well as the direct sfc and weight effects, were included in the evaluation. A summation of the advantages estimated in this Task is presented in Figure 5. It should be emphasized that the improvements presented are estimates. Experience has shown that not all improvements are achievable, and only a portion of the totals presented in Figure 5 should be counted on.

The mixed-flow, composite nacelle provided the largest potential for reduced energy consumption and improved aircraft economics. The estimated improvements were 3.9% fuel usage and 1.5% DOC for the CF6-50 powered aircraft with fixed payload.

Other features requiring technology development were estimated to provide the following gains (again for the CF6-50 engine in a fixed-payload aircraft):

Design Change	ΔFuel Usage	ΔDOC
Improved Fan Blade Design	-0.3%	-0.1%
Composite Fan Blades (range depends upon efficiency level achieved)	+0.3 to -0.6%	-0.1 to -0.5%
Composite Frame	-0.2%	-0.2%
New Compressor Casing Coating*	-0.3%	-0.1%
Hydrodynamic Seals	-0.5%	-0.1%
Ni76 HPT Blades (versus R125)	-0.6%	+0.1%
MCrAly HPT Shroud Lining*	-0.2%	-0.1%
HPT Clearance Control*	-0.4%	-0.2%

All of the above require technology development to achieve the magnitude of gains indicated, and it must be pointed out that the degree of success in each case is subject to some uncertainty. The technology-dependent items vary in the amount of development required and the chance of meeting predicted characteristics. For example, composite fan blades and hydrodynamic seals are in the high risk category, while the improved HPT shroud material can almost be put in the design improvement category.

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*Improves performance retention.

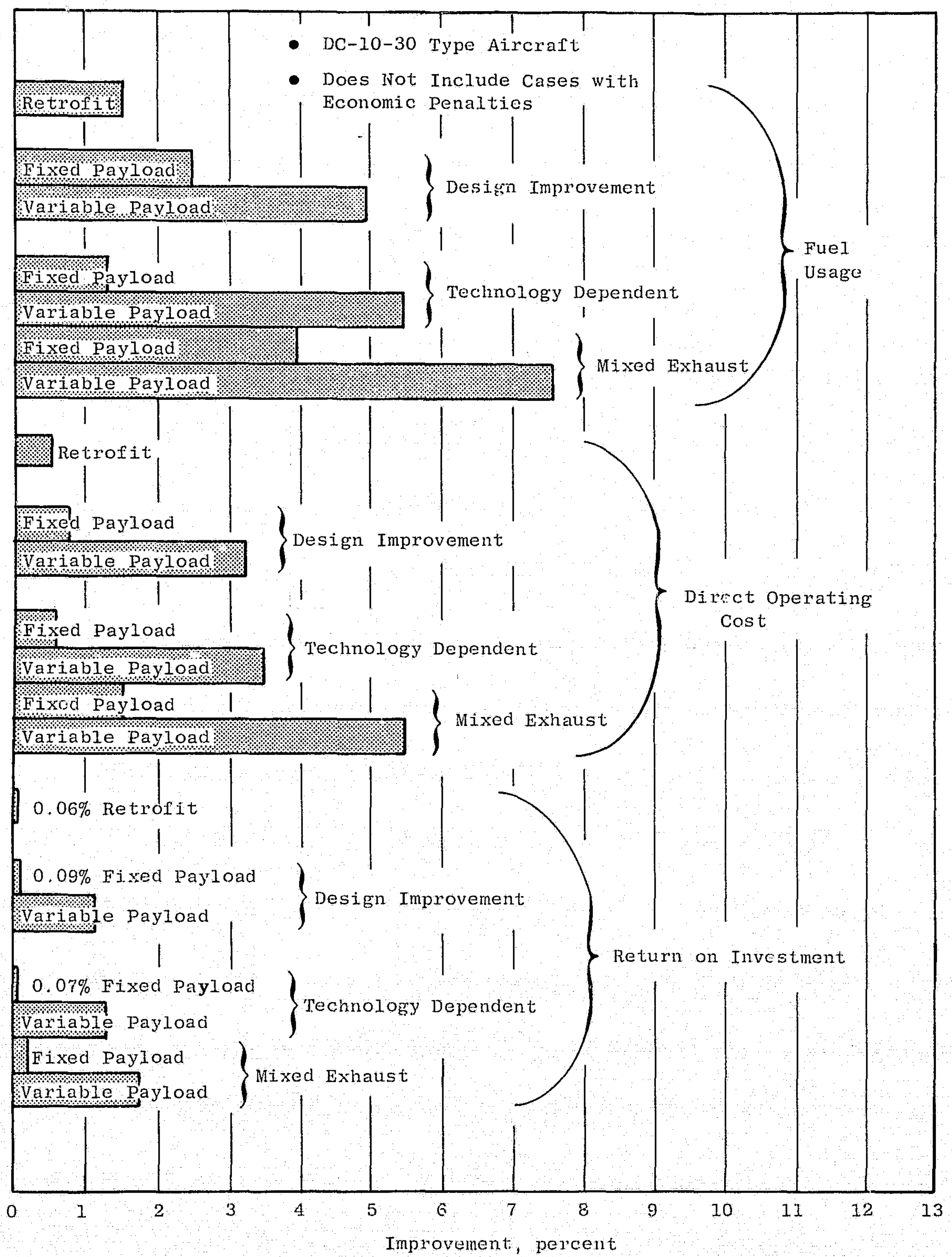


Figure 5. Summation of Improvements for CF6-50 Engines.

SECTION IV

TASK II - NEW ENGINES

A. Approach

The design of future turbofan engines, for entry into service in the 1985 time period, was studied to make choices on cycle and technology features. During the Task II effort, variations in turbine inlet temperature, overall pressure ratio, fan pressure ratio, bypass ratio, installation type, and the application of advanced materials and design concepts were studied for payoff in an advanced turbofan engine.

A reference design was identified based on in-house studies prior to the start of the STEDLEC study. The key features of the advanced turbofan, used as reference in Task II, are listed in Tables XIV and XV. A mixed-flow installation with takeoff cycle parameters of 1538° (2800° F) turbine rotor inlet temperature (T_{41}), 38:1 overall cycle pressure ratio, and a bypass ratio of 8.1 at M_N 0.8, 10,670 m (35,000 ft) Max. Climb was selected based on previous studies.

The effect of each cycle variation, or advanced technology feature, on sfc, weight, initial cost, and maintenance cost was estimated in the framework of the reference engine. Those features which showed potential for improvement in energy consumption were considered. The effects of each feature were then determined in terms of aircraft fuel usage and economics. The Task II effort was organized into a series of relatively independent studies by comparing each feature with a reference design involving current, or nearer-term technology, as summarized in Table XVI.

B. Evaluation Procedure

A 5560 km (3000 nmi)/200-passenger domestic trijet and a 10,190 km (5500 nmi)/200-passenger intercontinental quadjet were designed using advanced aerodynamics and structural weights consistent with the aircraft presented in Reference 6. The trijet field length was specified at 2600 m (8500 ft) for study (high lift devices used at takeoff). The aircraft design study, of Reference 6, specified a slightly shorter field length of 2400 m (8000 ft). Key data are given in Tables XVII and XVIII for the two aircraft. The baseline aircraft were "flown" over full-design and average-range/55% load-factor missions. The average-range/55% load-factor mission is more important in considering aricraft economics and total fuel consumed by the aircraft fleet.

Direct operating cost (DOC) was calculated using the ATA (American Transport Association) formula modified by General Electric engine maintenance factors. In the General Electric modification, reverser maintenance was identified as a separate item, and engine maintenance labor and material reflected General Electric experience. The detailed formula differences from ATA are given in Table XIX. The indirect operating costs were calculated

Table XIV. Reference Task II Advanced Engine, Design Size.

Altitude/Mach No.	0/0	0.8/10,670 m (35,000 ft)	
Rating	Takeoff	Max. Climb	Max. Cruise
Day, Std + ° C (° F)	15 (27)	10 (18)	10 (18)
Fn, N (lb)	172,600 (38,800)	43,800 (9,850)	40,700 (9,150)
Overall Pressure Ratio	32	38	36
T_{41} - hot day, ° C (° F)	1538 (2800)	1470 (2680)	1427 (2600)
$W\sqrt{\theta}/\delta$; kg/sec (lb/sec)	590 (1300)	653.2 (1440)	639 (1410)
Fan Pressure Ratio	1.51	1.65	1.61
Fan $U_T/\sqrt{\theta}$, m/sec (ft/sec)	442 (1450)	487.7 (1600)	476 (1560)
Booster Pressure Ratio	2.45	2.75	2.65
Core Compressor Pressure Ratio	12.7	14	13.7
Core Airflow, $W\sqrt{\theta}/\delta$ kg/sec (lb/sec)	29.5 (65)	31.8 (70)	31.3 (69)
Mixed Flow, 75% mixer effectiveness			

Table XV. Baseline Task II Advanced Engine Design Features.

Fan Tip Diameter, m (in.)	2.15 (84.5)
Fan Radius Ratio	0.38
Fan Design	Unshrouded Composite
No. of Boosters	3
No. of Core Compressor Stages	9
Core Compressor Radius Ratio (r/r)	0.68
Combustor Type	Double Dome, Low Emissions
No. HPT Turbine Stages	1
Cooling	Advanced Film Bore-Entry Supply
No. of LPT Stages	5 + OGV (2-3 cooled)
Avg. LPT Work Coeff. $gJ\Delta h/2U_p^2$	1.8
Exhaust	Mixed
Nozzles	Fixed Convergent-Divergent
Length (Flange to Flange), m (in.)	2.5654 (101)

Table XVI. Low Energy-Consumption Study; Features for Task II Evaluation.

Item	Feature	Reference for Comparison
1	T_{41}, Cycle Pressure Ratio and Turbine Technology	Baseline Engine Cycle
2	Fan Pressure Ratio Optimization	Baseline engine and installation
3	Mixed Flow	Separate-flow installation
4	High Tip Speed Composite Fan	Tip-Shrouded Ti Fan with Advanced Aerodynamics and Midspan Ti without Advanced Aerodynamics
5	Compressor Clearance Control	Current design approach applied to baseline engine
6	HPT Clearance Control	Current design approach applied to baseline engine
7	Bore-Entry Cooling for HPT	Compressor-Discharge Air for HPT Cooling
8	Ceramic HPT and LPT Vanes and HPT Shrouds	Metal Vanes Current Shroud Design
9	Eutectic Turbine Blades (HPT and LPT)	Nickel-base, Directionally Solidified, Casting Alloy
10	Advanced Film/Impingement Cooling in HPT BLade	Current film cooling
11	Integrated Composite Nacelle and Thin Inlet and Pylon Accessories	Metal nacelle with conventional inlet and bottom-mounted accessories
12	Lower Source-Noise LP Turbine	Conventional LP turbine with exhaust suppression
13	Inlet Bulk-Absorber Treatment	Honeycomb sandwich suppression lining in inlet
14	Composite Fan Frame	Metal Fan Frame

Table XVII. Baseline Aircraft.

	Trijet	Quadjet
Design Range, km (nmi)	5560 (3000)	10,190 (5500)
Average Mission Range, km (nmi)	1300 (700)	3700 (2000)
Design Payload No. of Pass.	200	200
Cruise Altitude, m (ft)	10,670 (35,000)	10,670 (35,000)
Cruise Mach Number	0.80	0.80
Design Field Length, m (ft)	2600 (8500)	2600 (8500)
TOGW, kg (lb)	101,000 (223,000)	145,000 (320,000)
SLS Takeoff Fn/Engine N (lb)	89,000 (20,000)	93,000 (21,000)
Wing Aspect Ratio	12	12
Cruise C_L, average	0.50	0.55
Cruise L/D, average	17	18
Takeoff C_L, average	2.75	2.75

Table XVIII. Base Aircraft Design Weight Distribution.

	Domestic Trijet	International Quadjet
Design Range, km (nmi)	5560 (3000)	10,190 (5500)
No. Passengers	200	200
TOGW, kg (lb)	101,200 (223,000)	145,100 (319,800)
SLS Takeoff Fn/Engine, N (lb)	88,960 (20,000)	93,400 (21,000)
Total Structural, kg (lb)	40,950 (90,280)	51,200 (112,900)
Total Powerplant, kg (lb)	5620 (12,400)	7920 (17,460)
Operating Equipment, kg (lb)	10,200 (22,500)	11,570 (25,500)
Operating Weight, Empty, kg (lb)	56,800 (125,200)	70,700 (155,840)
Fuel Burned	21,100 (46,500)	47,300 (104,290)
Reserve Fuel	4550 (10,030)	7570 (16,680)
Design Payload	18,600 (41,000)	19,500 (43,000)

Table XIX. Engine Maintenance Formulae.

	GE Modification	ATA (1967)
Engine Labor	$(0.55 + 2 \times 10^{-6}$ Fn) \$/Fl hr	$(2.4 + 1.08 \times 10^{-4}$ Fn) h/Fl Hr
	+ $(0.45 + 2 \times 10^{-6}$ Fn) \$/Fl Cycle	+ $(1.2 + 1.2 \times 10^{-4}$ Fn) \$/Fl Cycle
Engine Material	25×10^{-6} CE \$/Fl hr	25×10^{-6} CE \$/Fl hr
	+ 10×10^{-6} CE \$/Fl Cycle	+ 20×10^{-6} CE \$/Fl Cycle
Reverser Labor	0.24 \$/Fl hr	---
	+ (0.24 \$/Fl Cycle	---
Reverser Material	5×10^{-2} Cr \$/Fl hr	---
	+ 5×10^{-6} Cr \$/Fl Cycle	---
	Fn = SLS T/O Thrust, lb CE = Engine Cost, \$ CR = Reverser Cost, \$ Fl = Flight hr = Hour	

using the methods of R.F. Stoessel (Reference 7). The ROI was calculated using a discounted cash flow method.

Typical data for the aircraft/missions are listed in Table XX. The DOC breakdown for the two average missions is given in Figure 6 in order to illustrate the contribution of the propulsion system to direct operating cost. Increments in powerplant sfc, weight, initial price, parts price, installation price, mainenance cost, and flight hours were taken individually for each aircraft/mission. The fuel consumed, and the economic factors, were recalculated to obtain mission trade factors on each powerplant variable. The resulting mission trade factors for the part-range/55% load factor, representing an average mission, are given in Table XXI as they were used in Task II. The mission trade factors for full range/full payload did not differ greatly. Mission trade factors for other fuel costs were used in a few instances to test the sensitivity of the study results to fuel costs. The mission trade factors for other ranges and fuel costs are presented in Table XXII.

The advanced technology feature was defined, in each case, in sufficient detail to permit an estimate of the sfc, weight, initial price and maintenance costs in order to determine the total effect of its implementation in the baseline engine.

The resulting change in weight, sfc, and cost factors were then scaled into the engine size appropriate to the mission under consideration. These changes were used with the mission trade factors to obtain the potential fuel and DOC savings. The scaling exponents and key procedural assumptions used are summarized in Table XXIII.

C. Baseline Engine and Installation

A baseline propulsion system was defined, at the beginning of the study, in order to provide data for the baseline aircraft design and mission analyses described under Evaluation Procedure. At the outset of Task II, this engine cycle and the technology employed was believed to be a reasonable selection for the 1985 advanced turbofan. In some cases, the results of Task II changed the parameters selected for use in the Task III engine. The key cycle and design parameters for the baseline Task II engine are given in Tables XIV and XV. Compared to current high bypass engines with a maximum takeoff turbine inlet temperature of 1316° C (2400° F) and 32:1 overall pressure ratio at maximum climb, the baseline engine was selected with 1538° C (2800° F) at takeoff and 38:1 at maximum climb conditions. It featured a compact core compressor driven by a single-stage, high-pressure turbine. The fan was a composite, high tip speed, advanced aerodynamic design driven by an advanced, highly loaded, five-stage, low-pressure turbine.

The baseline Task II installation, illustrated in Figure 7, was a long-duct/mixed-flow type with a thin inlet for lower drag. The installation incorporated accessories in the pylon, employed extensive use of composites, and utilized noise treatment aimed at FAR 36 minus 10 EPNdB. The benefit of each element of the installation design was evaluated in the Task II study

Table XX. Baseline Aircraft Fuel and Economic Data.

	Trijet	Quadjet	Quadjet
Design Range, km (nmi)	5560 (3000)	10,190 (5500)	10,190 (5500)
Range, km (nmi)	1300 (700)	10,190 (5500)	3700 (2000)
Block Speed, m/sec (mph)	180.7 (351)	227.5 (443)	213.6 (415)
Passengers - Design	200	200	200
Load Factor, %	55	100	55
Fuel Cost, $\$/m^3$ (¢/gal)	79 (30)	119 (45)	119 (45)
Fuel, kg (lb)	5780 (11,640)	47,310 (104,300)	16,012 (35,300)
DOC - $/km ($/nmi)	1.76 (3.26)	1.94 (3.60)	2.06 (3.82)
- ¢/Seat-km (¢/Seat-mi)	0.88 (1.63)	0.97 (1.80)	1.03 (1.91)
Fuel/Pass. km, kg/Passenger-km (mi, lb/Passenger mi)	0.037 (0.15)	0.023 (0.095)	0.039 (0.16)
Fuel/Aircraft-Year, Millions kg/Year (Millions lb/Year)	44.5 (20.2)	82.0 (37.2)	65.5 (29.7)

Domestic Trijet
Range = 1,300 km (700 nmi)
Load Factor = 55%
DOC = 1.76 \$/km (3.26 \$/nmi)
DOC = 0.79 ¢/Seat/km (1.46 ¢/Seat/nmi)
Fuel at \$79/m^3 (\$0.30/gallon)

International Quadjet
Range = 3,700 km (2,000 nmi)
Load Factor = 55%
DOC = 2.06 \$/km (3.82 \$/nmi)
DOC = 1.03 ¢/Seat/km (1.91 ¢/Seat/nmi)
Fuel at \$119/m^3 (\$0.45/gallon)

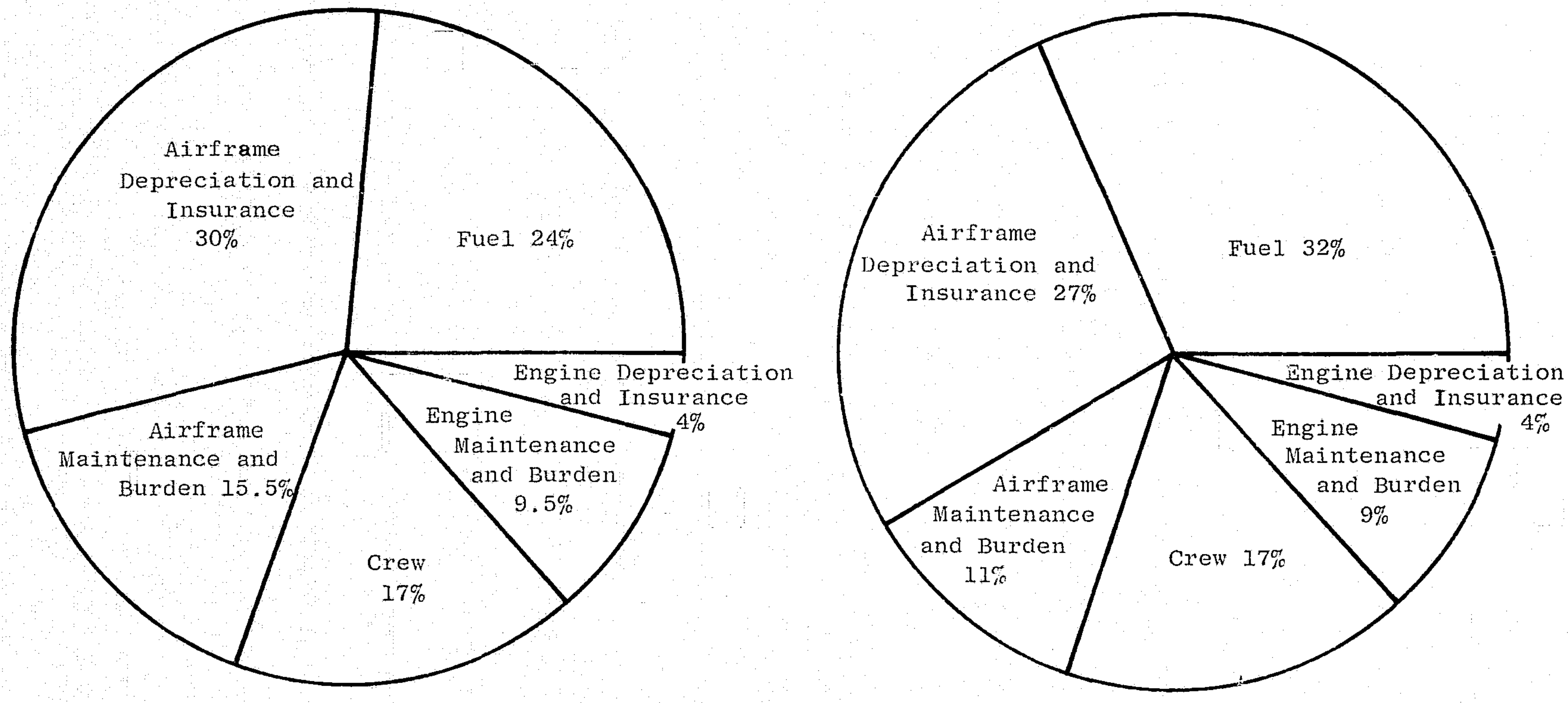


Figure 6. DOC Breakdown, Trijet and Quadjet.

Table XXI. Mission Trade Factors, Average Mission.

	Trijet		Quadjet	
Range, km (nmi)	1300 (700)		3700 (2000)	
Load Factor, %	55		55	
Fuel Cost, $/m^3 (¢/gal)	79 (30)		119 (45)	
Change (per engine)	Aircraft Δ DOC, %	Aircraft Δ Fuel Used	Aircraft Δ DOC, %	Aircraft Δ Fuel Used
1% sfc	+0.39	+1.09	+0.71	+1.44
45.36 kg (100 lb) Engine or Installation	+0.17	+0.26	+0.22	+0.31
$10,000 Engine Initial Price	+0.073	---	+0.060	---
$10,000 Engine Parts Price	+0.070	---	+0.065	---
$10,000 Installation Price	+0.073	---	+0.060	---
$1.0 Maint. Cost/Flight hr.	+0.23	---	+0.24	---

Table XXII. Mission Trade Factors.

No. Engines	3				3				3				4				4			
Design Range, km (nmi)	5560 (3000)				5560 (3000)				5560 (3000)				6480 (3500)				10190 (5500)			
Mission Range, km (nmi)	1300 (700)				1300 (700)				5560 (3000)				3700 (2000)				10190 (5500)			
Load Factor, %	55				55				100				55				100			
Fuel Cost, $\$/m^3$ (¢/gal)	79 (30)				106 (40)				106 (40)				119 (45)				119 (45)			
Δ%	DOC	W_f	TOGW	ROI	DOC	W_f	TOGW	ROI	DOC	W_f	TOGW	ROI	DOC	W_f	TOGW	ROI	DOC	W_f	TOGW	ROI
+1% sfc	0.39	1.09	0.47	0.11	0.44	1.09	0.47	0.13	0.56	1.22	0.48	0.30	0.71	1.44	0.87	0.25	0.80	1.56	0.87	0.41
45.36 kg (+100 lb) Engine or Installation	0.17	0.26	0.32	0.061	0.17	0.26	0.32	0.058	0.17	0.22	0.34	0.16	0.22	0.31	0.39	0.902	0.23	0.32	0.39	0.17
+$10,000 Engine Initial Price	0.73	–	–	0.045	0.068	–	–	0.044	0.059	–	–	0.18	0.060	–	–	0.037	0.055	–	–	0.11
+$10,000 Engine Parts Price	0.070	–	–	0.011	0.065	–	–	0.012	0.066	–	–	0.015	0.065	–	–	0.014	0.064	–	–	0.012
+$10,000 Installation Price	0.073	–	–	0.045	0.068	–	–	0.044	0.059	–	–	0.18	0.060	–	–	0.037	0.055	–	–	0.11
+$1.0 Maint. Labor Cost/ Flight hr	0.23	–	–	0.080	0.47	–	–	0.082	0.55	–	–	0.12	0.24	–	–	0.12	0.54	–	–	0.10

DOC – Direct Operating Cost
W_f – Fuel Used
TOGW – Takeoff Gross Weight
ROI – Return on Investment

Table XXIII. Evaluation Procedure.

Constant Payload and Range, Variable Gross Weight.
Baseline A/C 5,560 km (3000 nmi)/200 PAX Trijet 10,190 km (5500 nmi)/200 PAX Quadjet
Mission Trade Factors for Engine Changes Determined.
Baseline Engine Pressure Ratio 1.65 Fan, Mixed Flow with Advanced Technology.
Effects of Changes in Installed Engine Characteristics Determined for Each Engine Variation Studied.
Effects of Engine Price Related to Production Cost - 1974 $.
Individual Parts Replacement Rates Considered for Engine Maintenance Costs.
Engines Scaled to Thrust Required by Baseline Aircraft.
Engine Scaling Exponents, Weight - 1.25, Price - 0.55
Installation Scaling Exponents, Weight - 1.1, Price - 0.80

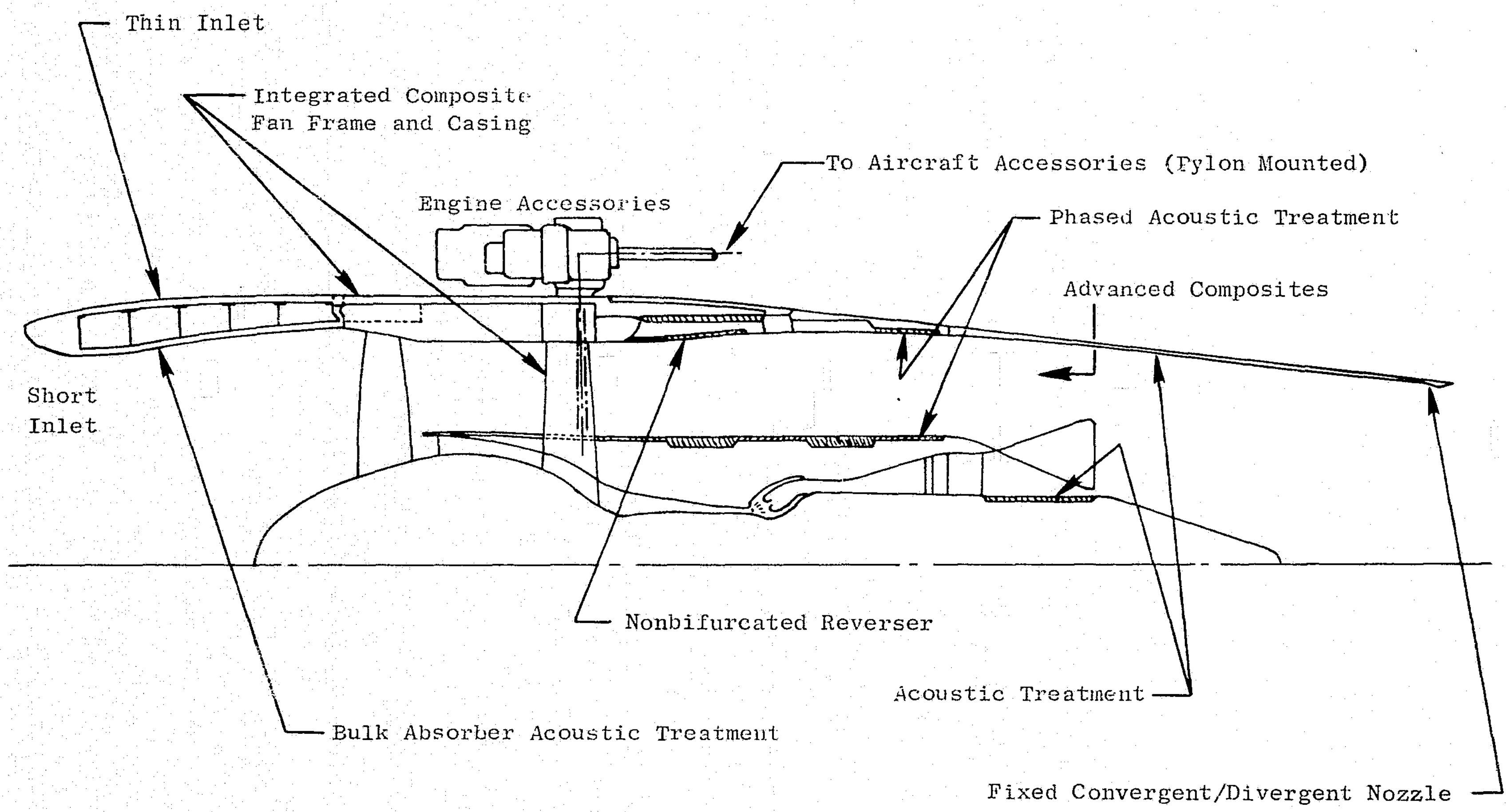


Figure 7. Baseline Nacelle.

and the results are discussed in Section G. Figure 8 illustrates the baseline installation compared to a CF6-50 DC-10 installation.

D. Cycle Selection Studies - Turbine Temperature and Cycle Pressure Ratio

The selection of turbine inlet temperature, cycle pressure ratio, and turbine technology level were considered together in the evaluation because of the interrelation in setting the turbine cooling flow requirements.

The fan pressure ratio was held constant for that portion of the cycle study at a baseline value of 1.65 at the maximum climb design point. That effectively held engine specific thrust and propulsive efficiency constant. In the studies described in Section E, the fan pressure ratio was varied while core engine parameters were held constant. Bypass ratio was allowed to vary in both portions of the study.

Both a current level and an advanced level of cooling technology was defined for the purpose of that study. The pertinent material and cooling technology selections are tabulated in Table XXIV. Allowable turbine blade and vane temperatures were set at each technology level for equal design life based on mission turbine-inlet temperatures, blade stress levels, and consideration of the turbine aerodynamic design. Takeoff turbine inlet temperatures were varied between 1316° C (2400° F) and 1649° C (3000° F) for the advanced technology case, while the current technology considered only the range of 1316° C (2400° F) to 1427° C (2600° F). Overall cycle pressure ratios, at the altitude maximum climb design point, were varied between 25:1 and 45:1.

Cooling flow extraction locations (compressor stage number) were set by the pressure requirements in each engine. The cooling flow for the low pressure turbine was removed at the compressor casing, and the HP blade flow was bled at the compressor hub and introduced into the turbine through a bore-entry system. Using consistent turbine cooling flow calculation procedures, the cooling flows for the HPT and LPT were estimated as illustrated in Figure 9. The cooling flow requirements of the multistage LPT became an overriding factor at the higher turbine inlet temperatures considered.

The resulting sfc trends associated with turbine temperature are illustrated in Figure 10 for cycles with constant fan pressure ratio and with exhaust systems matched for mixed flow. Weight and engine price trends were prepared for the range of engines and are presented in Figure 11. Those trends were not smooth versus cycle pressure ratio because of two opposing trends involving changes in the number of stages. The core components got smaller and lighter as precompression was added, but weight increased due to more booster stages on the low speed spool. At 1538° C (2800° F) takeoff T_{41}, the two effects canceled between 38:1 and 45:1 overall pressure ratio and resulted in no weight difference. The effect of higher turbine temperatures, at constant cycle pressure ratio, resulted in a smaller core size for a given fan pressure ratio. Hence, a weight and cost reduction for a given level of materials and cooling technology was realized.

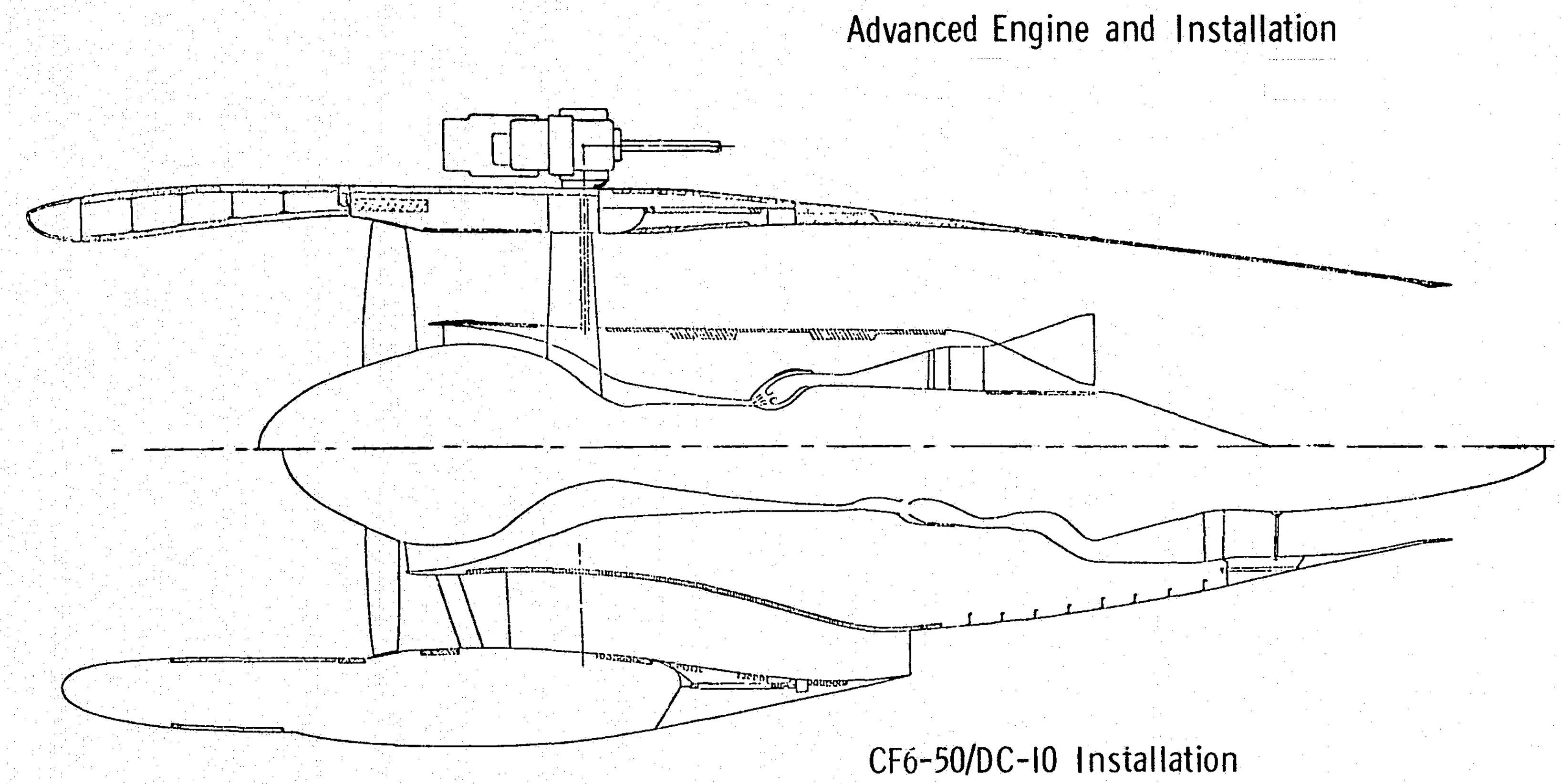


Figure 8. Installation Comparison.

Table XXIV. Cycle Selection Study Technology Definitions.

Technology	Current (1)*		Advanced (2)	
	Material	Cooling	Material	Cooling
HPT Vane	Nickel-Base, Thorium-Stabilized Alloy	Current Film	Ceramic	Convection modified for ceramic
HPT Band	Oxidation-resistant, Nickel-Base Alloy	Impingement film	Nickel-Base Alloy with very high oxidation resistance	Impingement and film
HPT Blade	Nickel-Base Casting	Current film	Nickel-Base Casting, DIrectionally Solidified	Impingement-film cooling
HPT Shroud	Cooled Porous Design		Ceramic	
LPT Vane	Cobalt-Base, High Temp Alloy	Current film	Ceramic	None
LPT Blade	Current Nickel Base	Convection	Nickel-Base Casting, Directionally Solidified	Improved convection

*Representative of F101 and Transport Derivatives thereof - More Advanced Than CF6.

