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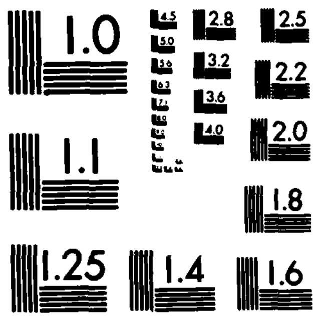

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AGARD LECTURE SERIES No.148

Engine-Airframe Integration for Rotorcraft

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Lecture Series No.148

ENGINE-AIRFRAME INTEGRATION FOR ROTORCRAFT

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INTRODUCTION/OVERVIEW
L.S 148 ENGINE-AIRFRAME INTEGRATION FOR ROTORCRAFT

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The Propulsion and Energetic Panel has emphasized that, in general, AGARD has stressed the technology advancements of high-speed fixed wing aircraft. Some attention has been given to STOL aircraft, but little specialization has been directed toward rotorcraft. It is hoped that this lecture series will be the first step in an AGARD focus of attention on this most fascinating and unique aircraft and, particularly, the integration of its propulsion systems.

The advent of the turbine engine powered rotorcraft in the 1950s, signified a major technology improvement of the then emerging rotorcraft industry. The significant gains in power-to-weight, specific fuel comsumption, and overall aircraft performance paved the way for major improvements in rotorcraft utilization. However, the rotorcraft itself created a unique and somewhat severe environment in which the turbine engine had to operate. Engine installation problems were encountered that before had never been considered: rotorcraft vibrations and sand/dust ingestion were quickly identified as "unique" operational conditions; torsional stability of the "free-power turbine"/engine control system/main-rotor, tail-rotor/transmission system seemed to plague each new generation of rotorcraft; complex dynamic conditions existed that rapidly promoted structural fatigue; exhaust gas reingestion was encountered in some new found flight regimes (backward, sideward flight, etc.). The list of engine-airframe/integration topics could be expanded to a greater number of specific areas than could be covered in this overview or indeed by this Lecture Series.

One of the primary concerns and problems encountered by the early rotorcraft designer was the lack of specific technical rotorcraft design and test literature. Several decades of fixed wing emphasis had developed a considerable amount of detailed technical literature. Initial attempts were to apply this existing fixed-wing data; but, in general, only small selective areas were applicable. To a somewhat lesser degree, that situation remains today. The engineering expertise required to address rotorcraft engine-airframe integration and design and test, encompasses all the major engineering disciplines; aerodynamics, thermodynamics, dynamics, and structures. This Lecture Series will present several papers addressing different areas of engine-airframe integration considerations which have been developed over the last ten years.

In the first lecture, Mr. Frawley will present an approach to the design of propulsion systems which has major impact during rotorcraft preliminary design relative to aerodynamics and engine power available. Areas of specific emphasis include: engine air induction systems, engine exhaust and cooling systems, and engine bleed air extraction systems.

Professor Dini, in the second lecture, will cover a number of compatibility problems encountered by turbine-powered rotorcraft. Avoidance of fatigue cracking in airframe structure will be discussed. The usage of computer finite-element analytical programs during the preliminary design stage will be presented. In addition, the usage of composite material in selected areas of rotorcraft drive systems will be considered from an overall weight, cost savings, reliability, and safety standpoint.

In the third lecture, Mr. Ballard will address current state-of-the-art concepts for engine inlet protection systems and infrared protection. His lecture will review several options and methods for selecting the best overall helicopter integrated designs.

The fourth lecturer, Mr. Ray, will present both past history and current principles associated with propulsion system component design and qualification specifications. Of particular interests are the current trend of "tailoring" requirements for specific application and lessons learned in recent U.S. military rotorcraft programs.

The fifth lecture by Mr. Early is basically a step forward in that it is from an engine manufacturer's viewpoint to impact the basic engine design based on advanced rotorcraft designs, mission requirements, fuel prices, and advancing engine technology. The lecture investigates changes in engine cycle on engine performance, size, weight, engine cost, and rotorcraft operating cost. Advantages and disadvantages of different engine cycles will be presented.

The sixth lecture by Mr. de la Servette concentrates on the engine air intake, addressing rotorcraft requirements such as pressure drop, distortion, and, sand/dust and foreign object protection. Specific areas such as snow protection and the special constraints imposed on this intake design will be discussed.

The seventh lecture by Dr. Brandon/Mr. Enders will address one of the most significant engine-airframe compatibility considerations--adequate torsional stability margin. Discussion of the torsional stability interface focuses on requirements both present and those currently in development. The subject of requirements is dealt with in a framework of analysis and testing. Current practices were surveyed and new trends in analytical modeling and testing will be presented.

The eighth and final lecture by Mr. Ray expands from his first lecture on propulsion components to the rotorcraft system specification and propulsion systems qualification requirements. This lecture generally presents the "requirer's" view of propulsion system requirements as seen today by the U.S. military. Emphasis on state-of-the-art design philosophy relative to "fully integrated" design concepts will be highlighted.

ENGINE-AIRFRAME INTEGRATION CONSIDERATIONS FOR PRELIMINARY AIR VEHICLE PERFORMANCE ANALYSIS

by

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SUMMARY

This paper deals with helicopter engine/airframe integration issues from a preliminary design viewpoint with emphasis on those areas where an impact on aircraft power available and/or aerodynamics is involved. The areas of the helicopter propulsion system specifically addressed include: The engine air induction system considering aerodynamic performance, anti-icing, exhaust gas re-ingestion and inlet particle separation; the engine exhaust system considering aerodynamic performance, engine compartment cooling and exhaust ejector systems, and lastly, engine bleed air extraction as it relates to environmental control systems and the trade studies associated with same. These subsystems are discussed in terms of basic design guidelines, interaction with other helicopter subsystems, system trade studies, and first order analytical design approaches adequate for preliminary design purposes. The paper stops short of a discussion of the detail design associated with each subsystem.

INTRODUCTION

A considerable amount of literature has been devoted to the techniques to be utilized in the design of fixed wing aircraft propulsion systems. The rotary wing aircraft propulsion designer has, by contrast, had to rely on utilization of the applicable portions of fixed wing literature supplemented by the experience he or his co-workers have obtained in actual helicopter design.

Many questions arise and must be answered in planning and executing the preliminary design of a helicopter powerplant system. It is, of course, a major system of the overall aircraft, and hence must be consistent with and supportive of the aircraft requirements. It is clear that such basic elements as who the customer is, what his mission requirements are, and what follow-on market opportunities may exist will strongly influence the design approach. These will determine aircraft preliminary design basics such as weight, size, speed, power required, range, and endurance of a new helicopter and, of course, its powerplant.

These questions include:

- Who is the intended primary customer or market, military or civil?

Each U.S. military service works to its own set of regulations and specifications which sometimes parallel each other, but often don't. Military customers usually specify, in great detail, the required aircraft and powerplant characteristics, often including engines. Civil aircraft must comply with yet another set of rules which, in the United States, are the Federal Aviation Regulations. The aircraft manufacturer must at his own risk, define similar characteristics that he perceives will produce a marketable design and simultaneously comply with the regulations. Very often, an initial military market for a given model will, desirably, be followed by a secondary civil market or an additional military or foreign market. Design conflicts due to differences in rules, regulations and missions must be minimized.

- What are its missions and their requirements?

If military, is it intended for a combat or non-combatant role? This will strongly influence engine and component number and location, redundancy, fire and explosion suppression, ballistic resistance of powerplant and fuel system, signature suppression, life support and other characteristics. Gross weight, range, payload, endurance and speed of the aircraft will affect engine size and number, specific fuel consumption requirements and fuel capacity as well as designs of subsystems such as engine air induction, cooling and exhaust.

Environmental conditions of operating altitude, climate and such local conditions as marine or desert environment will strongly influence engine sizing, ice and sand protection, corrosion protection, heating, ventilation and air conditioning and secondary power system. One engine inoperative (OEI) requirements, particularly if substantial operation at very low altitudes is in the mission, will also strongly affect engine size, number and ratings. Crashworthiness of fuel systems, mounts, controls and accessories has become a major consideration in the design of powerplant systems, both civil and military.

But these broad attributes apply across the board to any aircraft including fixed wing - what makes the helicopter so special? The answer lies, of course, in its unique ability to take off and land in a space no bigger than itself, climb and descend vertically, fly horizontally in any direction, hover stationary over a point for durations limited only by fuel or display great agility when required. These unique capabilities have always imposed special requirements on the helicopter powerplant designer, even in the old piston engine days. The success of the overall helicopter system design rests very substantially on how well the designer reconciles the sometimes conflicting demands of his various subsystems with each other and integrates the whole powerplant system into the aircraft.

This paper attempts to pull together the engine installation/interface experience of one major rotary wing manufacturer with emphasis on those areas of the propulsion system which have requirements peculiar to helicopter operation. Due to the large number of propulsion related systems, the present endeavor has been limited to the preliminary design aspects of rotorcraft design primarily focusing on those areas where an impact on aircraft power available and/or aerodynamics is involved. This paper thus covers the following areas of rotorcraft propulsion system design and stops short of a discussion of the detail design associated with each:

- Engine Air Induction Systems
    - Performance
    - Anti-Icing
    - Re-Ingestion
    - Inlet Particle Separation
- Engine Exhaust and Cooling Systems
    - Performance
    - Ejectors
- Engine Bleed Air Extraction
    - Environmental Control

INLET DESIGN

The requirements of high ram pressure recovery, low engine compressor distortion, and low external drag for efficient cruise flight predominate in the early aerodynamic design of the engine air induction system. Ordinarily, relatively simple one-dimensional analyses are adequate for preliminary designs.

Kuchemann and Weber (Reference 1) define the inlet capture area for maximum efficiency in terms of the range of inlet to free stream velocity ratio (Vi/Vo) as $0.40 \le Vi/Vo \le 0.65$, within which local velocity gradients and flow separation are minimized. The specific value selected by the designer within this range is influenced strongly by mission considerations and when combined with engine power, airflow, and cruise speed, defines the inlet area.

Kuchemann and Weber classifies lip geometry into classes A, B and C for purposes of design. Class A exhibits a large nose radius, relative insensitivity to variations in inlet velocity ratio and airflow direction and relatively high external drag. Class C possesses a very small nose radius and minimum external drag but also much greater sensitivity to velocity ratio and directional variations. With the great speed, attitude, and directional change capabilities of the helicopter this sensitivity of Class C inlet lips would result in excessive pressure loss and distortion at off-design conditions and is not used. As is often the case throughout the helicopter design, inlet lip design is a compromise and usually settles on the intermediate Class B.

