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Contingency Power Concepts for Helicopter Turboshaft Engine

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CONTINGENCY POWER CONCEPTS FOR HELICOPTER TURBOSHAFT ENGINE

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Abstract

Twin helicopter engines are often sized by power requirement of safe mission completion after the failure of one of the two engines. This study was undertaken for NASA Lewis by General Electric Co. to evaluate the merits of special design features to provide a 2 1/2 minute Contingency Power rating, permitting an engine size reduction. The merits of water injection, cooling flow modulation, throttle push and an auxiliary power plant were evaluated using military life cycle cost (LCC) and commercial helicopter direct operating cost (DOC) merit factors in a "rubber engine/rubber aircraft" scenario.

Notation

CRP	Contingency rated power
DOC	Direct operating cost
GW	Gross weight
HPT	High pressure turbine
IRP	Intermediate rated power
LCC	Life cycle cost
MC	Maximum continuous
NH	Compressor RPM
OEI	One engine inoperative
OWE	Operating weight empty
SLS	Sea level static
TOP	Takeoff power
T41	Turbine inlet temperature

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Introduction

Suppliers of military and civil helicopter engines have been aware of the need for higher contingency power ratings for future rotorcraft. These suppliers have both civil and military engines currently qualified with CRP using conventional approaches. For example, the GE T700-401 (LAMPS) engine is designed and qualified to meet a 2 1/2 minute contingency power requirement, 10% greater than intermediate rated power. However, savings in both DOC and LCC can be achieved by still higher contingency power ratings.

An incentive to provide higher contingency power for a given set of OEI requirements is downsizing the engine which allows reductions in rotorcraft size reductions as well as savings in operating costs. However, the design features required to provide contingency power, while maintaining constant life, involve penalties in powerplant SFC, weight and cost, which partially offset the downsizing benefit.

The results of a systematic study of several contingency power concepts performed by General Electric Company's Aircraft Engine Business Group for NASA Lewis in 1983* are reported in this paper. The scope of the study included a definition of turboshaft engine designs with varying contingency power capability to establish a parametric approach to the "cost" of including this feature. The changes in the engine design were established and included in rotorcraft design, performance and economic penalties in terms of life cycle cost (LCC) and direct operating cost (DOC) for the military and civil rotorcraft, respectively. Rotorcraft design support was supplied by Sikorsky Aircraft under subcontract to General Electric Company.

* NAS3-23705

The basic approach taken was to design all engines for equal life whether they have contingency power ratings or not. The design changes necessary to accomplish this objective result in penalties in propulsion system weight, cost and SFC. These were balanced against the advantages of using the smaller engine to obtain a quantitative evaluation of the net system gain.

Baseline Rotorcraft & Engines

Conventional helicopter designs were chosen for the baseline civil & military rotorcraft. These designs were judged to represent a significant segment of the rotorcraft market in the 90's. The levels of cruise design speeds are logical for the two missions. In the low speed case, one engine inoperative (OEI) takeoff is clearly sizing the engines while in the high speed case cruise power is close to becoming the sizing condition.

The civil rotorcraft is a 30 passenger commuter with 250 statute mile design range. Cruise speed is 160 kts at 3000 ft altitude and fuel reserve is sufficient for 25 statute miles plus 30 minutes at 160 kts. Power sizing criteria for the engines is determined by hover in ground effect (HIGE) with OEI at 1000 ft altitude, 82.4^{o}F ambient temperature (ISA + 15^{o}C) equivalent to Cat A design criteria. The rotorcraft design features are detailed in Table 1.

The military rotorcraft is a 24 troop marine assault transport (Helicopter Experimental Marine, HXM) with 200 nautical miles design radius; that is, 200 nautical miles out with troops and gear plus 200 nautical miles return without payload. Design speed is 180 kts and the power sizing criteria for the engines is determined by hover in ground effect with one engine inoperative at 3000 ft altitude, 91.5^{o}F (hot day) also equivalent to Cat A design criteria. The design features are also listed in Table 1.

The technology level of the engines was selected to be consistent with qualification in the late 80's. The baseline turboshaft engine for both the civil and military vehicles consists of a 20:1 pressure ratio class axial-centrifugal compressor, a two stage cooled high pressure turbine, an annular combustor, and a three stage uncooled free power turbine. The 100^{o}F lower turbine inlet temperature for the civil engine was chosen because of the need for more mission life than in the military engine. Contingency power of +15% over intermediate rated power (IRP) or takeoff Power (TOP) was available for the baseline engine at the high ambient temperature design point as being representative of what would be designed into a new engine. Smaller power increases are available at cooler ambient temperatures where corrected speeds

Table 1. Baseline rotorcraft Characteristics

	CIVIL COMMUTER	MILITARY TRANSPORT
Entry into Service	1990	1990
Type	Conventional Single Main Rotor Helicopter	
Gross Weight, lb	26000 Class	35000 class
Cruise Speed, kts	160	180
Payload, lb	6000	5760
Passengers	30	24 Troop & gear
Rotor dia, ft.	66	66
No. of Engines	2	2
SLS IRP SHP Class	3000	5000
Fuel Weight, lb	2500 plus	5500 plus
Empty Weight, lb	17000 plus	21000 plus

rather than turbine temperature, may be limiting. A summary of the engine cycle is provided in Table 2.

fuel consumption increase resulting from the turbine cooling flow and turbine efficiency changes.