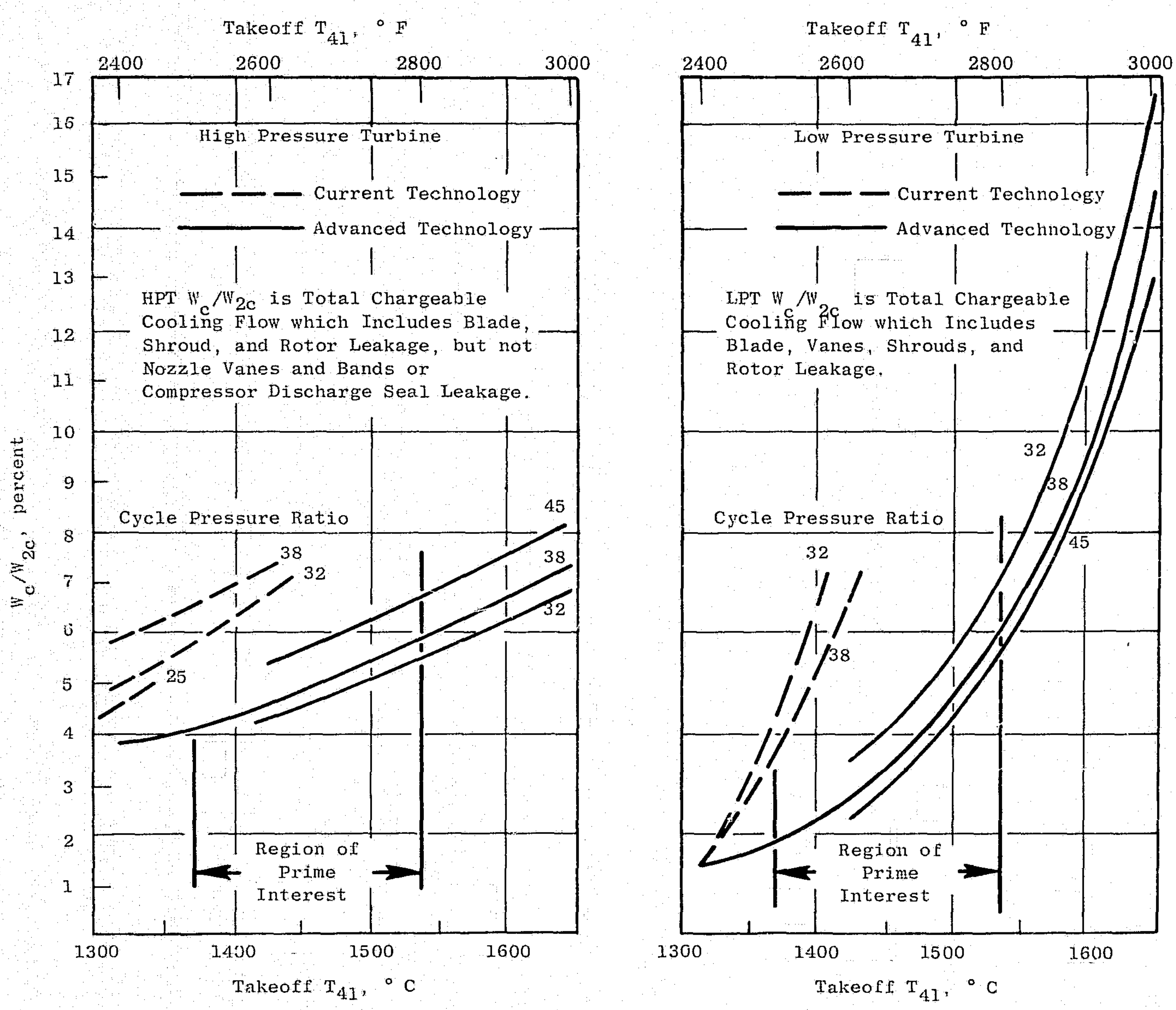


Figure 9. Cycle Selection Study; Turbine Cooling Flow Trends.

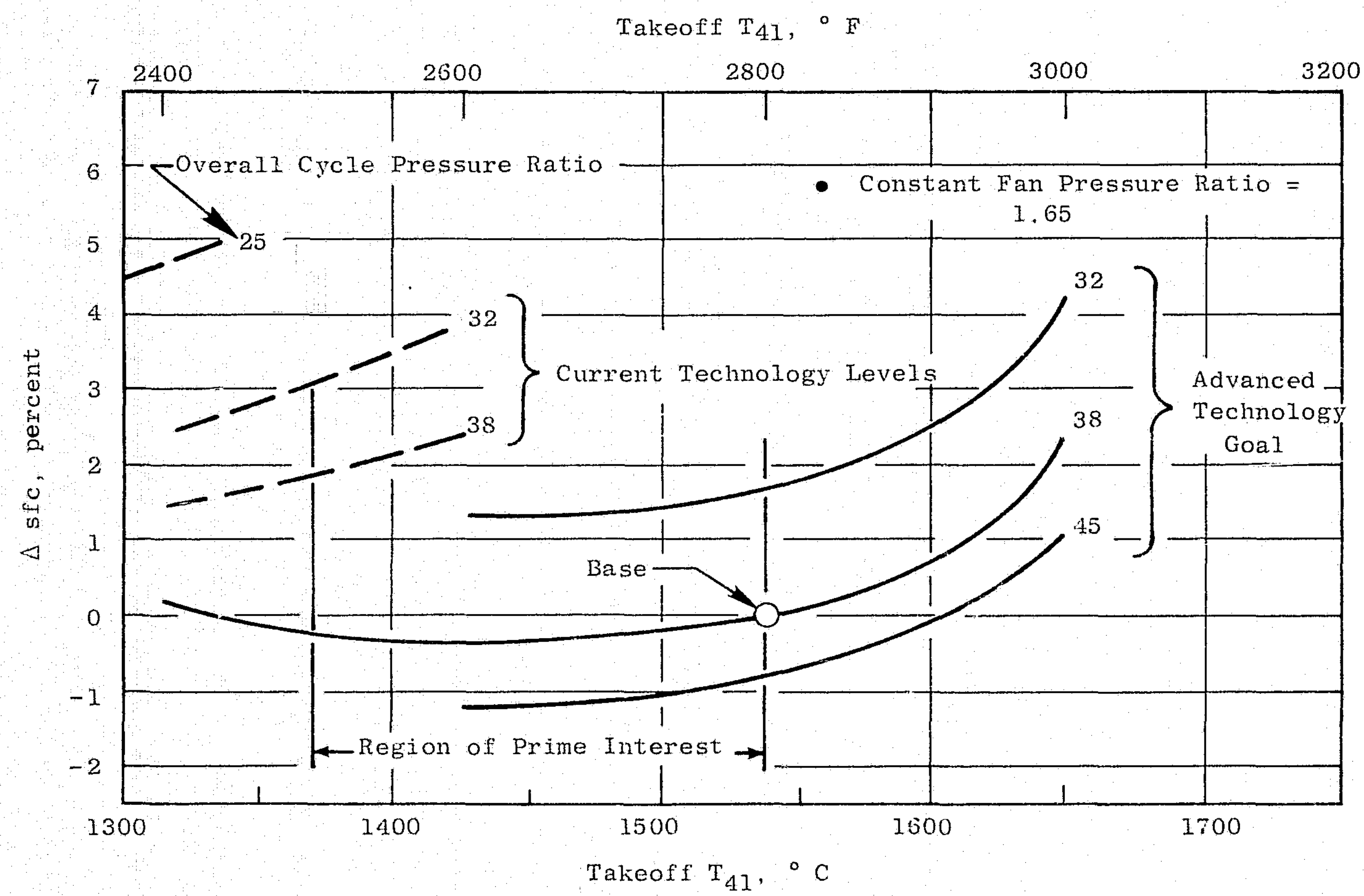


Figure 10. Cycle Selection Study; Installed sfc Trends.

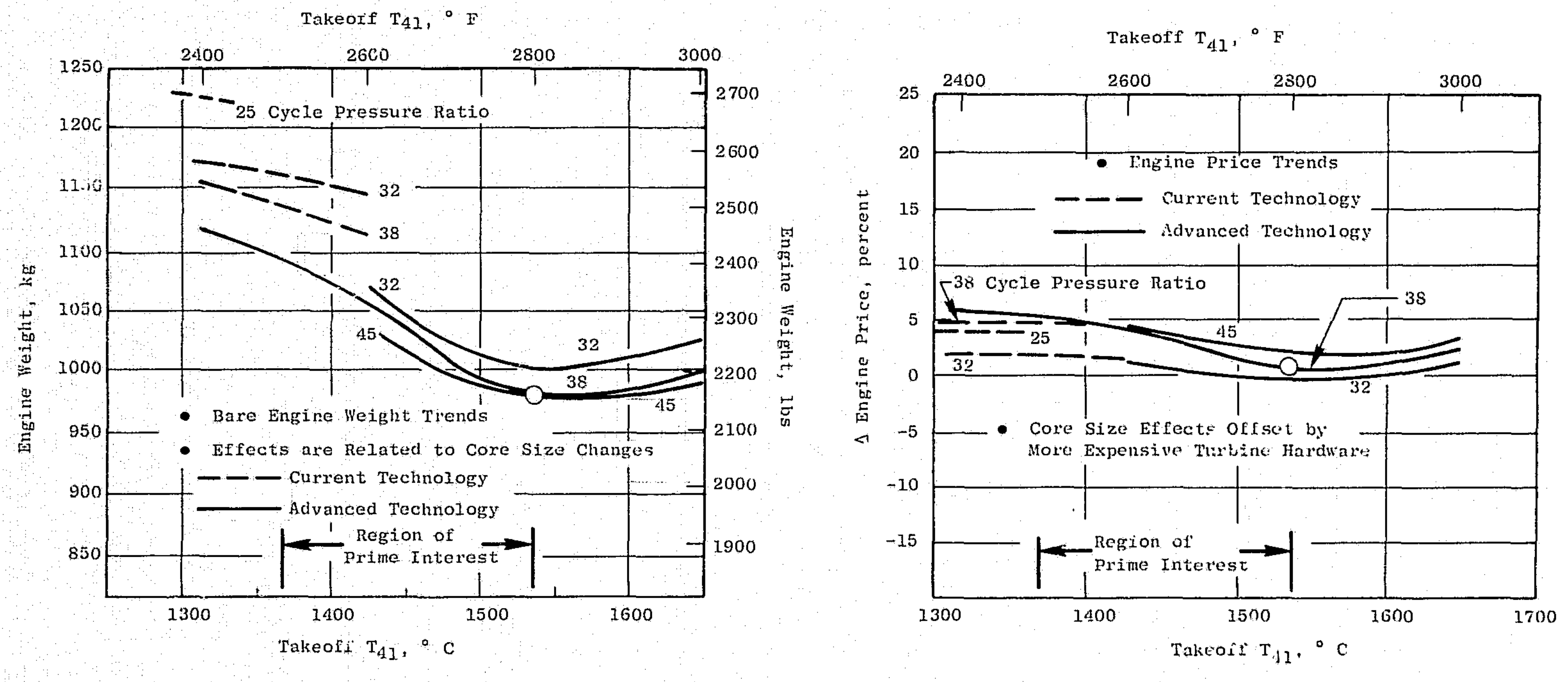


Figure 11. Cycle Selection Study; Weight and Price Trends.

The advanced technology cores were smaller and lighter than current technology cores for the same cycle conditions, however, the price was approximately the same because of the use of higher-cost, advanced turbine materials. The cooling flow penalties for higher turbine temperature would have increased, if current cooling technology had been employed, and the minimum sfc for a given cycle pressure would have been poorer than for the advanced technology (and also would have occurred at a lower T_{41}). The engine weights were also greater, because a larger core was required for a given thrust. The engine prices were almost the same because the lower-cost turbine balances the effect of the larger core size.

The resulting DOC and fuel-usage trends are presented in Figure 12 and Figure 13 for the 5560 km (3000 nmi) trijet on the 1300 km (700 nmi), 55% load-factor mission. A takeoff turbine inlet temperature of 1427° C (2600° F) provided minimum fuel consumption with a very small DOC penalty. A higher cycle pressure ratio of 45:1 (maximum climb at altitude) achieved up to a 1% improvement in DOC and fuel usage over the baseline level of 38:1 at 1538° C (2800° F) takeoff turbine inlet temperature. Similar trends were obtained for the 10,190 km (5500 nmi) quadjet presented in Figure 14 and Figure 15 for the 3700 km (2000 nmi) mission and 55% load factor. The effect of advanced turbine technology versus current turbine technology was a 3.2% fuel saving and a 1.5% DOC saving at an overall pressure ratio of 38 at maximum climb, and a turbine inlet temperature of 1427° C (2600° F) at takeoff.

If ceramic vanes were omitted from the advanced technology package, the DOC would have become insensitive to turbine inlet temperature selection, as illustrated in Figure 16. The absolute level of fuel burned increased 1.1% at 1538° C (2800° F) takeoff T_{41} for metal versus ceramic vanes, in both turbines (Figure 17). Ceramic vanes were not included in the recommended advanced technology Task III engine design because feasibility from the impact-damage standpoint had not been established.

Observations and conclusions as a result of those studies are summarized:

- Significant advantages (-2% in DOC and -3.5% in fuel for the trijet application) were achieved with a 1538° C (2800° F) takeoff T_{41}/ 38:1 maximum-climb pressure ratio/advanced turbine technology versus 1316° C (2400° F) takeoff T_{41} 32:1 maximum-climb pressure ratio/current technology.

- The range of interest for advanced engines is 1371° C to 1538° C (2500° F to 2800° F) takeoff T_{41}.

- LPT cooling requirements became dominant at 1538° C (2800° F) takeoff T_{41} and above.

- 1427° C (2600° F) takeoff T_{41} is recommended for initial rating, with growth to 1538° C (2800° F).

- A cycle pressure ratio of 38:1, at altitude, is recommended for initial rating (higher for growth). This choice was made to reduce

* 5,560 km (3,000 nmi) Design Range
* 55% Load Factor; 1,300 km (700 nmi) Mission

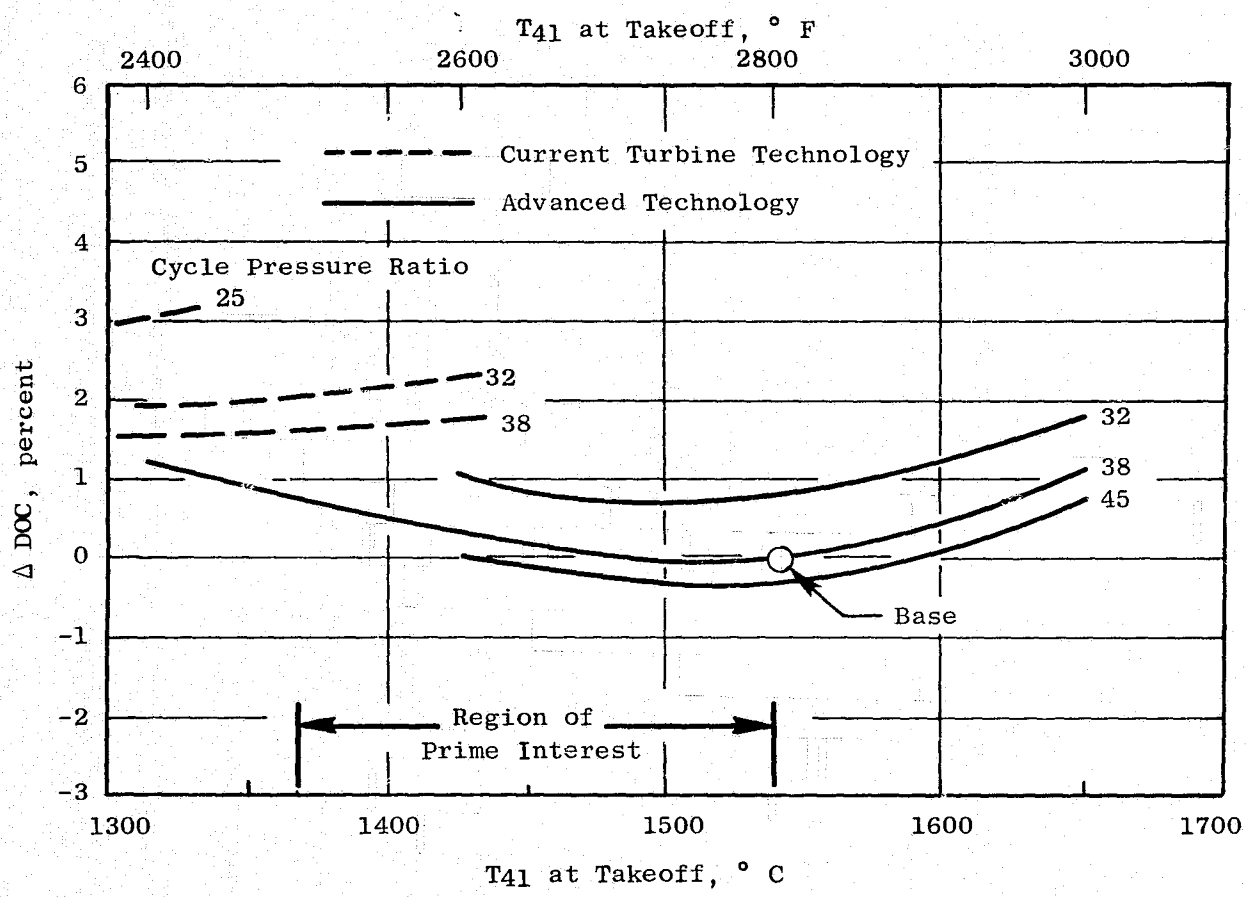


Figure 12. Cycle Selection Study; Trijet DOC Trends.

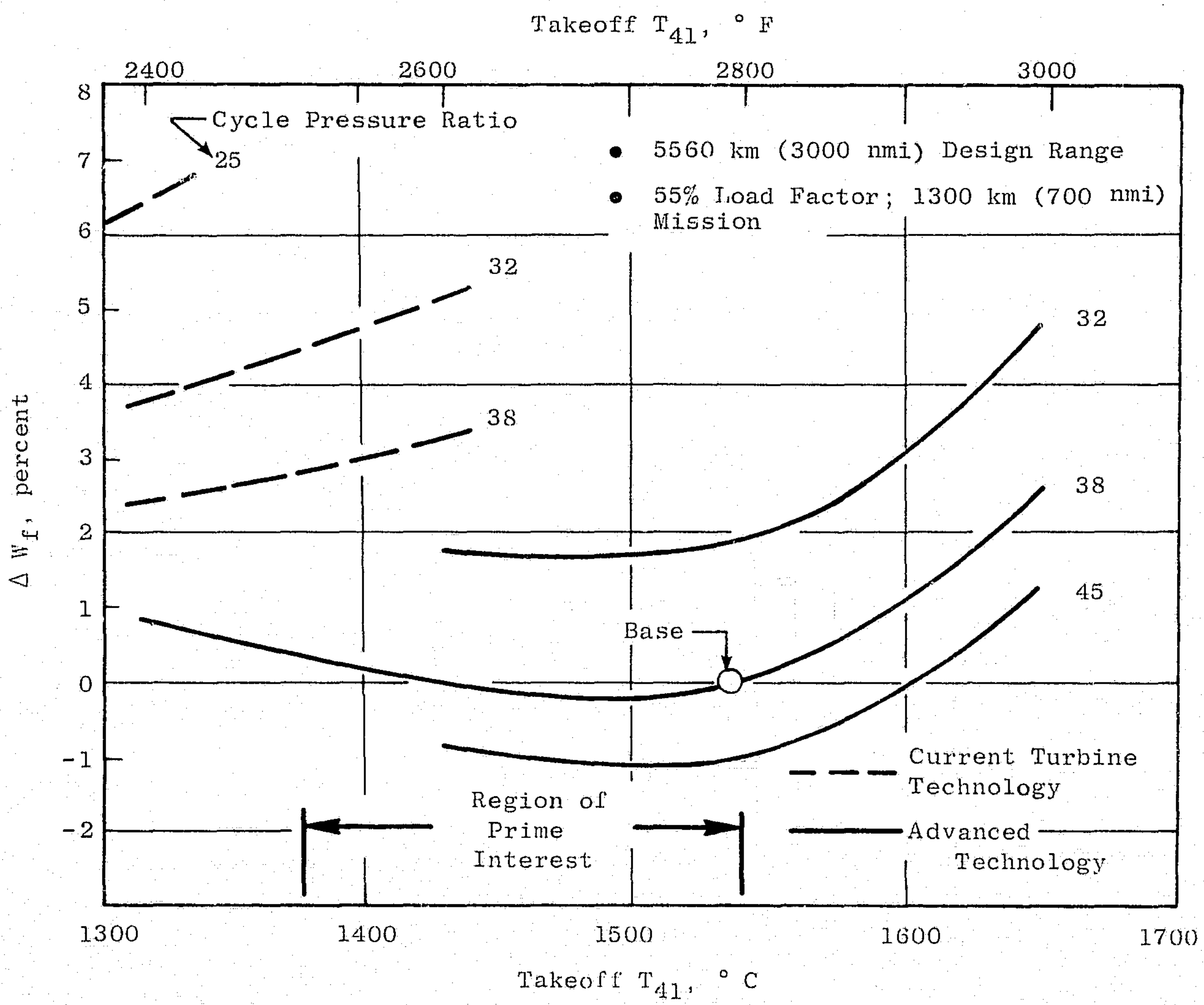


Figure 13. Cycle Selection Study; Trijet Fuel Flow Trends.

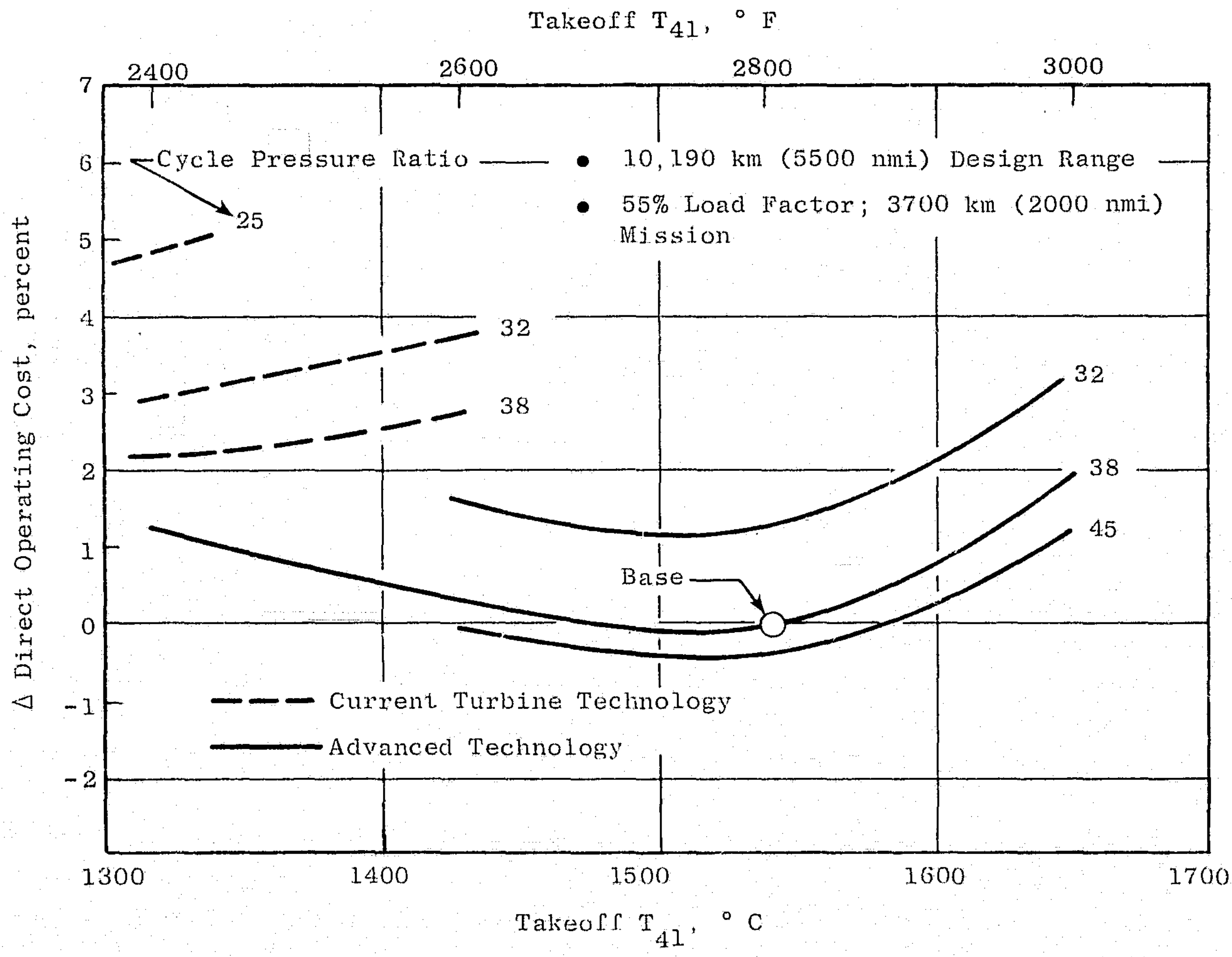


Figure 14. Cycle Selection Study; Quadjet DOC Trends.

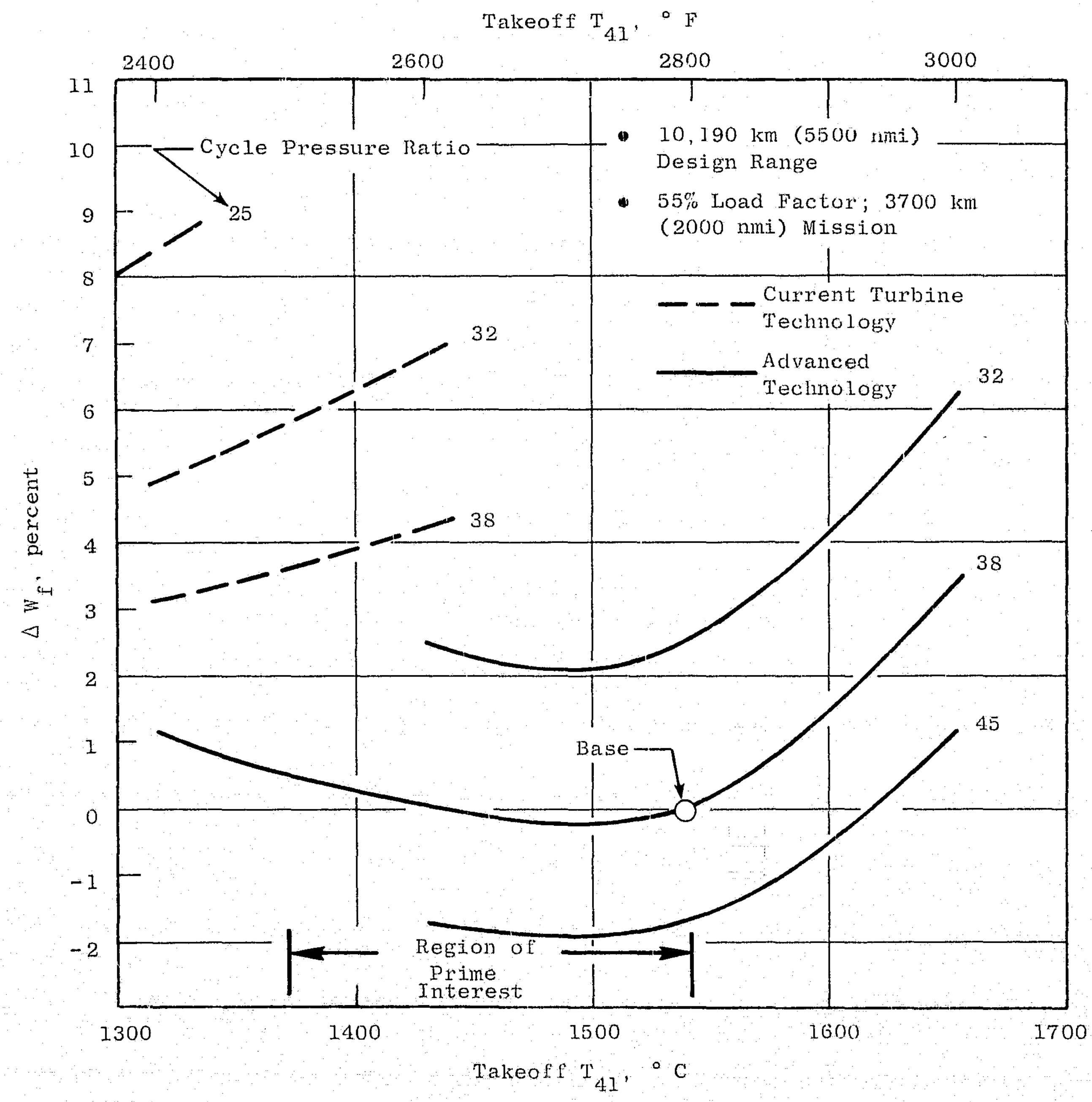


Figure 15. Cycle Selection Study; Quadjet Fuel Flow Trends.

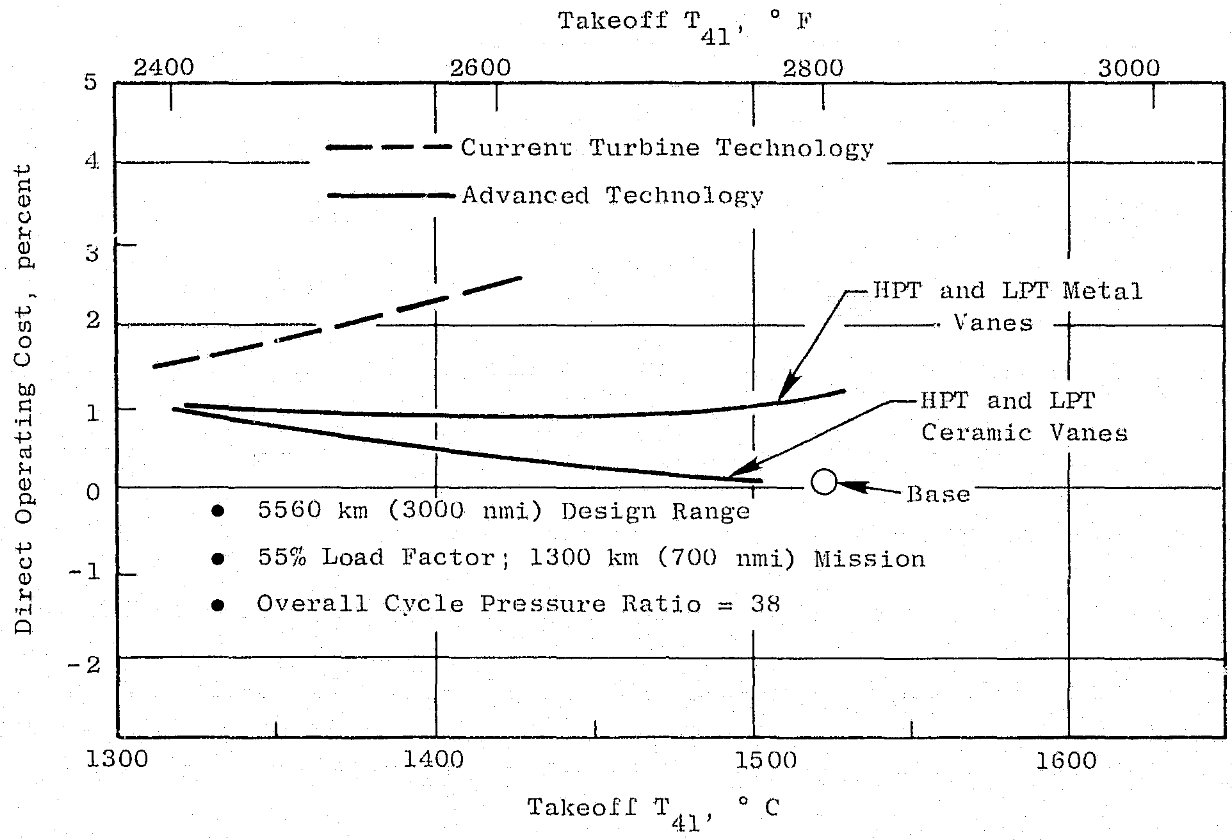


Figure 16. Cycle Selection Study; Effect of Ceramics in HPT and LPT Vanes on DOC Trends.

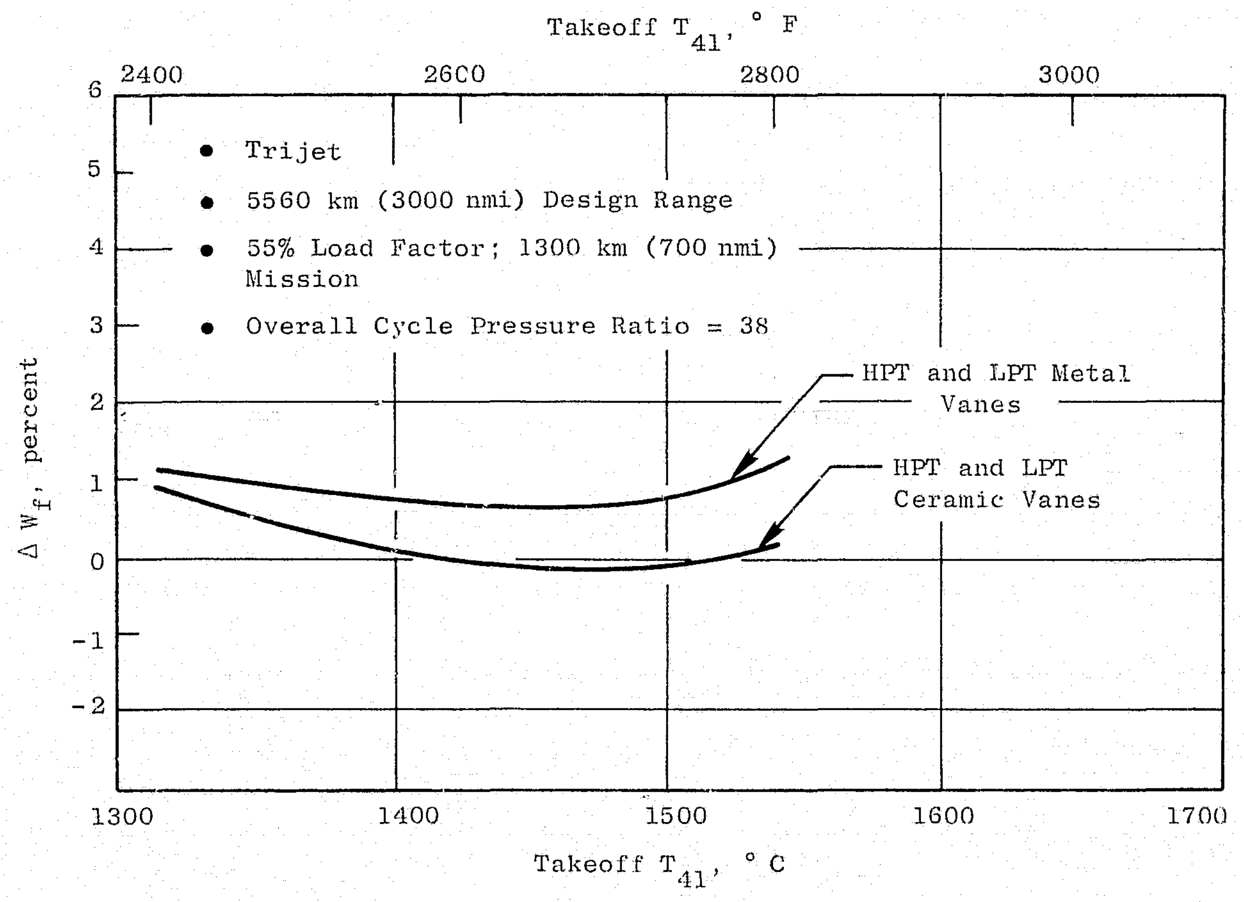


Figure 17. Cycle Selection Study; Effect of Ceramics in HPT and LPT Vanes on Fuel-Used Trends.

the development risk for the initial ratings of the advanced engine and to provide capacity for growth by boosting the core without encountering excessive pressure ratios.