Inlet lip total pressure loss is conservatively estimated, using Figure 1 (Reference 2), by the relationship

$$\Delta P_{tl} = K_t \cdot q_{th} \tag{1}$$

where ΔP_{tl} = total pressure drop of lip

q_{th} = $\frac{1}{2}\rho V_{th}^2$ = dynamic pressure at the inlet throat

K_t = lip loss coefficient from Figure 1

Figure 1 is entered assuming $A_2/A_1 = 0$ ($A_1 \sim \infty$), r is radius of inlet lip and D_2 is hydraulic diameter of the inlet throat.

Similarly, pressure losses associated with ducting from the entry lip region to the engine compressor face are estimated using the generalized equation

$$\Delta P = K_t \cdot q \tag{2}$$

and are related to duct length, bends, roughness, area changes, flow impediments, etc. Many references present tabulated loss coefficients for a great variety of geometries; two excellent ones are the SAE Manual and GE Data books (References 2 and 3). The loss coefficients, K_t, are percentages of available dynamic pressure, $q = 1/2\ \rho v^2$. Frequently, the designer will take advantage of the favorable pressure gradients and resulting decreased wall flow separation tendencies of a mildly accelerating duct, other conditions permitting. In this case, acceptable turning losses in a bend can be obtained using average duct velocity, $V = \bar{V}$, in Equation (2). Contrariwise, in determining pressure loss due to accelerating the flow, the maximum velocity, $V = V_{max}$, is used in Equation (2) and the loss is, thereby, a percentage of maximum dynamic pressure. Calculating pressure drop due to friction is somewhat more complicated because friction factor data given in the literature applies to fully developed duct flow which rarely occurs in relatively short helicopter inlet ducts. These factors must then be modified for non-fully developed flow and the Reynolds Analogy is used to accomplish this. This analogy relates the convective heat transfer coefficient, h, and the skin friction factor, f, in the equation

$$\frac{f}{2} = \frac{h}{\rho C_p V} \tag{3}$$

where ρ = density

C_p = specific heat at constant pressure

V = average velocity at cross-section

$f \equiv (\Delta P)/4(L/D)(\rho V^2/2)$

For assumed constant ρ, C_p and V, the friction factor, f, is evidently proportional to the convective heat transfer coefficient, h. The relationship between undeveloped and fully developed convection coefficients is given in Reference 4 as:

$$\frac{h_m}{h_\infty} = 1 + \frac{C}{(L/D)^n} \tag{4}$$

where h_m = coefficient for undeveloped flow at entrance

h_∞ = coefficient for fully developed flow

L = duct length

D = duct diameter

C, n = constants based on entrance condition

Reference 5 indicates that in the range of Reynolds Numbers between 26,000 and 56,000, for values of C = 1.4 and n = 1.0, and using the proportionality between f and h, when equation (4) is modified as:

$$\frac{f_m}{f_\infty} = 1 + \frac{1.4}{(L/D)^{1.0}} \tag{5}$$

the analytical results correlate well with experimental data.

In addition to pressure drop in the induction system, another source of engine performance degradation is heat transfer into the inlet system resulting in air temperature rise and its concomitant reduction in density and mass flow. A principal potential source of heat is engine exhaust gas reingestion (EGR), which may occur during hover in ground effect (HIGE). A secondary source of undesired heat may be an internal hot body, such as a gearbox. Detailed analysis is beyond the scope of preliminary design but some guidelines, based on experience, do exist. When in HIGE mode, engine exhaust gases tend to be entrained in the rotor downwash inner wake and swept up into the engine inlets with the upward flow. Locating the engine exhaust plane outside the boundary of the inner wake results in the exhaust gases being carried away with the outward flow of the outer wake, preventing any significant EGR. Studies (Reference 6), indicate that the rotor inner wake boundary location is a function of wheel height, Figure 2, which can be non-dimensionalized for preliminary design purposes. Figure 3 presents the location of the inner wake boundary as a percent of rotor radius plotted against the non-dimensional quantity Z/R where Z is rotor height above ground (wheel height plus wheel-to-rotor distance) and R is rotor radius. It is noted that this procedure gives preliminary results only which must be validated by wind tunnel and development testing of a particular design.

Acceptable limits of inlet pressure distortion are specified by the engine manufacturer to avoid performance loss, compressor surge or stall and structural failure. While no specific guidelines exist for quantifying how to achieve a duct design that will not exceed these limits, it is clear that simple, straight ducts without impediments or flow separation are highly unlikely to produce distortion. It is equally clear that increasingly complex systems in terms of sharp bends, roughnesses, adverse pressure gradients, etc. are likely to be increasingly prone to distortion and should be avoided. Success or failure of a design can only be determined by airflow testing of the system well after preliminary design is completed.

INLET ANTI-ICE

A helicopter may or may not require ice protection of its rotors, windshields and other susceptible components depending on the dictates of its mission. However, its engine air induction system has always been protected against icing based on the presumption of inadvertent entry into icing conditions. The detailed design of the ice protection system is beyond the scope of preliminary design but the determination of the method of ice protection and estimation of its energy requirements, if any, must be part of the preliminary design of the aircraft.

Meteorological design requirements of ambient temperature, liquid water content and droplet size are specified for a number of flight conditions in both military specifications and civil regulations. These requirements must be met with or without thermal energy. A number of applications have shown compliance without heat but most have required use of either a hot air source, such as compressor bleed air, or electrically energized heating mats on the inlet breeze surface. For those requiring heat, the duct surface temperature is customarily controlled to 40°F and, depending on duct geometry, flight speed and meteorological design conditions, usually requires significant heating capacity. If hot air is used as the heat source it may be necessary to supplement or replace engine compressor bleed air with a shaft driven load compressor. If electric power energizes the duct heater mats, the load is frequently sufficient to size the aircraft electric system generation capacity. It is evident that a reasonably accurate assessment of the type of inlet duct anti-icing, its heat load and energy source, must be made for a successful aircraft preliminary design.

The choice of electrical or bleed air anti-icing for the engine inlet potentially impacts several other aircraft systems and should generally be the subject of a trade-off in the preliminary design phase where sufficient flexibility exists to select the optimum approach.

The trade-off must be based on the particular aircraft operational requirements but a number of general interfaces/impacts must be addressed.

- Electrical systems impact aircraft generator sizing and consequently the aircraft gearbox design.

- Bleed air system flow requirements must be within engine bleed air capabilities or an alternative source of air provided. Anti-ice requirements at low powers, autorotation or ground idle, may exceed the engine's bleed limits or bleed energy availability despite more than adequate flow under other operating conditions. This is especially true of engines which incorporate integral particle separators which themselves may require bleed air for anti-icing.

- Aircraft power penalties for bleed air systems are generally more severe than those associated with electrical systems in that the energy conversion efficiency of a generator driven by the aircraft gearbox is much greater than that of engine compressor bleed.

- Electrical systems tend to be heavier than bleed air systems due to the increased generator weight and generator size may become a problem if the aircraft utilizes electro-thermal rotor de-icing which would operate at the same time.

- System reliability generally favors the bleed air system if the main engines can provide the required bleed energy. Additional bleed air sources such as load compressors negate much of this advantage unless they are already installed for other purposes.

Approximating the thermal energy requirements of the anti-ice system typically precedes the trade study which determines the type of system.

To approximate the anti-icing system energy requirements, the following procedure may be used:

The local heat flux at any wall temperature Tw is given by the sum of the convective, sensible and evaporative heat fluxes:

$$(Q/A)_{TOTAL} = (Q/A)_{conv} + (Q/A)_{sens} + (Q/A)_{evap} \tag{6}$$

where $$(Q/A)_{conv} = h_o(Tw - Taw) \tag{7}$$

$$(Q/A)_{sens} = WCp\ (Tw - Taw) \tag{8}$$

$$(Q/A)_{evap} = 2.9\ Lvh_o \left[\frac{P_{sn} - P_{H_2O}}{P_{amb} - P_{sn}}\right] \tag{9}$$

and

Lv = Heat of vaporization, BTU/lbm

h_o = External heat transfer coefficinet, BTU/hr ft² °F

W = Impinged moisture rate, lb/sec

Cp = Specific heat of imposed moisture, BTU/lbm °F

P_{H_2O} = Partial pressure of H_2O in air, PSIA

P_{amb} = Ambient pressure, PSIA

P_{sn} = Partial pressure of H_2O at surface temperature, PSIA

T_{aw} = Adiabatic wall temperature, °F

The adiabatic wall temperature is related to the local total temperature T_t and static temperature T_s by the recovery factor RF.

$$RF = \frac{T_{aw} - T_s}{T_t - T_s} \tag{10}$$

The recovery is a function of the local Prandtl number:

$RF = (Pr)^{1/3}$ for turbulent flow

$RF = (Pr)^{1/2}$ for laminar flow

The above analysis adequately defines a thermal energy requirement consistent with a design condition. For electrically powered systems, the results can be used directly to determine the impact on the aircraft's electrical system, i.e., generator size, weight, etc. A bleed air system, however, requires additional analysis as the energy requirement in terms of BTU/hr must be converted to an equivalent amount of bleed air flow (lbm/min or equivalent), consistent with the available air source temperature. Additionally, it is necessary to establish the adequacy of the design concept by verifying that the assumed amount of bleed flow is sufficient to maintain the inlet surface at 40°F and result in a manufacturable design, i.e., a design in which the heat exchanger gap heights can be controlled during fabrication. Gap height control is required since surface heating is a function of the internal convective coefficient which in turn depends on gap height and mass rate of flow. The anti-ice system heat exchanger analysis is best illustrated by the following example.

Assume that the individual heat loads are given as:

Convective, 30,000 BTU/hr
Sensible, 4,000 BTU/hr
Evaporative, 12,000 BTU/hr

which yields a total load of 46,000 BTU/hr. Further assume that the maximum amount of bleed air available (analogous to the conditions which resulted in the above heat load) as specified by the engine manufacturer is 0.45 lbm/sec at 250°F. The temperature lost by the bleed air heating the inlet surface is given by:

$$\Delta T = \frac{Q}{MCp} \tag{11}$$

where Q = Total heat load, BTU/hr

M = Bleed air flow, lbm/hr

Cp = Specific heat of air, BTU/lbm °F

therefore $$\Delta T = \frac{46,000}{(.45)(3600)(.24)}$$

$$\Delta T = 118°F$$

which results in a bleed outlet temperature of 132°F. The adequacy of the design concept is established by verifying that this outlet bleed temperature is sufficient to maintain the inlet breeze surface at 40 degrees F with a manufacturable and controllable bleed air passage height, typically not less than 0.1 inch.