Table 2. Cycle for baseline engines

	CIVIL	MILITARY
Sizing Condition, Alt/Mo/To	1000K/5/0/82.4°F	3000/5/0/91.5°F
SHP @ sizing Condition, TOP or IRP	2690	3940
CRP/TOP or CRP/IRP	1.15	1.15
Design Corrected Flow class, lb/sec	15	22
Cycle Pressure Ratio	20:1 Class	20:1 Class
Turbine Rotor Inlet Temperature at TOP or IRP, °F	2400°F	2500°F

Engine Sizing Procedure

The direct approach to increasing the contingency power ratio beyond 1.15 is by "throttle push" to greater physical rotor speeds and turbine inlet temperatures. Increases in rotor structural weight and turbine cooling flow are required to maintain engine life at the baseline engine level and these will have a detrimental impact on engine weight, power and fuel consumption. Beyond contingency power ratios of approximately 1.30, the compressor must be rematched to lower speeds at IRP or TOP in order to provide sufficient flow capacity, stall margin and compressor efficiency at CRP. This results in a larger, heavier engine (to maintain IRP) and the attendant reduced cycle pressure ratio at normal power settings is reflected in higher fuel consumption.

Fig. 1 shows how two different engines with CRP/TOP ratios of 1.15 (baseline), and 1.35 are matched in terms of compressor corrected airflow versus corrected speed for the civil application. This curve reflects a typical compressor characteristic and shows how increasing the CRP/TOP ratio reduces takeoff power to a lower corrected airflow.

Table 3 lists the increase in compressor flow size required to maintain constant takeoff power and the specific fuel consumption increase resulting from the turbine cooling flow and turbine efficiency changes.

Similar information is provided for the military engine in Table 4. Note that the compressor flow increase required (Δ%W2R) is 13.6% for the military engine versus 10.3% for the civil engine at CRP/IRP or CRP/TOP of 1.465. This is a result of the military engine operating at 100°F hotter turbine inlet temperature at IRP than the civil engine at TOP although both engines are limited to approximately the same turbine rotor inlet temperature at CRP.

Therefore, the increase in temperature from IRP to CRP is less for the military engine than TOP to CRP for the civil engine and this results in a

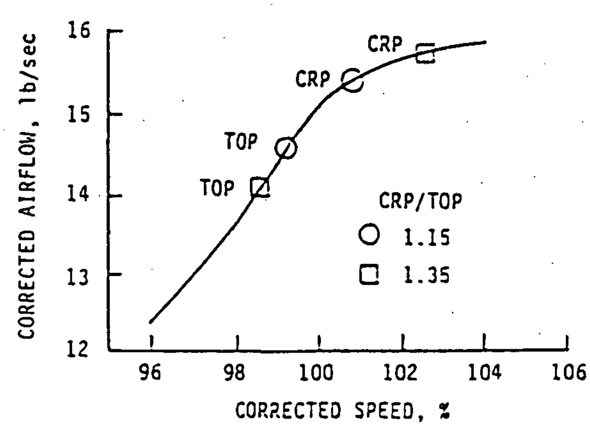


Fig. 1. Compressor flow-speed characteristic

Table 3. Cycle design point summary for civil rotorcraft at constant takeoff power.

	NOM. CRP/TOP = 1.15		NOM. CRP/TOP = 1.25		CYCLE REMATCHED NOM. CRP/TOP = 1.35		CYCLE REMATCHED NOM. CRP/TOP = 1.45	
	← BASE →		← THROTTLE PUSH →					
	TOP	CRP	TOP	CRP	TOP	CRP	TOP	CRP
Δ % SFC	BASE	---	+.72	---	+1.6	---	+2.8	---
Δ % Design Flow	BASE	---	+1.7	---	+5.3	---	+10.3	---
Δ T41, °F	---	BASE	---	+140	---	+215	---	+285
Δ % NH	BASE	+1.5	BASE	+3.2	-0.8	+3.2	-1.6	+3.2

Table 4. Cycle design point summary for military rotorcraft at constant intermediate rated power.

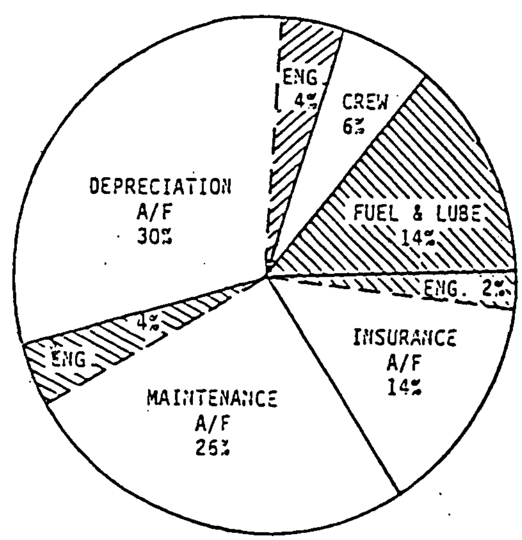
	NOM. CRP/IRP = 1.15		NOM. CRP/IRP = 1.25		CYCLE REMATCHED NOM. CRP/IRP = 1.35		CYCLE REMATCHED NOM. CRP/IRP = 1.45	
	← BASE →		← THROTTLE PUSH →					
	IRP	CRP	IRP	CRP	IRP	CRP	IRP	CRP
Δ % SFC	BASE	---	+.36	---	+1.0	---	+1.8	---
Δ % Design Flow	BASE	---	+1.0	---	+4.7	---	+13.6	---
Δ T41, °F	---	BASE	---	+125	---	+205	---	+205
Δ % NH	BASE	+1.1	BASE	+2.6	-0.9	+2.6	-2.9	+0.7