E. Fan Pressure Ratio and Exhaust Type Studies

Separate- and mixed-flow engines were studied over a range of fan pressure ratios from 1.55 to 1.80. For this portion of the study, a 1538° C (2800° F) takeoff turbine inlet temperature, 38.1 maximum-climb cycle pressure ratio and advanced turbine technology were employed. For the separate-flow engines, the primary to fan exhaust jet velocity ratio was initially selected at 1.65; a value representative of modern separate-flow, turbofan engines. In a separate study, the primary stream energy extraction was varied.

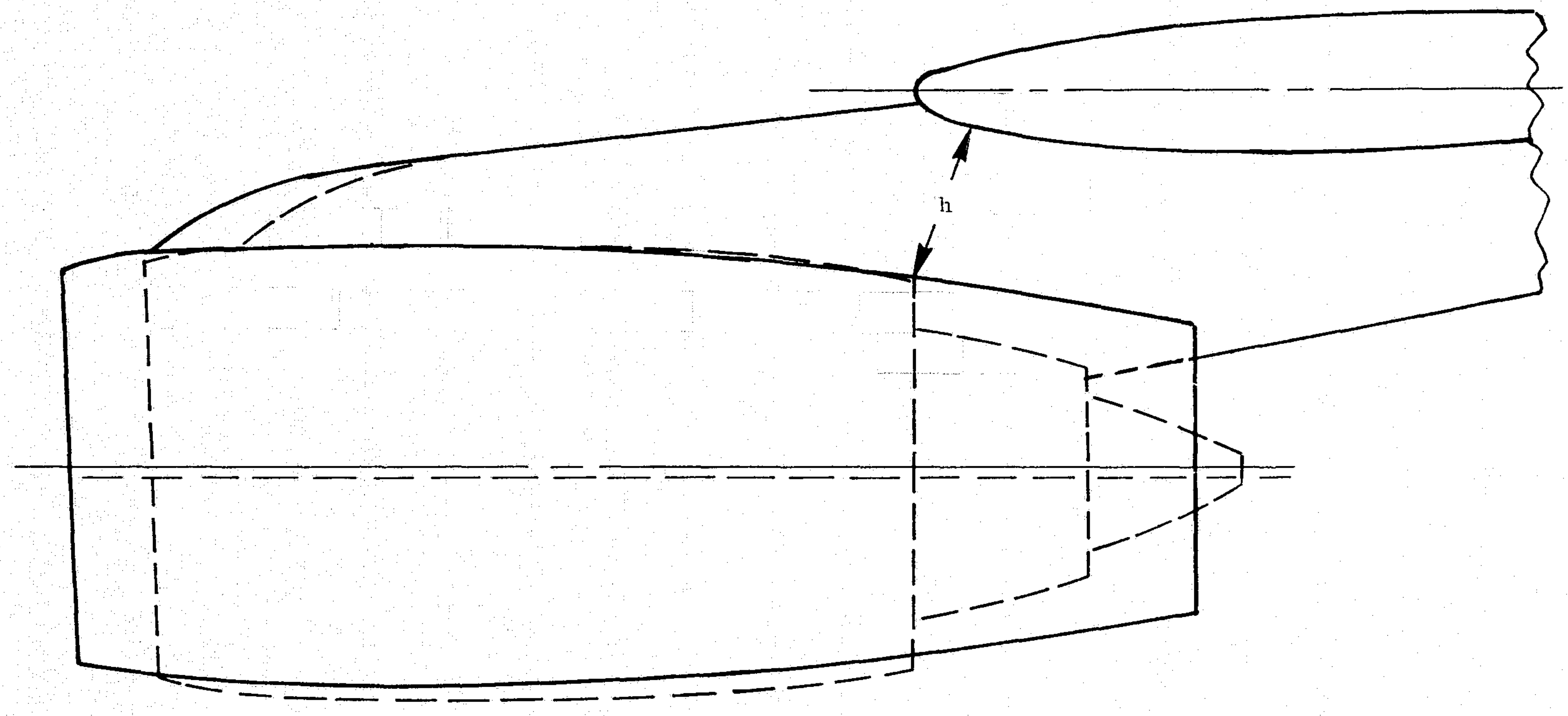
Some of the pertinent cycle parameters are tabulated in Table XXV for the three mixed-flow engines at fan pressure ratios 1.55, 1.65, and 1.80, and for the two separate-flow engines at fan pressure ratios 1.65 and 1.80, with an exhaust velocity ratio of 1.65. The table also shows a separate-flow engine at a fan pressure ratio of 1.76 and velocity ratio of 1.50. That engine had the same specific thrust and fan diameter as the 1.65 fan pressure ratio, mixed-flow, baseline engine but with the extraction selected for minimum fuel usage in a separate-flow cycle.

The relationship of the separate- and mixed-flow nacelles, relative to the wing, is illustrated in Figure 18. The spacing parameter "h" was held constant to achieve negligible interference drag for both installations. Nacelles were defined for each case with approximately the same overall noise level. The effect of the pylon drag, weight, and cost differences between installations was also considered in the study. Installed and bare-engine sfc trends are illustrated in Figure 19 for the engines in Table XXV, except for the separate-flow engine of exhaust velocity ratio 1.5 which is discussed later in this section. Installed weight and price trends for the same engines, derived by consistent procedures, are illustrated in Figure 20.

The resulting DOC and fuel usage trends for the trijet and quadjet are presented in Figures 21 and 22. The curves show that a higher fan pressure ratio yields a lower DOC, but an increase in fuel burned. The thrust lapse rate from takeoff to cruise altitude varied with fan pressure ratio. When the comparison between engines designed at several levels of fan pressure ratio was made, the engines could be scaled to hold takeoff or cruise thrust constant. The results are shown for both engine-sizing assumptions. The cruise-sized comparison favored the higher fan pressure ratio. The trijet mixed-flow installation showed a 4.2% fuel usage and 1.0% DOC advantage versus the separate flow at the baseline fan diameter, as illustrated in Figure 21. Comparison at a constant fan diameter is equivalent to constant specific thrust, since all engines are scaled to the same installed thrust in this chart.

Table XXV. Cycle Definitions, Mixed versus Separate Flow.

5560 km (3000 nmi), Trijet, Cruise Sized, M = 0.80, 10,670 m (35K ft), Fn_I = 20,000 N (4500 lb)						
	Mixed, η_{mix} = 175			Separate		
				V_9/V_{29}=1.65	1.65	1.50
Fan P/P MxCl	1.65	1.55	1.80	1.65	1.80	1.76
MxCr	1.61	1.51	1.75	1.61	1.75	1.74
Takeoff F_n sls, Flat to +15° C, N (+27° F, lb)	89,000 (20,000)	93,000 (20,910)	83,800 (18,830)	91,100 (20,470)	85,600 (19,250)	87,200 (19,613)
MxCl F_n, 0.8/10670 m (35K ft) MxCl Flat to +10° C, N (°18° F, lb)	22,700 (5100)	22,800 (5130)	22,500 (5060)	22,800 (5130)	22,600 (5080)	22,900 (5134)
MxCr +10° C, N (+18° F, lb)	21,000 (4740)	21,200 (4770)	20,900 (4700)	21,200 (4770)	21,000 (4730)	21,200 (4758)
Bypass Ratio MxCr	8.1	9.6	6.5	8.6	6.8	7.7
MxCl Corr, Fan Flow, kg/sec (lb/sec)	337 (742)	387 (853)	283 (625)	367 (809)	305 (673)	337 (742)
Fan Diameter, m (in.)	1.54 (60.6)	1.65 (65.0)	1.41 (55.6)	1.61 (63.3)	1.47 (57.7)	1.54 (60.6)
No. Booster Stages	3	4	2	3	2	2
No. LP Turbine Stages	5-1/2	6-1/2	4-1/2	5-1/2	4-1/2	5-1/2



<u>Assumpions/Constraints</u>

- h Held Constant
- Comparison at 1.65 and 1.80 Fan Pressure Ratio
- Same Level of Advanced Component Technology for Both Mixed and Separate Flow
- Separate Flow Velocity Ratio Set at 1.4 (SLS) and 1.65 at Maximum Cruise
- Both Separate and Mixed Flow Engines Designed for Similar Noise Levels
- Fan Reverser Only

Figure 18. Separate Versus Mixed Flow.

- 10,670 m/0.80 Mach/+10° C (35,000 ft/0.80 Mach/+18° F) 95% MxCr
- Constant Turbine Temperature and Cycle Pressure Ratio

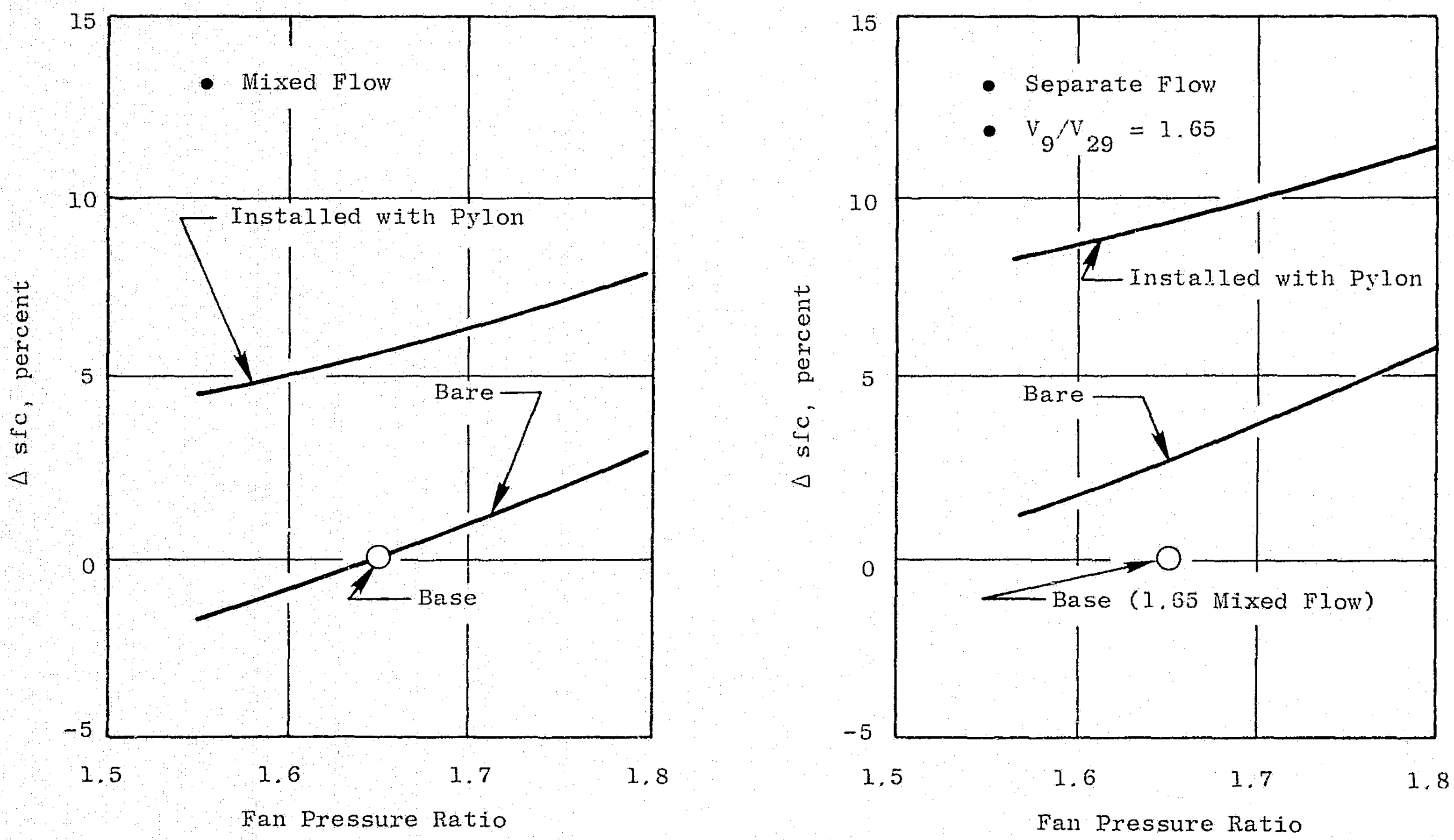


Figure 19. Bare and Installed sfc Trends Versus Fan Pressure Ratio.

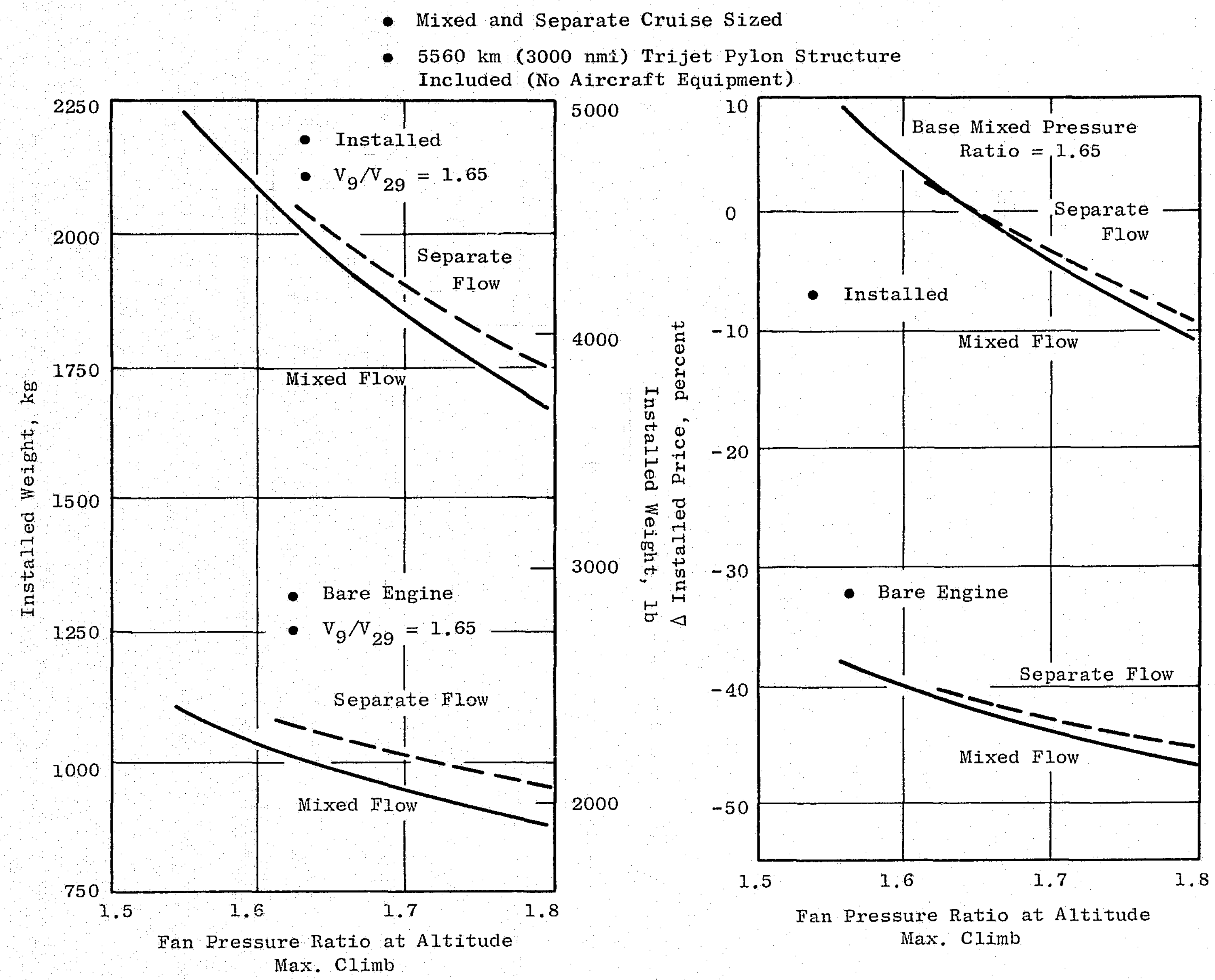


Figure 20. Installed Engine Weight and Price Trends.

- 55% Load Factor, 1300 km (700 nmi) Mission
- 10,670 m/0.80 Mach No./+10° C 95% MxCr
  (35,000 ft)/0.80 Mach No./+18° F 95% MxCr
- Fan Pressure Ratios (P/P) = 1.55, 1.65 and 1.80

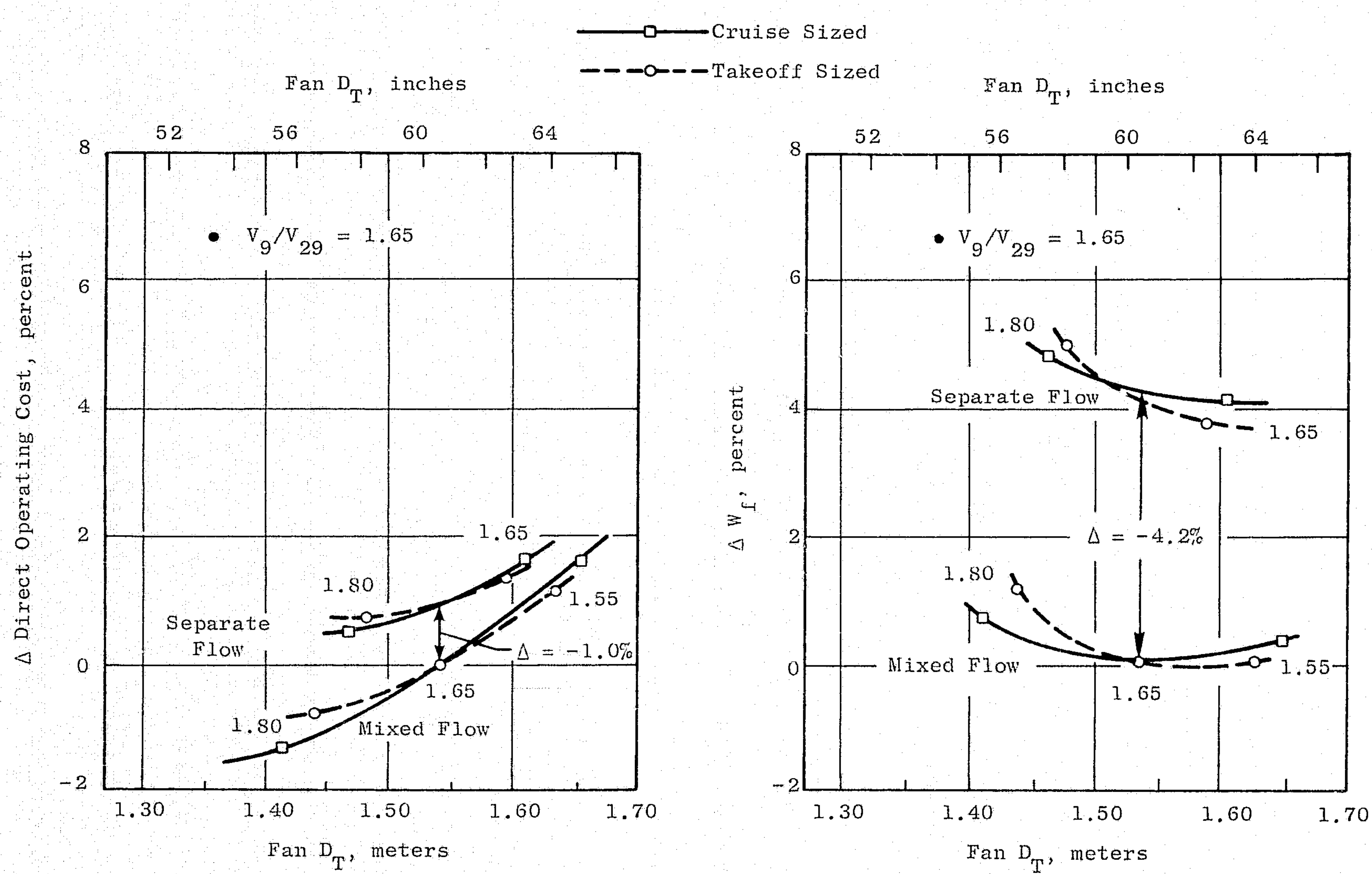


Figure 21. Separate Versus Mixed Flow, 5560 km (3,000 nmi) Trijet Mission.

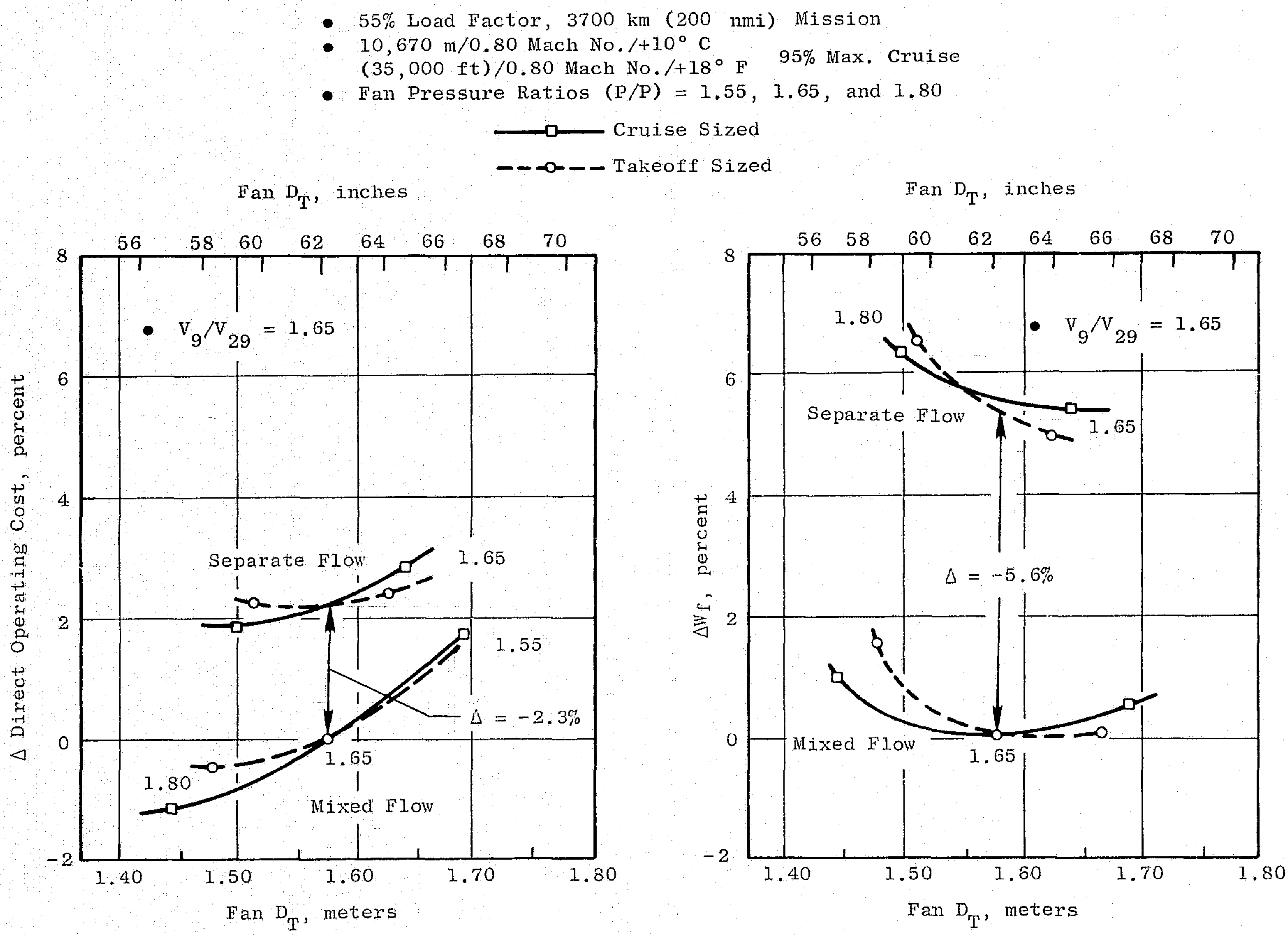


Figure 22. Separate Versus Mixed Flow, 10,190 km (5,500 nmi) Quadjet Mission.

The primary to fan exhaust jet velocity ratio was initially set at 1.65 at the M = 0.8, 10,670 m (35K ft) maximum climb design point for the separate-flow cycles discussed above. This velocity ratio was varied down to 1.1 (higher primary-stream extraction) for the separate-flow engines. The results are presented in Figures 23 and 24 for a fan pressure ratio of 1.65 and 1.80.

The best exhaust velocity ratio for minimum fuel consumed and DOC was approximately 1.5, when compared at constant fan diameter or specific thrust. For higher-extraction, separate-flow cycles, an additional low pressure turbine stage was required. There was a range of velocity ratios where either a four- or five-stage turbine, for fan pressure ratio 1.80, or a five- or six-stage turbine, for pressure ratio 1.65, could be used.

There was a significant turbine efficiency improvement when the four-stage LPT was replaced with a five-stage LPT, or the five-stage was replaced by a six-stage LPT. The resulting sfc improvement yielded lower mission fuel consumption for the same exhaust velocity ratio, as illustrated in Figure 24. Except for those higher-extraction, separate-flow engines where increased turbine staging was required, component efficiency differences between the mixed- and separate-flow engines were a minor part of the sfc differences.

For persepective, the two levels of core extraction are superposed on Figure 25. Compared to the best high-extraction, separate-flow cycle, with a five-stage LPT, the mixed-flow DOC and fuel-consumed advantage was reduced to 0.8% and 2.6% respectively at the baseline specific thrust. The distribution of DOC and fuel-consumed gains are presented in Table XXVI. The largest portion of the gain was due to sfc improvement.

To examine the case where interference drag for the mixed-flow nacelle might be higher than the separate-flow nacelle, in the nominal axial position illustrated in Figure 26, a possible solution may be to reposition the mixed-flow nacelle further forward of the wing. If the mixed-flow exhaust plane is relocated in the plane of the wing leading edge, the fuel saving for mixed versus separate flow will be reduced from 2.6% to 2.1%. The reduction in fuel saved will be due to the weight and drag (noninterference) increase of the longer pylon necessary to support the nacelle. Alternately, if the sensitivity to an interference drag difference between mixed- and separate-flow nacelle (in the nominal position) is examined, the effect of a 1% (of installed cruise thrust) drag increase reduces the fuel saved from 2.6% to 1.6%, as illustrated in Figure 27. Figure 28 illustrates the sensitivity to DOC and fuel saved for an increase in aircraft structural weight associated with moving the nacelle of the mixed-flow engine to a more forward location. If the increased moment arm of the installation results in a 136 kg (300 lb) wing weight increase per engine, the mixed-flow fuel saving will be reduced from 2.6% to 1.9%, while the 0.8% DOC saving will be reduced to 0.1%.

The following are observations and conclusions from the fan P/P (pressure ratio) and exhaust-type study:

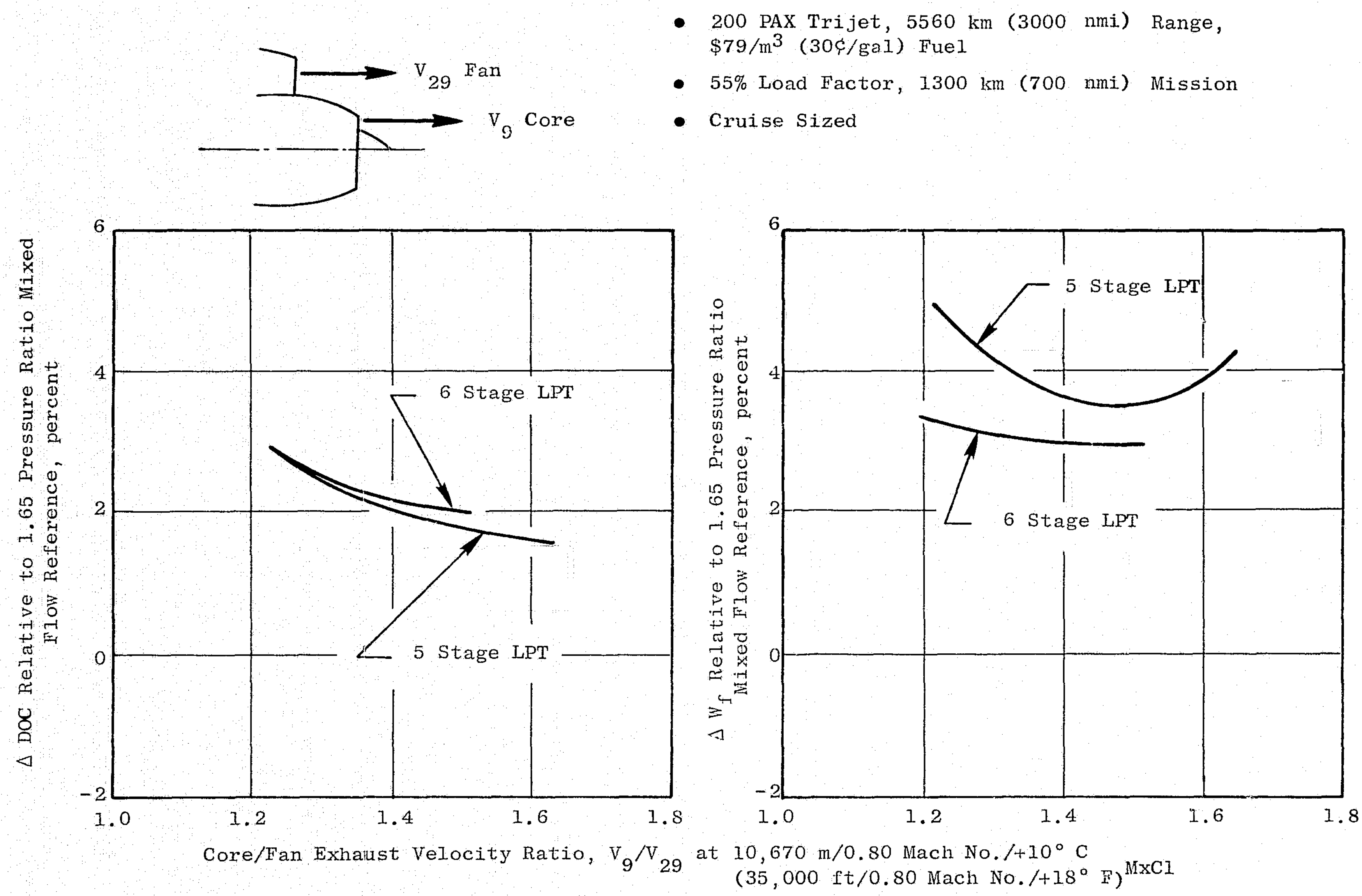


Figure 23. Core Extraction Study, Effect of Exhaust Velocity Ratio on 1.65 Pressure Ratio - Separate Flow.

- 200 PAX Trijet, 5560 km (3000 nmi) Range, \$79/m^3 (30¢/gal) Fuel
- 55% Load Factor, 1300 km (700 nmi) Mission
- Cruise Sized

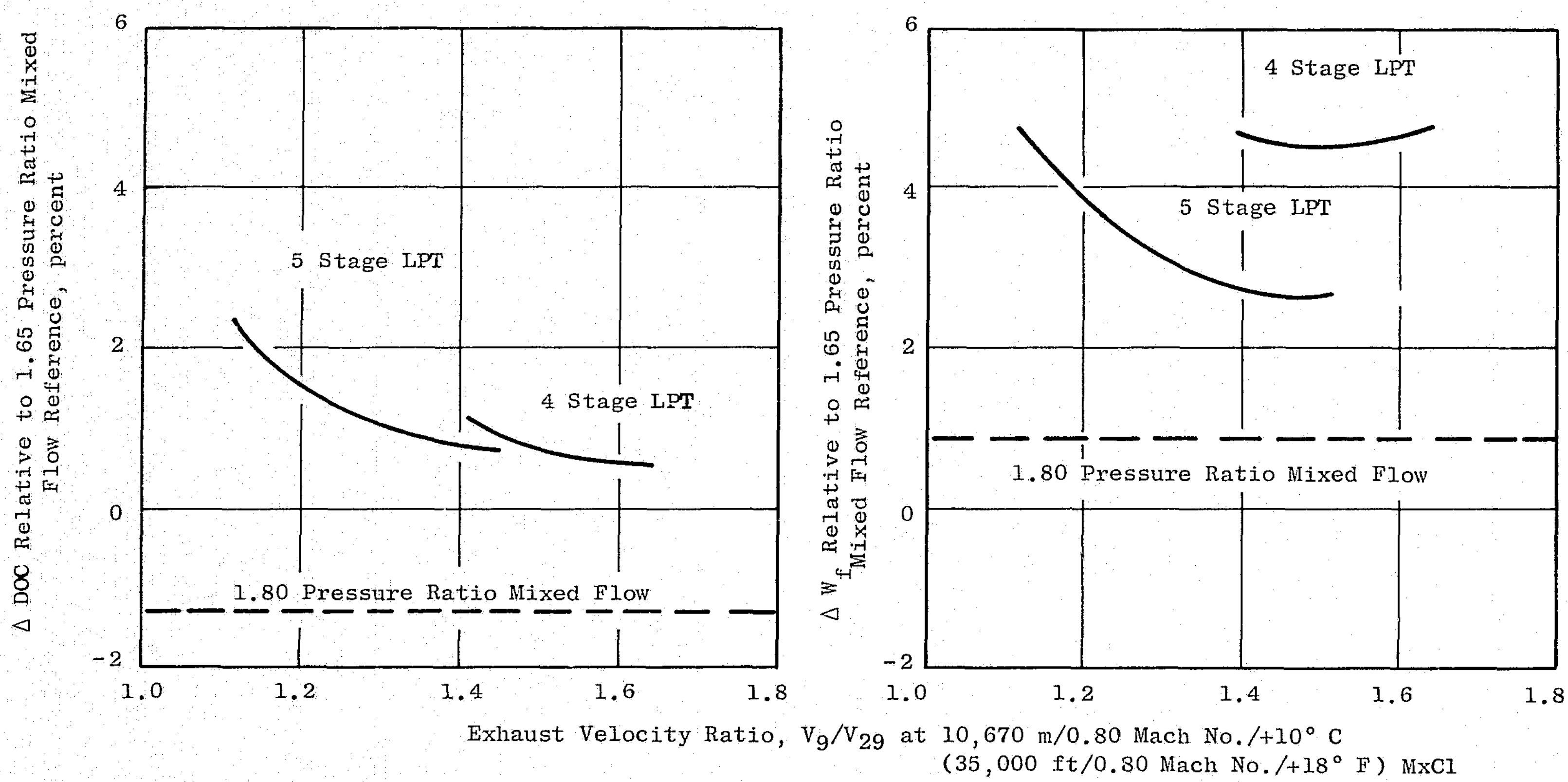


Figure 24. Core Extraction Study, Effect of Exhaust Velocity Ratio on 1.80 Pressure Ratio - Separate Flow.