An estimate of the local heat load at the point where the bleed air exits the heat exchanger is required. This is accomplished using the local air velocity to calculate the heat transfer coefficient (depending on geometry, flat plate theory or the convective coefficient relationships for short ducts as given by References 4 and 7 are adequate). For purpose of continuing with the example, the heat fluxes are assumed to be:

Convective, 800 BTU/ft^2 hr
Evaporative, 500 BTU/ft^2 hr
Sensible, 0

yielding a total flux of 1300 BTU/ft^2 hr. To attain this flux with a bleed air outlet temperature of 132°F and an inlet surface temperature of 40°F, the required internal heat transfer coefficient is given by :

$$h_i = \frac{(Q/A)}{T_{bl} - T_s} \tag{12}$$

where Q/A = local heat flux

T_{bl} = bleed air outlet temperature, °F

T_s = inlet surface temperature, °F

then

$$h_i = \frac{1300}{(132 - 40)}$$

$$h_i = 14.13 \text{ BTU/ft}^2 \text{ hr °F}$$

The turbulent heat transfer coefficient in non-circular channels is calculated form the conventional circular tube correlations by replacing the diameter with the hydraulic diameter and multiplying the result by 0.75 (Reference 4).

$$\frac{h_i D}{k_b} = .023 \left[\frac{MD}{\mu_b A}\right]^{.8} \left[\frac{\mu Cp}{k}\right]_b^{.4} (.75) \tag{13}$$

where $\frac{h_i D}{k}$ = Nusselt Number

$\frac{MD}{\mu_b A}$ = Reynolds Number

$\frac{\mu Cp}{k}$ = Prandtl Number

and the subscript b refers to the bulk or mixed mean fluid temperature, $1/2\ (T_{bl} + T_s)$.

Assuming that the width of the exit gap is 3.0 ft., substituting for hydraulic diameter, flow area, and the following parameters:

k_b = .016 BTU/hr ft °F

Pr_b = .706

μ_b = .046 lbm/hr ft

M = .45 lbm/sec

$D = D_H = \frac{4 \text{ Area}}{\text{Perimeter}} = \frac{2xl}{h + l}$ for rectangular sections

$A = xl$ where x = gap height and l = width

l = 3 ft

Equation (13) reduces to:

$$\frac{46.7576(x)}{3 + x} = \left[\frac{1}{3 + x}\right]^{.8}$$

which is solved by trial and error to yield a required gap height of 0.0267 ft. or 0.32 inches, well above the minimum that can be manufactured.

The procedure can be iterated until the desired combination of bleed flow and heat exchanger geometry is found. This analysis is intended as a basic substantiation of the adequacy of the design approach and to conservatively quantify the bleed air flow requirement. The final design is the subject of a much more detailed design analysis beyond the scope of this paper.

Engine Air Particle Separator (EAPS)

The need for an effective means of protection against the ingestion of foreign objects, particularly sand and dust, has arisen from the unique operational requirements of helicopters. Helicopter operation from unprepared sites has resulted in erosion of engine compressor blades, the effects of which generally result in a loss of power and/or a reduction in compressor surge margin. The rate of component erosion is directly related to the rate of sand and dust ingestion, the type of sand and dust, and the time spent in the environment.

The detail design of an EAPS system is typically not accomplished during preliminary design, however, not unlike the anti-icing system, the inclusion of an EAPS in the baseline aircraft design will impact the areas of installed engine performance, aircraft drag and weight. Engine performance penalties are realized as a result of the filtration mechanism i.e., imparting a force on the entrained particle in a different direction than the air (inertial separation). In practice, this results in an increased momentum loss due to turning, resulting in lower inlet recovery ratios.

Inertial separators are primarily of two types: the multiple tube cyclone type matrix, typified by the CH-53E design (Figure 4) and the large cyclone type used as integral components on some gas turbine engines as the T700-GE-700 engine (Figure 5). The matrix cyclone separators have demonstrated filtration efficiencies in the 93% range on coarse test sand and higher on coarser particles as compared to a level of 85% on the larger cyclone separator. The larger separator, because it is integral with the engine and does not offer a bypass mode, suffers from the disadvantage of retaining more operational penalties outside of the sand and dust operating environment. However, the larger separator has the advantage of being integral with the engine and, therefore, presents a minimum of installation difficulties; also its presence does not represent an installation loss from rated engine performance to the airframe designer.

Barrier type filters have essentially been discarded for aircraft use due to the relatively frequent cleaning requirements and the large surface area necessary to provide acceptable pressure drop.

Prediction of the pressure drop associated with the airframe mounted multi-tube inertia/separator system requires a computer procedure to balance the flow through each tube and to account for the friction and momentum losses associated with the collection duct upstream of the engine inlet.

The following guidelines are presented in closing the discussion on EAPS systems and will aid the designer in establishing EAPS requirements and losses during preliminary design. A helicopter air inlet sand and dust protection device should meet following criteria:

- Separate at least 90% by weight of all particles larger than 20 microns.
- Produce a power loss not greater than 2 to 3%. This correlates to a pressure drop of approximately 4 to 6 inches of water.
- Be compact, lightweight and compatible with the basic aircraft.
- Have by-pass provisions for ram recovery at forward flight speeds.

Compartment Cooling and Exhaust System

A forced air supply is usually required to absorb engine generated heat and maintain engine compartment air temperatures below specified values for reliable operation of engine accessories. The helicopter, which is not always the benefactor of ram air, must include in its basic design a means of inducing the compartment cooling flow in hover and rearward flight. An engine exhaust ejector is a convenient means for forcing a flow of compartment cooling air during engine operation. After shutdown residual heat is dissipated by free convection and usually removed through vents in the upper surface of the engine cowling. Air inlets are located in areas such that the cooling air is directed over the most critical components.

The design of the ejector cooling system must be coordinated with the design of the engine exhaust system, which, in turn, needs to meet the following objectives:

- Minimize pressure loss in order to reduce engine power loss.
- Prevent hot exhaust gas from flowing through the aircraft tail rotor and reducing tail rotor efficiency.
- Prevent reingestion of exhaust gas by the engine and loss of power due to inlet air heating.
- Prevent impingement of the exhaust on aircraft surfaces in order to minimize structural effects.
- Provide maximum possible thrust recovery compatible with the above objectives in order to minimize the momentum drag of the engine installation.
- Minimize engine-mounted exhaust duct length to prevent vibration-induced fatigue failures. Airframe mounted systems are generally recommended if appropriate ejector gap heights can be assured.

The analytical technique employed is based on SAE ARP No. 996, Reference 8. To simplify the analyses, the engine and cowling are assumed to be concentric cylinders (Figure 6).

The steady state energy balance for heat transferred from the engine is given as:

$$Q_e + Q_r = Q_a + Q_o \tag{14}$$

where:

Q_e = heat tranferred from the engine by convection

Q_r = Heat transferred from the engine by radiation

Q_a = Heat convected away by cooling airflow

Q_o = Heat transferred to outside air: equals the heat transferred to the cowl by convection and radiation, equals the heat transferred through the cowl by conduction, equals the heat convected from the cowl to outside air plus the radiant transfer from the cowl outer skin to the surroundings.

The expression for each of the above quantities is given as follows:

Convection from the Engine

$$Q_e = h_e A_e (T_e - T_a) \tag{15}$$

$$\text{and} \quad T_a = (T_{a_1} + T_{a_2})/2 \tag{16}$$

Radiation from engine to cowl

$$Q_r = \frac{\sigma A_e (T_e{}^4 - T_{ci}{}^4)}{\frac{1}{\varepsilon_e} + \frac{A_e}{A_{ci}}\left[\frac{1}{\varepsilon_c} - 1\right]} \tag{17}$$

Heat convected away by cooling airflow

$$Q_a = M_a\ C_p\ (T_{a_2} - T_{a_1}) = 2\ M_a\ C_p\ (T_a - T_{a_1}) \tag{18}$$

Heat transferred to outside air

$$Q_o = \frac{U_c\ (T_{ci} - T_{co})\ 2\pi L}{\ln\ (r_{co}/r_{ci})} \tag{19}$$

$$= h_{co}\ A_{co}\ (T_{co} - T_\infty\ ) + \sigma\varepsilon_c\ A_{co}\ (T_{co}{}^4 - T_\infty{}^4) \tag{20}$$

$$= h_{ci}\ A_{ci}\ (T_{ci} - T_a) + \frac{\sigma A_e (T_e{}^4 - T_{ci}{}^4)}{\frac{1}{\varepsilon_e} + \frac{A_e}{A_{ci}}\left[\frac{1}{\varepsilon_c} - 1\right]} \tag{21}$$

Where:

h = Film coefficient for forced convection, BTU/hr ft^2 °F

A = Surface Area, ft^2

T = Absolute temperature, °R

σ = Steffan-Boltzman Constant, BTU/hr ft$^2 \cdot$ °R^4

ε = Total emissivity

M = Weight flow, lbm/hr

Cp = Specific heat at constant pressure, BTU/lbm °F

U = Overall conductive heat transfer coefficient, BTU/hr ft^2 °F

L = Length, ft

r = radius, ft

Subscripts

a = air

i = inner surface

o = outer surface

c = cowl

e = engine

∞ = outside or free stream air

1 = inlet stations

2 = exit station

Inspection of the above equations reveals five unknown quantities: T_a, T_{ci}, T_{co}, Q_o and M_a. The calculation procedure involves making an initial assumption relative to M_a and utilization of engine data to calculate the remaining unknowns. Engine data is typically represented by curves of engine surface temperature vs. length as a function of heat rejection per unit length (Figure 7). This information is found in engine manufacturer model specifications for different operating environments and engine power settings. The condition corresponding to the maximum aircraft operating temperature and the 30 minute power rating, or equivalent, should be used in determining the cooling airflow required.

The solution is calculated between engine stations which represent sections of approximately constant geometry and engine surface temperature. The outlet conditions at section one become the inlet conditions at section two and so forth, until the entire length of the nacelle has been analyzed. The solution is repeated with variable M_a until the required values of cooling air and nacelle temperature are achieved.

Exhaust Ejector

The ejector system (Figure 8) is sized by use of the energy, continuity and momentum equations. The steady flow momentum equation also incorporates experimentally determined mixing constants. Several references (9 and 10) are available which treat the development of the ejector equation and present the correction factors needed to size the system.

It can be shown that the solution to the energy, continuity and momentum equations applied to the ejector problem is given by:

$$\frac{\Delta P}{q_p} = \frac{2}{(1 + A^*)}\left[1 + \frac{M^{*2}\, T^*\, (A^* - 1)}{2A^{*2}} - \frac{(1 + M^*)(1 + M^*T^*)C}{(1 + A^*)}\right] \qquad (22)$$

where M^* = M_s/M_p

M_s = secondary flow rate or required cooling flow rate

M_p = primary flow rate or engine exhaust gas flow rate

T^* = T_s/T_p

T_s = secondary or cooling flow rate temperature at the pumping plane

T_p = primary or exhaust gas temperature

A^* = A_s/A_p

A_s = secondary flow area (pumping plane, station 2, Figure 8)

A_p = primary flow area (pumping plane, station 2, Figure 8)

ΔP = ejector pressure rise or secondary flow path pressure loss $(P_a - P_{t2})$

P_a = ambient pressure

P_{t2} = stagnation pressure at secondary inlet (station 2, Figure 8)

q_p = primary dynamic pressure ($\frac{1}{2}\rho_p V_p{}^2$)

$$C = K_m + 2\frac{fL}{D_m} - \frac{N_D}{2}\left[1 - (A_m/A_e)^2\right] \qquad (23)$$

K_m = momentum correction factor, $f(A_m, A_e, W^*, T^*, L/D_2)$ See Figures 9 through 11, reproduced from Ref. 10

L = mixing duct length

D_m = mixing duct diamenter

A_m = mixing duct area

A_e = diffuser duct exit area

N_D = diffuser efficiency = $\frac{C_p}{Cp_i}$

C_p = pressure recovery

Cp_i = ideal pressure recovery = $1 - \frac{1}{(A_e/A_m)^2}$

Diffuser efficiency is typically determined via experiment for the various classes of diffusions, i.e., two dimensional, annular axisymmetric or conial. Reference 11 is one source of diffuser efficiency data presenting information on diffusers with rectangular, conical and annular cross sections.