relatively greater increase in compressor flow size required for the military engine. The absolute turbine temperature limit was set in order to keep the low pressure turbine blades uncooled.

As the contingency power ratio increases, the engine weight, specific fuel consumption, engine acquisition cost and maintenance cost all increase for a fixed IRP SHP level. These penalties are subtracted from the benefits of idealized engine and rotorcraft downsizing due to increased contingency power ratio. The net gain is measured in terms of decreased vehicle gross weight, mission fuel burned, direct operating cost (DOC) and life cycle cost (LCC).

Rotorcraft DOC and LCC Models

The DOC and LCC models were developed for the baseline rotorcraft by Sikorsky Aircraft. The DOC model assumptions for the civil commuter are listed in Table 5 and the resulting DOC distribution is provided in Fig. 2. Since the engine



1% DOC = 200 MILLION $ FOR FLEET OF 200 AIRCRAFT PER 2000 HRS/YEAR OVER 10 YEARS

Fig. 2. DOC distribution/commuter

maintenance, insurance and depreciation collectively represent only 10% of the total DOC distribution, engine downsizing results in a modest DOC reduction.

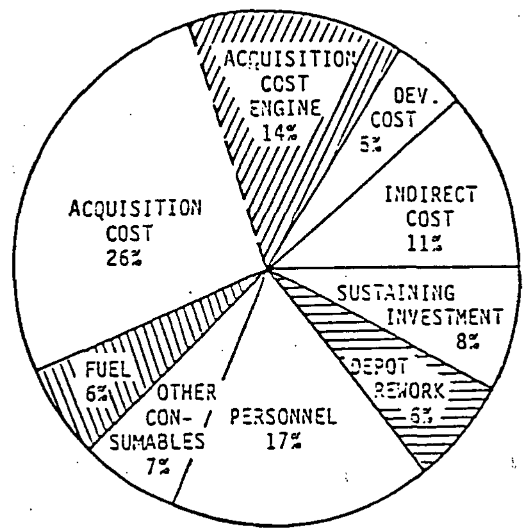
The military engine LCC model assumptions and resulting distribution of LCC are shown in Table 6 and Fig 3. Note that the % LCC changes are given from a base of the active weapon system LCC, not engine only.
Mission trade factors on a "rubber engine/rubber aircraft" basis relate changes in engine contingency power ratio, weight, SFC, cost and maintenance to changes in vehicle gross weight, LCC, DOC, acquisition cost, fuel burned, and OWE. These trade factors were used to compare the net benefit of the various contingency power concepts designed for equal life.

Table 5. DOC model assumptions - commuter.

1983 Dollars
Production 400 Units
Development Cost Amortization into Vehicle Price
Airframe Maintenance Burden, 1.5 X Direct Labor
Engine Burden 2 X Direct Labor
Insurance is 4% of Airliner Price
Aircraft Spares 15%
Engine Spares 30%
Straight-Line Depreciation, 10 Years to 25% Residual
Fuel $1.00/Gallon
Maintenance Labor $15/Hour
Utilization 2,000 Hours/Year
Crew Costs -- $35,000/Year/Pilot for Two Pilots (80 Hours/Month)
Off-Design Mission, 65% Load Factor

Table 6. LCC model assumptions - military transport.

1983 Dollars
Production 400 Units
Marine Assault Helicopter -- LCC Model
Fuel $1.00/Gallon, 2% Spillage
Utilization 360 Hours/Year Nominal Value
20 - Year Life Cycle



1% LCC = 100 MILLION $ FOR FLEET OF 400 AIRCRAFT FOR 20 YEARS

Fig. 3. LCC distribution/military transport

Novel Concepts - Descriptions - Design Changes, Operation, System Logistics

Several novel systems were proposed as alternative approaches to throttle push to satisfy the increased CRP demand. Each system was defined in sufficient detail to permit an evaluation of required engine system design changes. Seven systems were selected for preliminary screening and five of these were chosen for quantitative evaluation as shown in Table 7.

Table 7. Contingency power concepts

Initial Screening	Selected for Evaluation
Throttle Push	X
Cooling Flow Modulation (Rotor Only)	X
Cooling Flow Modulation (Rotor & Stator)	X
Water Injection at Compressor Inlet	
Water Injection into Turbine Cooling System	X
Water Injection into Compressor and Turbine Cooling System	X
Solid Propellant Powered Emergency Power Unit (EPU)	

Throttle Push

The throttle push concept is basically the same as the baseline engine design with the hot section redesigned and cooling flow system resized for constant engine life. Other design changes such as increased disk thicknesses for higher RPM capability are also required. No special add-on system or components are required except for control changes. These engines were evaluated and compared to the baseline engines to account for all the consequences of cycle changes required to obtain the higher contingency power ratios illustrated in Tables 3 and 4. The screening results indicated that this approach was promising enough to merit a quantitative evaluation.