- 200 PAX Trijet/5560 km (3000 nmi) Range
  \$79/m^3 (30¢/gal) Fuel
- 55% Load Factor, 1300 km (700 nmi) Mission
- Cruise Sized

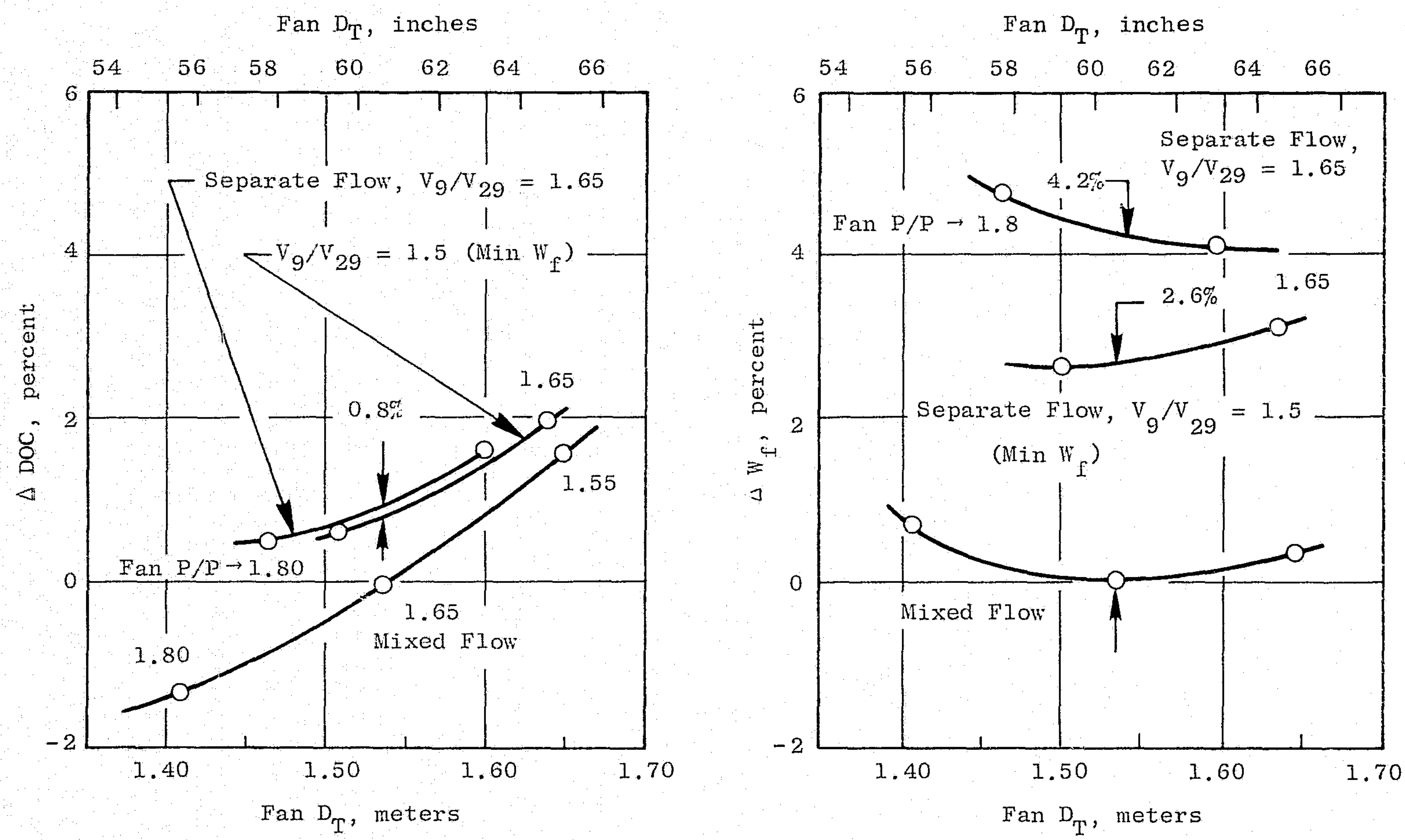


Figure 25. Core Extraction Study, Comparison of Separate Versus Mixed Flow.

Table XXVI. Engine Evaluation, Mixed versus Separate Flow.*

Fan Diameter = 1.54 m (60.6 inches) Trijet A/C Cruise Sized, 55% Load Factor, 1300 km (700 nmi) Mission 0.8 Mn/10670 m (35K ft)				
			Δ DOC %	ΔW_f %
Fan Pressure Ratio	1.65 Mixed	1.76 Separate		
Velocity Ratio	-	1.50		
Δ η_{mix} %	75	-		
Δ Engine Weight, kg (lb)	-54 (-118)		-0.2	-0.3
Δ Nacelle Weight, kg (lb)	+88 (+195)		+0.3	+0.5
Δ Engine Price, 1000 $	+4		0	-
Δ Nacelle Price, 1000 $	+25		+0.2	-
Δ sfc Bare, % 95% Max. Cruise	-2.5		-0.10	-2.7
Δ Drag, % F_n	-0.1		-0.1	-0.10
Total			-0.8	-2.6

*High extraction version of separate flow, V_9/V_{29} = 1.5

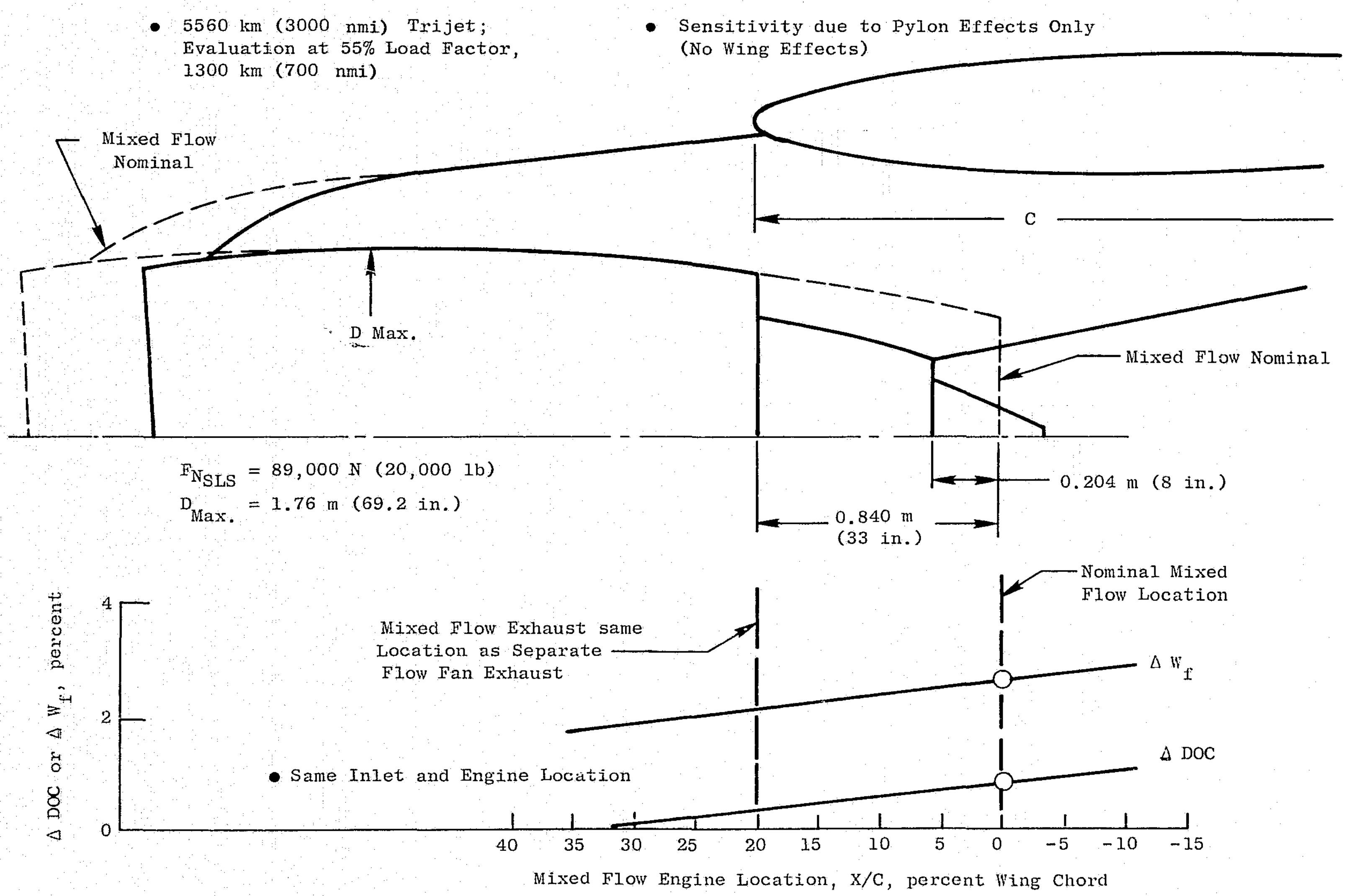


Figure 26. Separate Versus Mixed-Flow Sensitivity to Engine Location.

- 5560 km (3000 nmi); Trijet, Mission
- 55% Load Factor, 1300 km (700 nmi) Mission
- 10,670 m/0.80 Mach No./+10° C (35,000 ft/0.80 Mach No./+18° F) 95% Max. Cruise

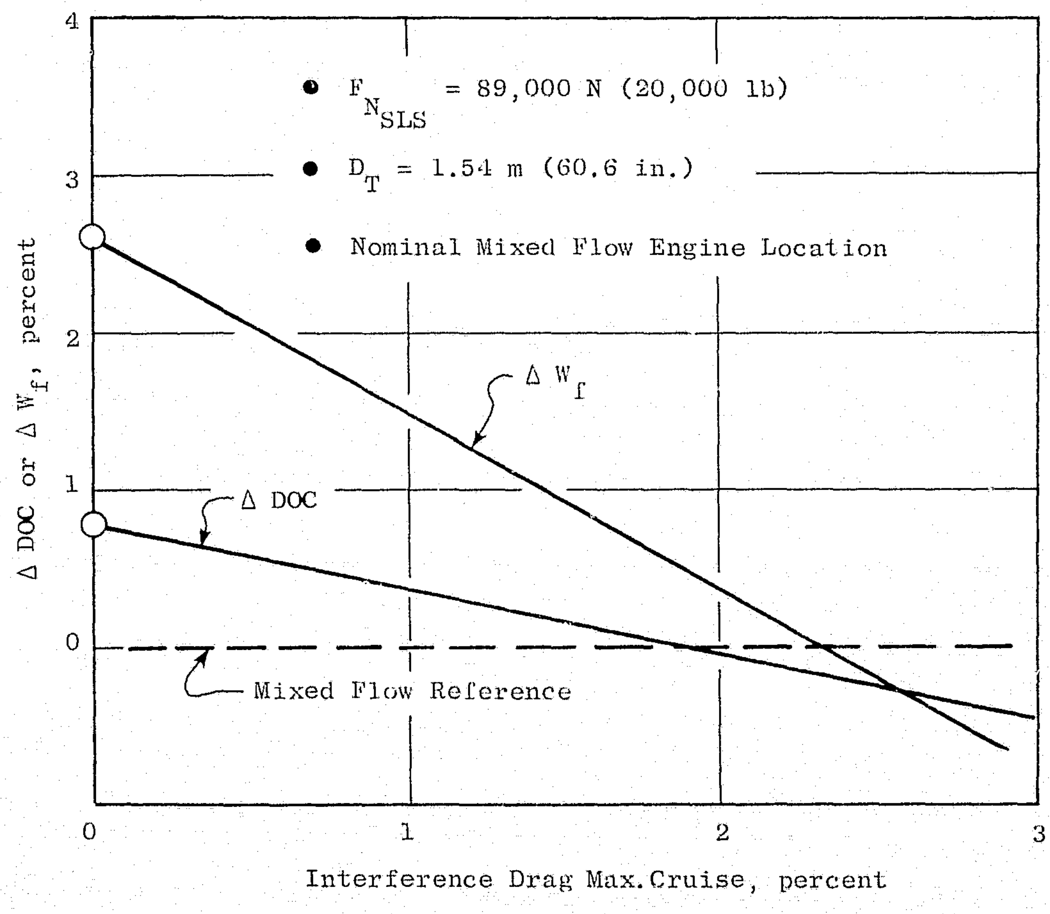


Figure 27. Separate Versus Mixed Flow Sensitivity to Interface Drag.

- 5560 km (3000 nmi) ; Trijet, Mission
- 55% Load Factor, 1300 km (700 nmi) Mission
- 10,670 m/0.80 Mach No./+10° C (35,000 ft/0.80 Mach No./+18° F) 95% Max. Cruise

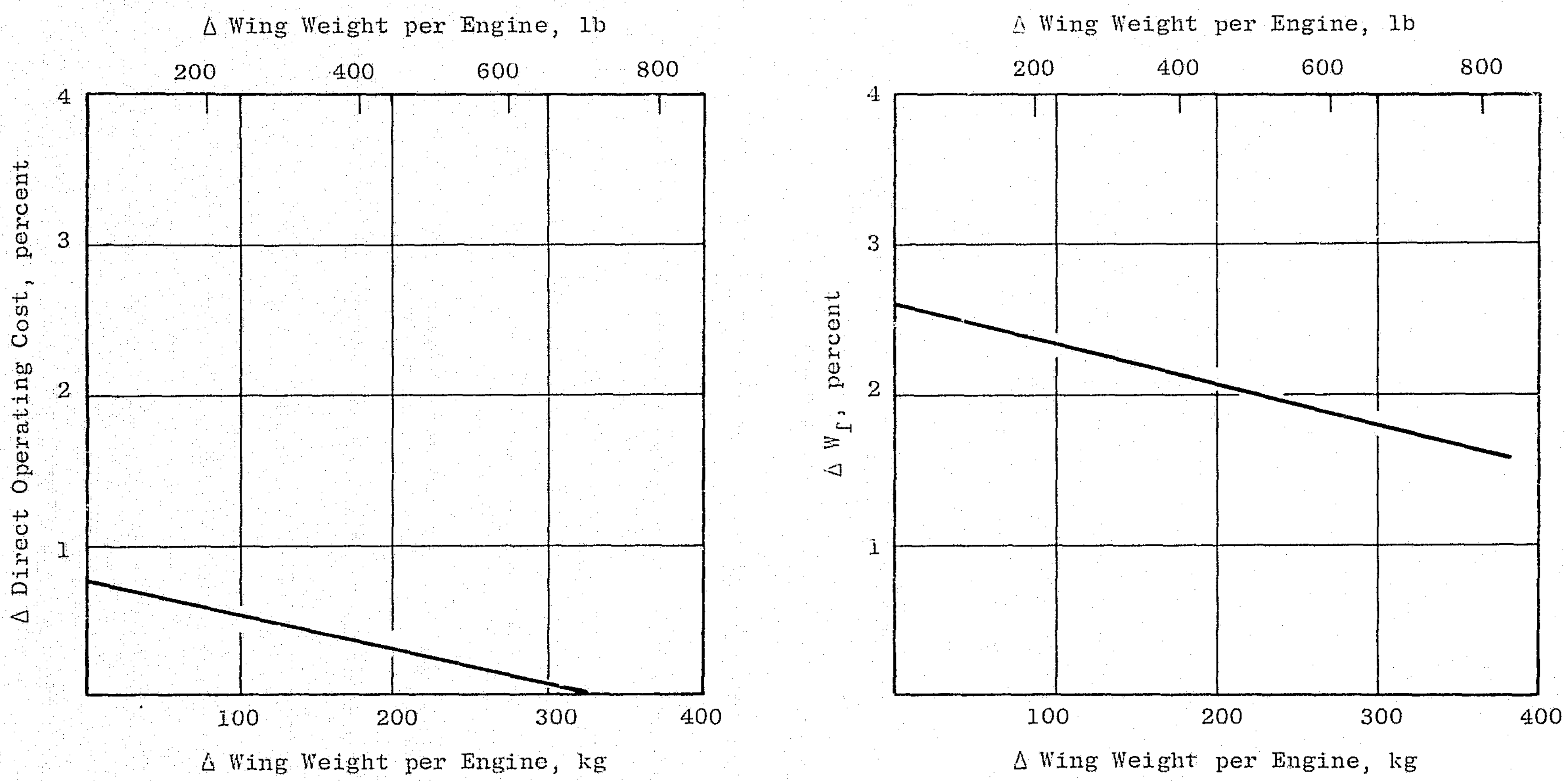


Figure 28. Separate Versus Mixed Flow Sensitivity to Wing Weight.

- A higher fan pressure ratio (up to 1.8) yielded improved DOC and a small increase in fuel usage.

- A fan pressure ratio of 1.7 is recommended for initial rating (increase with growth).

- Mixed flow has potential of major payoff in DOC and fuel usage. For example, improvements of 2.6% in fuel and 0.8% in DOC in the trijet average mission were estimated.

- The sfc advantage of mixed exhaust was the dominant factor in the comparison to separate flow. For instance, the base sfc contributed a 2.7% fuel reduction. The effect of installed weight and drag only slightly altered this result to 2.6%.

- The potential interference-drag problem requires attention, but the mixed-flow sfc advantage appears large enought to overcome possible penalties associated with different nacelle location.

F. Basic Engine Technology Evaluation

The use of composites in the fan blade was considered, in combination with an advanced high tip speed aerodynamic design. It was compared to a current-technology design for reference, and also to a titanium, high-speed, advanced aerodynamic design in order to evaluate the material effect by itself. The design features are presented in Table XXVII and the resulting weight, price, sfc changes, DOC, and fuel-usage benefits are listed in Table XXVIII. The high tip speed, advanced aerodynamic benefits were about half the total benefit of the advanced fan design with composites contributing the remainder. The sensitivity to variations in design input, such as blade aspect ratio, composite cost level, and fan efficiency differences, are illustrated in Figures 29, 30 and 31.

The substitution of a composite fan-frame design for a conventional metal frame was also evaluated. The results are presented in Table XXVIII.

An improved compressor clearance control design was substituted for the current technology design and a comparison of the two was made. The design utilized a casing material with a low thermal expansion coefficient and casing thermal insulation and cooling features. The advanced design allowed lower running clearances for a criteria of no significant rubs during engine transients. The improvements resulting from the compressor clearance reduction are illustrated in Table XXIX.

Improved high pressure turbine clearances were obtained by the application of an on-off, HPT shroud-cooling system in this study. The shroud support diameter remained small during steady-state operation with cooling on. The cooling flow was turned off during engine transients, causing the casing to run hotter than would otherwise have been the case, and therefore maintained adequate clearances to prevent rubs. The benefits obtained are presented in Table XXIX.

Table XXVII. Advanced Fan Design Features.

	Reference Midspan Shroud CF6-50 Type	High Speed Tip Shrouded	High Speed Composite
Number of Blades	38	44	30
Material	Ti	Ti	Composite-Hybrid
To F_n, N (lb)	88,960 (20,000)	88,960 (20,000)	88,960 (20,000)
Corrected Flow, kg/sec (lb/sec)	337 (743)	338 (745)	337 (744)
$U_T\sqrt{\theta}$, m/sec (ft/sec)	421 (1380)	488 (1600)	488 (1600)
Performance Effects for Constant Tip Clearance			
Δ Fan Efficiency, %	Base	+0.8	+1.1
Δ Fan Hub Efficiency, % (Includes Boosters)	Base	+0.3	-0.7
Staging			
N_{Boost}	4	3	3
Core	Same	Same	Same
N_{LPT}	6	5	5

Table XXVIII. Payoff Summary of Higher Fan Tip Speed and Advanced Materials.

5560 km (3000 nmi) Trijet Evaluation at 55% Load Factor, 1300 km (700 nmi)							
	Technology Description		Installed →				
Application	Advanced	Reference	Δ Weight kg (lb)**	Δ Price $1000**	Δ sfc %	Δ DOC %	Δ Fuel Used %
Fan Blade	Composite High Tip Speed	CF6 Type Fan	-118 (-260)	-46	-0.75	-1.38 (-1.85)	-1.49 (-1.95)*
Fan Blade	Ti High Tip Speed	CF6 Type Fan	+2.3 (+5)	-27	-0.88	-0.71 (-1.01)	-0.95 (-1.26)*
Fan Frame/Case	Composite Integrated with Nacelle	Steel Conventional Construction	-70 (-155)	-21	0	-0.42 (-0.53)	-0.44 (-0.52)*

*10,190 km (5500 nmi) Quadjet

**Includes cycle effect

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• Base: Composite High Speed Fan Vs. Reference: 38-Bladed Midspan Shrouded Fan

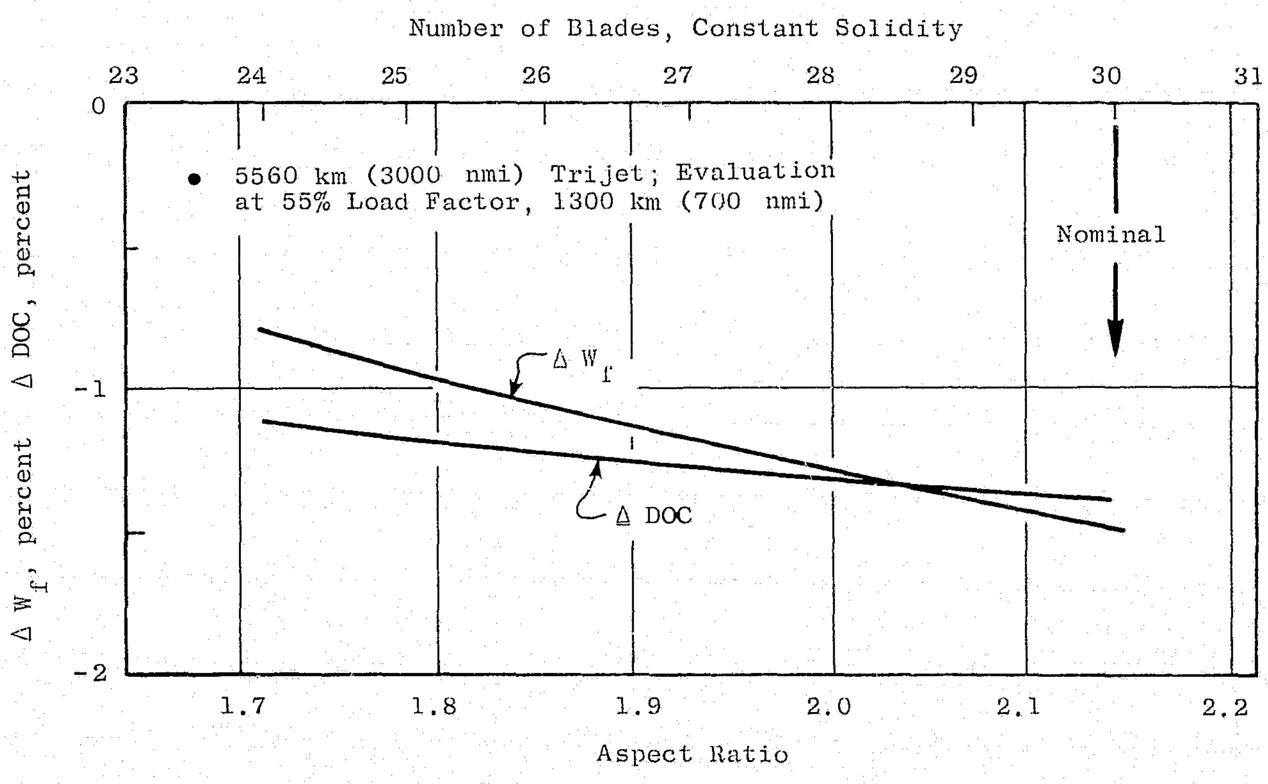

Figure 29. High Tip Speed Composite Fan Sensitivity Analysis; Aspect Ratio.

• Base: Composite High Speed Fan Vs. Reference: 38-Bladed Midspan Shroud Fan

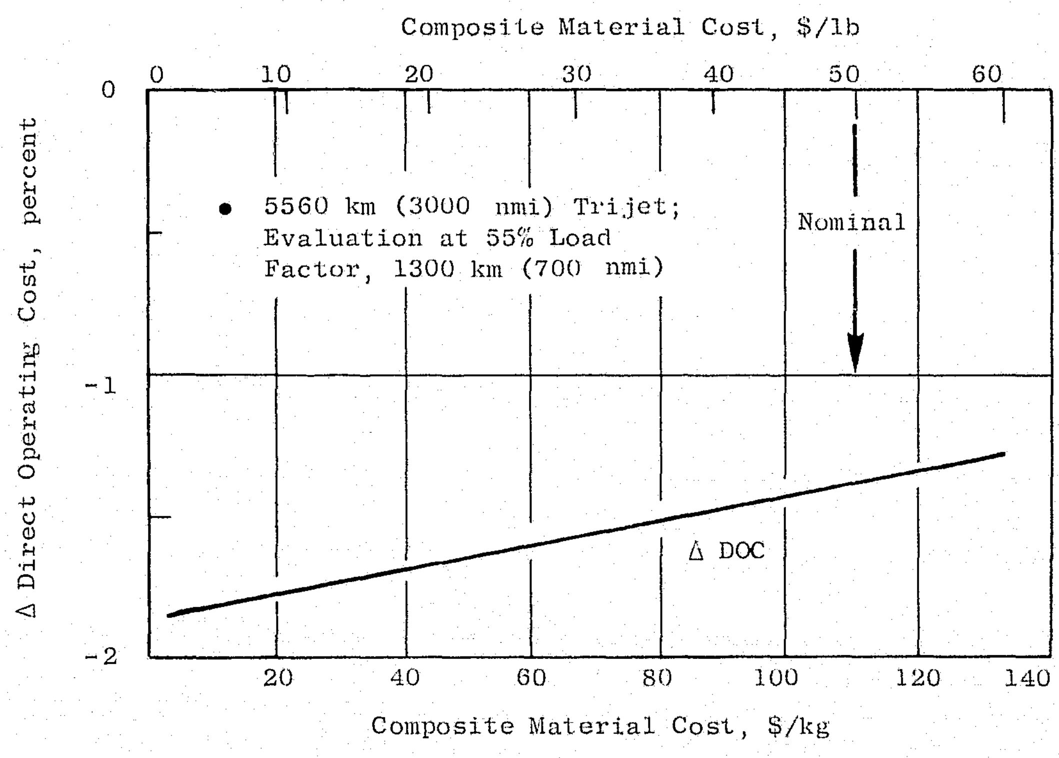

Figure 30. High Tip Speed Composite Fan Sensitivity Analysis; Composite Material Cost.

Base		Reference
Composite High Speed Fan	Vs.	38-Bladed Midspan Shrouded Fan

- 5560 km (3000 nmi) Trijet; Evaluation at 55% Load Factor, 1300 km (700 nmi)

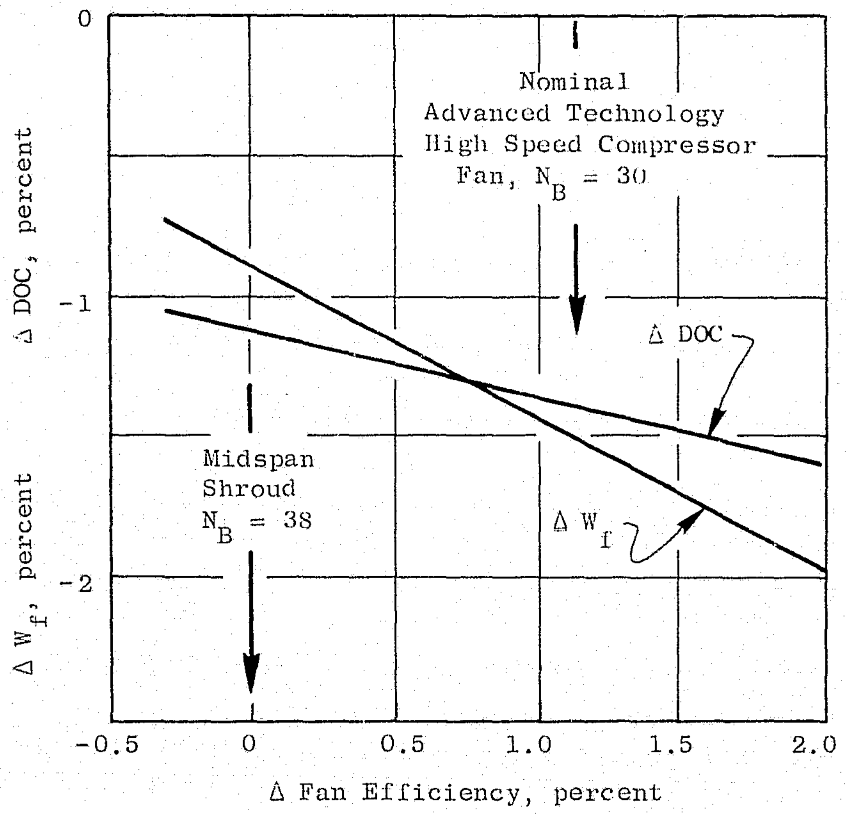

Figure 31. High Tip Speed Composite Fan Sensitivity Analysis; Fan Efficiency.

Table XXIX. Clearance Control Payoff.

5560 km (3000 nmi) Trijet Evaluation at 55% Load Factor, 1300 km (700 nmi)

Application	Technology Description		Installed →				
	Advanced	Reference	Δ Weight kg (lb)**	Δ Price $1000**	Δ sfc %	Δ DOC %	Δ W_f %
Compressor Clearance Control	Inco 903 Cooled Casing Insulated	Inco 718 Uncooled Casing Uninsulated	-5.9 (-13)	0	-0.32	-0.15 (-0.26)	-0.38 (-0.50)*
HPT Clearance Control	Active Cooling System	Passive Cooling Sys	-6.8 (-15)	+4	-1.0	-0.39 (-0.73)	-1.13 (-1.49)*
HPT Cooling Supply	Bore Entry	Rim Entry	-4.5 (-10)	-2	-0.06	-0.06 (-0.09)	-0.09 (-0.12)*

*10,190 km (5500 nmi) Quadjet

**Includes cycle effects

Supplying the HPT blade-cooling flow from the compressor midstage to HPT rotor hub (bore entry) resulted in a small advantage, as listed in Table XXIX.

The use of ceramics was evaluated in the HPT and LPT vanes and HPT shrouds. The results (Table XXX) indicated a net benefit for all applications. However, the feasibility of ceramic vanes from an impact-damage standpoint has not yet been established. For this reason, ceramic vanes were not used in the Task III design, reserving them for later engine growth. Figure 32 illustrates the impact of changes in parts usage and cost level on the benefits of ceramic vanes. Ceramic shrouds were used in the Task III design since shrouds do not have the same impact problem as vanes (They are not out in the flow path nor in front of rotating parts). Figure 33 illustrates the effect of changes in ceramic shroud cost and parts usage.

The use of eutectic turbine-blade materials in the HPT and LPT resulted in a fuel saving, but also a DOC increase (due to the high production costs of eutectic blades) as presented in Table XXXI. The use of full internal impingement cooling with multiple inserts, relative to the simpler monolithic film-cooled design, resulted in a small fuel saving, but a substantial DOC increase (again due to production cost increase of the blades). A brief summary of the conclusions and observations from the basic engine technology studies are listed below:

- Composite fan blades; a high potential payoff area:
    - Survivability of high tip speed composite fan blades to bird strike not established.
    - Safety (containment) an additional advantage beyond DOC.
- Composite frame/case integrated with nacelle; a major weight and cost reduction item.
- Core compressor clearance control; will be applied in any new engine design.
- Active (on-off cooling) HPT clearance control; worth pursuing.
- Bore-entry cooling; modest payoff, but considered good design approach.
- Ceramics; high potential payoff area
    - Feasibility from impact-damage standpoint not established.
    - Shrouds would be first area of application
    - Uncooled LPT ceramic vanes would be next area of attention.
- Cost of eutectic turbine-blade material; barrier to its use.

Table XXX. Ceramic Materials Payoff.

5560 km (3000 nmi) Trijet, Evaluation at 55% Load Factor, 1300 km (700 nmi)							
	Technology Description		Installed →				
Application	Advanced	Reference	Δ Weight kg (lb)**	Δ Price $1000**	Δ sfc %	Δ DOC %	Δ W_f %
LPT Vane	Ceramic	Mar M 509 Metal	-35 (-77)	-16	-0.66	-0.69 (-0.98)	-0.91 (-1.20)*
HPT Vane	Ceramic	MA 754	0	-5	-0.15	-0.31 (-0.39)	-0.18 (-0.23)*
HPT Shroud	Ceramic	Poroloy	-8 (-18)	-8	-0.24	-0.53 (-0.67)	-0.31 (-0.41)*

*10,190 km (5500 nmi) Quadjet
**Includes cycle effects

Table XXXI. Advanced Materials and Cooling Payoff.

5560 km (3000 nmi) Trijet, Evaluation at 55% Load Factor, 1300 km (700 nmi)							
	Technology Description		Installed →				
Application	Advanced	Reference	Δ Weight kg (lb)**	Δ Price $1000**	Δ sfc %	Δ DOC %	Δ W_f %
HPT Blade	Eutectic	Advanced Nickel-Base DS Casting	-12 (-26)	+10	-0.16	+0.34 (+0.33)	-0.24 (-0.32)*
LPT Blade	Eutectic	Advanced Nickel-Base DS Casting	-22 (-48)	+14	-0.60	+0.10 (-0.07)	-0.78 (-1.03)*
HPT Blade Cooling	Film and Full Impingement	Film Cooling, Cold Bridge	+8 (+17)	+14	-0.11	+0.49 (+0.51)	-0.17 (-0.22)*

*10,190 km (5500 nmi) Quadjet
**Includes cycle effects

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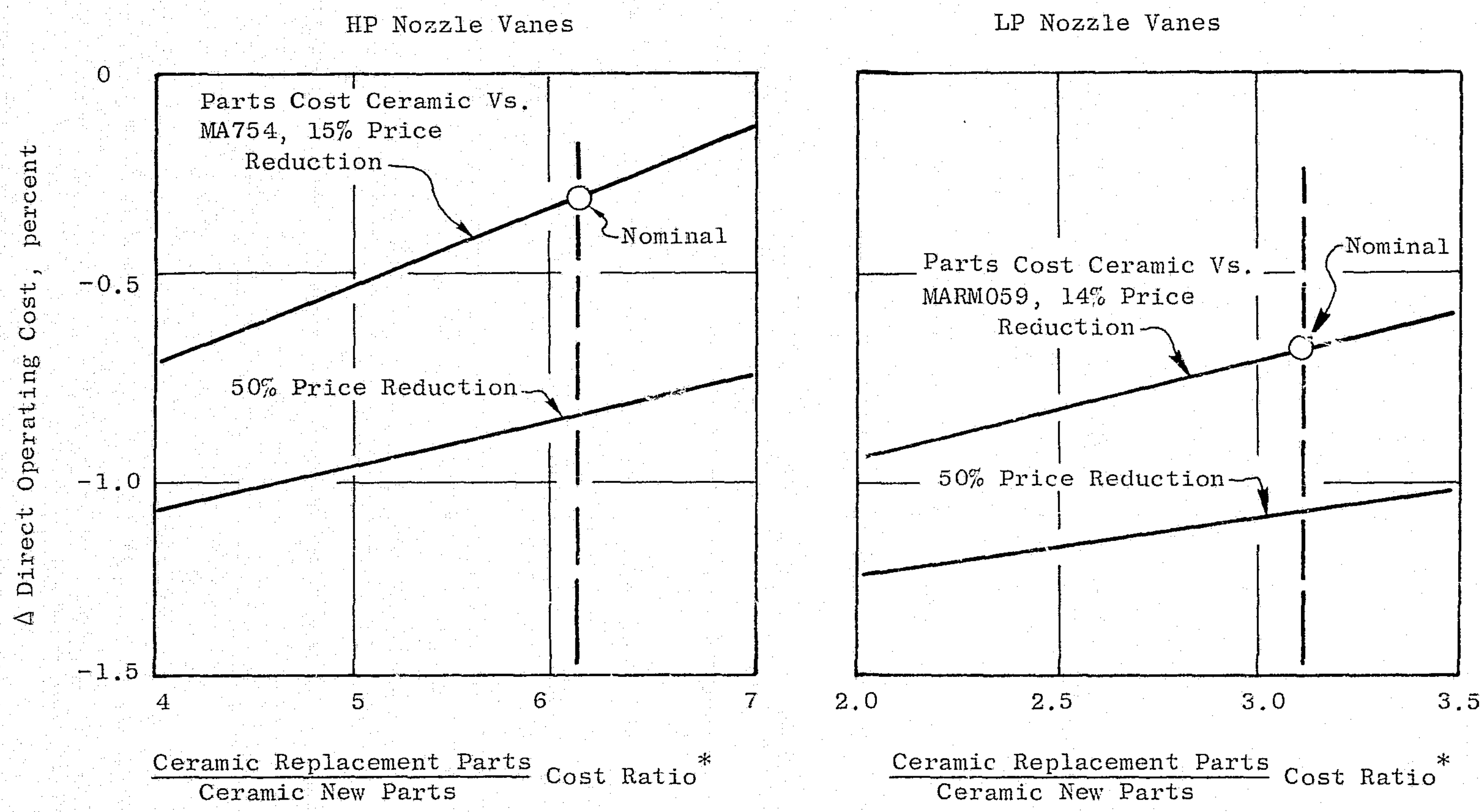
- 5560 km (3000 nmi) Trijet; Evaluation at 55% Load Factor, 1300 km (700 nmi)



* Indincates the frequency of required ceramic part change in 36,000 hours engine life

Figure 32. Sensitivity Analysis; Replacement Parts Rate.