The procedure by which the exhaust ejector is sized is iterative and involves a matching process whereby an ejector geometric solution is defined when the ejector operating line intersects the secondary flow path resistance curve at the desired value of M_s (Figure 12).

The ejector operating line is determined from the ejector equation for an assumed geometry. Geometric constraints are defined based on the aircraft layout and considerations mentioned earlier. A series of operating lines (different ejector geometries) are usually required to converge on a solution and there will most likely be several geometries that satisfy the cooling flow requirements. The geometry which produces the smallest and lightest package along with a minimum performance penalty is obviously the solution sought. Available space and weight constraints typically limit the length of the exhaust/ejector system. For the class of ejector used in engine compartment cooling applications (annular, axisymmetric or conical), the exhaust system designer is usually restricted to values of L/D_m between 1.0 and 2.0. If adequate cooling flow can not be achieved within these limits of L/D_m, the designer has the option of nozzling the primary exit area to induce a higher primary stream momentum, or adding a diffuser downstream of the mixing duct, which provides for increased pumping by reducing the static pressure at the pumping plane. Both solutions will penalize aircraft performance. Reducing the primary nozzle exit area reduces power available via increased backpressure on the engine power turbine. The addition of a diffusing duct downstream of the mixer will add weight and may also provide for a lower momentum stream exiting the aircraft. This lower velocity exit flow will induce a momentum drag penalty which reduces aircraft performance. Conversely, if the aircraft exit nozzle is reduced in cross section (to reduce the momentum losses) ejector performance (pumping) will suffer.

To achieve the optimum exhaust ejector design, the designer must be aware of all the potential interactions between the subsystem being considered and other design and operational aspects of the aircraft and conduct the appropriate trade-offs.

Upon completion of ejector system trade studies, the installation loss attributable to the design is a fallout of the ejector sizing calculation. The static pressure at the primary nozzle exit or the pumping plane is known. Assume that the static pressure is constant across station 2 (Figure 8), such that the total pressure in the primary nozzle is calculated as follows:

$$P_{tp} = P_{s2} + \frac{1}{2} \rho_p V_p^2 \tag{24}$$

where: P_{tp} = primary nozzle total pressure

P_{s2} = static pressure at Station 2

ρ_p = primary nozzle density

V_p = primary nozzle velocity

In order to quantify the pressure downstream of the engine power turbine, the loss from this point to the primary nozzle exit is needed such that:

$$P_{t_{pte}} = P_{tp} + \Delta P_t \tag{25}$$

where: $P_{t_{pte}}$ = total pressure at power turbine exit

P_{tp} = primary nozzle exit total pressure

ΔP_t = tailpipe pressure loss

The tail pipe pressure loss can be estimated by assuming frictional losses as follows:

$$\Delta P_t = \frac{4fL}{D_H}\bar{q} \qquad (26)$$

where: $4f$ = friction factor (Fanning)

L = tailpipe length

D_H = average hydraulic diameter of tailpipe

$\bar{q}$ = average dynamic pressure in tailpipe $(q_1 + q_2)/2$

If the tailpipe is classified as a diffuser, Reference (11) should be consulted to determine the recovery of the device. The diffuser recovery is stated as follows:

$$C_p = \frac{P_{s_2} - P_{s_1}}{q_1} \qquad (27)$$

where: C_p = recovery coefficient

P_{s_2} = static pressure at Station 2

P_{s_1} = static pressure at power turbine exit or tailpipe inlet

q_1 = dynamic pressure at power turbine exit

Reference (11) presents data on recovery coefficient, C_p, for a variety of diffuser types and geometries. Assuming that C_p is found from available data, P_{s2} and q_1 are both known quantities, then P_{s1} is easily calculated. With knowledge of P_{s_1} and q_1, the total pressure at the power turbine exit or tailpipe inlet is simply:

$$P_{t1} = P_{s1} + q_1$$

The first method of estimating P_{t1} is acceptable in the absence of known tailpipe recovery performance. The second method is recommended and represents a more accurate approach, however, it presupposes knowledge of tailpipe recovery.

Regardless of the method used, the installed performance penalty, in terms of engine power loss can be estimated by assuming that pressure loss is roughly proportional to power loss on a one to one basis i.e., 1.5% in pressure is equivalent to 1.5% in power. The actual relationship varies from engine to engine. If an engine computer deck is available to the designer, the exhaust backpressure or tailpipe characteristic is often an input quantity. The latter is clearly a more accurate approach and is strongly recommended.

An additional factor in the successful operation of a low loss engine exhaust ejector is the exhaust flow swirl angle variation as a function of engine power. This swirl can cause ejector backflow with exhaust gas entering the engine compartment under high swirl conditions when the gas flow is low and the swirl angle high such as typically occurs at flight or ground idle conditions. The magnitude of this problem is a function of the engine swirl angle range, the engine exhaust frame design, and the aircraft exhaust system configuration. The exhaust frame on some engines incorporates struts which act as flow straighteners to reduce the exhaust swirl entering the aircraft exhaust system. The aircraft exhaust system back pressure, which is a function of the diffusion rate and the degree of turning also impacts the propensity for backflow. A secondary ejector duct which has high diffusion to reduce losses and maximize pumping and which is subject to back pressure in a tail wind or in rearward flight will also increase the likelihood of backflow.

The preliminary designer must assess the potential for ejector backflow and, if necessary, include a deswirl primary exhaust duct which contains sufficient turning vanes to straighten the exhaust flow prior to its entering the ejector mixing plane. Such a deswirl duct will impact engine exhaust losses by increasing the back pressure on the engine turbine but it may be a more palatable solution to a backflow problem than increasing the velocity at the ejector primary by nozzling the exhaust.

Engine Bleed Extraction

Engine bleed extraction is discussed in terms of those air vehicle subsystems potentially requiring engine bleed air for operation. As was shown earlier, the engine air induction anti-ice system is one system which can utilize bleed air or electrical power in providing the required inlet surface temperature. The preliminary design engineer is concerned with this trade-off as one solution reduces engine cycle efficiency and impacts fuel flow, hence, range while the other solution adds weight to the air vehicle in the form of larger electric generators. The later constitutes a penalty, in the form of added weight, which is carried by the air vehicle at all times while the former is clearly an interim penalty occurring only when the anti-ice system is operating. As discussed earlier, the choice of anti-ice system is related to air vehicle design requirements such as gross weight, range, speed, etc.

The anti-ice system, however, represents but one system that can potentially effect engine bleed air extraction and hence power available, gross weight and aircraft performance. An additional subsystem which should be the subject of a trade study during preliminary design is the air vehicle environmental control system.

The increasing sophistication of helicopter avionics and electrical systems results in increased heat generation in avionic equipment bays. For reliable operation of avionic equipment, electric fans have been added to cool the equipment convectively. In some installations, the SH-60B for example, the heat rejection and temperature requirements are such that ambient air cooling is inadequate, within reasonable fan size and air flow levels. In other cases, an additional requirement for dry air, i.e., air exhibiting a relative humidity of 50% or less, precludes the use of ambient air cooling.

Cooling for avionic and electrical equipment often represents only a portion of the total air vehicle cooling requirement. Executive and VIP helicopters typically require conditioned air delivery to the cabin and cockpit in addition to avionic and electrical equipment cooling. The importance of including the ECS in the air vehicle preliminary design should not be underemphasized.

As mentioned earlier, the trade study centers around the use of bleed air energy from the engine or electrical energy from the aircraft generators. The system requiring engine bleed air is appropriately called an air cycle system, similar to those used on commercial airliners. The air cycle system has the advantage of being a complete environmental control system, i.e., it incorporates a heating mode as well as an air conditioning or cooling mode. Air cycle systems generally are lighter in weight than the alternative vapor cycle system and also offer fewer logistic concerns as freon and charging carts are not needed. Lastly, leakage within the air cycle system will allow system operation at a reduced capacity indefinitely, where as a freon leak occurring in a vapor cycle system will eventually cause system shut down as the closed loop freon systems require a minimum charge of working fluid to operate properly.

At first glance, it would appear that air cycle systems are the obvious choice, fewer logistic concerns, higher reliability, lighter weight and integral heating capability. The determining factor, however, is the power utilization. During hot weather operation, the effects of bleeding the engine can be much more costly than during cold day operation since turbine engine output power is significantly reduced at higher ambient temperatures. Trade studies conducted by Sikorsky have shown that bleeding compressor air for air cycle system operation can reduce engine power available substantially. For example, an air cycle ECS utilizing 40 pounds/minute of engine bleed air results in a reduction of power available of approximately 160 SHP on a twin engine aircraft with two nominally 1500 shaft horsepower engines. This translates to a reduction in lift capability of approximately 800 lbs (assuming a power loading of 5 lbs/SHP). The weight of the air cycle system which meets the cooling requirement is a nominal 90 lbs, while the weight of the vapor cycle system plus an aircraft heating unit is a nominal 230 lbs. The generator energy needed to drive the vapor cycle system compressor is in 20 to 25 SHP range. The following table summarizes the performance effects of the two candidate systems:

TABLE 1

System	Weight	Power Utilization	Max Delta Lift*
Air Cycle	90 lbs	160 SHP	890 lbs
Vapor Cycle	230 lbs	25 SHP	355 lbs

*assumes 5 lbs/SHP

Table 1 is therefore interpreted as showing a 535 lb lift advantage for the vapor cycle system.

The trade study serves to quantify the costs associated with engine bleed air extraction on a hot day. It must be emphasized that the deltas established above are limits which occur, in practice, only when the engine has reached its thermal limit. At power levels where the increased power is available within the thermal limit, penalties take the form of increased specific fuel consumption and reduced range. Realistically, one way around the take-off gross weight limitation is simply to shut off the ECS during take-off. Current aircraft, for example, utilize an air cycle ECS which incorporates a temperature override feature that closes the engine bleed air actuator automatically when the pilot demands intermediate rated power.