Cooling Flow Modulation (Rotor Only)

In this concept, an electrically operated modulating valve is installed into the rotor cooling supply so that stage 1 HPT rotor blade cooling flow can be adjusted from its maximum at contingency power level to the lower level adequate during the remainder of its mission. This eliminates the over cooling penalty in the throttle push concept where cooling flows are set by blade coating limits at contingency and then remain the same at other power levels. This concept was selected for detailed study since the screening results were favorable.

Cooling Flow Modulation (Rotor and Stator)

This concept has the added feature of also increasing the cooling flow to the stg 1 turbine shroud but only when contingency power is required. Besides reducing the SFC at powers $\leq$ takeoff power by lower cooling flows, this feature also takes on the function of an active clearance control, by shrinking the shroud together with the blade, thus improving turbine efficiency at contingency vs. the rotor only cooling flow modulation system. Screening results also indicated merit in this concept worthy of quantitative evaluation.

Water Injection into the Compressor Inlet

This is a well-known concept for increasing engine power, where a water/antifreeze mixture is sprayed through nozzles into the engine inlet, increasing engine flow and reducing compressor work. However, the water/air ratio is limited by the adverse effect of water on compressor stall margin. Also, the buildup compressor clearances must be opened up so that rubs do not occur when the water is turned on at contingency power. This causes an SFC penalty during dry operation. In addition, the weight of water, tank, and nozzles becomes a sizeable penalty, further aggravated by the complexity and logistics problem of supplying demineralized water. As a result of these screening results the concept was eliminated from further consideration.

Water Injection into the Turbine Cooling Flow

In this concept, HPT blade temperatures during contingency operation are controlled by reducing the coolant temperature by water evaporation rather than by an increased flow as in the cooling flow modulation system. The weight of water required per contingency event is much less than that of the compressor water concept but all the other system components (pump, tank, plumbing, valves and controls) are still required. In addition, some losses will be experienced as a result of the stage 1 HPT blade shrinkage which would increase tip clearance and thereby cause some reduction in turbine efficiency. Nevertheless, screening results were favorable and this system was carried into the quantitative phase of the study.

Water Injection into Compressor and Turbine Cooling System

This combination was proposed to determine if using one common water tank, and water system to serve two functions could result in a synergistic advantage. This combination concept was also carried through the quantitative phase of the study.

Solid Propellent Powered Emergency Power Unit

During the screening study, a separate prime mover consisting of a turbine, a 3:1 gear reduction, and overrunning clutch and a solid propellant cartridge were defined for weight and size estimates. The weight estimates for a 2 1/2 min. duration system were so large, that there was no net benefit. The system was not competitive for durations greater than 30 seconds. Consequently, the concept was eliminated from further consideration.

Duty Cycles, Life Analysis and Limits

Qualification Tests and Duty Cycles

In order to calculate relative life usage for the various engine components studied, it was necessary to establish the power levels and usage in the engine qualification test as well as during the mission for both the civil and military engines.

Design changes for each contingency power concept were made as required to ensure engine life equal to the baseline engine life. These changes (primarily cooling flow increases) resulted in the design penalties included in the evaluation of each system.

For the civil engine, the test conditions are based on the FAA engine certification requirements of 25 cycles, each of 6 hours duration. Contingency power is reached twice during each cycle giving a total time at contingency of 125 minutes. The civil mission assumed is a combination of commuter and training segments, based on in-service experience with commercial operators. The total mission life is 2,000 hr/year for 10 years or 20,000 hours, of which 500 hours is allocated to training flights. Only one 2 1/2 minute contingency power operation is considered in the revenue part of the mission, while it is assumed that contingency power is reached for 30 seconds once in each of 1,050 training flights, giving a total time at contingency power of approximately 9 hours for the composite mission.

The test conditions for the military engine were based on the T700 "LAMPS" engine test requirements, which were thought to be representative for any future military application. The 300 hour test consists of fifty cycles, including four where contingency power is reached, for a total contingency time of 10 minutes. The military engine mission was derived from the time weighted average of eleven typical missions for the HXM Marine Assault helicopter. The total mission life is 7,200 hours over 20 years. Approximately 45 minutes of this time is spent at contingency power, consisting of 144 usages in the pilot proficiency part of the mission.

A summary of time at contingency power together with those at other powers is shown in bar chart form in Fig. 4 for the civil and military baseline engines.