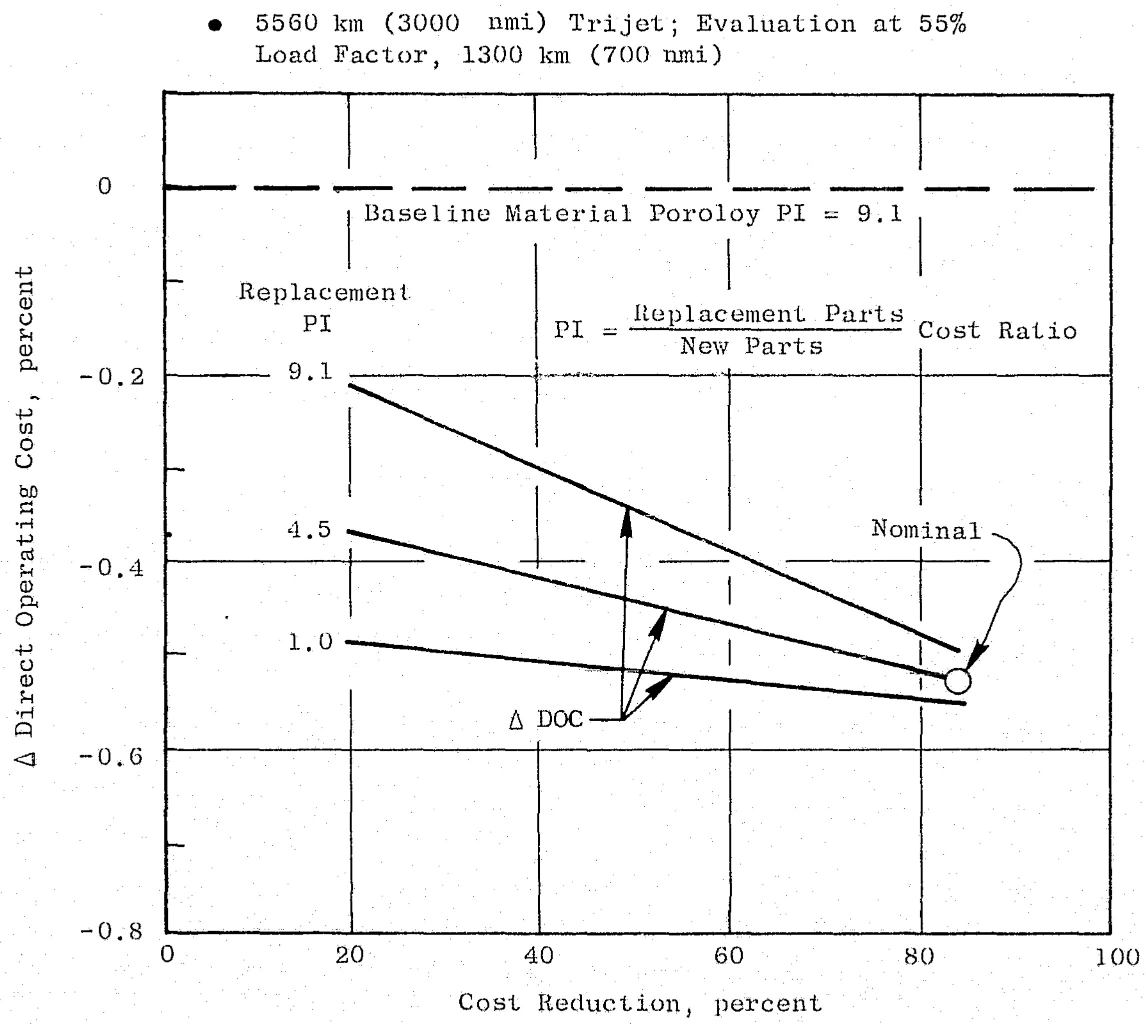


Figure 33. Sensitivity Analysis; Effect of Cost and Parts Replacement on Economic Benefit of HPT Ceramic Shroud.

- Elaborate blade-cooling designs must consider cost performance trades.

G. Installation and Acoustic Technology Evaluation

The baseline composite nacelle is illustrated in Figure 7 with the advanced technology features indicated. The payoff summary of Table XXXII indicates a substantial DOC and fuel-saved benefit for the integrated composite nacelle. The installation improvement features depended upon advanced lightweight composites, aerodynamic improvements in the nacelle design, and the sfc improvement.

The composite nacelle showed improvements from four sources; composite substitution for metal, thin inlet for lower drag, integration of fan frame with the engine, and moving the accessory pod into the pylon for lower drag. The breakdown of the benefits is given in Table XXXIII. As shown, the composite material accounted for over half of the DOC and onethird the fuel saving benefit. Moving the accessories to the pylon, to eliminate the nacelle bulge, yielded about one-fourth of the DOC and one-third of the fuel reduction. The results of the mixed- versus separate-flow study are covered in Section E, but are repeated here since they can also be considered installation technology.

The results from preliminary noise studies indicated that turbine noise reduction would be needed to meet FAR minus 10 EPNdB. Two alternate ways to obtain an extra 5 PNdB of turbine suppression were considered, as illustrated in Figure 34: 1) the high frequency source-noise LPT and 2) a spool piece extension plus splitter. Figure 35 illustrates that additional suppression can be obtained with no penalty by utilizing the higher source-frequency LP turbine. The result is shown in tabular form in Table XXXII.

The benefit of utilizing bulk absorber versus aluminum honeycomb-sandwich, inlet-suppression lining for equal suppression of 8 PNdB was substantial, as listed in Table XXXII, because of the large weight saving due to a shorter inlet length with bulk absorber treatment. This benefit can also be taken as increased suppression of 3 PNdB, at a constant DOC and fuel-used penalty, as illustrated in Figure 36.

A brief summary of observations and conclusions relating to advanced nacelle technology are listed below:

- Composites in the nacelle offer major payoff
- Pylon-mounted accessories have payoff
- Small diameter outer cowl has payoff

$\left.\right\}$ Complementary benefit of integrated nacelle

- Turbine noise suppression; LPT with high blade-passing frequency is a good approach.
- Bulk-absorber inlet treatment; significant advantage over conventional inlet treatment.

Table XXXII. Installation Technology Payoff Summary.

5560 km (3000 nmi) Trijet/10,190 km (5500 nmi) Quadjet

Evaluation at 55% Load Factor, 1300 km (700 nmi)/3700 km (2000 nmi)

Description	Technology: Advanced	Technology: Reference	Aircraft	Δ Weight kg (lb)**	Δ Price $1000**	Δ sfc %	Δ DOC %	Δ W_f %
Mixed Flow	Mixed Cruise Sized Baseline Engine	Separate Same Fan Size	Trijet	+35 (+77)	+29	-2.6	-0.8	-2.6
			Quadjet	+37 (+81)	+36	-2.6	-1.4	-3.6
Composite Nacelle	Composite Thin Inlet Fan Frame Accessories in Pylon	Metal Thick Inlet Nonintegral Accessories in POD	Trijet	-150 (-330)	-87	-0.85	-1.52	-1.77
			Quadjet	-159 (-350)	-109	-0.85	-2.03	-2.31
Turbine	High Frequency LPT	Spool piece extension + splitter	Trijet	-29 (-63)	-3	-0.24	-0.18	-0.38
	Turbine noise Suppression Δ = -9 PNdB		Quadjet	-32 (-71)	-5	-0.24	-0.31	-0.56
Inlet Suppression	Bulk Absorber L/D = 0.68	AL Honeycomb sandwich	Trijet	-32 (-70)	-10	-0.13	-0.25	
		L/D = 92	Quadjet	-33 (-73)	-13	-0.13	-0.34	-0.42
	Δ = -8 PNdB							

** Cycle effects included

Trijet 5560 km (3000 nmi)/200 PAX 79. $/m^3 (30¢/gal) fuel, engine F_n = 89,000 N (30,000 lb)
Quadjet 10,190 km (5500 nmi)/200 PAX 119. $/m^3 (45¢/gal) fuel, engine F_n = 93,410 N (21,000 lb)

Table XXXIII. Separation of Nacelle Features.

5560 km (3000 nmi) Trijet, Evaluation at 55% Load Factor, 1300 km (700 nmi)

	Features				Benefits				
	Material	Inlet Lip Thickness	Accessory Location	Fan Frame Integration		Weight kg (lb)	Price $1000	Drag N (lb)	Total Effects
Reference	Metal	Thin	Pylon	Yes	Δ	-100 (-220)	-59	0	
					Δ DOC %	-0.36	-0.43	0	-0.80
Replacement	Composite	Thin	Pylon	Yes	Δ W_f %	-0.56	0	0	-0.56
Reference	Metal	Thick	Pylon	Yes	Δ	-14 (-30)	-7	-80 (-18)	
					Δ DOC %	-0.05	-0.05	-0.15	-0.25
Replacement	Metal	Thin	Pylon	Yes	Δ W_f %	-0.08	-	-0.42	-0.50
Reference	Metal	Thick	Pylon	No	Δ	-18 (-40)	-11		
					Δ DOC %	-0.07	-0.08	-	-0.15
Replacement	Metal	Thick	Pylon	Yes	Δ W_f %	-0.11	-	-	-0.11
Reference	Metal	Thick	POD	No	Δ	-18 (-40)	-10	-98 (-22)	
					Δ DOC %	-0.07	-0.07	-0.18	-0.33
Replacement	Metal	Thick	Pylon	No	Δ W_f %	-0.10	-	-0.51	-0.61
	All Effects Combined								
Reference	Metal	Thick	POD	No	Δ	-150 (-330)	-87	-178 (-40)	-
					Δ DOC %	-0.55	-0.64	-0.33	-1.52
Replacement	Composite	Thin	Pylon	Yes	Δ W_f %	-0.85	-	-0.92	-1.77

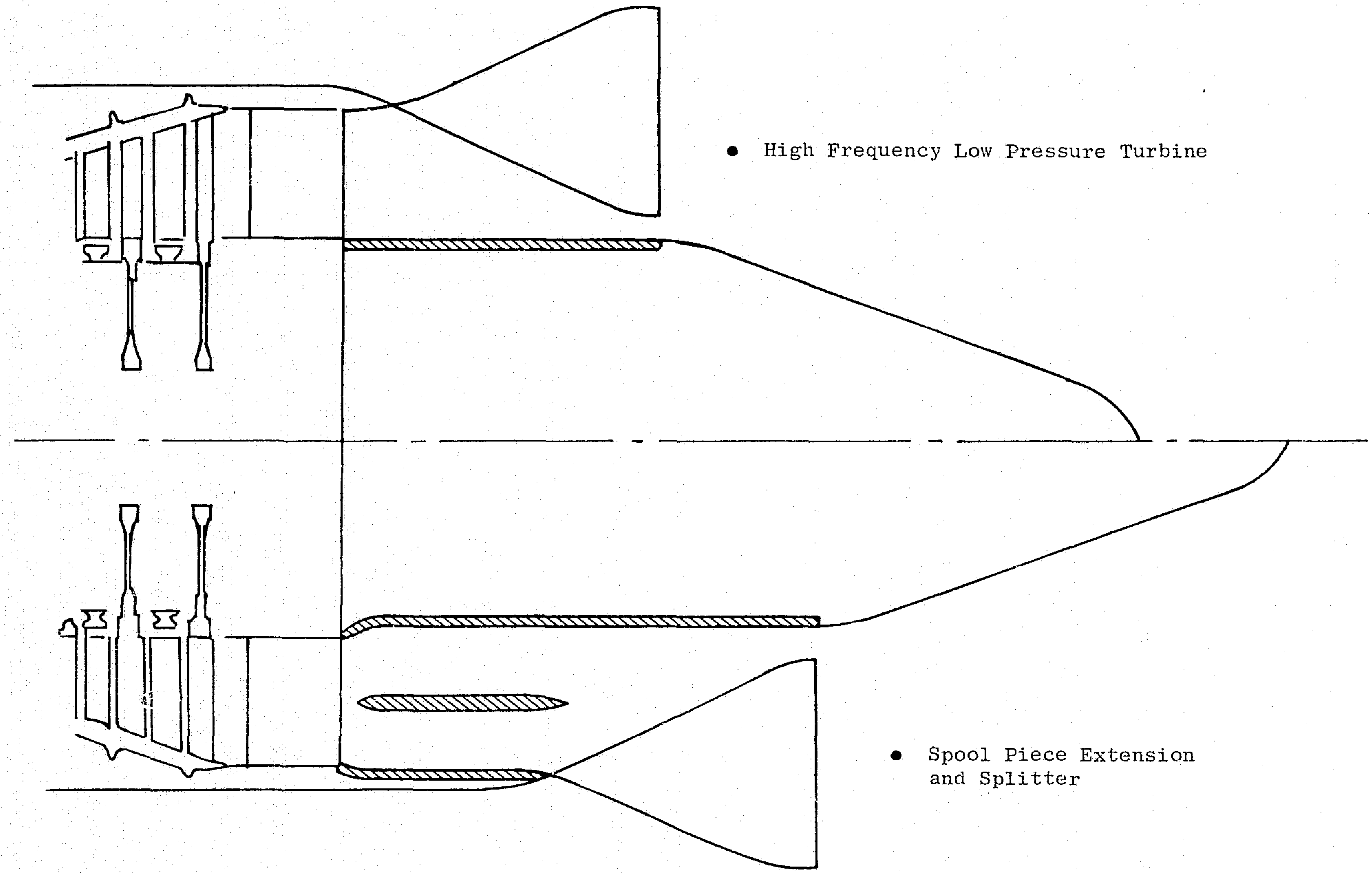


Figure 34. Turbine Noise Suppression.

- 55% Load Factor; 1300 km (700 nmi) Mission
- 10,670 m/0.80 Mach No./+10° C (35,000 ft/0.80 Mach No./+18° F) 95% Max. Cruise

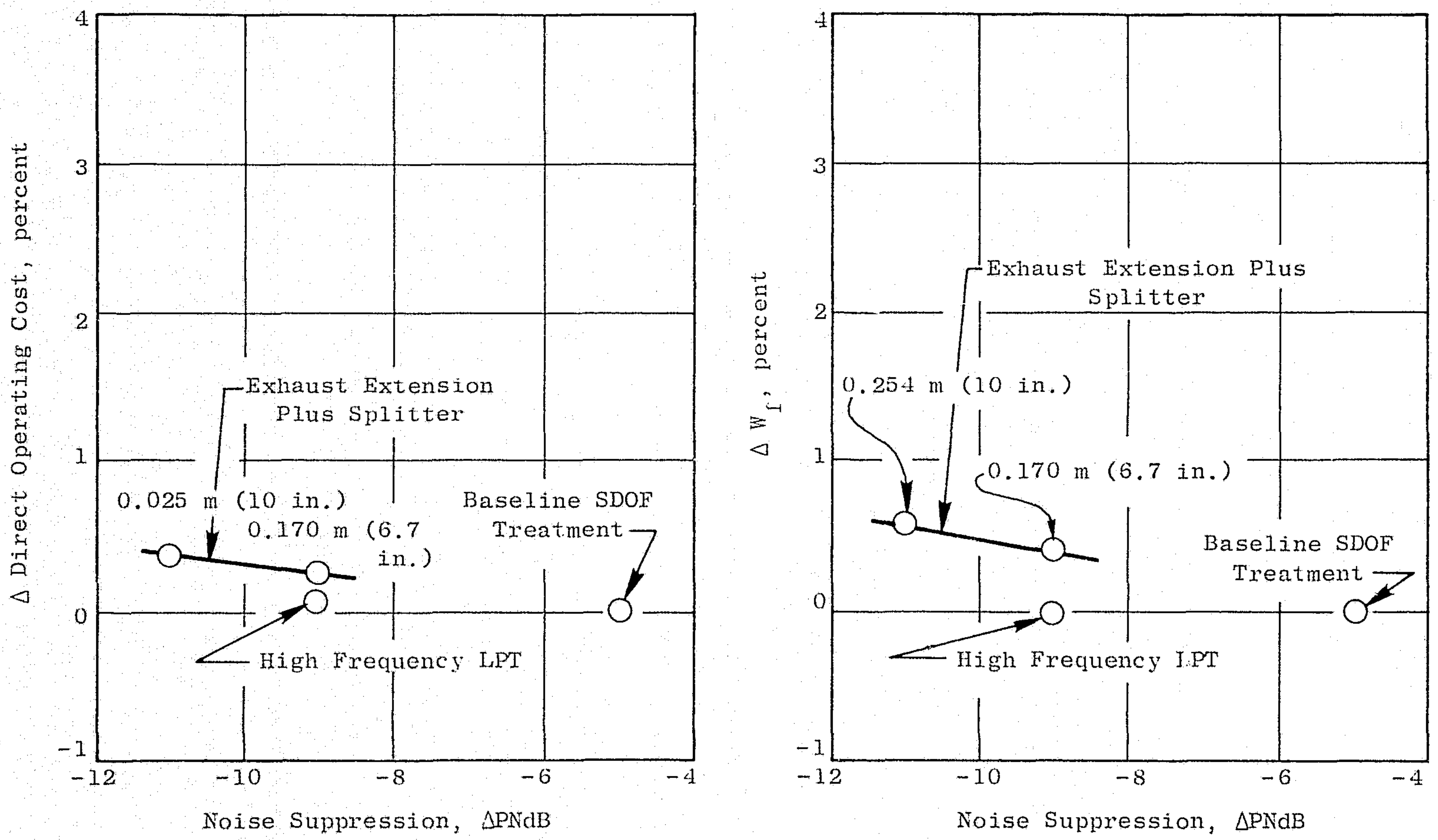


Figure 35. Turbine Noise Suppression; \$79/m^3 (\$0.30/gallon) Fuel, 5560 km (3000 nmi) Trijet Mission.

- 5560 km (3000 nmi) Design Range Trijet
- 55% Load Factor 1300 km (700 nmi) Mission

L = Inlet Length
D_T = Fan Tip Diameter

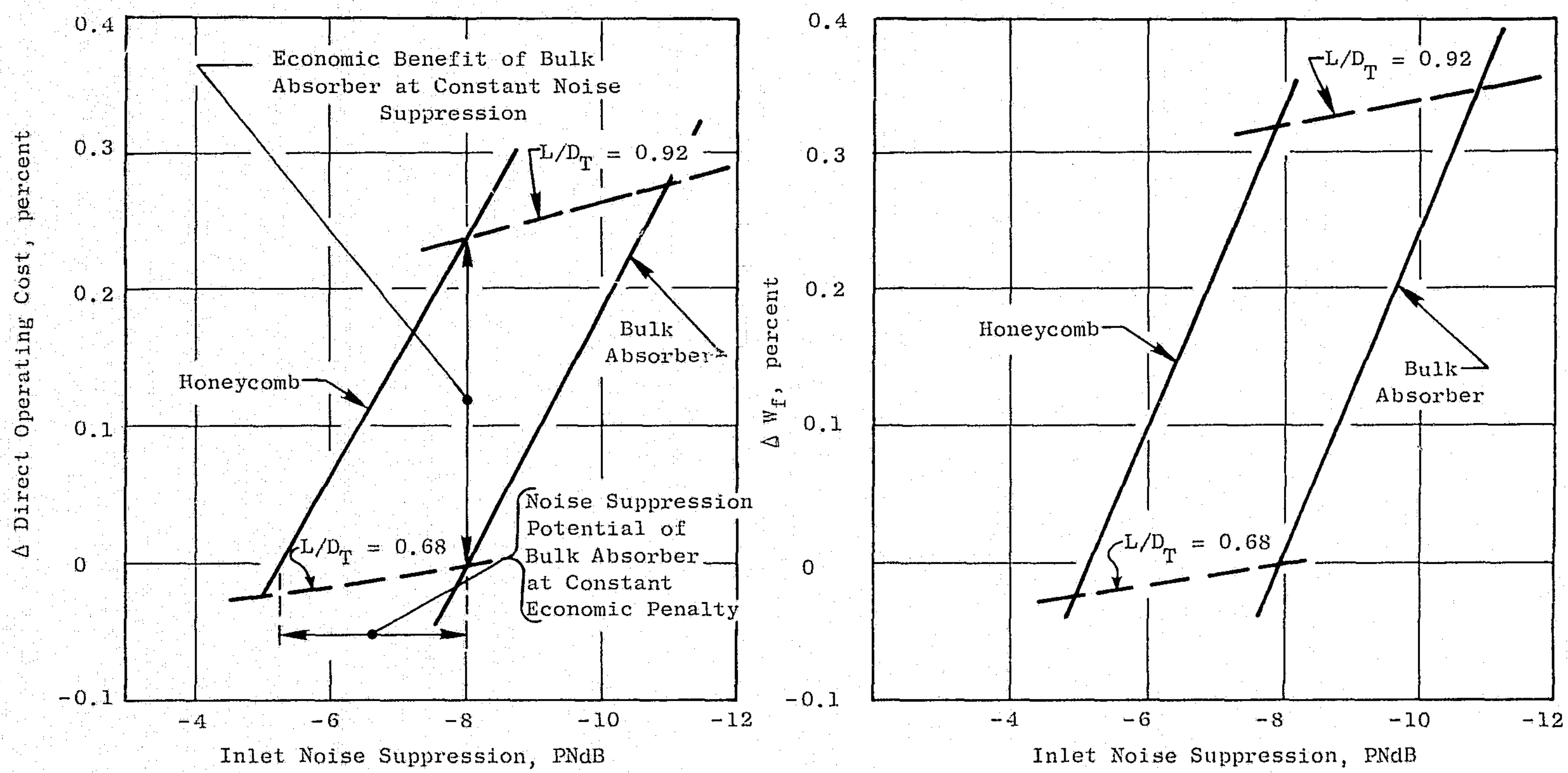


Figure 36. Economic Penalties of Inlet Noise Suppression.

H. Summary of Results - Task II

Parametric cycle studies and evaluation of advanced design and technology features were carried out for new advanced turbofans with a technology level consistent with 1985 introduction into service. The evaluation was made for two new advanced aircraft: a transcontinental design and an intercontinental design. The significant results and conclusions from these studies are summarized as follows:

Turbine Inlet Temperature and Cycle Pressure Ratio

1. Significant advantage (DOC and fuel usage) was achieved with the combination of higher T_{41} and cycle pressure ratio with advanced turbine technology.

2. The resulting range of interest in takeoff T_{41} for an advanced engine was 1371 to 1538° C (2500 to 2800° F).

3. LPT cooling requirements became dominant at 1538° C (2800° F) and above.

4. 1427° C (2600° F) is suggested for initial rating, growth to 1538° C (2800° F).

5. A cycle pressure ratio of 38:1 at altitude (31:1 at takeoff) is recommended for initial rating, higher for growth.

6. The magnitude of advantage estimated for the selected core cycle and turbine technology versus CF6 level was 1.3% DOC and 3.5% fuel usage for the transcontinental aircraft.

Fan Pressure Ratio and Exhaust System

1. A higher fan pressure ratio (up to 1.8) yielded improved DOC (small penalty in fuel usage).

2. A fan pressure ratio of 1.7 (altitude) is recommended for initial rating, increase with growth.

3. The sfc advantage of mixed exhaust was a dominant benefitial factor for advanced technology nacelles at constant noise.

4. Mixed flow has the potential for major payoff in DOC and fuel usage. Estimates were 1% DOC and 4.2% fuel saved versus a CF6-type, separate-flow cycle and 0.8% DOC and 2.6% fuel saved versus a high extraction, advanced, separate-flow cycle.

5. The potential interference drag problem requires attention, but the mixed-flow sfc advantage is large enough to overcome possible penalties associated with different nacelle locations.

Basic Engine Technology

The technology payoff items investigated in Task II for use in the engine and installation are summarized in Table XXXIV. This permits a comparison, on one chart, of the relative merits of all items. This is considered separately from cycle selection issues, such as fan pressure ratio, turbine inlet temperature, overall pressure ratio, and mixed versus separate flow.

The conclusions reached for basic engine technology features are as follows:

1. Composite fan blades are a high potential payoff area
   - Bird-strike feasibility is not established for high tip speed
   - Safety is an additional advantage beyond DOC improvement
2. Composite frame/case integrated with nacelle; major weight and cost reduction item.
3. Core compressor clearance control will be applied in any new engine design.
4. Active (on-off design) HPT clearance control is worth pursuing.
5. Ceramics are a high potential payoff area.
   - Feasibility from impact-damage standpoint is not established.
   - Shrouds are the first area of application.
   - Uncooled LPT ceramic vanes are the next area for attention - higher risk.
6. Cost of eutectic turbine-blade material is currently barrier to its use.
7. Elaborate blade-cooling designs must consider cost and performance trades.

Summary - Nacelle Technology

The following is a brief summary of the evaluation of nacelle technology features for the transcontinental aircraft.

Table XXXIV. Engine Technology Payoff Summary.

5560 km (3000 nmi) Trijet/10190 km (5500 nmi) Quadjet

Evaluation 55% Load Factor, Average Range

Description	Technology Advanced	Technology Reference	Aircraft	Δ Weight Installed kg (lb)	Price* Installed $1000	Δ sfc Installed %	Δ DOC %	Δ W_f %
Fan Blade	Composite	CF6 Type Fan	Trijet	-572 (-260)	-46	-0.75	-1.38	-1.49
	High Tip Speed		Quadjet	-612 (-278)	-56	-0.75	-1.85	-1.95
Fan Frame/Case	Composite	CF6 Type Fan	Trijet	-340 (-155)	-21	0	-0.42	-0.44
	Integrated with Nacelle		Quadjet	-360 (-165)	-26	0	-0.53	-0.52
Compressor Clearance Control	Inco 903, Cooled Casing, Insulated	Inco 70, Uncooled, Uninsulated	Trijet	-29 (-13)	0	-0.32	-0.15	-0.38
			Quadjet	-31 (-14)	0	-0.32	-0.26	-0.50
HPT Clearance Control	Active Cooling System	Passive Cooling Sys.	Trijet	-33 (-15)	+4	-1.0	-0.39	-1.13
			Quadjet	-35 (-16)	+4	-1.0	-0.73	-1.49
HPT Cooling Supply Without Clearance Control Credit	Bore Entry	Rim Entry	Trijet	-22 (-10)	-2	-0.06	-0.06	-0.09
			Quadjet	-24 (-11)	-2	-0.06	-0.09	-0.12
LPT Vane	Ceramic	MAR M 509 Metal	Trijet	-170 (-77)	-16	-0.66	-0.69	-0.91
			Quadjet	-180 (-82)	-19	-0.66	-0.98	-1.20
HPT Vane	Ceramic	MA754	Trijet	0 (0)	-5	-0.15	-0.31	-0.18
			Quadjet	0 (0)	-7	-0.15	-0.39	-0.23
HPT Shroud	Ceramic	Poroloy	Trijet	-40 (-18)	-8	-0.24	-0.53	-0.31
			Quadjet	-42 (-19)	-9	-0.24	-0.067	-0.41
HPT Blade	Eutectic	Advanced, Nickel-Base DS Casting	Trijet	-57 (-26)	+10	-0.16	+0.34	-0.24
			Quadjet	-62 -28	+13	-0.16	+0.33	-0.32
LPT Blade	Eutectic	Advanced, Nickel-Base DS Casting	Trijet	-106 (-118)	+4	-0.60	+0.10	-0.78
			Quadjet	-110 (-50)	+17	-0.60	-0.07	-1.03
HPT Blade Cooling	Film and Full Impingement	Film Cooling Cold Bridge	Trijet	-37 (-17)	+14	-0.11	+0.49	-0.17
			Quadjet	-40 (-18)	+18	-0.11	+0.51	-0.22

*Includes cycle effects on weight and price

	ΔDOC, %	Δ Fuel Usage %
Advanced, Mixed-Exhaust System versus High-Extraction, Separate-Flow Design	-0.8	-2.6
Composite Construction versus Conventional Metal	-0.8	-0.6
High D_{Max}/D_{HL} Inlet versus CF6 Type Inlet	-0.3	-0.5
Pylon-Mounted Accessories versus Fan Case Mounted	-0.4	-0.6
High Aspect Ratio and Spacing LPT versus Splitter and Spool Piece for Reduced Turbine Noise	-0.3	-0.3

The conclusions reached are as follows:

1. Composites in the nacelle have major payoff
2. Pylon-mounted accessories have payoff — Complementary Benefit in Integrated Nacelle
3. Small diameter outer cowl has payoff
4. Turbine noise suppression: LPT with high blade passing frequency is a good approach.
5. Bulk-absorber inlet treatment: significant advantage over conventional inlet treatment.

SECTION V

TASK III - REFINED ANALYSIS

A. Baseline Engine and Installation

In Task III, the results of the Task II evaluation were applied to define a new, baseline, advanced-technology engine. Features and modifications that were found to be a reasonable compromise between direct operating cost, fuel economy and engine growth potential were incorporated.

The significant cycle and technology items included in this design are as follows:

Cycle	Features
T_{41}: 1427° C (2600° F) (takeoff)	Unshrouded composite fan blades
Overall Pressure Ratio: 38:1 (MxCl)	Composite frame and nacelle Long-duct, mixed-flow exhaust
Fan Pressure Ratio: 1.7 (MxCl)	Clearance control systems Advanced noise-suppression systems Highly loaded four-stage LPT

The engine and installation layouts for the Task III engine are illustrated in Figures 37 and 38. They are quite similar to the Task II baseline engine described earlier, except for the specific cycle and component changes recommended as a result of Task II studies. Certain component configurations defined in Task II were altered slightly due to optimization studies carried out in Task III.

B. Cycle and Component Aerodynamic Definition and Performance

From the parametric cycle work done in Task II, it was decided to make two major changes from the Task II basline cycle. Turbine inlet temperature at take-off power was lowered to 1427° C (2600° F) from 1538° C (2800° F), with the higher value being reserved for growth. Fan pressure ratio was raised from 1.65 to 1.71 at the maximum climb design point at M = 0.8, 10,670 m (35,000 ft).

An overall definition of the Task III engine cycle is given in Table XXXV. This is the "design size" that will be referred to in later section. A summary of engine component characteristics is presented in Table XXXVI.

A primary goal of the study was to define an engine with lower fuel consumption than a current modern turbofan. As illustrated in Figure 39, the Task III basline engine (scaled) has an installed sfc advantage (including

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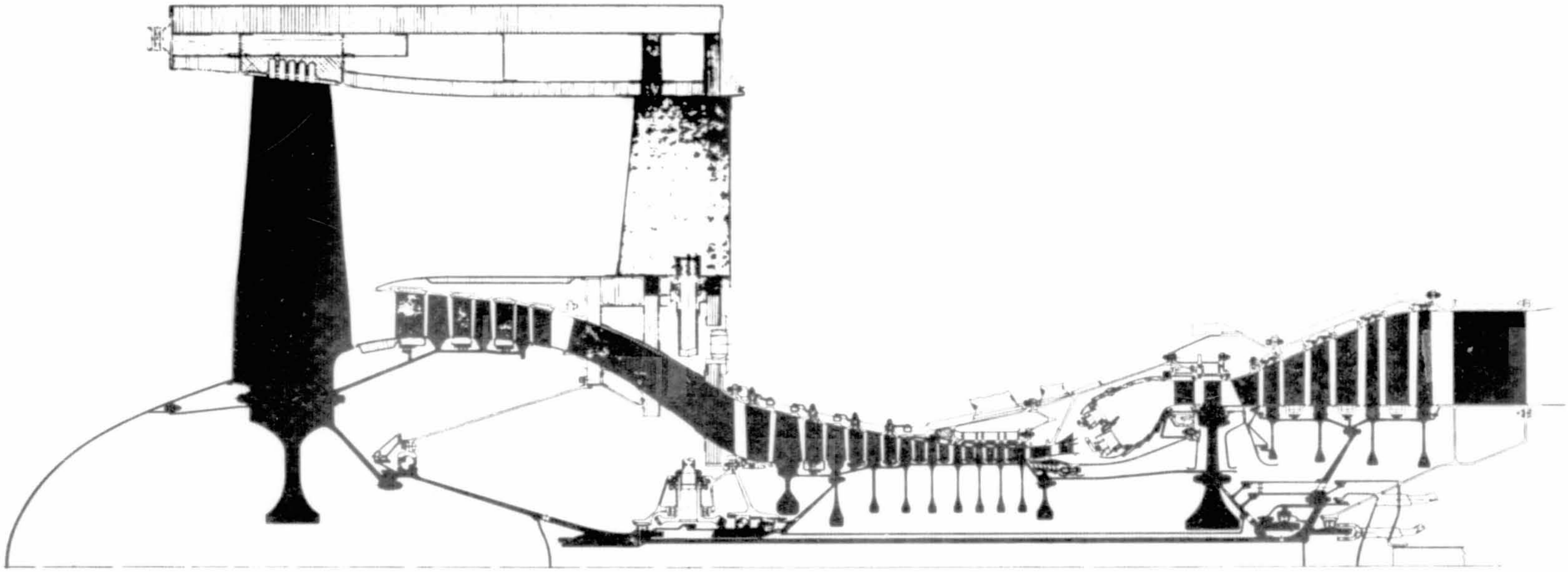

Figure 37. Bare Engine Cross Section; Task III.

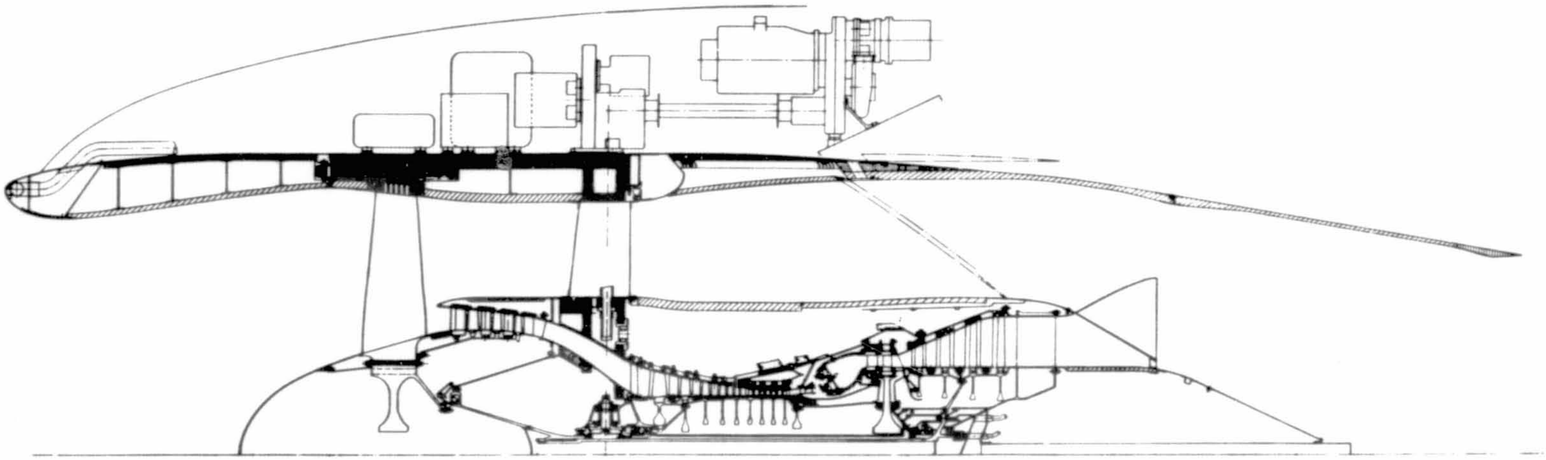

Figure 38. Installation; Task III.