The trade study between air cycle and vapor cycle cooling systems is often guided by the quantity of cooling needed. Should the cooling air flow requirement fall below that allowed by the engine manufacturer, then clearly bleed air is a viable source. The aircraft preliminary designer has one additional option, that being to design a shaft driven load compressor into the aircraft dynamic system. In so doing, ground cooling can be accomplished by engine bleed followed by load compressor supplied air thereafter. The combination of an air cycle environmental control system supplied by air from a transmission driven compressor provides a light weight, reliable system at reasonable power, fuel and gross weight penalties. Table 1 is expanded below to include the air cycle ECS supplied by air from a transmission driven compressor. The compressor and gearbox components are assumed, conservatively, to weigh 50 lbs and require a nominal 50 SHP.

TABLE 2

System	Weight	Power Utilization	Max Delta Lift
Air Cycle (Engine Bleed)	90 lbs	160 SHP	890 lbs
Air Cycle (Compressor)	140 lbs	50 SHP	390 lbs
Vapor Cycle	230 lbs	25 SHP	355 lbs

Table 2 shows that an air cycle system with a load compressor provides a viable alternative to the vapor cycle if a load compressor installation is acceptable within the total aircraft design approach.

In summarizing the discussion on the ECS trade study, the choice of the type of system is primarily driven by the quantity of cooling air flow required. In this example the case where the cooling system has been defined as a baseline subsystem of the aircraft, the choice of an air cycle ECS supplied by air from a transmission driven compressor should be optimal for moderate cooling loads (20 to 40 lb/min systems). If minimal cooling loads exist (less than 20 lb/min capacity), the weight and complexity of the shaft driven compressor may not be justified. As cooling loads increase (greater than 40 lb/min), vapor cycle systems tend to be the optimal based on their high efficiencies.

Owing to the importance of ECS air flow requirement and its impact on the ECS trade studies, a simple method, adequate for preliminary design, is presented below to quantify the cooling flow requirements for both avionics equipment bays and cabin/ cockpit environment.

Cooling Analysis

The following general approach is applicable to all compartments on the aircraft. Heat is transferred from the environment to the aircraft by conduction, convection, infiltration of ambient air, and solar radiation through windows. Heat from crew members and occupants as well as avionic systems should also be considered. The following analysis illustrates the heat balance procedure:

$$Q_{out} = Q_{in} = Q_c + Q_{inf} + Q_s + Q_{crew} + Q_a \tag{28}$$

where

Q_{out} = Total heat removed from the area in question by the ECS, BTU/hr

Q_{in} = Total heat gained by the area in question, BTU/hr

Q_c = Heat gained due to conduction and convection from the environment, BTU/hr

Q_{crew} = Body heat from occupants, BTU/hr

Q_{inf} = Heat gained due to infiltration of ambient air, BTU/hr

Q_{solar} = Heat gained by solar radiation through the aircraft windows, BTU/hr

$Q_{avionics}$ = Heat generated by avionics, BTU/hr

Determination of Q_c:

$$Q_c = \left(\sum_1^n U_j A_j\right)(T_a - T_c) \tag{29}$$

where

T_c = Desired compartment temperature, °F

T_a = Ambient air temperature, °F

$\sum_1^n U_j A_j$ = Summation of overall thermal conductance multiplied by area for each exterior surface, BTU/hr, °F

The conductance, U_j, for each exterior surface is calculated from:

$$U_j = \frac{1}{\frac{1}{h_i} + \frac{\Delta X}{k} + \frac{1}{h_o}} \tag{30}$$

where h_i, h_o = convective heat transfer coefficient for inside and outside surface of area element A_j, BTU/hr °F ft²

ΔX = Wall thickness of area element A_j, ft

k = Wall thermal conductivity of area element, A_j, BTU/hr ft °F

The calculation of the convective coefficients is configuration specific. The equations of reference 7 for a cylinder in cross flow can be used to model the aircraft nose section or cockpit while the flat plate equation of reference 7 can be used to model the sides of the aircraft.

Determination of Q_{crew}:

The metabolic heat rejection of crew and occupants is primarily a function of activity level. Reference 12 should be consulted for detail, however, the heat rejection for a passenger can be adequately accounted for by assuming Q_{crew} equal to 400 BTU/hr person while a somewhat higher 700 BTU/hr person is more appropriate for the pilots.

Determination of Q_{inf}:

The infiltration heat load is given by

$$Q_{inf} = M_{inf}\, C_p\, (T_a - T_c) \tag{31}$$

where M_{inf} is the infiltration flow rate expressed in lbm/hr. The method described in reference (13) can be utilized to calculate M_{inf}.

Determination of Q_{solar}:

Reference 12 should be utilized in determining the solar load transmitted through the aircraft transparencies.

Determination of $Q_{avionics}$:

The heat load due to avionic equipment is given by:

$$Q_{avionics} = \sum_{1}^{n} (E_{avionics}\, (N_A)(DC)) \tag{32}$$

where $E_{avionics}$ = Avionics input power is BTU/hr

N_A = Component efficiency

DC = Component duty cycle

The total cooling load is then balanced against the ECS output energy as follows:

$$Q_{ECS} = Q_{TOTAL} = Q_{OUT}$$

$$Q_{TOTAL} = M_{ECS}\, Cp\, (T_c - T_{ECS}) \tag{33}$$

The ECS flow required is quantified by substituting the desired environmental temperature T_c, and the delivery temperature T_{ECS}, into equation (33).

Once the ECS cooling flow requirement has been quantified, the equipment trade study can be conducted to determine the optimal system for the aircraft in question and its particular mission requirements.

CONCLUSION

We have attempted in this presentation to review the approach to the preliminary design of those powerplant subsystems which have the greatest impact on aircraft preliminary design in the areas of power available and aerodynamics. Many other subsystems such as fuel, fire protection, engine mounting, cowling, starting, etc., must also be addressed before a complete aircraft preliminary design can be established. Development of the preliminary design approaches outlined herein will, however, provide the preliminary aircraft designer with those power and aerodynamic impacts necessary to advance the helicopter design to the next stage.

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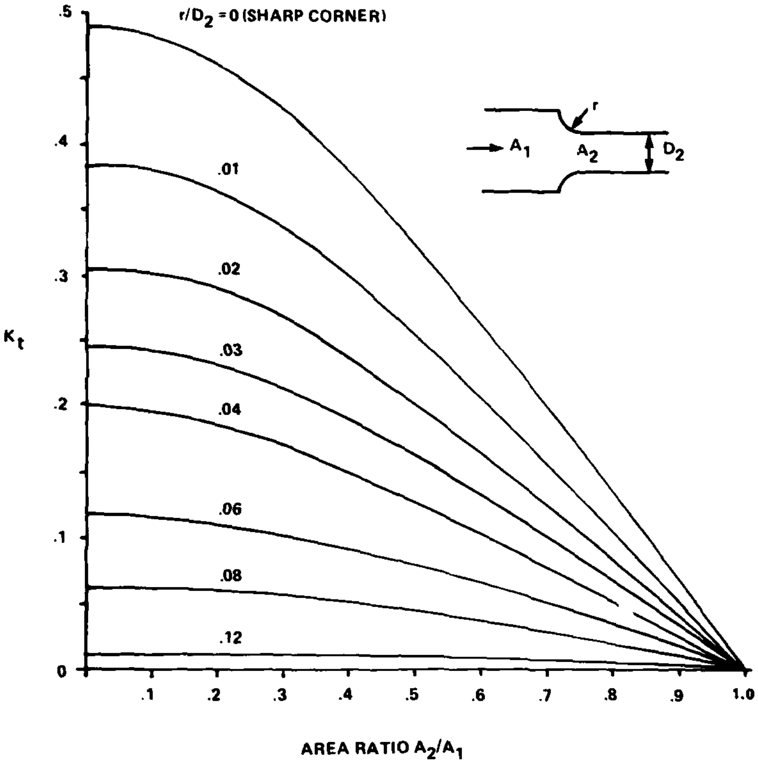

FIGURE 1. PRESSURE LOSS COEFFICIENT VS. AREA RATIO (REF.2)

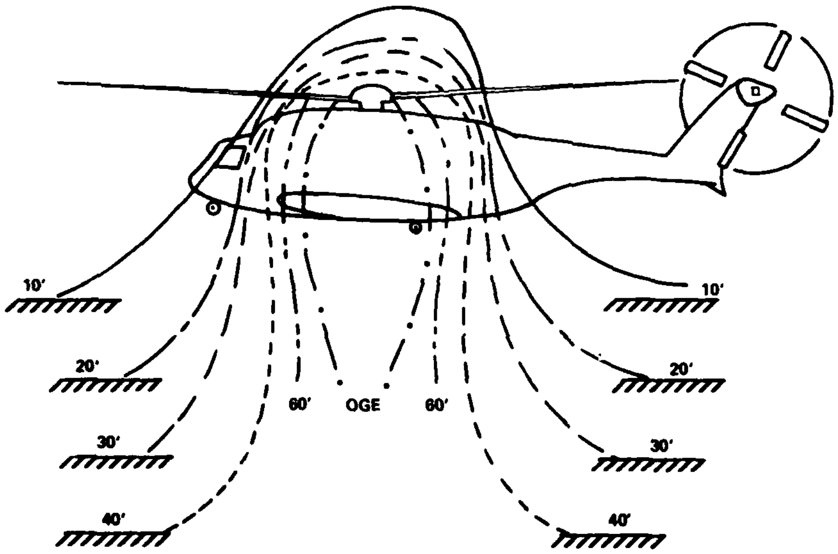

FIGURE 2. VARIATION OF ROTOR INNER WAKE BOUNDARIES WITH WHEEL HEIGHT (REF. 8)