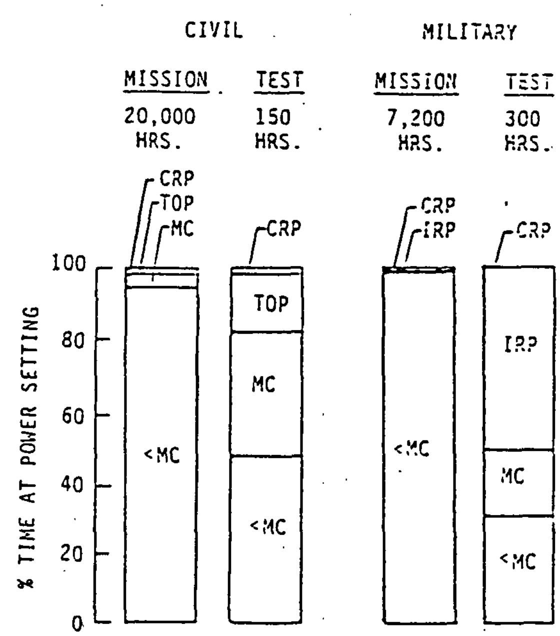


Fig. 4. Time at power level for baseline engine

At contingency power ratios greater than the 1.15 of the baseline engine the apportionment of power usage is unchanged for the tests, but has to be revised for the missions due to downsizing the engines which results from higher CRP/IRP ratios. In this case, the time spent at contingency power is unchanged, while the remaining power spectrum has more time at relatively higher powers than the baseline engines.

Components Studied For Effects Of Contingency Power

Fig. 5 lists the components which are affected by speed or temperature at contingency power and are considered to be important in this study. The cold parts, compressor disks and impeller, are only affected by an increase in speed from that of the baseline engine. Therefore, the stress levels in these components at contingency power may be brought back to and can be maintained at their original values by the addition of material. Higher gas temperatures both inside and outside the combustor liner can be accommodated by more elaborate cooling air patterns and distribution without adversely affecting the temperature profile at the combustor discharge. In the high pressure turbine, the nozzle and bucket airfoils are maintained at their temperature limitations by the addition of cooling air and redesign of the airfoil cooling passages. The first stage nozzle inner and outer bands require an additional cooling system for all contingency power operation.

The first stage shrouds need additional impingement cooling at contingency power to stay below or at limiting temperatures. There is no problem with the cooling plates as there is enough cooling to keep the cooling plates at or below acceptable temperature limits when additional air is required for bucket cooling. The high pressure turbine rotor disks require additional material at higher contingency ratios to allow for speed increases above the baseline engine level. The low pressure turbine blades are uncooled. Consequently, when higher contingency ratios are encountered, a material change is required to retain the uncooled blade feature. To allow for higher flowpath gas temperatures at contingency power, the interturbine frame has an additional air insulation shield around the oil passages in the struts to prevent oil coking.

Limits

The high pressure turbine rotor blade airfoils are limited by either the maximum surface temperature or the stress level, which are not necessarily at the same location. The integrity of the airfoil coating during the life of the blade determines the maximum allowable temperature. Creep rupture determines the life of the blade. In the case of the Stage 1 bucket, the coating temperature becomes more limiting than the creep rupture life requirements with increasing contingency power ratio, such that the creep rupture life used decreases with increasing contingency power ratio.

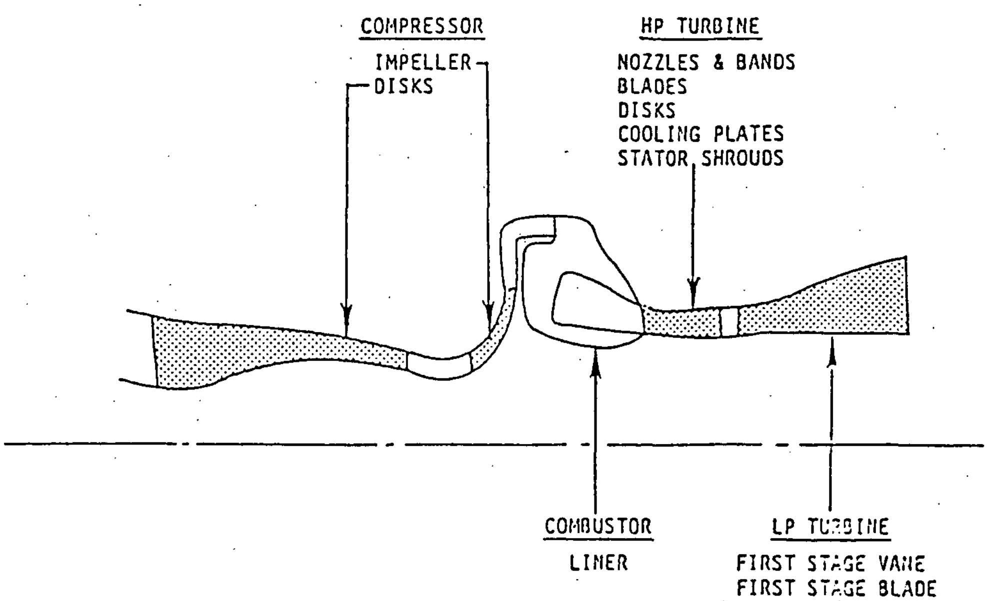


Fig. 5. Components studied.

In the case of the nozzle airfoils, the limits are determined by the airfoil coating allowable temperature or by the creep rupture stress. The nozzle blade bands are limited by the coating temperature.

The limit on the Stage 1 rotor shroud is the temperature at which the shroud segment bonding material starts to oxidize which would lead to segment separation.

The cooling plates attached to the high pressure turbine rotor disks direct cooling air into the blades on the forward sides of the disks. The rim temperature is limiting as the thermal gradient is approximately equal to the rim temperature minus the cooling air temperature and this gradient determines the rim hoop stress.