Table XXXV. Engine Cycle Definition.

Design Point at MxCl Mn = 0.8, 10,670 m (35K ft)		
	Baseline-STEDLEC Task III	
Takeoff - Hot Day		
- F_n, N (lb)	147,700	(33,200)
- T_{41}, Turb. Rotor Inlet (Average Cycle)	1427° C	(2600° F)
- Fan Pressure Ratio (P/P)	1.52	
- Bypass Ratio Mixed Flow - 75% Effectiveness	7.5	
Mn = 0.80, 10,668 m (35K ft)		
- MxCl F_n, N (lb)	38,900	(8740)
- $W\sqrt{\theta}/\delta$ at MxCl, kg/sec (lb/sec)	568	(1253)
- Fan $U_T/\sqrt{\theta}$ at MxCl, m/sec (ft/sec)	494	(1620)
- Bypass Ratio at MxCl	6.9	
- MxCr F_n N (lb)	35,600	(8010)
- Booster P/P at MxCr	2.65	
- Core Comp. $P/P/W\sqrt{\theta}/\delta$ kg/sec (lb/sec) at MxCr	13.6/30.4	(13.6/67.1)
- Overall P/P at MxCr	35.9	(38:1 at Design Point)
- T_{41} at MxCr - hot day	1327° C	(2420° F)
- T_{41} at MxCl - hot day	1371° C	(2500° F)

Table XXXVI. Engine Component Definition.

	Baseline-STEDLEC Task III
Fan Tip Diameter, cm (in.)	200 (78.8)
Fan r/r	0.38
Fan Design	Unshrouded Composites
No. of Boosters	3
No. of Core Compressor Stages	9
Core Compressor r/r	0.68
Combustor Type	Double Dome, Low Emissions
No. HPT Stages	1
Cooling	Film Impingement; Bore-Entry Supply
No. LPT Stages	4 + OGV (1 Cooled Blade)
Average LPT Work Coefficient $\frac{gJ\Delta h}{2\ Up^2}$	1.63
Nozzles	Fixed Convergent/Divergent
Length (Flange to Flange), cm (in.)	259 (102)

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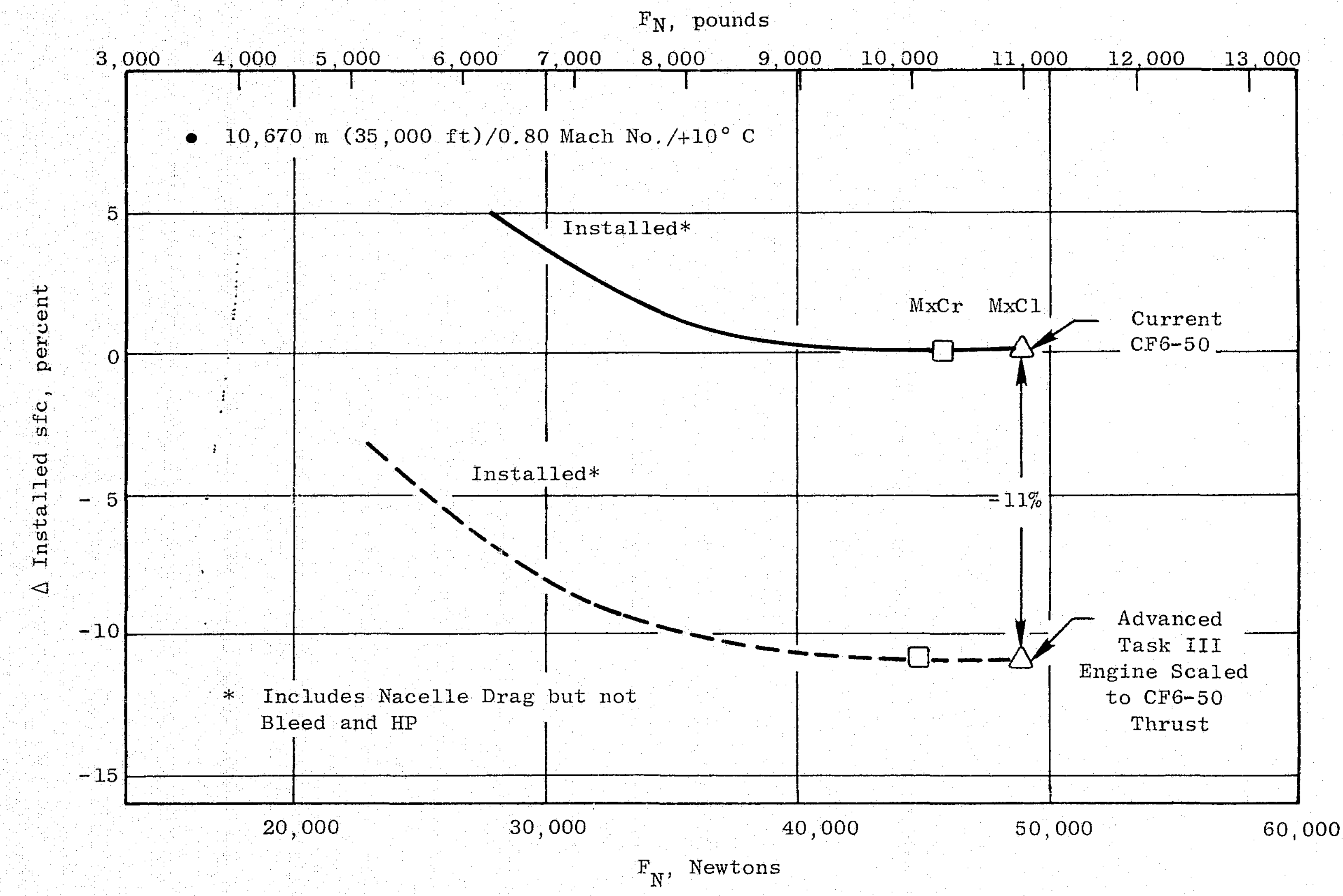


Figure 39. Advanced Engine sfc Comparison.

nacelle drag) of over 10% in the cruise thrust range of interest when compared to the CF6-50.

Prior to selecting the final LP system configuration, presented in Table XXXVII, a fan tip speed and LPT optimization study was carried out using the evaluation procedure of Task II. The study took into account fan and LPT efficiency variations, weight, cost, and the effect on the booster. Results of this study are illustrated in Figure 40. As the fan tip speed was reduced below 494 m/sec (1620 ft/sec), the fan efficiency improved at the constant fan design pressure ratio of 1.7. This was due to lower blade shock and compressibility losses. The low pressure turbine flowpath was close coupled to the high pressure turbine; as a result the four-stage, low pressure turbine loading increased, and its efficiency decreased, as the fan tip speed was reduced. At some fan tip speed, 480 m/sec (1575 ft/sec) for example, a fifth stage had to be added because no reasonable four-stage design was possible. When the stage number was increased, the low pressure turbine efficiency increased, improving sfc by 0.8%, but also increasing the engine weight and price. Based on a minimum DOC, a corrected fan tip speed of 494 m/sec (1620 ft/sec) and a four-stage LPT were chosen. Although the 480 m/sec (1575 ft/sec) fan with five-stage turbine gave minimum fuel usage, that advantage was overcome by the increase in price, weight and complexity; as indicated by the DOC result.

The booster compressor design is summarized in Table XXXVIII. A three-stage design was selected to provide adequate stall margin at the available tip speed. Relatively low blade and vane aspect ratios were chosen for mechanical strength and resistance to aero/mechanical vibrations.

The high pressure compressor design, presented in Table XXXIX, had a high tip speed of 431 m/sec (1415 ft/sec) at the first rotor. The high speed was necessary to produce a 14:1 pressure ratio in 9 stages. The compressor design included advanced aerodynamic blading design and thermal insulation of the compressor casing for improved clearance control on the back stages.

The double-dome combustor was designed to meet both low idle HC and CO emissions as well as low NO_X emissions at high power settings. This was accomplished by operating only on the outer fuel nozzles at idle and low speeds with a high level fuel-air ratio. At high power the secondary inner dome, with lean stoichiometry for low NO_X emission, was activated. Low emissions was the primary requirement of the double-annular combustor described in Table XL. It was also expected that better profile and pattern factors could be achieved with this design than with current combustor designs.

Figure 41 illustrates the results of a study made on the HPT for changes in diameter and loading. Tip clearance, radius ratio, weight, cost and efficiency effects were taken into account for this study. Based on these results, a turbine work coefficient of 0.87 and a configuration almost identical to the Task II engine HPT were selected. When the turbine pitch diameter was increased by 5%, the turbine loading parameter was reduced by 10% (example, from 0.87 to 0.79). There was an adverse effect on tip clearances

Table XXXVII. Fan Aerodynamic Design.

	MxCl/MxCr
Fan Pressure Ratio	1.71/1.65
Diameter, cm (in.)	200 (78.8)
Hub Radius/Tip Radius	0.38
U_T/θ_2, m/sec (ft/sec)	494/480 (1620/1573)
$W\sqrt{\theta_2}/\delta_2$ A, kg/sec-m^2 (lb/sec-ft^2)	1.82/1.778 (43.3/42.2)
No. Blades	28
No. Vanes	40(1)
Shroud Type	Unshrouded
Solidity, Tip/Hub	1.65/2.60

(1) Vane Frame

• 200 PAX/5,560 km (3,000 nmi) Trijet; Evaluation at 55% Load Factor, 1,300 km (700 nmi)

• Fan Pressure Ratio = 1.70, 10,670 m (35,000 ft)/ 0.8 Maximum Climb

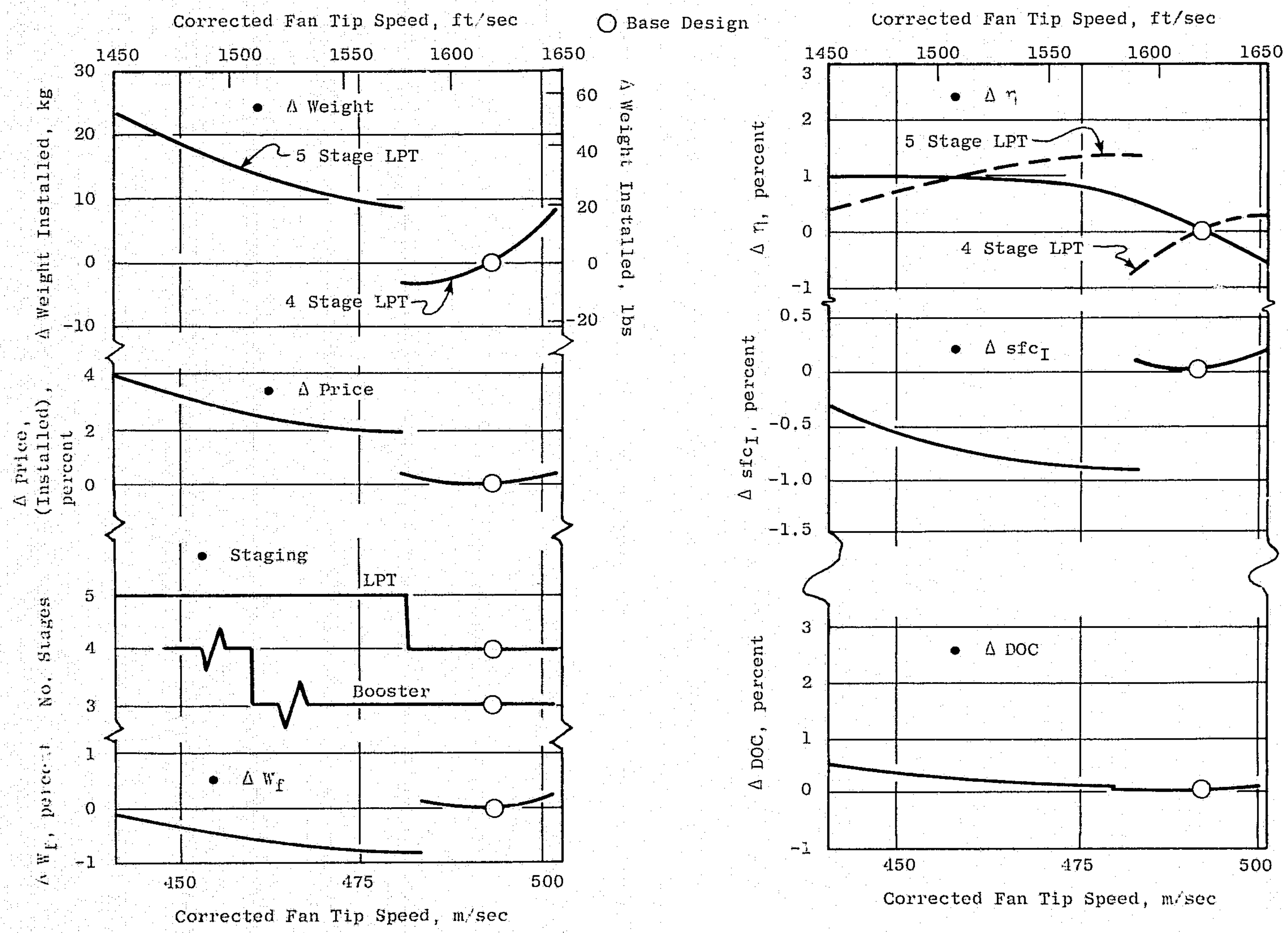


Figure 40. Fan Tip Speed Selection.

Table XXXVIII. Booster Aerodynamic Design.

	MxCl/MxCr
Pressure Ratio, Fan Hub (Booster Inlet)	1.61/1.55
No. Booster Stages	3
Pressure Ratio, Booster Stages	1.71
Average P/P/Stage	1.20
$U_H/\sqrt{\theta_2}$ Boost Rotor Stage 1, m/sec (ft/sec)	212/206 (694/674)
Average Blade Aspect Ratio (AR)	2.0
Average Blade Pitchline Solidity	1.3
Average Vane Aspect Ratio (AR)	1.8
Average Vane Pitchline Solidity	1.7

Table XXXIX. Compressor Aerodynamic Design.

10,670 m (35K ft) 10.8 Mn	
	MxCl/MxCr
W_2 Corrected, kb/sec (lb/sec)	31.0/30.3 (68.4/67.1)
Pressure Ratio	14/13.6
No. Stages	9
Corrected Tip Speed, m/sec (ft/sec)	431/427 (1415/1400)
r_H/r_T 1st Rotor Inlet	0.68
Corrected Flow, kg/sec (lb/sec)	17.1 (37.8)
C_L/L Last Stage, %	2.0

Table XL. Combustor Design.

	SLS Takeoff
P_3, N/m^2 (psia)	2,900,000 (433)
T_3, ° C (° F)	586 (1086)
Combustor Exit Temp. ° C (° F)	1479 (2695)
$T_{Exit} - T_3$, ° C (° F)	876 (1609)
M_3	0.294
V_{Dome} Outer/Inner, m/sec (ft/sec)	5.18/31.4 (17/103)
Space Rate, joule/hr-N/m^2-m^3 (Btu/hr-atm-ft^3)$x10^6$	2.76 (7.5)
Comb. Length/Dome Height	3.2/3.9
Profile Factor	0.10
Pattern Factor	0.25
Liner Cooling, %	17.7
Type	Double Annular

- 200 PAX/5,560 km (3,000 nmi)
- Trijet; Evaluation at 55% Load Factor, 1,300 km (700 nmi)
- Single Stage HPT, P/P = 3.9
- 10,670 m (35,000 ft)/0.8 Max. Climb

○ Base Design

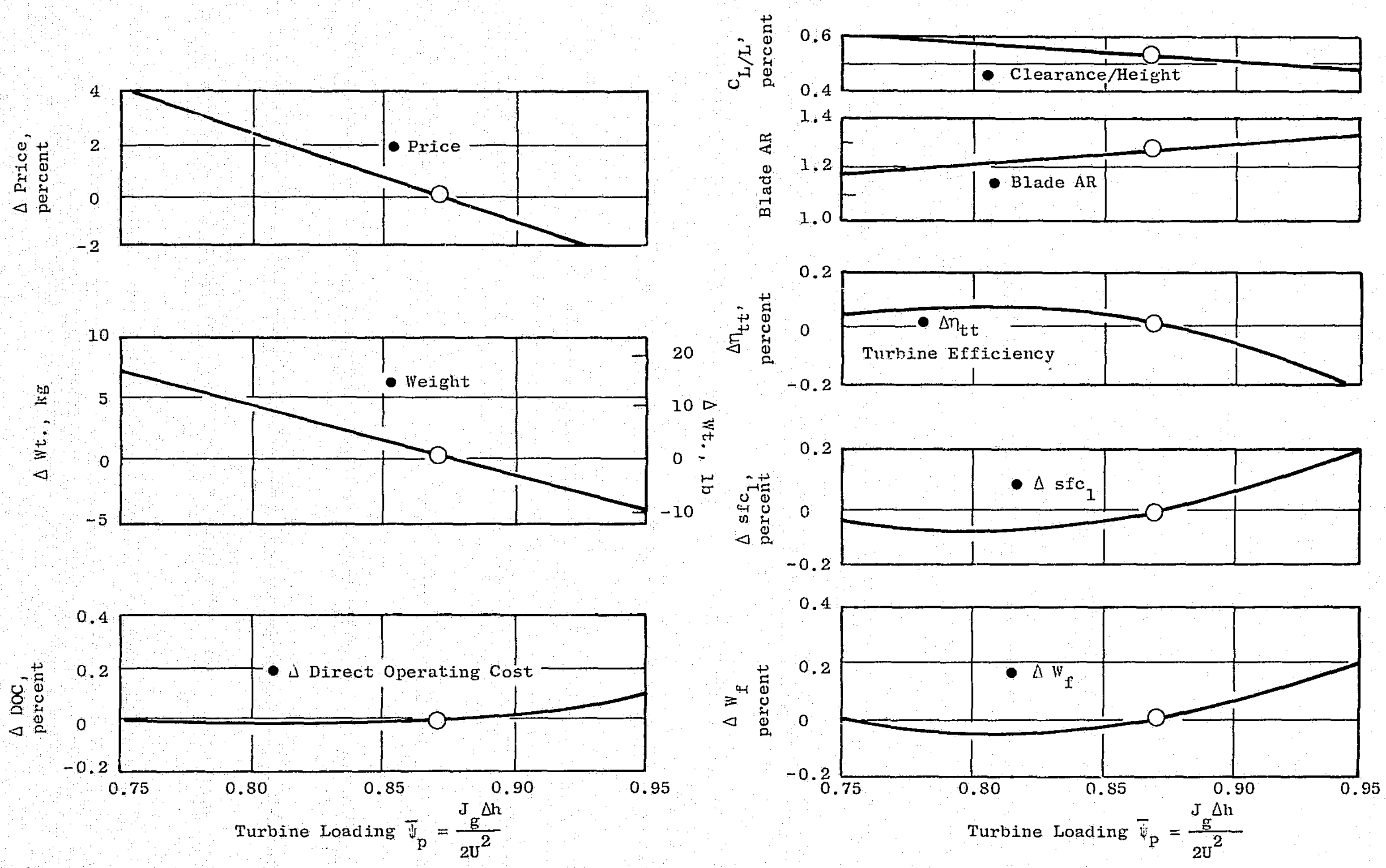


Figure 41. HPT Loading Study.

due to a smaller blade height at a larger diameter, an increase in secondary flow losses to lower blade aspect ratio, and finally an efficiency increase due to lower turbine loading. Overall there was a small net improvement in turbine efficiency. But there was also a turbine weight and price increase due to an increased number of turbine blades at the larger pitch diameter.

Table XLI presents a general description of the HPT (high pressure turbine). The highly loaded, single-stage turbine was advantageous, from a cooling standpoint, for a high temperature engine. Active clearance control was employed to achieve better efficiencies than currently available from this type of HPT deisgn.

The LPT (low pressure turbine), described in Table XLII, was a highly loaded design that utilized results from recent NASA-sponsored, low speed turbine programs. The LPT counterrotates from the direction of the HPT to reduce the turning required of the stage 1 LPT vane. Two stages of vane cooling and one stage of blade cooling were required.

The turbine cooling design features are listed in Table XLIII. Some of the advanced features (such as Ni76XB blade material, and ceramic HPT shrouds and bands) have been included. Others, such as ceramic turbine vanes, are not included but should be considered for later growth of the engine.

One important consideration in any new engine design is whether the engine is capable of growing significantly in thrust for later application. Table XLIV presents the growth goals for this engine (20 to 25%), and the methods available to achieve them.

C. Engine Design

Design features evaluated in Task II, and determined to have economic payoff, were incorporated into the Task III engine (Figure 37) with the exception of ceramic HPT and LPT vanes (it was felt the necessary technology and development would not be in place in time to permit a 1985 certification). The Task III design features are compared to the CF6-50 in Table XLV.

Composites were used extensively in the fan rotor, frame, and casing with significant weight and cost savings. A wide-chord fan blade, without shrouds, was designed to satisfy aeromechanical stability requirements. Use of composites in the fan case and frame permitted an integration of the two with the nacelle in the area over the fan. The fan-frame struts were designed to perform the outlet guide vane function, allowing a more compact engine layout. The fan blade containment design was based upon a Kevlar-type material.

The booster mechanical design was conventional, with titanium used for the blades and vanes. Low aspect blading was chosen to provide necessary stall margin and to enhance the FOD (Foreign Object Damage) resistance of the blading.

Table XLI. Hp Turbine Aerodynamic Design.

10,670 m (35,000 ft)/0.8 Mn	
	MxCr
Δ h, joule/kg (Btu/lb)	476,000 (205)
$\overline{\psi}_P = \dfrac{J_g \Delta h}{2\ U^2}$	0.87
No. Stages	1
Overall Pressure Ratio	3.82
Clearance Control	Active
CL/L %	0.5
Rotor Cooling Supply	Stage 5 Bore Entry into Turbine (Compressor Source),
Leaving Mach No.	0.48
Leaving Swirl	20°
Tip Shroud	No

Table XLII. LP Turbine Aerodynamic Design.

10,670 m (35,000 ft)/0.8 Mn	
	MxCl/MxCr
No. Stages	4 1/2
Δh, joule/kg (btu/lb)	456,000/437,000 (196/188)
P/P	5.7/5.6
$\psi P - \dfrac{gJ\Delta h}{2U^2}$	1.63/1.61
Tip Shroud	Yes
Rotation	Counterrotating with Core
Interturbine Frame	No
Cooled Blades	Yes (1 stage)

Table XLIII. Turbine Cooling System.

	High Pressure	Low Pressure	
Stage	1	1	2
Blade Material	Ni76XB	Ni76XB	Ni76XB
Cooling Technology	Impingement/Film	Improved Convection	Uncooled
Vane Material	MA754	MARM 509	MARM 509
Cooling Technology	Impingement Film	Simple Film	Simple Film
Band Material	Ceramic	MARM 509	MARM 509
Shroud Material	Ceramic	Metallic Honeycomb	Metallic Honeycomb

Table XLIV. Growth Considerations.

20 to 25% thrust increase required, takeoff and cruise, - same diameter		
Initial cycle and configuration - select so that no barriers to growth exist (such as jet noise, temperature limitation, shaft torque, etc.)		
Elements of growth necessary		Growth Cycle
- Core energy	+93° C (+200° F) turbine temperature added booster stage	1538° C (2800° F) T_{41} ~44:1 cycle P/P
- Thrust porducing capability	+8 to 10% fan pressure ratio plus some flow within same fan diameter	~1.85 fan P/P
Technology features for growth		
- Ceramic vanes	Feasibility needs to be established	
- Eutectic turbine materials and more exotic cooling	Cost trades change in growth context	

Table XLV. Engine Design Features.

	Advanced Turbofan Task III	Current Engine CF6-50
Fan	Composite, Unshrouded High Tip Speed, Advanced Aerodynamics Composite Fan Blade Containment Composite Fan Frame Integrated Fan-Exit Guide Vanes	Titanium Midspan Shroud Low Tip Speed Armor Steel Metal Frame Separate Struts and Exit Guide Vanes
Compressor	9 Stages High Tip Speed, Advanced Aerodynamics Clearance Controlled Rear Casing Rugged, Wide-Chord Blading	14 Stages Current Technology
Combustor	Double-Dome Annular Low Emissions	Single-Dome Annular
High Pressure Turbine	High P/P, High Loading Single Stage Advanced Aero. and Tip Clearance Control	2-Stage Turbine, Low Loading Current Technology
Low Pressure Turbine	4-1/2-Stage High Loading, Cooled No Interturbine Transition Low Source Noise	4-Stage Low Loading Transition Duct
Exhaust	Advanced Mixer 75% Effectiveness	Separate

Titanium and Inconel blading was employed in the core compressor. The wide-chord design was an element of the high stage-loading concept and also increased tolerance to blade erosion. The rotor was cooled by booster discharge air. An inside diameter extraction system was used to provide HPT blade-cooling air. Compressor clearances in the last four stages were maintained by using a double casing, of a cooled and insulated, low-expansion alloy design (See Task II for a description of the benefits). Variable stators were necessary in the forward stages for flow matching.

A double-annular combustor, with primary and secondary burning zones for low emissions was utilized in the Task III engine. Film cooling was employed in a machined-ring design for improved reliability.

The HPT was a high tip speed, single-stage design utilizing an advanced, bore-entry, cooling-supply system, an advanced, directionally solidified, blade alloy, and an active clearance-control system for the blade/shroud interface. The improvements due to the active clearance-control system were evaluated in Task II. Table XLIII describes the cooling technology and materials used in the blades and vanes. Ceramics were utilized for the HPT blade shroud and vane inner and outer bands.

A close-coupled, highly loaded LPT with 4-1/2 stages was employed in the Task III engine. Two stages of vanes and one stage of blading required cooling, due to the high LPT inlet temperatures. A directionally solidified blade material was used in the first two stages, and a high temperature nickel alloy was used in the first three stages of vanes. The OGV function was combined with the struts in the exhaust frame to remove the swirl coming out of the last rotor stage.

Figure 42 illustrates an engine schematic, consistent with the Task III design, with overall dimensions and CG (center of gravity) location identified. The estimated bare-engine weight in the design size was 2040 kg (4500 lb), yielding a thrust-weight (uninstalled take-off thrust) ratio of 7.6. Figure 43 is a comparison, by component, of the Task III engine weight versus a scaled CF6-50.

D. Installation Design

The long-duct, mixed-flow installation used in Task II was retained for Task III (illustrated in Figure 38).

Extensive use of composites in the cooler part of the nacelle and fan duct produced significant weight and cost savings, as described in Task II. Strength and weight calculations were based on a honeycomb sandwich construction with graphite polymeric composite surfaces. The surfaces consisted of multiple layers of prepreg material oriented at 0.79 to 1.05 rad (45° of 60°) with conductive strips for lightning and static electric discharge. Leading-edge anti-icing was accomplished through an aluminum-sheet, leading edge annulus blown with hot compressor discharge air. An epoxy coating was used to reduce normal surface erosion.

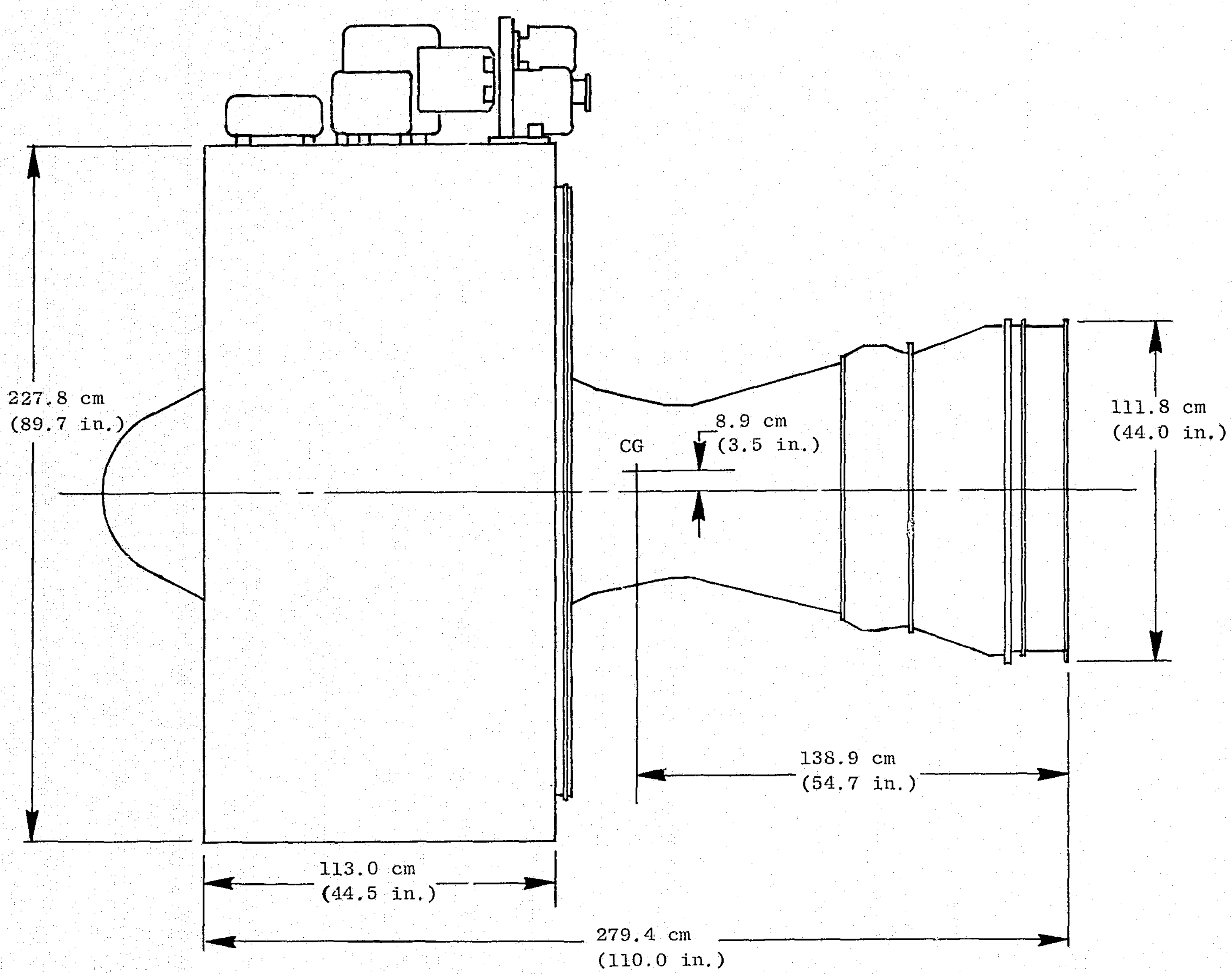
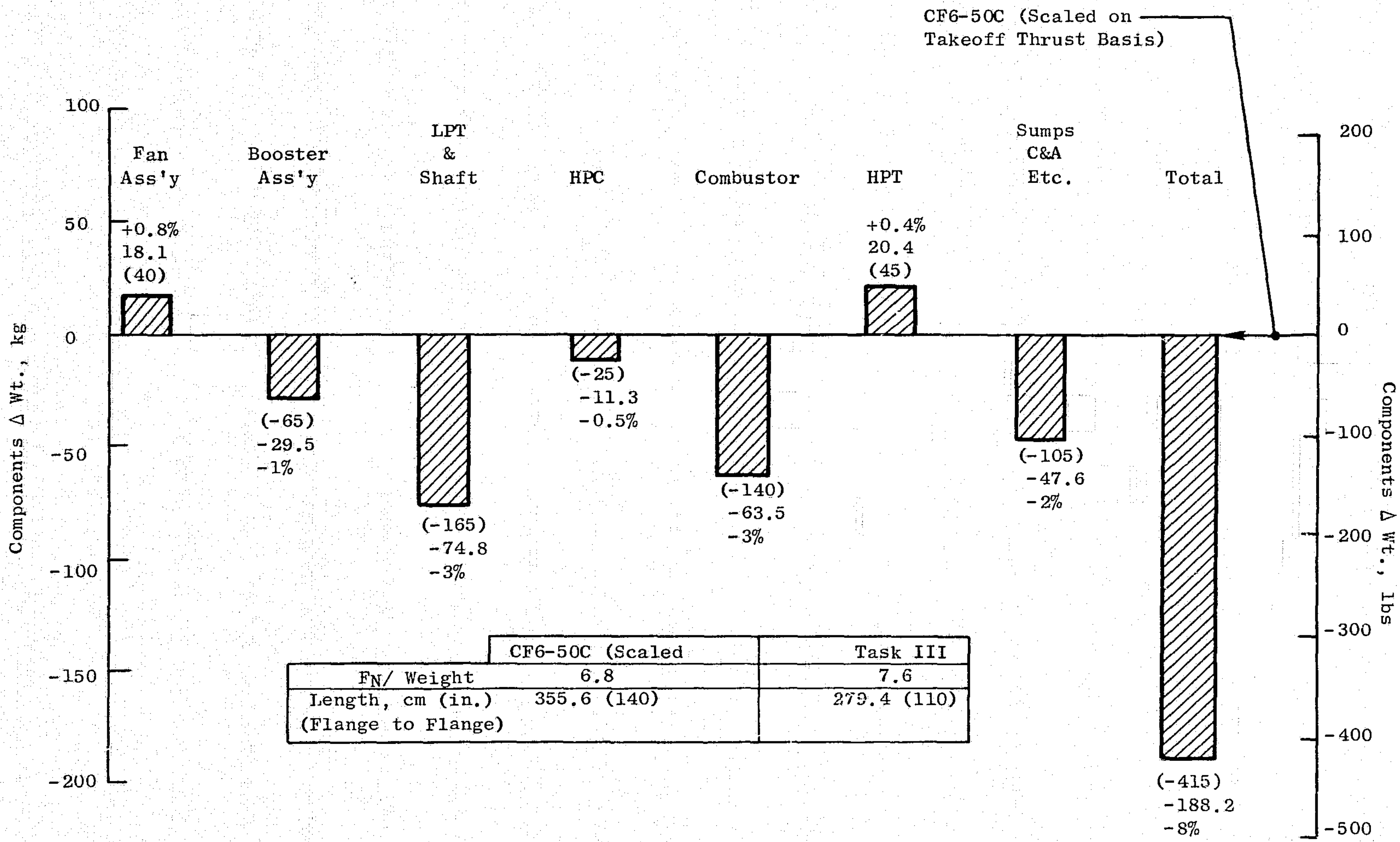


Figure 42. Design Size Overall Engine Dimensions.



	CF6-50C (Scaled	Task III
F_N/ Weight	6.8	7.6
Length, cm (in.) (Flange to Flange)	355.6 (140)	279.4 (110)

Figure 43. Weight; Baseline Engine Versus CF6-50C (Scaled).

A conventional, nonredundant mounting system was employed on the Task III engine, as illustrated in Figure 44. Thrust is taken out by a yoke mounted to the front frame of the core engine.