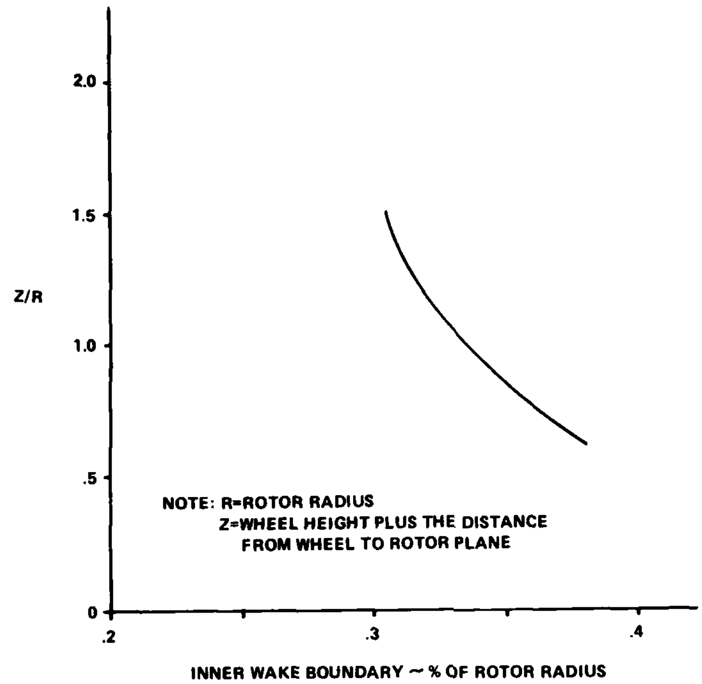

FIGURE 3. ROTOR INNER WAKE BOUNDARY LOCATION

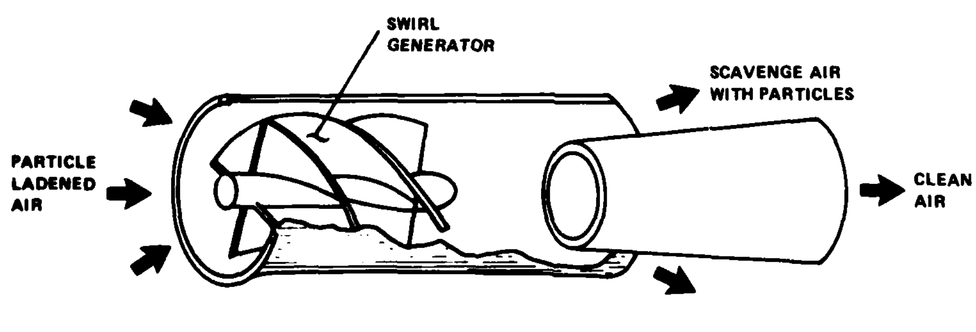

FIGURE 4. TYPICAL VORTEX TUBE PARTICLE SEPARATOR

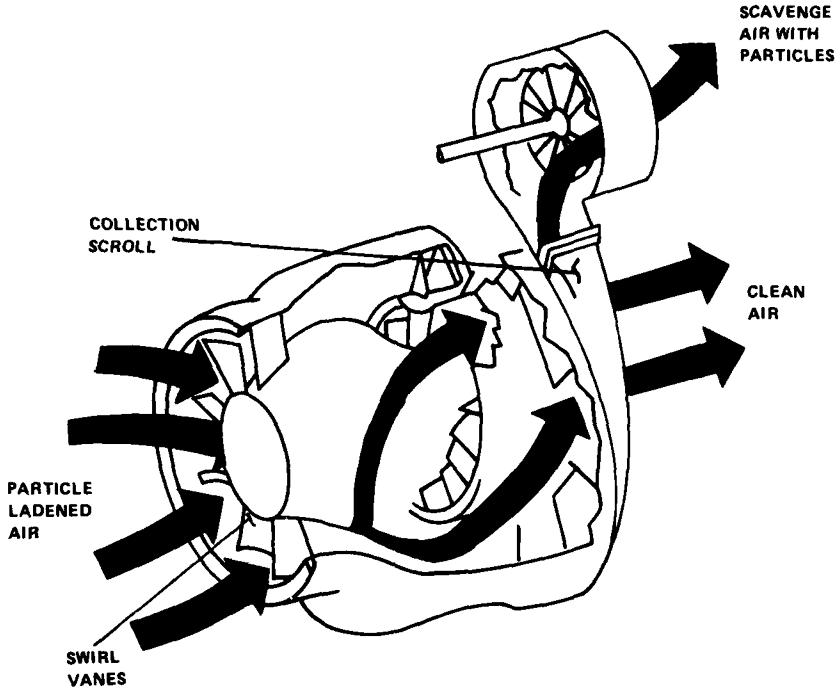

FIGURE 5. CYCLONE TYPE PARTICLE SEPARATOR

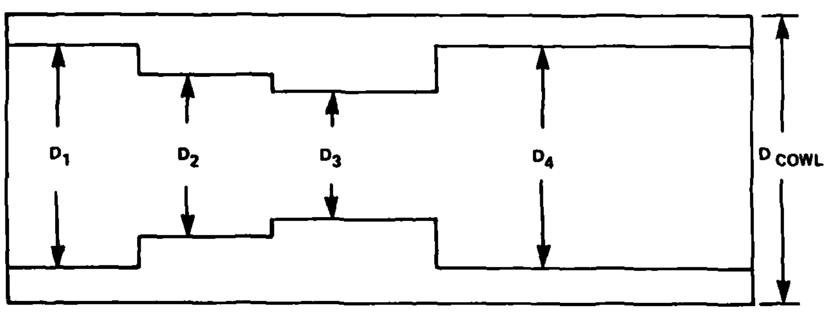

FIGURE 6. ENGINE COMPARTMENT COOLING MODEL

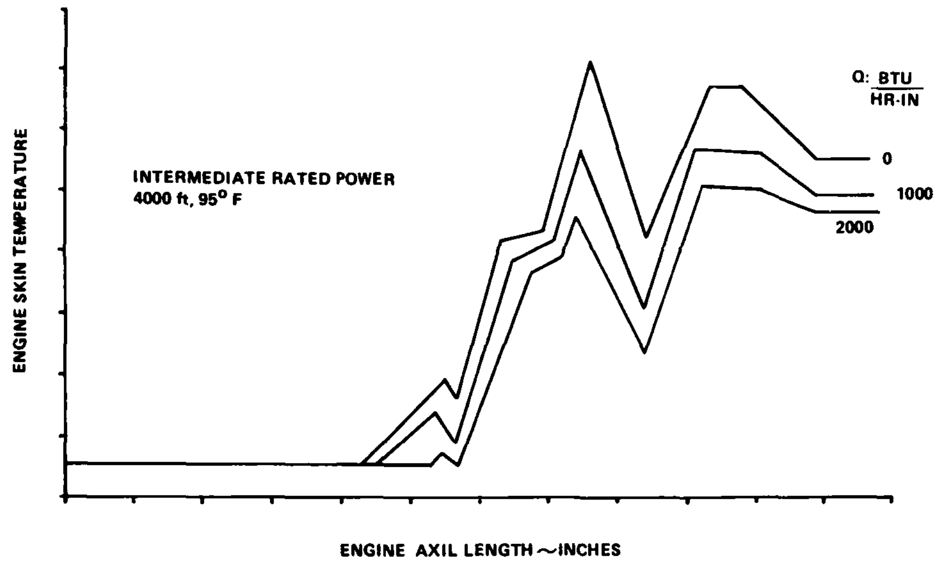

FIGURE 7. TYPICAL ENGINE SKIN TEMPERATURE VS. AXIAL LENGTH AS A FUNCTION OF HEAT REJECTION RATE

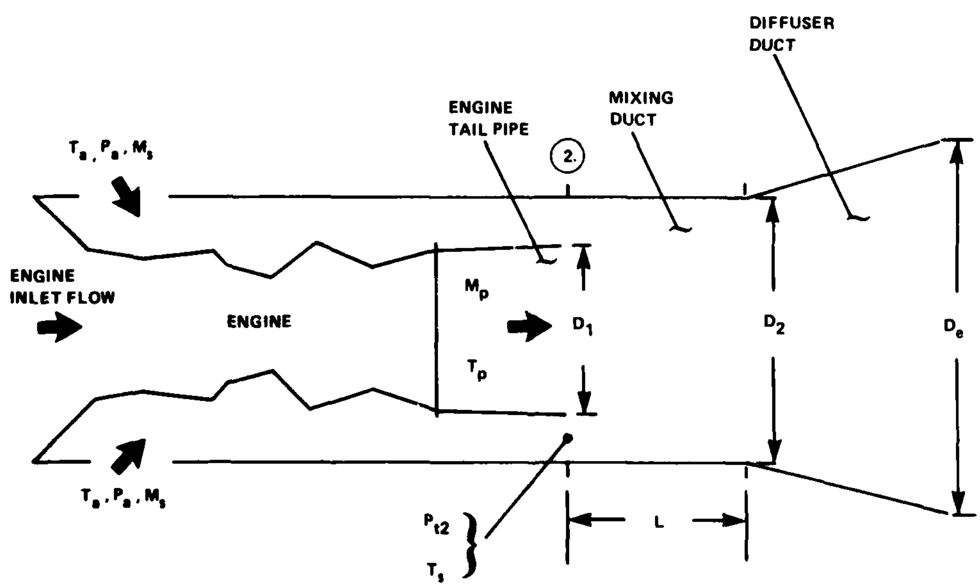

FIGURE 8. ENGINE COMPARTMENT/EJECTOR SCHEMATIC

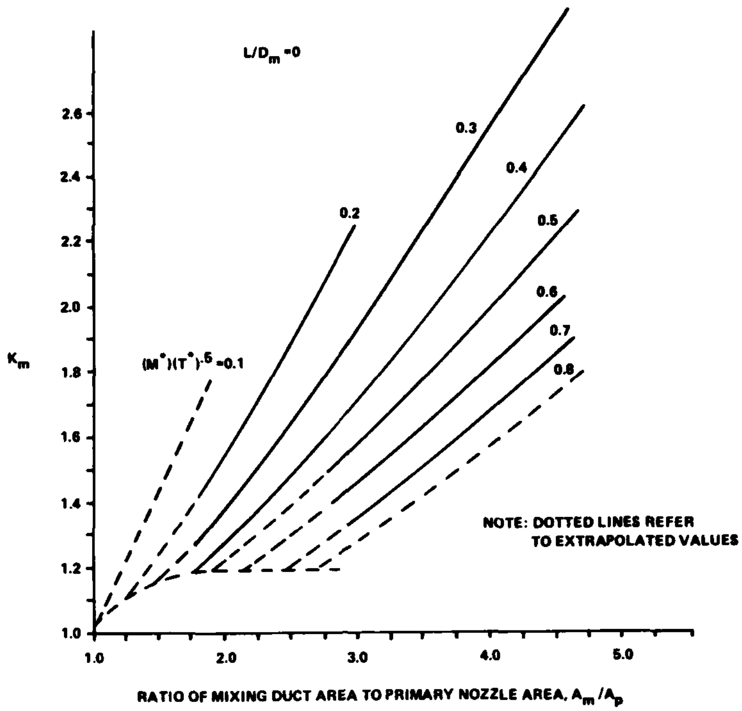

FIGURE 9. MOMENTUM CORRECTION FACTOR (REF. 8)