Creep rupture is limiting in the Stage 2 disk dovetail posts due to the difficulty in obtaining enough temperature differential for cooling purposes between the metal and cooling air.

The limiting consideration for the axial and centrifugal compressor rotors and also the high pressure turbine rotor is the burst margin whenever the speed exceeds that of the baseline engine.

Life Analysis

Component temperatures were generally calculated from a turbine inlet gas temperature, which included an allowance for combustor temperature distortion, and a cooling effectiveness, or the relative temperature of an uncooled component.

Creep rupture stress and temperature limiting locations were based on analyses performed on the baseline engine, as was the derivation of cooling effectiveness curves versus cooling air flow for the components studied.

For the baseline engine first stage bucket, it was determined that 100% life is used during the mission in the civil rotorcraft but during the qualification test for the military uses, for which the power spectra are shown in Fig. 4. The relative life usage for the civil and military missions and tests are shown in Fig. 6.

Fig. 7 shows an example of the life situation for the Stage 1 bucket military qualification test. Power usage has been divided into three categories; contingency, intermediate and part power. The part power component is a combination of all life usages for powers below intermediate allowing for different flight conditions, and is expressed as a percentage of intermediate rated power.

Whereas 100% of the life is used at the baseline contingency power ratio of 1.15, only 22% of the bucket life is used at a 1.35 contingency power ratio due to the increased cooling air required to satisfy the blade coating temperature limitation at contingency power, resulting in much lower life usage at intermediate and part powers. A small part of the reduction in life usage at intermediate power is due to a lower metal temperature which results from the lower cooling air temperature for the rematched engine at a contingency power ratio of 1.35.

In the life calculations, a speed increase at contingency power above that of the baseline engine is accounted for by decreasing the allowable metal temperature .

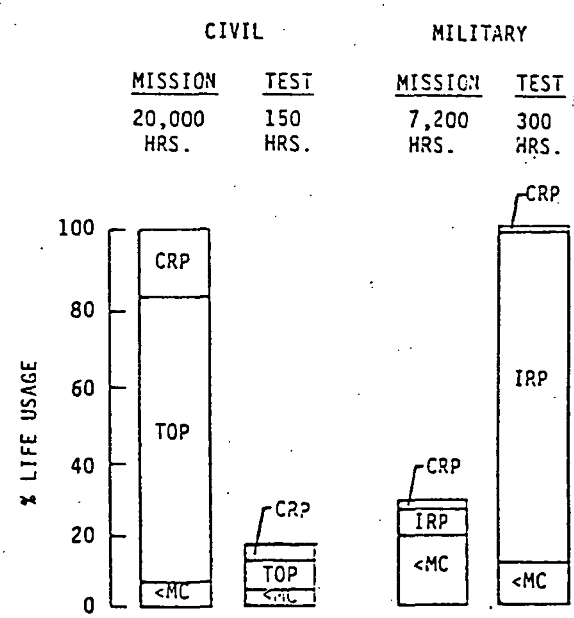


Fig. 6. Stage 1 bucket life usage/baseline engine

Results and Summary

The engine design penalties were compared to the rotorcraft benefits for each of the concepts and the net LCC or DOC reduction was then plotted vs. the CRP/IRP or CRP/TOP ratio for each engine considered.