An advanced, thin-nacelle concept was employed to reduce nacelle drag. Figure 45 illustrates the Task III inlet compared to a current CF6-type conventional inlet. A higher value of D_{HL}/D_{Max}, consistent with a cruise Mach number of 0.8, was selected, however, the internal contraction ratio was maintained. Placement of the engine and airframe accessories gearbox in the pylon allowed the usual nacelle accessory "hump" to be eliminated. A drag reduction of 0.8% of cruise thrust resulted from a combination of the thin nacelle and placement of the accessories in the pylon.

The mixer design was an advanced, 21-lobe system (Figure 46). Mixer effectiveness was estimated at 75%, which is a projection of what can be obtained with development effort. From the exit plane of the mixer aft, the engine nozzle is composed of steel honeycomb for light weight, stiffness and sound suppression. Backflow and pluming of hot gases occured during reverse operation, requiring the high temperature nozzle material.

The reverser was of the cascade type, but simplified and reduced in weight from current cascade reversers. Composites were employed where possible. During reverse operation, cooling air circuits were opened to allow cool outside air to flow into the hot recirculating exhaust gases to protect the composite duct walls from overtemperature. The reverser design provided approximately the same reverser effectiveness as the CF6-50 fan reverser.

A component-by-component comparison of installation weight with a scaled CF6-50C is given in Figure 47. Combined with the reduction in base engine weight, an installed weight savings of 12% was achieved when compared to a CF6-50 scaled to the same take-off thrust, and 24% when scaled to the same cruise thrust. An installed weight of 2940 kg (6600 lb) was estimated for the Task III engine for an installed take-off thrust/weight ratio of 5.1, compared to 4.5 for the CF6-50C on a scaled basis.

E. Noise and Emissions

Noise levels of the installed Task III engine were estimated to meet the FAR36 minus 10 EPNdB requirements established for this study in the host aircraft defined in Task II. The suppression performance penalty was estimated to be about 0.1% sfc with an installed cost and weight penalty of about 1.5%. Low noise levels were achieved through a combination of suppression and source-noise reduction assuming continued development of low noise technology.

Fan blade to fan stator spacing was set at 1.7 blade chord lengths to reduce pure-tone noise due to wake interference. In the LPT, the number of blades in the last two stages was increased, along with the blade to vane spacing, to reduce pure-tone noise and to increase the passing frequencies to less objectionable levels.

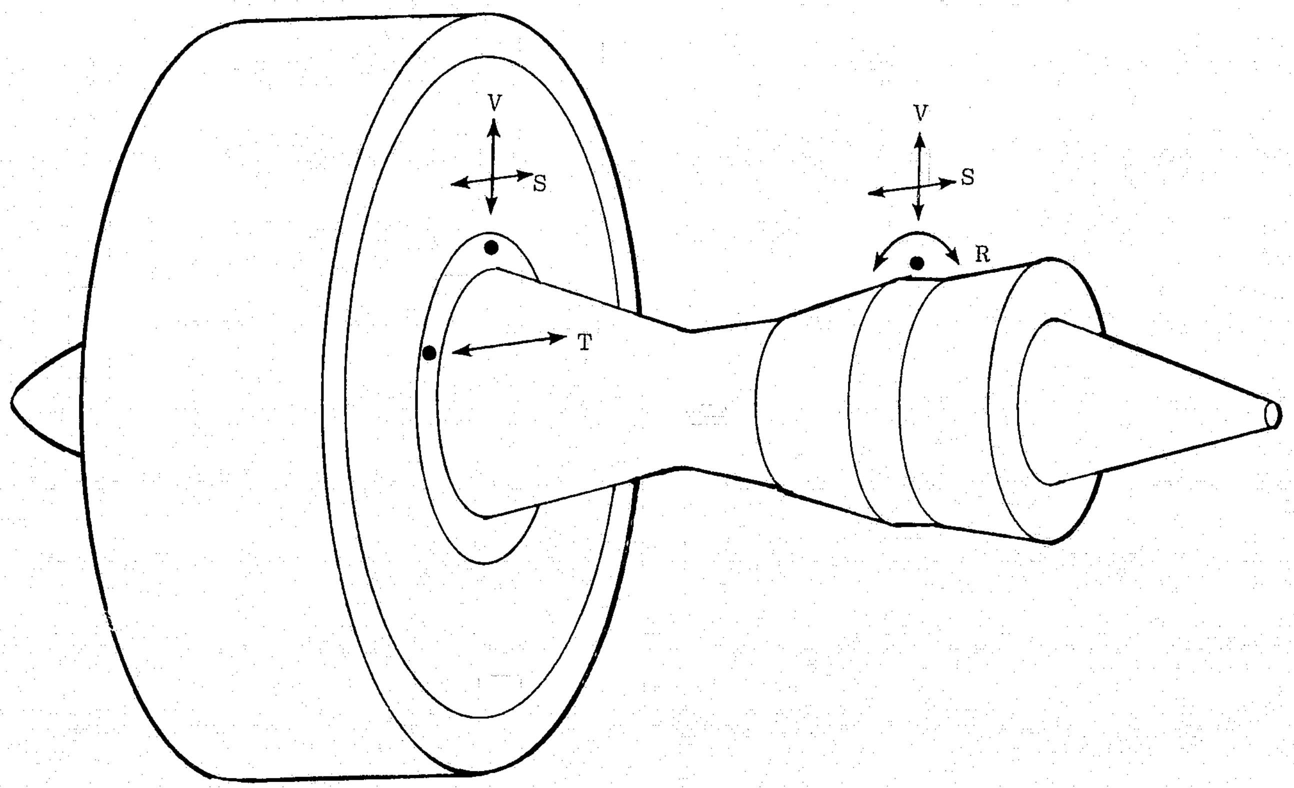


Figure 44. Engine Mounting Schematic.

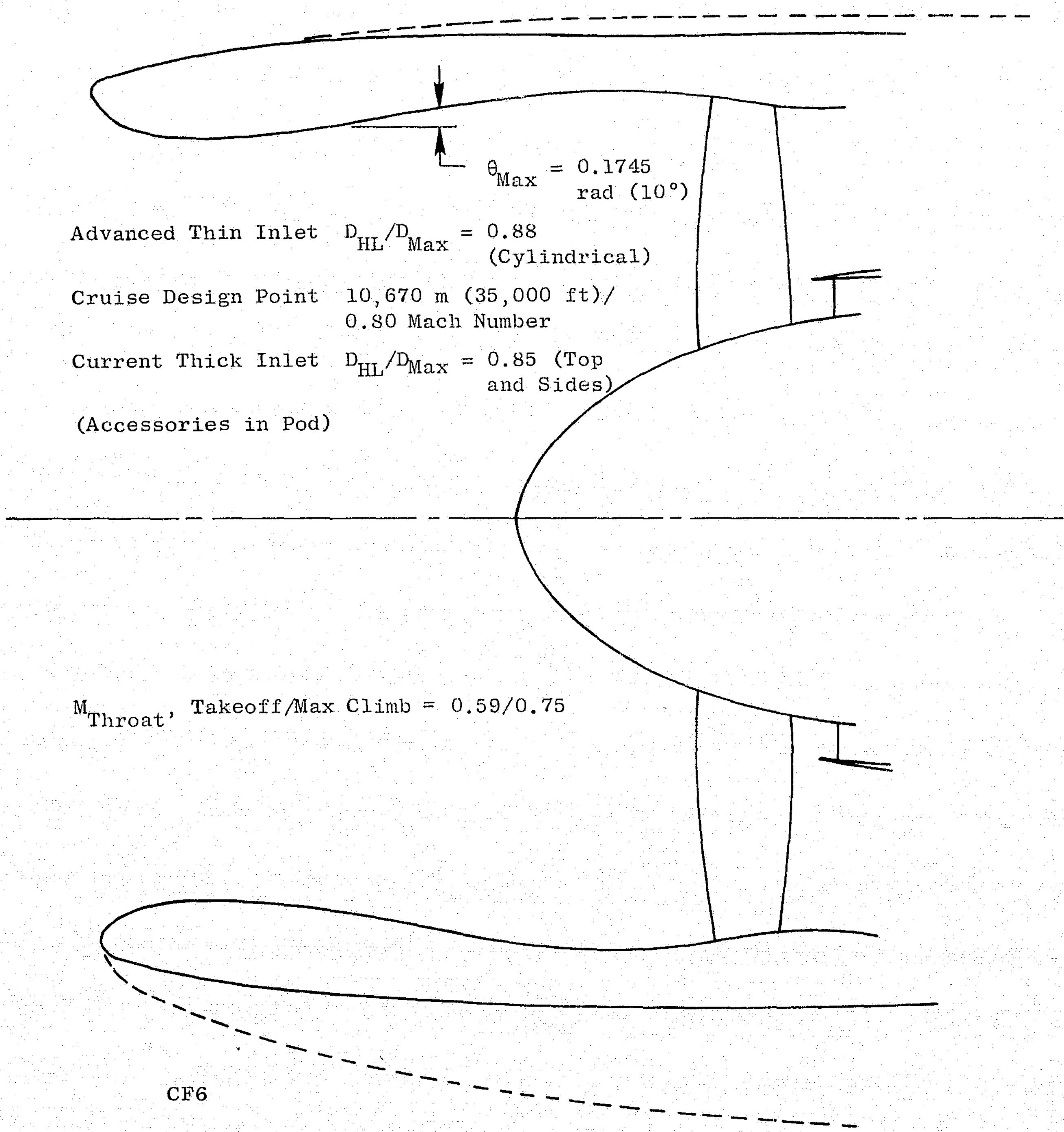


Figure 45. Inlet Comparison.

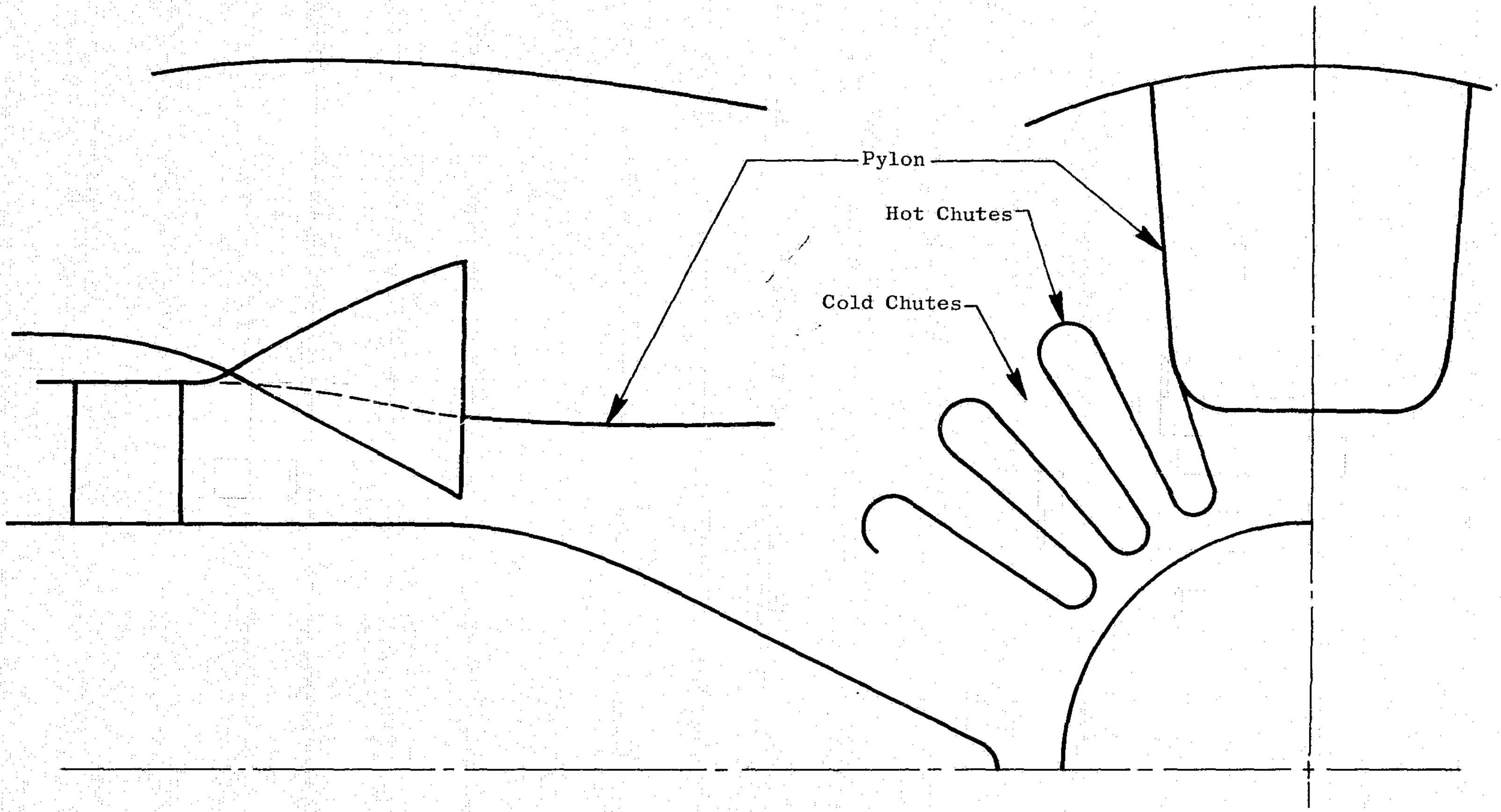


Figure 46. Advanced Mixer Design.

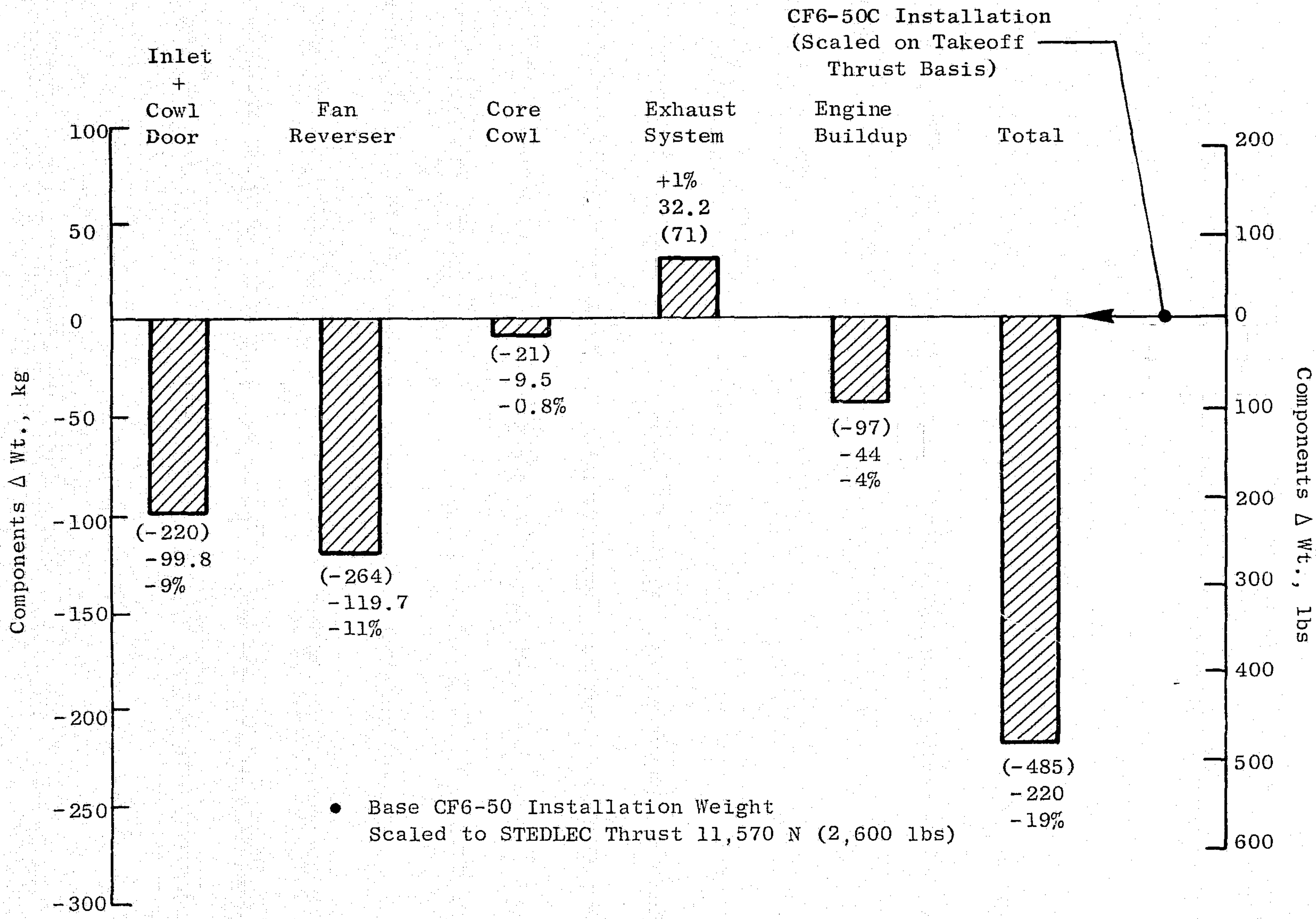


Figure 47. Installation Weight Versus CF6-50C (Scaled).

Suppression was accomplished by using bulk acoustic-suppression liners in the inlet and phased SDOF (Single Degree of Freedom; tuned for a particular frequency) treatment in the fan duct. A Kevlar-based bulk absorber treated to reduce wetting and moisture abosrption problems was used. A limited amount of honeycomb suppression liner was employed in the turbine exhaust and in the hot section of the mixed-flow exhaust liner.

Figures 48 and 49 present the estimated suppressed and unsuppressed source-noise levels, by component, for a typical approach condition and for takeoff with no cut back. Only forward- and aft-radiated noise levels are supplied since sideline noise was not a limiting factor. At takeoff, a noise level of FAR36 minus 10 EPNdB was achieved. During approach, a system noise level of just under FAR36 minus 10 EPNdB achieved, with the aircraft contribution being about the same as the engines. The aircraft contribution was estimated from DC-10 noise patterns. No margins or tolerances that would be necessary to certify an aircraft to a given noise level were included in these estimates.

Emission estimates were made for an advanced, double-annular combustor design, which employed the concept of a primary burner for low power and idle, and an added secondary burner for high power. During idle and low power settings, the primary burner employed a rich mixture, and low air velocities, to reduce CO and HC emissions. At higher power levels, the secondary burner cut in and maintained a leaner mixture and high burner velocities for low smoke and reduced NO_X.

Table XLVI presents the predicted emission levels of the Task III combustor, along with 1979 and 1981 EPA requirements. Assuming continued development of such a combustor, it is projected that the requirements could be met with the exception of NO_X (3.7 versus 3.0). Currently, no acceptable way to meet the NO_X emission requirement has been identified. Again, no margins that might be necessary for certification were included in these estimates.

F. Economic Factors

Estimates of the production cost for the Task III engine and installation were made and compared to the cost of a scaled CF6-50 engine. In Figure 50, relative cost changes (as a result of cycle, technology, and materials) are shown for both the basic engine and installation. The basic engine cost was higher, due primarily to the extensive use of more expensive high temperature materials. However, the installation items were lower due to the use of composites, resulting in a slightly lower installed engine cost.

Another economic factor was the impact on maintenance cost of the Task III engine design. Although a quantitative estimate of the maintenance cost per flight hour was not made, many features were incorporated into the engine to decrease maintenance. The basic engie layout was simple, and involved a relatively small number of parts. Table XLVII presents a few of the design features which should reduce maintenance cost. These features were separated into two major groups, primary failure prevention, and secondary failure prevention.

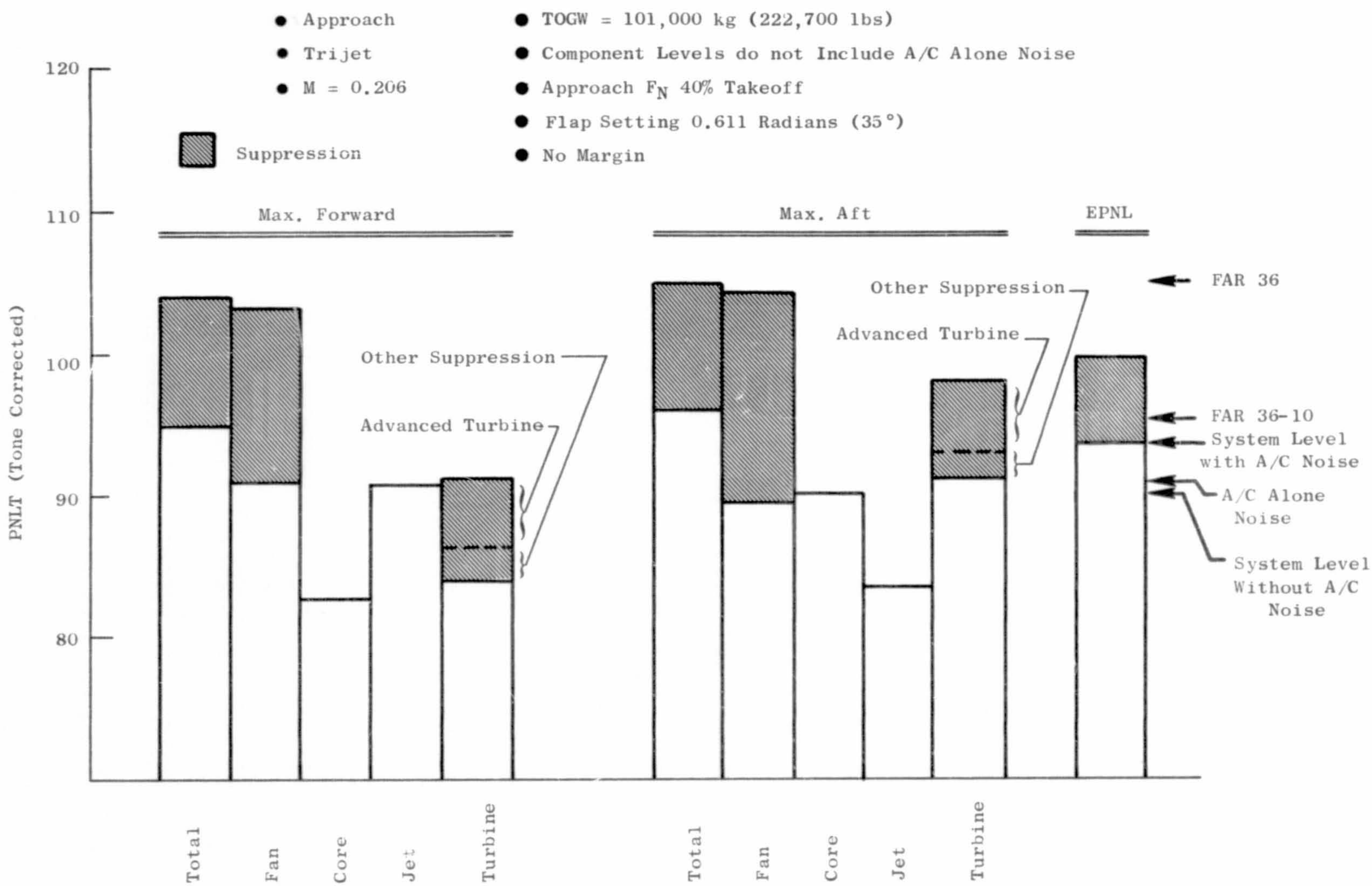


Figure 48. Baseline Noise Levels.

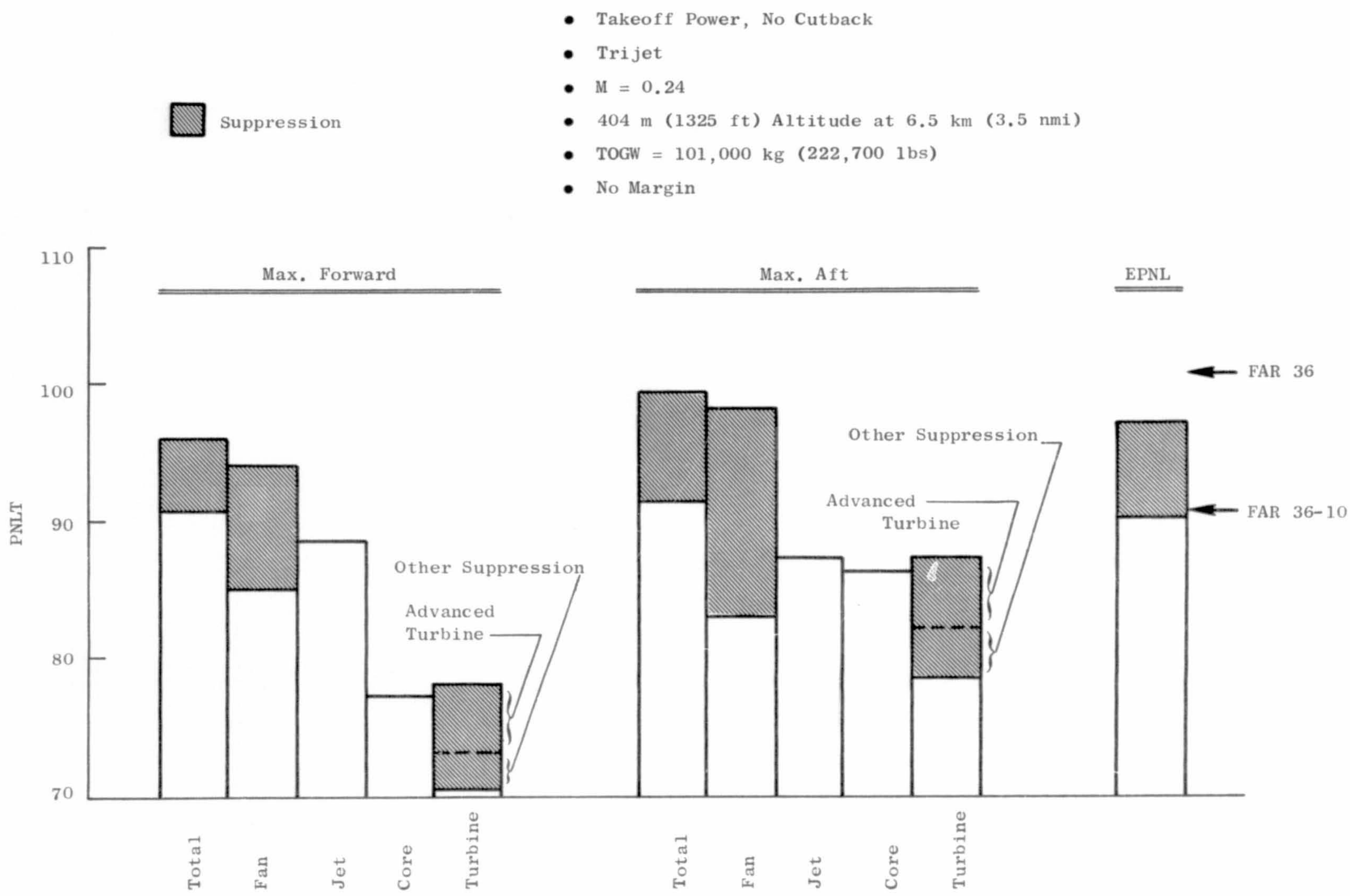


Figure 49. Baseline Nacelle Noise Levels.

Table XLVI. Emissions Estimates*.

	STEDLEC Task III	EPA Requirements T2 Class Turbine Engines 1979	1981
CO, kg/1000 N F_n-hr/Cycle (lb/1000 lb F_n-hr/Cycle)	$\leq$0.31 (3.0)	0.44 (4.3)	0.31 (3.0)
HC	$\leq$0.04 (0.4)	0.08 (0.8)	0.04 (0.4)
NO_X	0.38 (3.7)	0.31 (3.0)	0.31 (3.0)
AIA smoke at TO, %	$\leq$ 20		20

* No Margins

Table XLVII. Design Features for Improved Reliability and Lower Maintenence Costs.

- Primary Failure Prevention
  - Shrouded Compressor
  - Clearance Monitoring
  - More Rugged Blading
  - Rolled-Ring Combustor Liner
  - Condition Monitoring Ports
  - Remote Accessory Mounting
  - Optical HPT Blade Temperature Monitoring
  - Main-Shaft Bearing Monitoring
  - Ceramic Shrouds
- Secondary Failure Prevention
  - Composite Fan Blades
  - Chip Detection
  - Vibration Monitoring - Engine and Bearings
  - Double-Insulated, Aft Sump

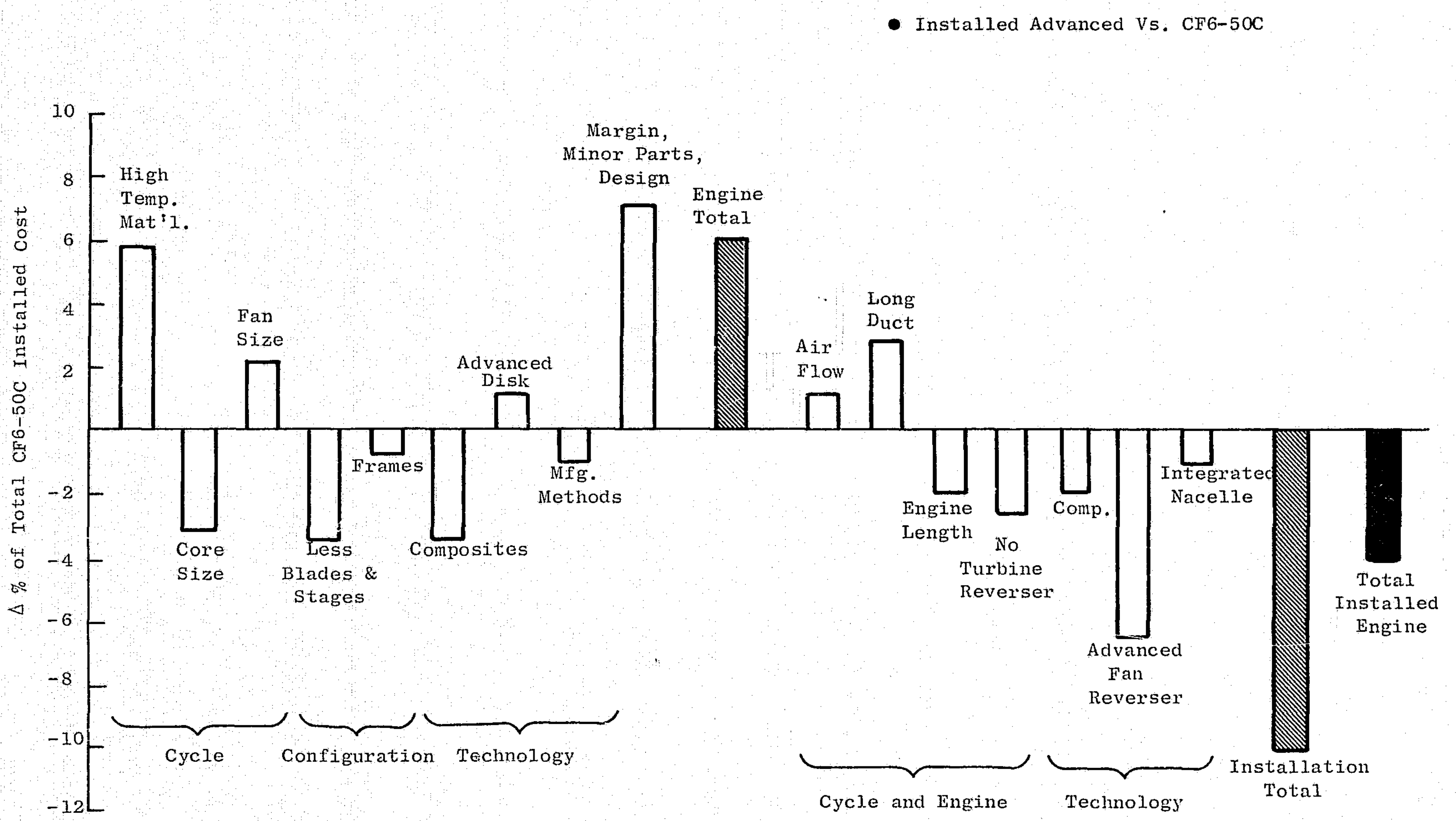


Figure 50. Relative Cost Changes.

The goal for the installed engine cost of the advanced engine was to match that of the CF6-50, scaled to same thrust, and compared at the same point in production run. The goal for maintenance costs was an improvement of 15% versus the scaled CF6-50.

The effects of meeting the engine technical goals, discussed above, are illustrated in Figure 51. The fuel saved ranged from 13.5 to 17.5%, if all the goals are met in the advanced propulsion system. As illustrated in Figure 39, the installed sfc improvement was predicted at 11%, 1% better than the 10% goal. The installed weight reduction was 24%, at the same cruise thrust as the CF6-50, which exceeded the 20% installed weight reduction goal indicated in Figure 51.

The DOC reductions estimated were just under 6% for the transcontinental aircraft, and 10% for the intercontinental aircraft; based on installed sfc and weight improvements only. Although the installed production engine cost was estimated as being the same for the advanced and CF6-50 engines, the engine market price may well be higher because of development cost and maturity effect. However, engine market pricing differences were beyond the scope of this study and are not included in Figure 51. The contribution of the 15% maintenance cost reduction on DOC is 1 to 1-1/2%, as illustrated in Figure 51. The installed sfc reduction contributed the largest portion toward the DOC and ROI benefits of the advanced aircraft.

G. Summary of Results - Task III

A preliminary design of the advanced technology engine identified in Task II was carried out. In order to illustrate the magnitude of improvement which could be achieved with a new engine incorporating advanced technology, comparisons were made with the CF6-50C engine, which is believed to be a good current engine from which to measure improvement in technology and performance. An improvement in installed sfc (including nacelle drag) of just over 10% was estimated in this study. It must be emphasized that this includes the effect of advanced technology in terms of component performance, cooling, and materials technology. A reduction in installed weight of 12% below that of the CF6-50C, scaled to the same take-off thrust, was obtained. Since the advanced engine ratings were set to provide relatively high cruise thrust, the resulting weight reduction becomes 24% when compared at the same cruise thrust. It was also estimated that the advanced engine, plus nacelle, would have a comparable cost level to a scaled CF6-50C at the same point in the production run.

The effects of installed engine improvements were estimated for the advanced aircraft defined for this study. An approximate 6% improvement in DOC, and 13% improvement in fuel usage, were shown for the transcontinental trijet. The corresponding improvements for the intercontinental quadjet were 10 and 17%.