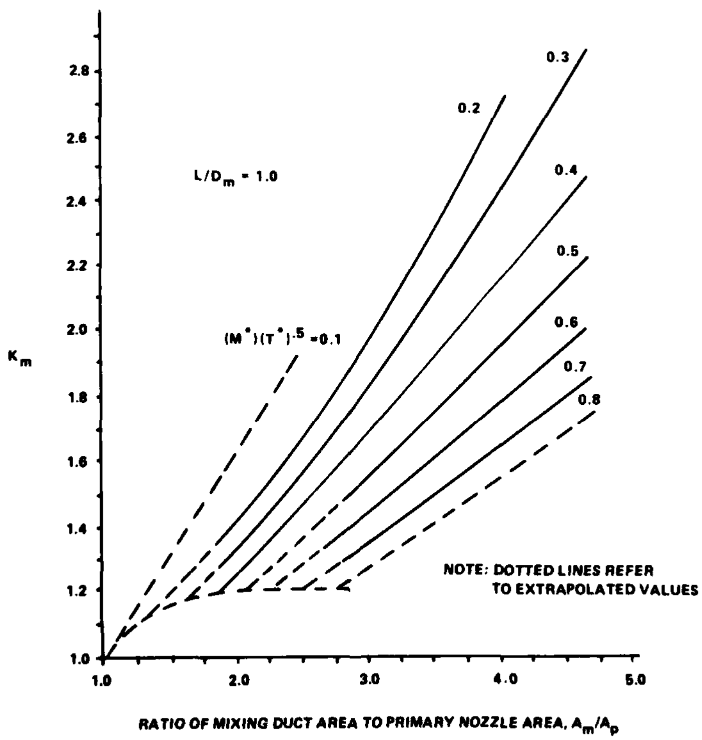

FIGURE 10. MOMENTUM CORRECTION FACTOR (REF. 8)

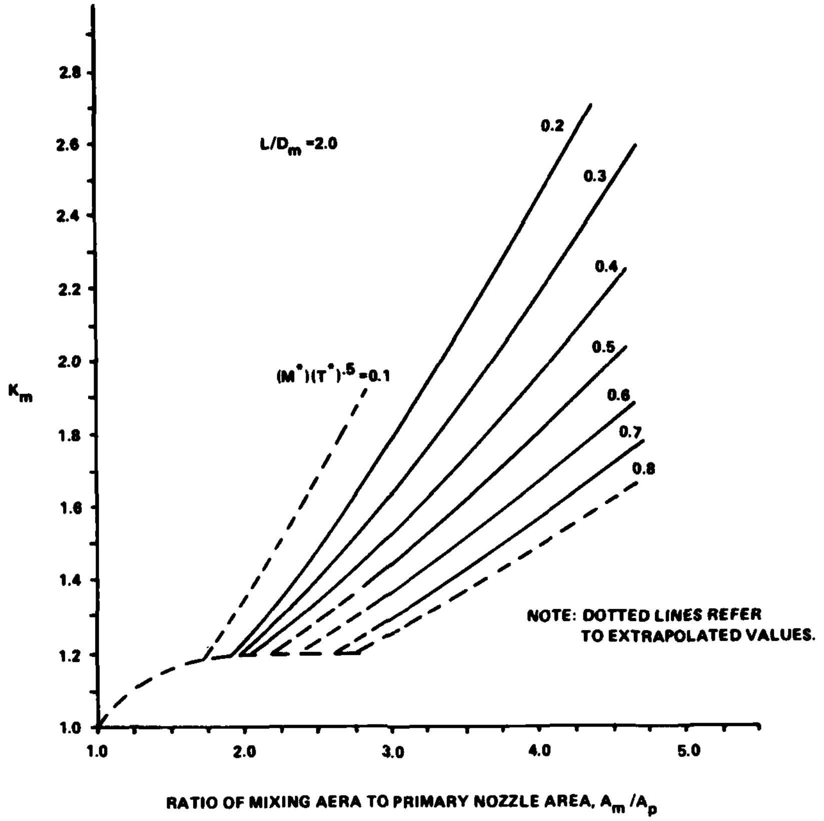

FIGURE 11. MOMENTUM CORRECTIO FACTOR (REF. 8)

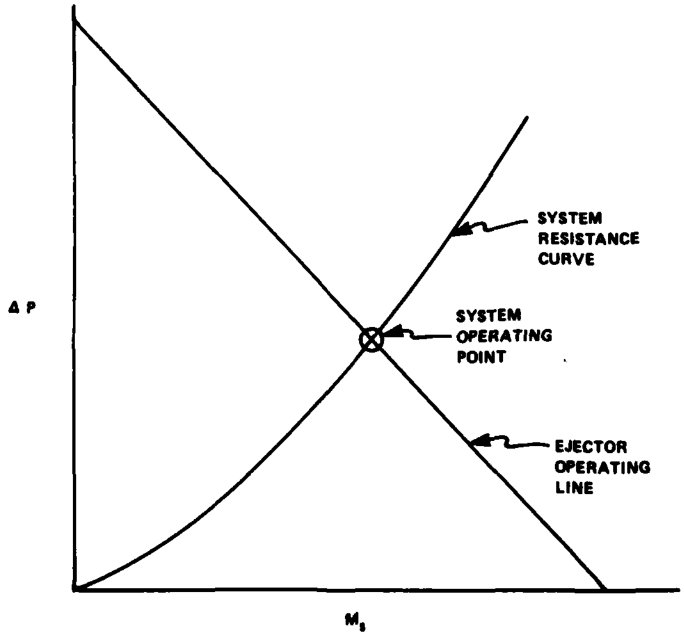

FIGURE 12. EJECTOR/SYSTEM MATCHING

ENGINE-TRANSMISSION-STRUCTURAL AIRFRAME MEMBERS COMPATIBILITY

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SUMMARY

The sophistication of current turbine-powered helicopters have introduced many new problems involving the compatibility of the major multidegree of freedom systems: rotor, control, airframe, drive train, and powerplant. The solutions to the various problems were obtained from the application of basic engineering principles.
This lecture covers the engine installation in the helicopter, particularly the A 129, i.e.: power plant attachment, oil cooling system, engine control and monitoring, electric starting system, location of components and leading particulars, and anti-vibration devices.
The design of the transmission system is dictated by the configuration of the helicopter. An aerodynamic analysis is conducted on the helicopter mission in order to obtain complete spectra of the rotor loads and moments as well as the maneuver loads and transient for maximum transmission reliability. When the interfaces with the engine (s), rotors, accessories, mounts, shafting, and controls are finalized, design criteria or specification can be prepared that defines the requirements that the helicopter and transmission system must meet.
Respondence to the imposed requirements is achieved through integration of the transmission system into the helicopter, principally through the adaption of design, confining new technology to those components, which, because of their nature, could not be conventionally developed.
To avoid a considerable amount of fatigue cracking in the structure of the airframe, because of compatibility with rotor and drive, the solution is to detune the crown frame structure by stiffening the frames and by providing proper synchronizing shaft support mount stiffening. Careful control of the complete transmission installation in the airframe is necessary to prevent unwanted distortion, avoiding failure on the connecting shaft that is transmitting torque to the aft transmission.
The next generation of rotary-wing aircraft will use composites and elastomeric bearings in the drive system, not only the structural airframe members but also as rotating, dynamic load-carrying, members. The application of fiber reinforced resin composites to selected areas of the helicopter drive system will produce weight and cost savings as well as improvements in reliability and safety. These improvements which have been demonstrated in airframe applications can be transferred to structural components in the drive system, with high weight saving.

INTRODUCTION

Two configurations of the current turbine-powered helicopter are in general use:
- a single main rotor with a tail rotor;
- a tandem configuration with twin controrotating rotor of equal size and loading so that the torques of the rotors are equal and opposing.

Many new problems, involving the compatibility of the major multidegree of freedom systems (rotor control, airframe, drive train, and powerplant), have been introduced in the sophistication of the design features including reliability, maintenability and survivability characteristics. The solutions to the various problems were obtained from the application of basic engineering principles.

A key element is the propulsion system, consisting of the engine and the rotor transmission, together with their respective interface to the airframe. Future progress in helicopters will be placed by the advancing technologies in gas turbines and transmission design and manufacture that will further increase their reliability, efficiency, and decreased specific weight, and volume, all at an acceptable cost. There are now available a number of well-developed free turbine engines ranging from 420 to 4,400 horsepower that are specifically designed for use in helicopter propulsion systems. They are used in single or multiengine installations for efficient helicopters with gross weights ranging from about 3,000 to well over 100,000 pounds.

Current transmission systems have reached a relatively high state of development. Allowable surface compressive stresses and gear root bending stresses have increased over

10 and 35 percent, respectively, over second generation design values. Progress is being made in reducing transmission structure and airborne noise by use of such approaches as higher contact ratio gears.
In both configurations, single and tandem rotor, the transmission loads are a function of power and speed. The engine power is determined from the maximum performance requirements of the mission, such as hover at 4,000 ft (i.e., out of ground effect) at 95°F. The input speed to the transmission is fixed by the output speed of the turbine, while the rotor speed is determined by the speed of the rotor blade. Thus, the overall drive train reduction ratio can be determined for a given rotor diameter.
Finally, a through aerodynamic analysis of the helicopter mission is conducted, to obtain complete spectra of the rotor loads and moments as well as the maneuver loads and transient for maximum transmission reliability.
Transmission systems have a reduction ratio in the region of 80 : 1 to 100 : 1, to reduce the gas-turbine engine speed to the main rotor. This ratio is achieved in either three or four stages of gearing, where each stage is either an epicyclic gear assembly, a spiral bevel gear pair or a helical gear pair. The trend to planetary gear trains is well established as it provides maximum torque in a highweight and compact gear reduction. In the planetary design, practical reductions vary from 2.15 : 1 to 7 : 1 for a single stage (sun gear input, ring gear fixed, cage output). In current practice the planetary seldom has a reduction ratio greater than 4.7 : 1. There are times when two planetaries are used in series to obtain higher reduction ratios.

There is much room for improving significant areas of current engine-transmission systems, i.e.: engine and transmission reliability; engine control system reliability; engine and transmission size and weight; propulsion system health monitoring; noise and vibration levels; inlet air filtration; engine/transmission/airframe integration.
About 50% of the cases of unreliability in modern turbine - powered helicopters are occurring in the engine - transmission systems. Of these, 35% are attributed to the engine or its associated subsystems and 14% to the transmissions. Over two-thirds of the helicopter accidents due to material failure are normally due to engine and its systems. Interruptions of operation due to material problems occur once every 400 flight hours; about 30% of such interruptions are caused by the engine-transmission systems. Two thirds of these power interruptions in flight are caused by the partially open-loop hydraulic or pneumatic mechanical engine control systems. Pilot's power demands to the ambient conditions and engine requirements may be matched by closed-loop scheduling of fuel flow and variable geometry microelectronics features. Operational recording devices and maintenance techniques can be developed to continuously monitor propulsion system health.

While there are structural and aerodynamic relationships between the propulsion and support systems of the conventional fixed-wing aircraft, the helicopter closes the gap completely. The propulsion system is also the support system and the complexities inherent in each are compounded because of the integration. To further emphasize the system interdependence, it needs only be remembered that the propulsion-support system is also the means by which the control of the helicopter is accomplished.
The rotor blades of the propulsion-support system are: or hinged at their roots and have movement about three axes; or fiberglass rotor blades of high elasticity attached to short very stiff hub without flapping and lagging hinges. Such dynamic systems are subject to vibrations, stresses and moments. The great centrifugal loading involved will tend to magnify, in the articulated rotor, seemingly small errors in calibration, torques, adjustments, and other similar processes. In the rigid rotor system, high moment capability are determined by the flapping stiffness of the rotor system.