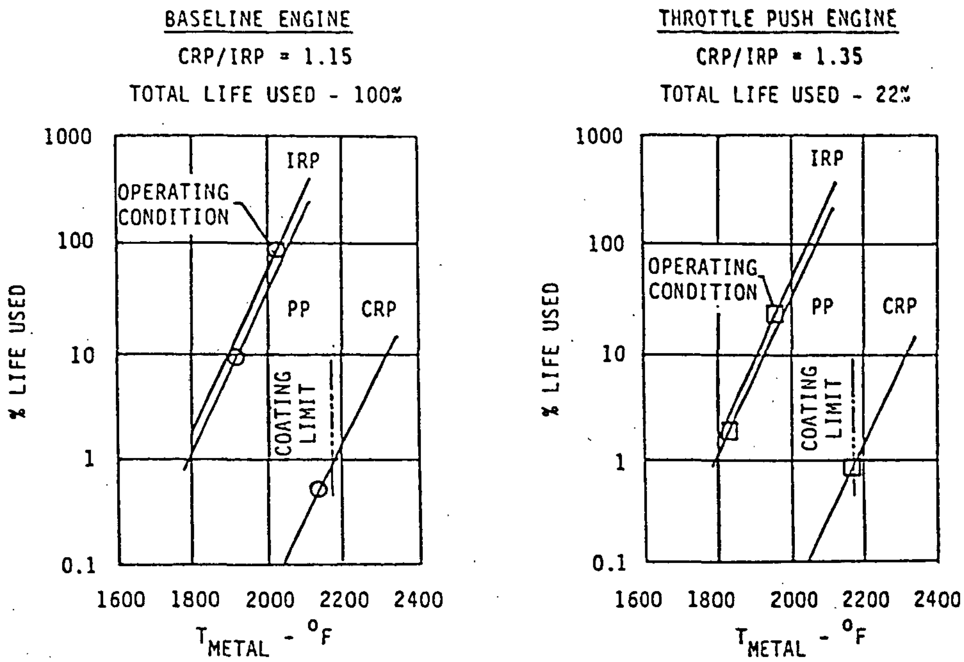


Fig. 7. Stage 1 bucket life usage in military qualification test

The results for LCC are plotted on Fig. 8 vs. CRP/IRP. The results for throttle push, cooling flow modulation and water into the turbine cooling system cluster in a relatively narrow band. The net LCC reduction achievable is in the range of 2.2% to 2.4% at the maximum useful CRP/IRP ratio of 1.40. Combined water injection into the compressor and turbine cooling system is poorer than the other concepts, primarily due to additional water weight and clearance effects. The net advantage of designing for higher CPR/IRP ratios diminishes beyond 1.30 for all concepts. This is caused by the rematching of engines to lower speeds at IRP in order to provide for compressor flow capacity at CRP. The small gain in net benefit beyond the breakpoint in the net benefit curves suggests that a balanced engineering design should not aim at a CRP/IRP ratio higher than 1.30.

The engine design penalties due to SFC, weight, cost and maintenance erode the idealized benefits attainable by increasing contingency power ratios as shown on Fig. 9 for the throttle push engines.

Results for the civil rotorcraft are shown on Fig. 10 with similar trends. Throttle push and cooling flow modulation were competitive systems with water into the turbine cooling system showing up somewhat poorer, especially at lower CRP/TOP ratios.