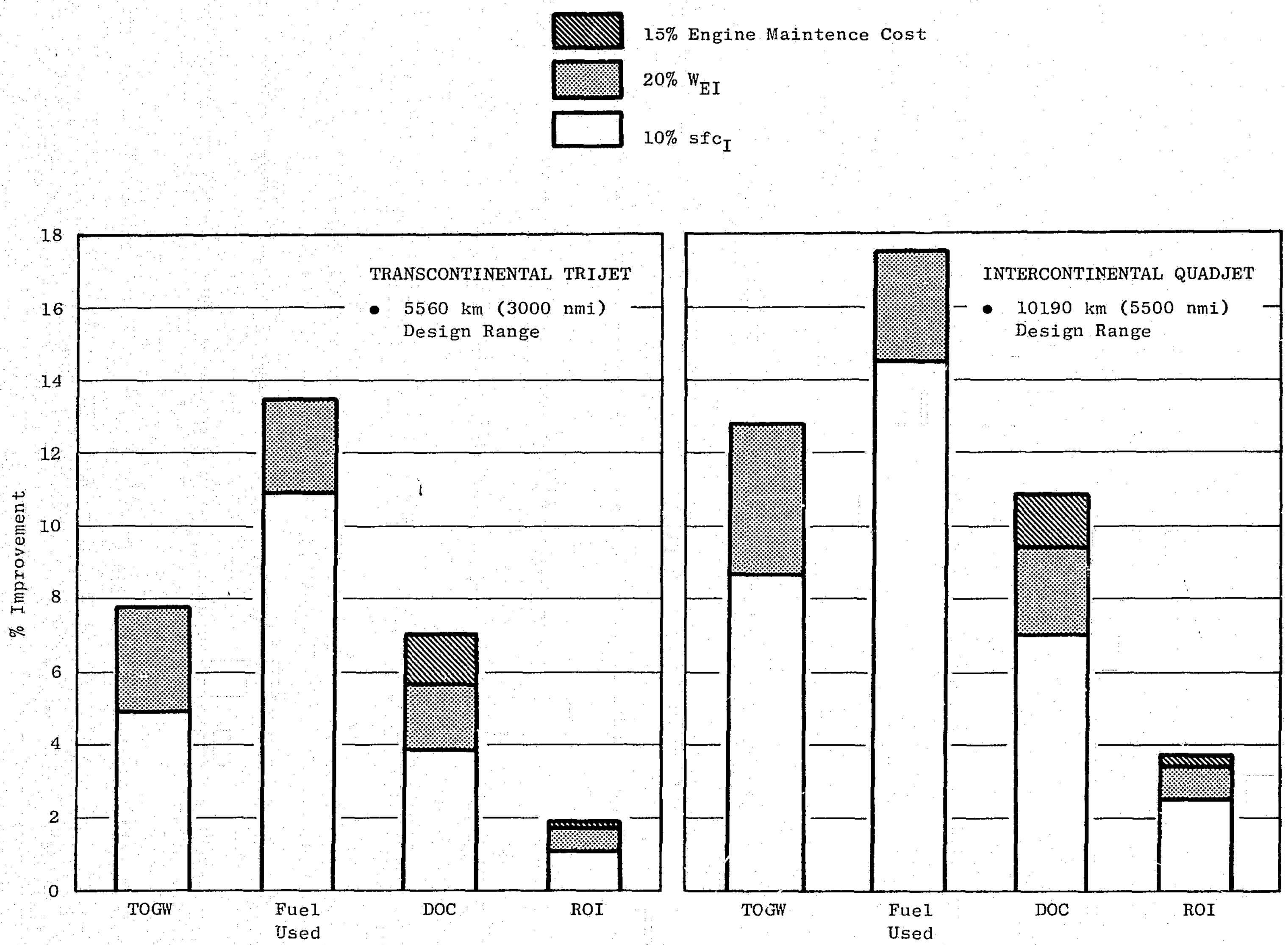


Figure 51. Effect of Advanced Engine Goals on Aircraft Merit Factors.

The advanced Task III engine was projected to meet the noise goal of FAR Part 36 minus 10 EPNdB for the aircraft defined in this study, utilizing the advanced noise-reduction technology identified.

The proposal 1981 EPA emissions requirements will require advanced combustor-technology features, but no acceptable approach to meeting the NO_X requirement has yet been identified.

SECTION VI

RECOMMENDED TECHNOLOGIES

Quantitative data, related to payoff of specific technology design features, was presented in Tasks I and II. Figure 52 is presented to summarize the payoff of the technology advances employed in the advanced engine relative to the technology of the current CF6 engines. Results of the STEDLEC study were supplemented by results from previous studies in constructing Figure 52. The two most significant advances, from an energy conservation standpoint, are the high thermal efficiency cycle (high turbine temperature and cycle pressure ratio) combined with advanced turbine technology and the advanced, mixed-exhaust system. All the advances, however, contribute to an improvement in aircraft DOC and are therefore important in justifying a new transport engine.

A. Energy Consumption

The technology needs, directed at reducing the energy consumption of advanced technology engines, are summarized below:

1. Engine technology allowing economical use of high thermal efficiency cycles:

    a. Advanced materials

    - Ni-base blades

    - Ceramics for static flowpath parts

    b. Engine arrangement for minimum cooling

    - Single-stage core turbine

It is believed that effort to advance technology is justified in all areas shown. The improvement indicated for high thermal efficiency cycles is large, provided that hot parts design technology can be developed for reasonable initial and maintenance costs. Improved, directionally solidified, Ni-base blade material not only will allow reduced cooling flows, because of the higher allowable metal temperature, but has the promise of improved low cycle fatigue characteristics. Ceramics are a good candidate for turbine shrouds, but the feasibility of alternate methods of designing ceramic vanes with impact-damage resistance must be explored.

2. Improved component performance:

    a. Fan; high tip speed fan compatible with composite blades

    b. Core compressor; high tip speed, compact, rugged design

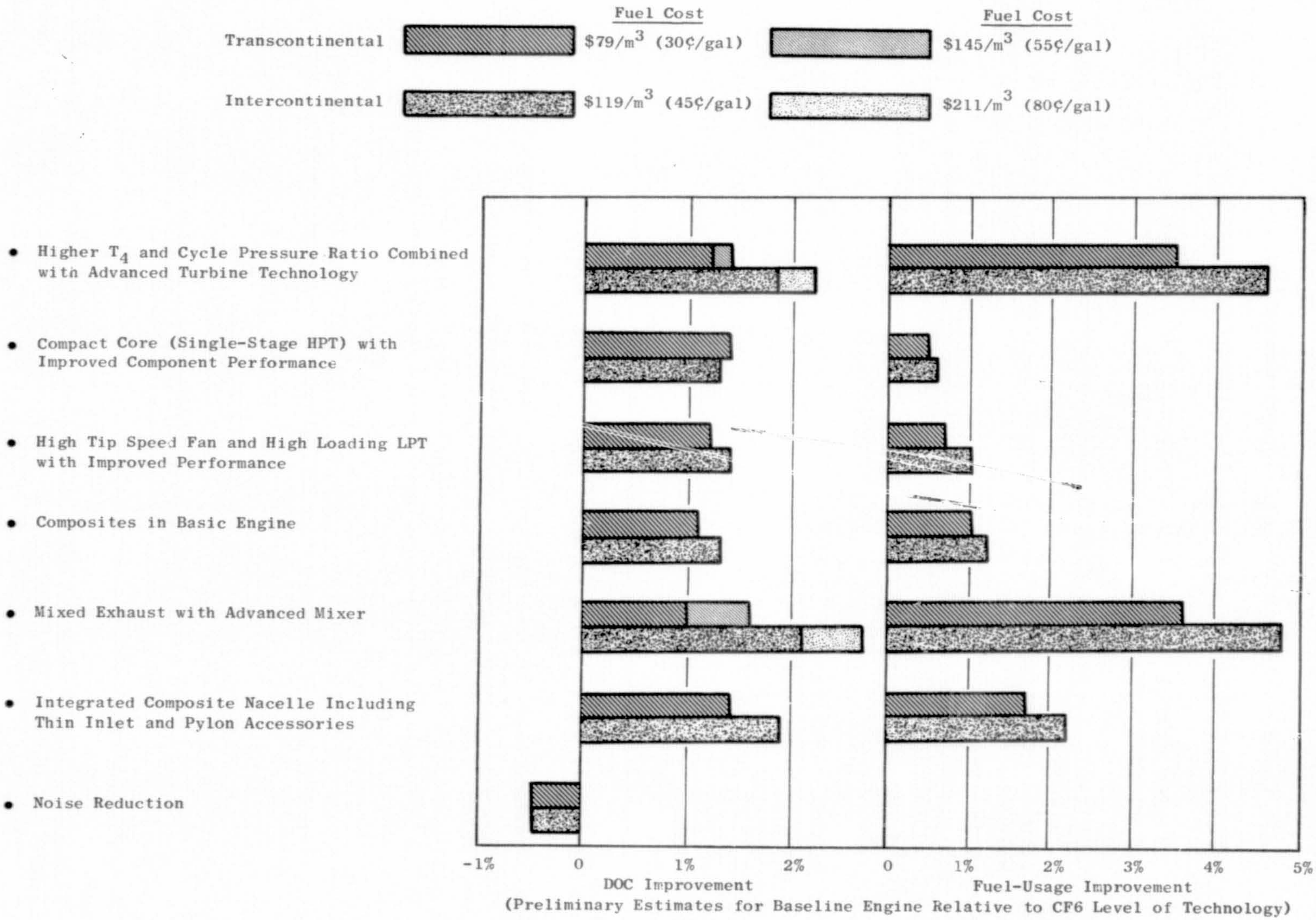


Figure 52. Technology Payoff Areas.

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c. Core turbine; single-stage, high pressure ratio design

d. Fan turbine; high work-coefficient design

e. Engine design and materials for clearance control

f. Engine design for efficient handling of cooling and leakage flows

g. Design features for reduced deterioration

Improved component performance is necessary in order for the compact arrangement, selected for the advanced 1985 engine, to provide the performance level projected in this report. If based on correlations of current technology, the designs selected would have lower component performance than the CF6. Figures 53 thru 56 illustrate the technology needs for each of the major components. An improvement in efficiency is projected, in each case, beyond that based on correlation of current technology.

The aerodynamic performance of the high tip speed fan must be obtained with low aspect-ratio blading compatible with composite construction. The core compressor has a relatively small number of stages (9) for its 14:1 pressure ratio, but an improvement in efficiency through a combination of aerodynamic refinement and clearance control is needed. The single-stage, high pressure ratio, core turbine provides an advantage in cooling air consumption, but improvements in the basic aerodynamics of the blade and end-clearance control are needed. The highly loaded fan turbine allows the relatively high bypass engine to be configured with a low number of stages, but development of the basic aerodynamic capability is necessary.

The basic engine must be designed to allow the full aerodynamic capability of the components to be achieved. This includes design features to maintain control of clearances during engine transients and to minimize the mechanical distortion of the engine under various operating conditions. Efficient means of handling the cooling and leakage flows, including use of advanced seal concepts, must also be a requirement of the engine design. Reduced in-service performance deterioration can also be obtained by the use of clearance-control design features, the use of erosion-resistant shrouds and coatings, and the use of rugged, low aspect-ratio blading, particularly in the high pressure section of the engine.

3. Cycle selection for good propulsive efficiency:

a. Fan pressure ratio and bypass ratio; balance of factors

b. Mixed-flow exhaust technology

The fan pressure-ratio/bypass-ratio combination was selected to balance energy consumption, DOC, noise, and growth capability. The mixed-flow exhaust system was a high-payoff feature, but development is needed to assure that the

Technology Need	Payoff
• High Tip Speed with Improved Efficiency	Engine Simplicity and Cost (Number of Stages)
• High Specific Flow with Good Efficiency	Engine Compactness Growth
• Low Aspect Ratio, Unshrouded and Stable Aero (Mechanical) Design	Allow Use of Composite Blades
• High Hub Loading	Reduces Booster P/P Req'd
• Integrated OGV/Frame	Compactness

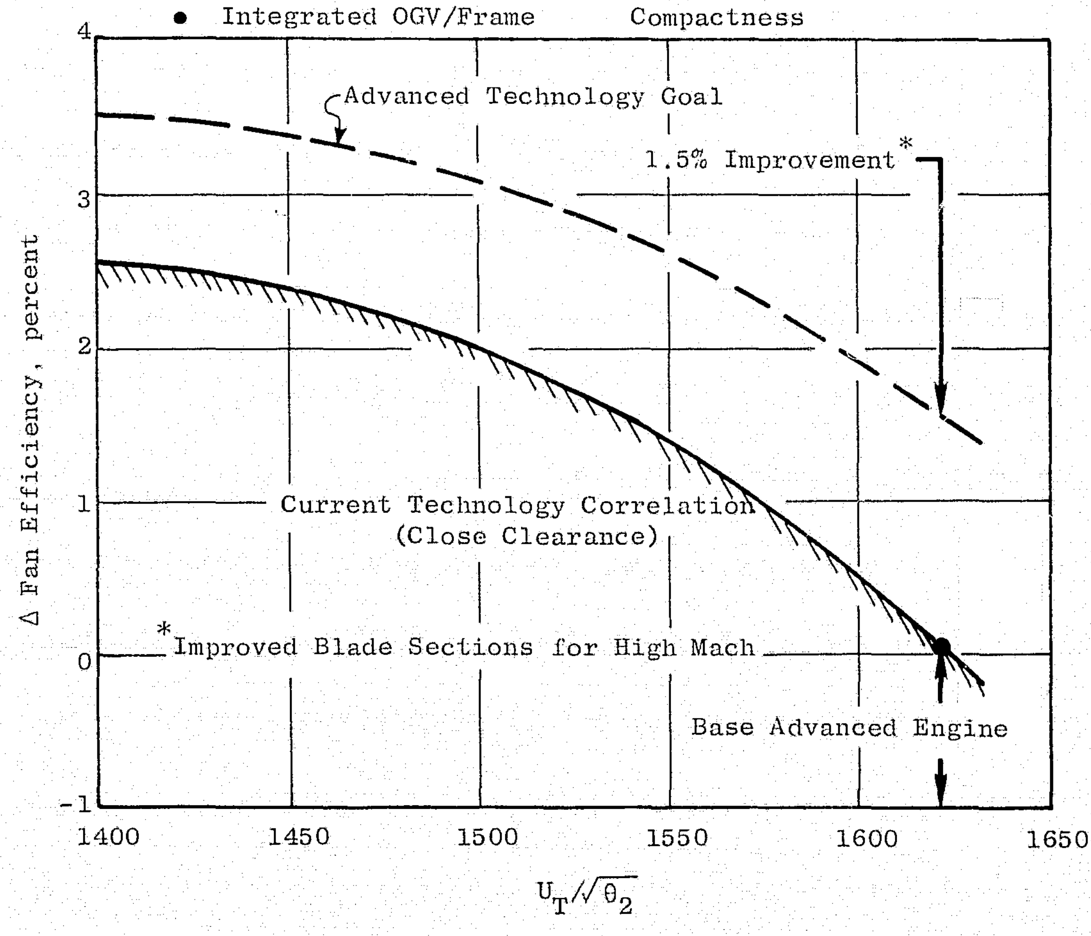


Figure 53. Fan Aerodynamic Technology Needs.

Technology Need	Payoff
• Improved Efficiency	sfc
• High Tip Speed and Radius Ratio	Engine Compactness and Cost (Number of Stages)
• Rugged Blades (Chord and Thickness)	Deterioration Maintenance Cost
• Variable Stator Boosters	Cycle Flexibility Elimate Booster Bleed

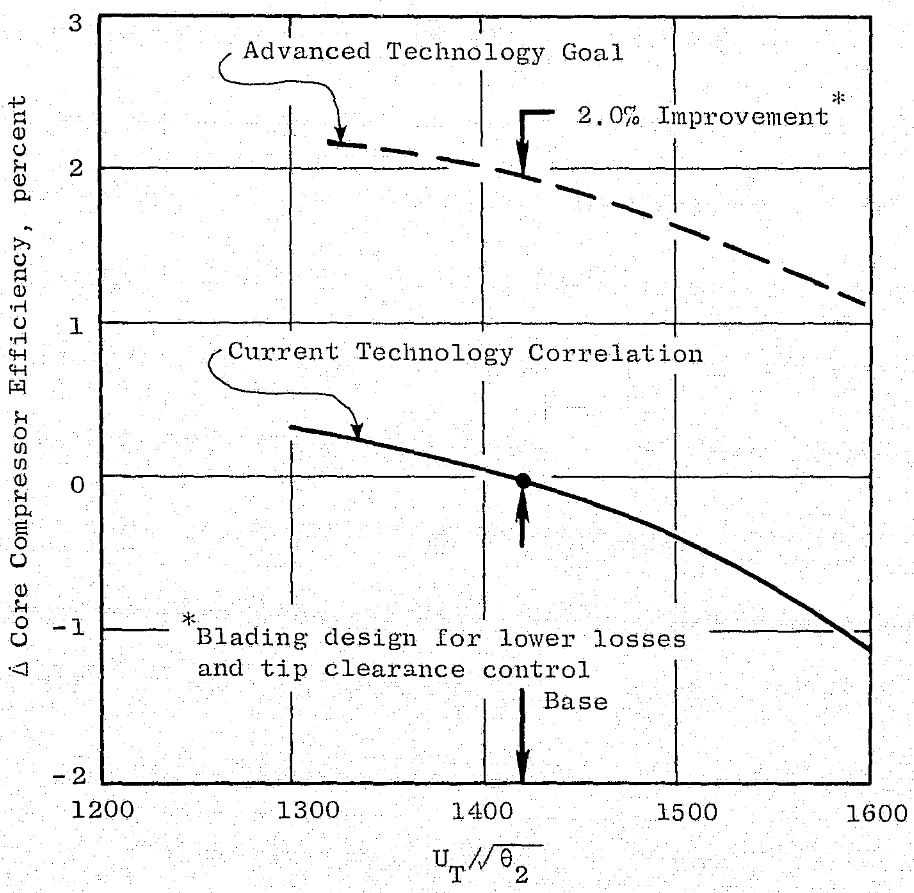


Figure 54. Compressor Aerodynamic Technology Needs.

Technology Need	Payoff
• Improved Efficiency Single Stage	sfc Engine Cost LPT Matching
• Cooling Design	Efficient Use of Cooling Air in Blade and Vane Longer Life Maintenance (Blade and Vane)

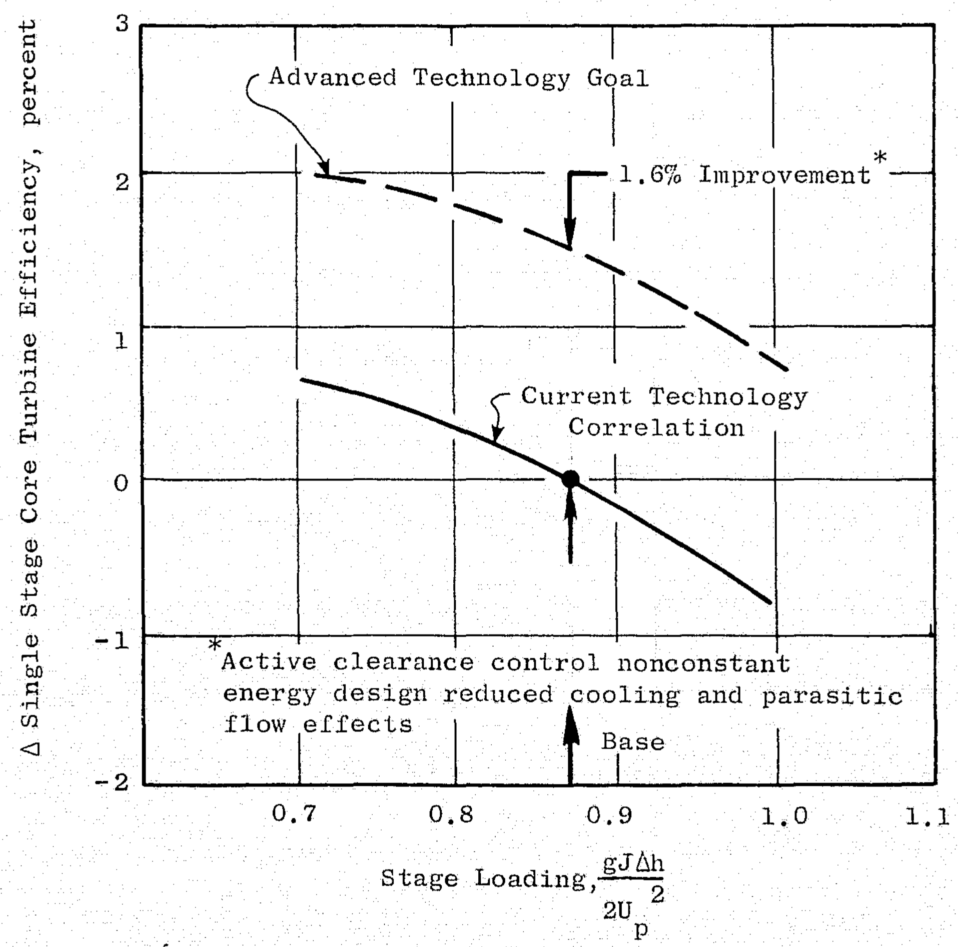


Figure 55. Core Turbine Aerodynamic Technology Needs.

Technology Need	Payoff
• High Loading with Good Efficiency	Cost and Compactness (Number of Stages)
• Blades Suitable for Cooling (Front Stages)	Higher T_4 Cycle for sfc
• High AR and Axial Space (Rear Stages)	Noise
• Integrated Exit Guide Vane/Frame	Compactness
• LP Blade Cooling	Efficient Use of Parasitic, sfc

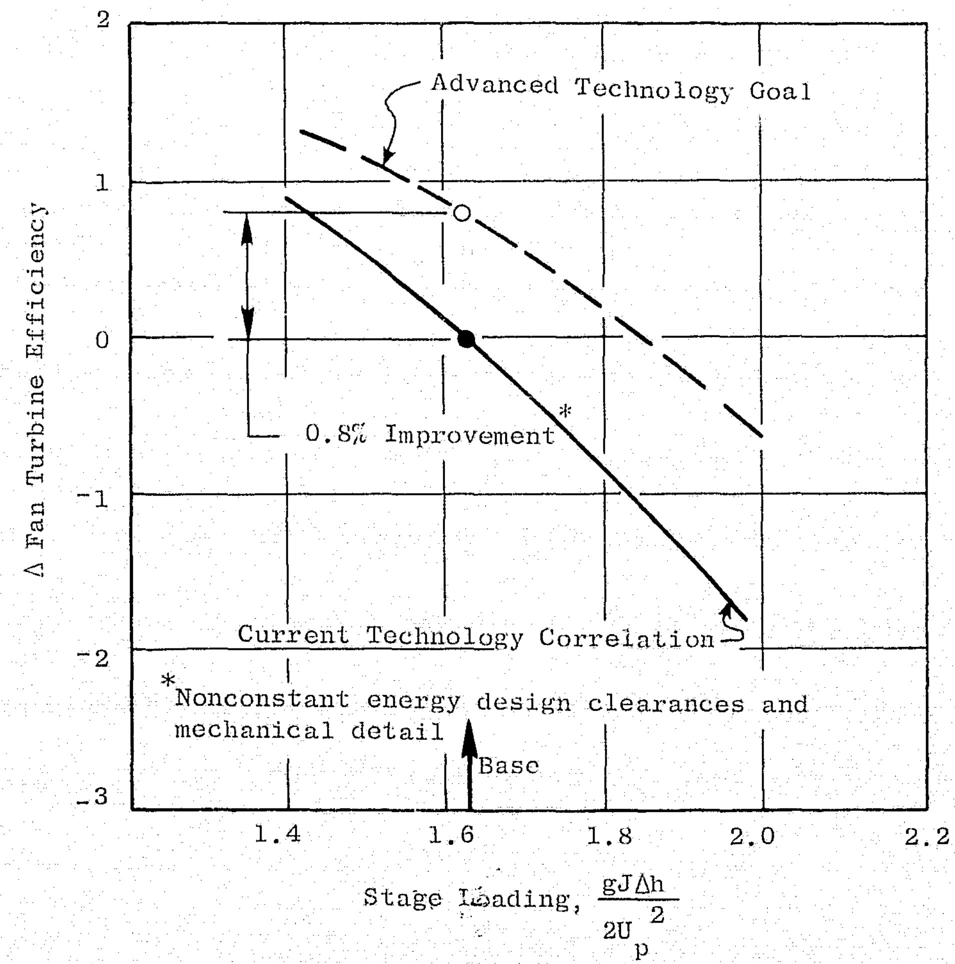


Figure 56. Fan Turbine Aerodynamic Technology Needs.

estimated levels of mixing effectiveness and losses can be achieved in an engine environment. Figure 57 illustrates the technology factors involved in the mixer for the advanced engine.

4. Minimum installation losses:

   a. Compact engine layout

   b. Efficient nacelle layout for M = 0.8 cruise

   c. Improved noise-suppression techniques

The nacelle aerodynamic design should be laid out to take advantage of the short length of an advanced engine and provide a minimum nacelle diameter, with accessories located in the pylon. Wind tunnel verification of such a layout, including interference effects, will be necessary. Improvements in noise-suppression techniques are necessary to meet the expected noise requirements for 1985 aircraft with normal-length inlets, and without the use of high loss splitters in the inlet or exhaust systems.

B. <u>Aircraft Economics</u>

In addition to improved fuel usage, a new engine must provide an improvement in aircraft economics in order to justify the development of such an engine. The technology features directed at reducing energy consumption, discussed above, are of course contributors to improved economics. In addition, there are additional technology needs directed primarily at improving economics, as summarized below:

1. Longer life and lower cost designs for high turbine temperatures:

   a. Design approach to balance all factors

   b. Materials technology

The hot parts of the engine are the major contributors to engine maintenance costs, and technology for long life and lower parts costs is necessary to make the high temperature, high pressure-ratio cycle acceptable to the airlines.

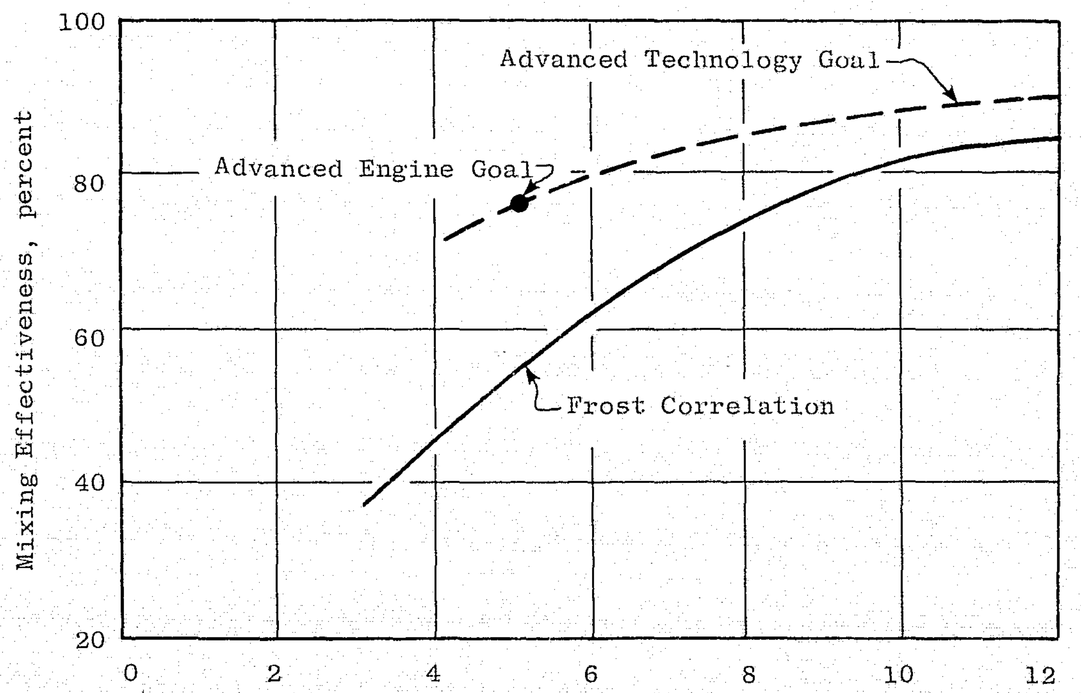
2. Engine design for minimum number of major parts:

   a. Advanced component aerodynamic technology - small number of stages

   b. Two-frame, five-bearing arrangement

The advanced engine has been laid out with a relatively small number of major parts, such as turbine stages and frames. The advanced component aerodynamic designs allow such an engine arrangement, but require development to provide appropriate efficiencies as described in the previous section.

Technology Need	Payoff
• Higher Mixing Effectiveness	sfc
• Shorter Mixer	Mixer Weight and Cost , Lower Losses
• Data Base (Analytical and Experimental)	Optimized Designs



$$\text{Mixing Parameter} = \frac{\text{Perimeter X Length}}{(\text{Equivalent Diameter})^2}$$

Figure 57. Mixed Exhaust Aerodynamic Technology Needs.

3. Use of composites in fan blade and frame

Composites in the fan section and nacelle can provide a substantial improvement in economics and, indirectly, fuel usage because of weight reduction. Much development work will be required to make composites practical. The fan blade, in particular, requires effort relative to the bird-strike problem.

4. Nacelle technology:

a. Composites; integrated with fan static parts

b. Improved reverser

The reverser system has historically been an area of concern, in airline service, in terms of installation components. The proposed, mixed-flow exhaust system will allow the elimination of the core reverser; and the proposed, advanced-concept, Task III fan-reverser system will result in a simpler, lighter, and more reliable configuration.

5. Advanced digital controls

Digital controls can be a contributor to improved DOC through the flexibility for additional functions, their possible integration with aircraft power management (including better engine protection and reduced pilot workload), and as an element of condition-monitoring system.

C. <u>Environmental and Safety</u>

Most of the above technology items are applicable in some degree to current engines or growth models of these engines.

A new engine must meet the noise and emissions requirements that will be in existence at the time it goes into service. The estimates shown in this report for the advanced engine assume that continued effort is applied and progress is made in both noise and emissions technology. Specific design features were described in Task III, but continued basic technology work in these areas is required.

It is also expected that a new engine will incorporate additional safety features which should be given attention from the technology standpoint. For example, composite blades (once they are developed for bird-strike capability) should have an advantage because of their tendency to fail in small fragments. Disc design technology for reduced chance of failure is another area where effort is justified.

SECTION VII

CONCLUSIONS

This report summarizes the results of the Study of Turbofan Engines Designed for Low Energy Consumption (STEDLEC). Design improvements and technology features were considered for the CF6 family of engines and evaluated in DC-10 type aircraft. Advanced technology features suitable for a new engine, which could enter service in 1985, were also evaluated in terms of the potential for improving energy consumption for advanced, subsonic transports. A specific design was laid out and compared to a current-technology, high bypass turbofan. Overall conclusions drawn from this study are as follows:

- CF6 Engine

Design changes categorized as design improvements, not requiring significant technology development, were identified which indicated a potential for 1-1/2 to 2% improvement in sfc. Basic engine design changes, which required technology development of various degrees, were identified which indicated a potential for 2 to 3% sfc improvement. The long-duct, mixed-flow design using composites was predicted to have the potential for 3 to 3-1/2% installed sfc imrovement. The changes studied will require substantiation thru rig and engine testing. Based on past experience, the total of the estimated improvements will normally not be achieved. In addition, certain items, particularly on a retrofit basis, do not show any advantage in aircraft economics.

- Advanced Engines

1. An increase in turbine temperature and cycle pressure ratio to 1427° C (2600° F) and 38:1 respectively, can provide a significant advantage in fuel usage in an engine incorporating advanced turbine technology. A corresponding improvement in DOC will also be obtained, provided that suitable life and endurance is developed into the advanced design hot section of the engine.

2. The mixed-exhaust system has the potential for a significant improvement over a separate-flow cycle and exhaust system. The fan pressure-ratio/bypass-ratio choice of 1.7 and 7:1 respectively represents a balance between DOC, energy consumption, and growth capability.

3. Component performance improvements, both aerodynamic and mechanical-design related features, such as clearance control, are necessary in order that the estimated fuel usage and DOC improvements can be achieved with the compact engine design shown.

4. Advanced materials, including composites in the fan section and ceramics in the hot flowpath static parts, have the potential for significant improvements, but much technology work is required before they can be incorporated in an engine design.

5. For a typical advanced-engine design incorporating the above technology, an improvement in installed sfc of over 10% is estimated, combined with a reduction in installed engine weight of about 20%. This results in an improvement in energy consumption of 13 to 17% for the transcontinental and intercontinental 0.8 Mn aircraft respectively.

6. Technology improvements are necessary in all areas of the engine and installation, in order to provide the following requirements for an advanced engine; a) a major improvement in energy consumption, b) sufficient improvement in aircraft economics to justify a new engine, and (c) environmental characteristics suitable for an all-new engine. An agressive research and development program directed at the technology of such an engine is necessary.

NOMENCLATURE/SYMBOLS

AIA	Aircraft Industry Association
ATA	American Transport Association
CDP	Compressor Discharge Pressure
CG	Center of gravity
C_L/L	Clearance ÷ blade height
DHL	Inlet highlight diameter, m (ft)
DHL	Inlet highlight diameter, m (ft)
$D_{Max.}$	Nacelle maximum diameter, m (ft)
DOC	Direct Operating Cost
D_T	Fan tip diameter, m (ft)
EPA	Environmental Protection Agency
F_m	Installed net thrust (net thrust minus drag), N (lb)
Fn or F_n	Net thrust, N (lb)
FOD	Foreign Object Damage
HPT	High Pressure Turbine
L	Inlet length, m (ft)
LPT	Low pressure turbine
M or Mn	Mach number
MA754	Dispersion-strengthened alloy
MARM509	High temperature nickel alloy
MxCl	Maximum climb rating
MxCr	Maximum cruise rating
N_B	Number of fan blades

Ni76XB	Advanced, directionally solidified, Nickel alloy
NiCrAly	Turbine shroud filler material
NO_x	Oxides of nitrogen
PAX	Passenger
PI	Cost ratio, replacement parts ÷ new parts
P/P	Pressure ratio
R95	Current, high temperature, Ni-base alloy (disc material)
R125	Current, high temperature, Ni-base alloy (blade material)
r_H	Hub radius, m (ft)
ROI	Return on investment
r/r	Radius ratio
r_T	Tip radius, m (ft)
SDOF	Single Degree of Freedom
sfc	Specific fuel consumption, kg/N-hr (lb/lb-hr)
sfc_I	Installed sfc, kg/N-hr (lb/lb-hr)
STEDLEC	Study of Turbofan Engines Designed for Low Energy Consumption
T_{41}	Turbine rotor inlet temperature, ° C (° F) or ° K (° R)
TO or T/O	Takeoff (Take-off power)
TOBL	Takeoff-Balanced Field Length
$U_H/\sqrt{\theta_2}$ B	Hub speed corrected to booster-rotor inlet conditions, m/sec (ft/sec)
$U_T/\sqrt{\theta_2}$	Corrected tip speed, m/sec (ft/sec)
V_9	Primary-jet velocity, m/sec (ft/sec)
V_{29}	Fan jet velocity, m/sec (ft/sec)
W_{2C}	Core compressor flow, kg/sec (lb/sec)
W_c	Turbine cooling flow, kg/sec (lb/sec)

W_{EI}	Weight of installed engine, kg (lb)
W_f	Fuel flow, kg/sec (lb/hr)
$W\sqrt{\theta}/\delta$	Standard day corrected airflow
η_{mix}	Mixer effectiveness, $\frac{\text{Ideal thrust, mixed - Actual thrust, mixed}}{\text{Ideal thrust, mixed - Thrust, separate flow}}$
$\overline{\psi}_p$	Mean loading of turbine stage

REFERENCES

1. NASA CR-121016, "Propulsion System Studies for an Advanced High Subsonic, Long Range Jet Commercial Transport Aircraft," by General Electric Company, prepared for NASA, November, 1972.

2. NASA CR-134495, "A Study of Engine Variable Geometry Systems for an Advanced High Subsonic Long Range Commercial Transport," by M.A. Compagnon, General Electric Company, prepared for NASA, October, 1973.

3. NASI-13356, "Conceptual Design Study of Advanced Acoustic-Composite Nacelles," by Douglass Aircraft Company/General Electric Company, prepared for NASA, April 2, 1975.

4. NASA CR-134696, "Study of the Costs and Benefits of Composite Materials in Advanced Turbofan Engines," by C.A. Steinhagen, C.L. Stotler and R.E. Neitzel, General Electric Company, prepared for NASA, October, 1974 under Contract No. NAS3-17775.

5. NASA CR-134702, "Cost Benefit Study of Advanced Materials Technology for Aircraft Turbine Engines," by E.W. Ross, R.P. Johnston and R.E. Neitzel, General Electric Company, prepared for NASA under Contract No. NAS3-17805, November 1974.

6. NASA CR-132608, "Fuel Conservation Possibilities for Terminal Area Compatible Aircraft," prepared under Contract No. NAS1-12018 by Boeing Commercial Airplane Company, for NASA, March 1975.

7. "A Proposed Standard Method for Estimating Airline Indirect Operating Expense," Report No. LW700-500R, May 1970 by R.F. Stoessel.