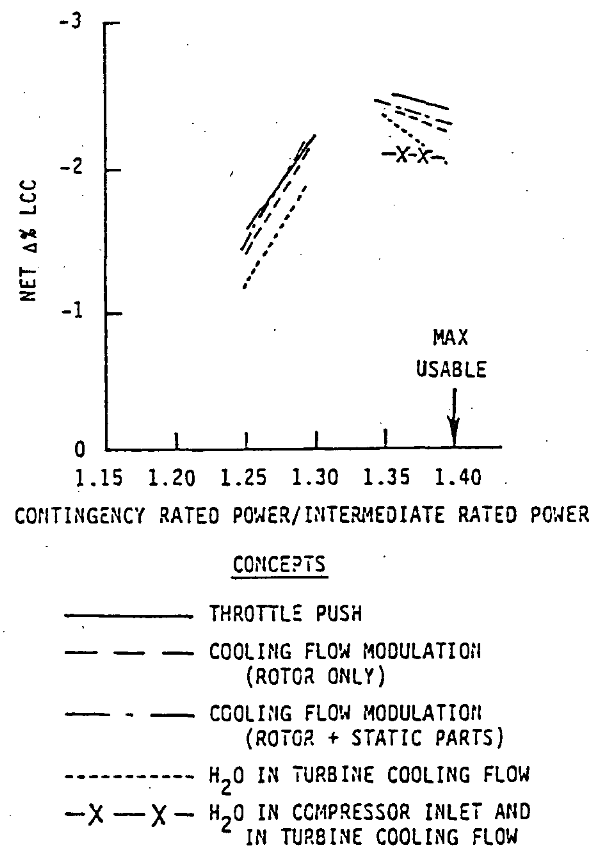


Fig. 8. Net LCC benefits/military rotorcraft

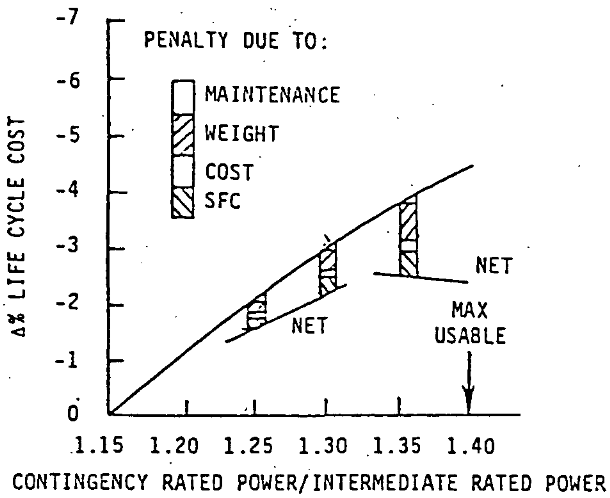


Fig. 9. LCC penalties - throttle push engine - military rotorcraft

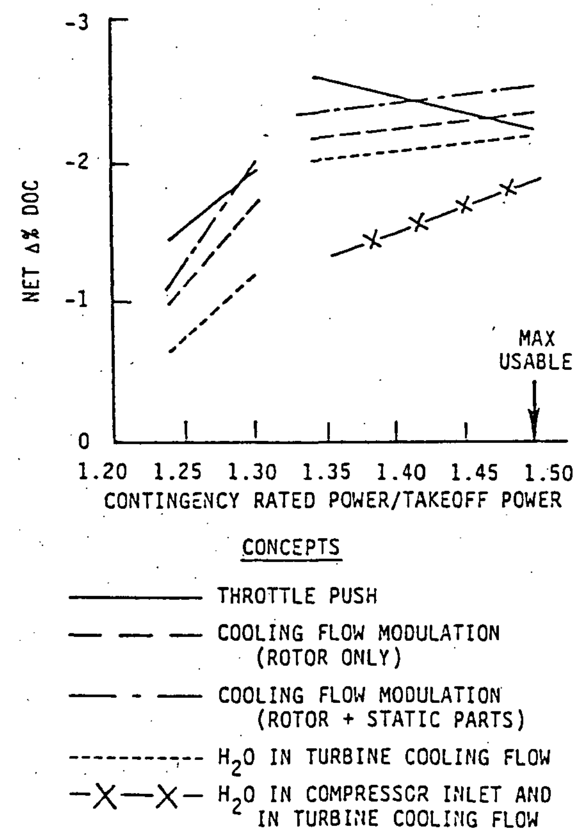


Fig. 10. Net DOC benefits/civil rotorcraft

The reversed slope of the throttle push concept is due to the more rapidly increasing fixed cooling flow penalties set at contingency power. The combined compressor inlet and turbine cooling water injection system was again poorer than the others as in the military rotorcraft. The effect of rematching at higher CRP/TOP ratios is also pronounced for the civil helicopter. At the highest CRP/TOP ratio of 1.50, cooling flow modulation of rotor and stator has the largest DOC reduction (2.7%). However, the DOC improvement between CRP/TOP ratio of 1.35 and 1.50 is modest, even for the best system, and may not justify setting the design objective at the highest ratio.

Proposed FAA Certification

There is an AIA proposal being circulated in the Aircraft Industry for a new 30 sec. "hot shot" rating with the objective of securing FAA review and approval. The concept is to set a rating which can be used only once in service, with perhaps some minor damage to engine components, but no signs of imminent failure. The engine would require inspection and repair after each use. The qualification test proposed, shown in Table 8, will set the design life inasmuch, under these circumstances, there would be no use of contingency power during training missions. Design limits were increased to the criteria of some damage allowed for the HPT nozzle bands and shroud bond coating.

This permitted cooling flow reductions and reduced the engine penalties. Fig. 11 compares the engine DOC benefits for engines designed to the proposed certification vs. the current certification system.

The net benefit in DOC increases by up to 0.8% at CRP/TOP = 1.50. These results and conclusions apply directly only to design options available in a new engine yet to be sized. The 30 sec. "hot shot" rating may show still more payoff vs. an engine redesigned to the current ratings if applied to derivatives of current engines.

Research and Technology Directions

A review of the engine design barriers to increased contingency power capability with lower penalties led to a list of Research and Technology needs for NASA planning.

Table 8. Proposed FAA rules

Results of AIA & SAE Committee Studies.	
Proposed Changes in OEI Helicopter Rules in Industry Review.	
Petition to FAA by Fall '83.	
New Rule Published '85-'86	
30 Second OEI	125% of Normal T/O
2 Minute OEI	110% of Normal T/O
Continuous Enroute OEI	100% of Normal T/O
30 Minute Certification Test Covers OEI Ratings.	
"No Sign of Imminent Failure" after Certification Test, but Degree of Allowable Damage not Established.	
Inspection required after in Service use.	
Military Services may Adapt Similar Ratings in Future.	

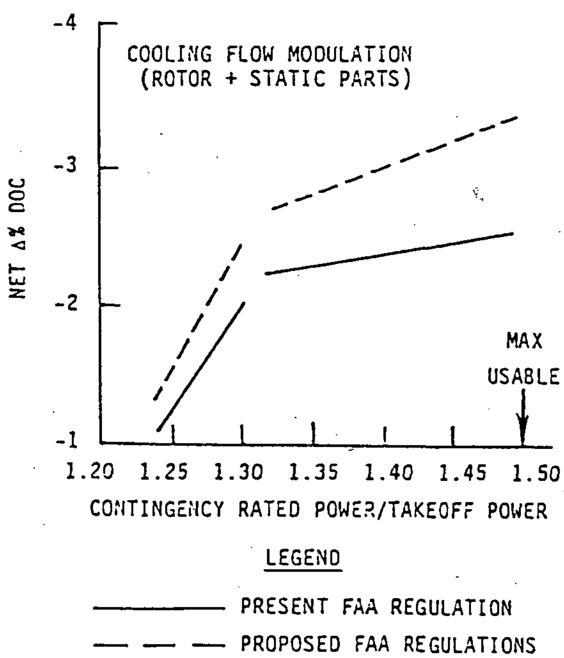


Fig. 11. Effect of proposed FAA regulations on DOC benefits

There are technology needs in materials development of improved turbine blade coatings, shroud bonding and short time creep rupture definition for materials such as Mono N4 used in the hot section parts.

Another need exists for development of a turbine cooling supply system for wider range of efficient cooling flow modulation. Research and Technology in these areas will also find application in other advanced engines.

A study was also recommended to apply these contingency power concepts to derivative engines.

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16. Abstract

Twin helicopter engines are often sized by power requirement of safe mission completion after the failure of one of the two engines. This study was undertaken for NASA Lewis by General Electric Co. to evaluate the merits of special design features to provde a 2-1/2 minute Contingency Power rating, permitting an engine size reduction. The merits of water injection, cooling flow modulation, throttle push and an auxiliary power plant were evaluated using military life cycle cost (LCC) and commercial helicopter direct operating cost (DOC) merit factors in a "rubber engine/rubber aircraft" scenario.

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