Engine-transmission-structural airframe members compatibility is considered a way of life in the helicopter technology. One of the earliest problems encounterd was that of ground resonance. Adequate system damping and control of the helicopter's natural frequencies have practically eliminated ground resonance as an operating problem. Engineers accomplished this by refinements resulting from intimate studies of blade dampers, oleo struts, spring rates, and so on, and by evaluating the compatibility of the rotor system, rotor damping, airframe inertia, oleo placement on the aircraft, and oleo damping characteristics.
With the sophistication of current turbine-powered helicopters, lowever, have come many new problems involving the compatibility of the major multidegree of freedom systems: rotor, control, airframe, drive train, and powerplant. Their dynamic interface is the most difficult to correct.
However, proper compatibility is usually achieved through: rotor/drive/engine system integration; establishment of the entire airframe structure resonant frequencies and forced-response stresses due to fluctuating aerodynamic pressure at rotor n/revolution frequencies; engine and rotor system torsional analysis to prevent engine/drive system torsional instability; minimization of self-excited vibrations of transmission-mounted components, considering rotor revolution and gear tooth mesh frequency excitations; potential vibrations and stresses of engine installations; compatibility analysis between the variuous systems.

An aerodynamic analysis has to be conducted on the helicopter mission in order to obtain complete spectra of the rotor loads and moments as well as the maneuver loads and transient for maximum reliability. When the interfaces with the engine (s), rotors, accessories, mounts, shafting, and controls, are finalized, design criteria or specification can be prepared that defines the requirements that the helicopter and transmission system must meet for overall compatibility.

POWER TRANSMISSION COMPATIBILITY

In a single main rotor, with a tail rotor, helicopter configuration, the power transmission is normally as in the sketches of figure 1, 2 and 3. Because of the free turbine engine, no clutch is required. The free wheel is in the engine output.

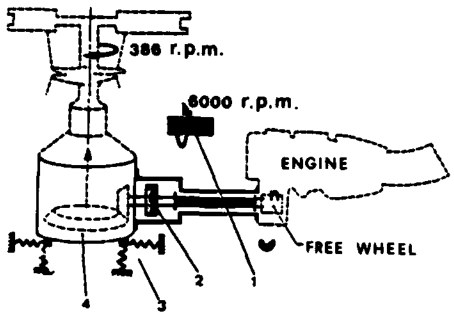


Fig. 1 - Engine to main gear box coupling

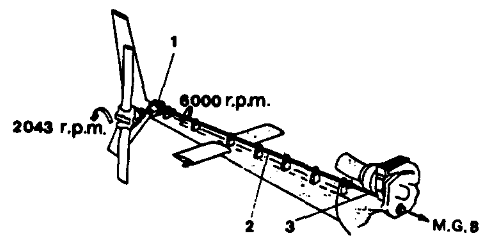


Fig. 2 - Tail rotor transmission system

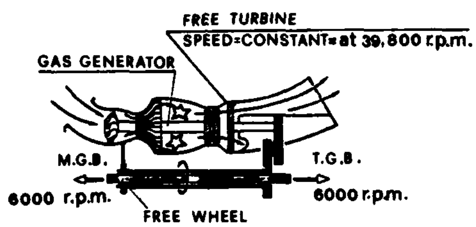


Fig. 3 - Engine installation

The main gear box is flexibly mounted to the airframe and moves at low amplitude in relation to the airframe. Therefore, the engine has to move with the main gear box to prevent introduction of excessive stresses into the drive shaft. This is accomplished by providing a rigid link between the engine and the main gear box, through the housing and barrel junction assembly. The engine may be thus attached at its forward end to the main gear box and only two attachment points are provided to allow the engine follow the main gear motion. Under these conditions the drive shaft transfers only the engine torque. All other loads are picked up by the housing and barrel junction assembly. Flexible couplings will allow any small misalignment between the engine power output shaft and the main gear box power input shaft to be compensated for.
The main gear box, figure 4, transmits the engine torque to the main rotor after reducing the rotational speed and changing its direction. It also transfers the reaction torque from the main rotor, through its casing, to the aircraft structure. The main gear box has a pressure lubrication system monitored by warning lights in the cockpit.
The main gear box gears and bearings, and the mast bearings, are lubricated and cooled by pressurized oil.
The purpose of a rotor brake is to quickly stop the rotor after engine shut down and to prevent the rotor from turning because of wind when the aircraft is parked.
The main gear box, which support the rotor mast assembly in which the rotor shaft is inserted, receives, from the shaft, vertical and horizontal periodical alternated loads. Any rigid attachment of the main gear box onto the transmission deck would transfer these vibrations to the structure. The solution consists in using a flexible suspension between the main gear box and the structure which absorbs (filters) the major parts of vibrations. The main gear box is suspended as a pendulum and oscillates about the intersection of some suspension bars.

The mechanical power transmission systems to the tail rotor, figure 2, includes the following components, starting at the engine rear power take off: forward coupling shaft, tail rotor drive shaft, and tail rotor gear box. The forward flexible coupling shaft absorbs engine/main gear box assembly motions resulting from the bi-directional main gear box suspension.

The power plant, figure 3, is mounted in a fire-proof compartment. At the forward end, it is linked to the main gear box through a flared coupling casing. Its attachment to the rear structure platform is achieved through rubber anti-vibration mounts. The engine bearings and gears are lubricated and cooled by oil circulation under pressure. The pilot has only one direct control of the engine at his disposal: the fuel flow control lever which is only used for engine starting shut-down and acceleration to governed r.p.m. (flight position). Another mechanical control acts automatically on the engine: the engine compensation control, figure 5. This governing control acts on the free turbine governor, automatically reacting to collective pitch variation, through centrifugal governor.
Engine power monitoring and engine starting system complete the engine installation.

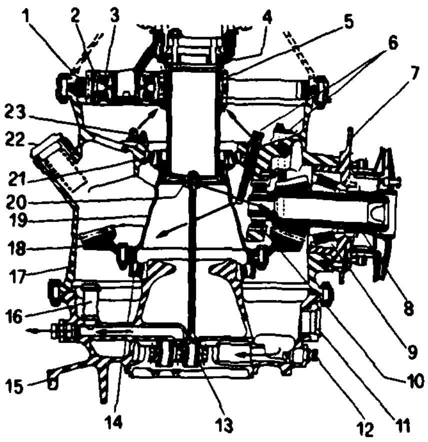


Fig. 4 - Main gear box components

1 - Fixed ring gear - case hardened steel
2 - Planet gear - case hardened steel
3 - Planet gear spherical bearing
4 - Thrust washer - used to secure planet gear cage to rotor shaft (chapter 4)
5 - Sun gear - nitrited steel
6 - Oil jets
7 - Power input casing
8 - Bevel pinion - case hardened steel
9 - Taper roller thrust bearing
10 - Cylindrical roller bearing
11 - Oil level sight gauge
12 - Magnetic drain plug
13 - Oil pump
14 - Taper roller bearing
15 - Lower casing - magnesium alloy
16 - Oil relief valve
17 - Main casing, ribbed, double wall - magnesium alloy
18 - Bevel ring gear - case hardened steel
19 - Vertical shaft
20 - Oil pump drive gear
21 - Taper roller bearing
22 - Oil filler plug
23 - Oil jet

The main rotor shaft, or mast, which is secured to, and driven by, the main gear box, drives the main rotor head, and, transmits the rotor lift to the structure, figure 6.

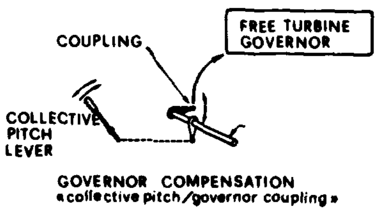


Fig. 5 - Engine controls

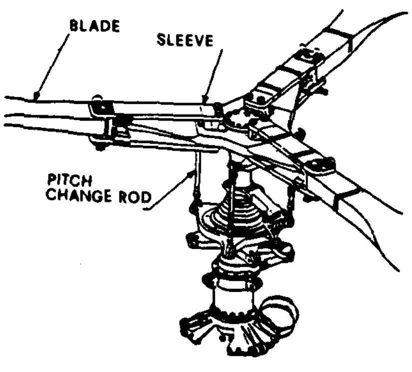


Fig. 6 - Main rotor

The main rotor head which is attached to the main rotor shaft carries the main blades from which it receives the lift. Is absorbs the forces induced by the rotation (centrifugal force, flapping and lead/lag loads).
The blades convert the engine mechanical power into aerodynamic forces (lift). Pitch change rods are used to change the blade incidence for all the blades, equal in collective pitch variation, and different in cyclic pitch variation, figure 7.

The main rotor blades are submitted to periodic alternate loads about the flap and drag axes. These loads cause stresses and reactions at the rotor head: bending and torsion moments applied to rotor shaft; shear loads on shaft.
Rotor head alternated stresses and reactions are regularly repeated every revolution at

each blade. Their frequency is equal to n ω, since there are three blades rotating at speed ω. Rotor head reactions induce, through the rotor mast, suspension bars and main gear box casing, vertical and horizontal vibrations which propagate in the structure. These vibrations are felt in the cockpit (particularly vertical vibrations). The main gear box bidirectional suspension is intended to filter such vibrations; but, some are passing through. Anti-vibration devices, operating on the resonator principle are used to improve the vibration level still further. If the resonator frequency is equal to the excitation frequency, the structure response is zero: the resonator breaks the vibrations.

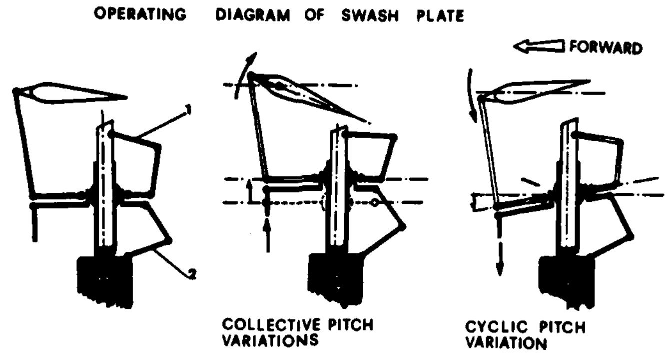


Fig. 7 - Operating diagram of swash plate

Vibration and overstress prevention systems - The propulsion system/structure interaction introduces vibration problems associated with the operation of the main rotor, cabin noise (frequency between 100 Hz and 10 kHz) and operation of the reduction gear box. The vibratory energies generated at the main rotor and at the gears introduce, respectively, mechanical-material fatigue and noise to the passengers via the air and the structure. These alternating stresses are periodic, and their fundamental frequency is the rotational frequency of the rotor. They are amplified or reduced in the rotor head, and are propagated to the fuselage depending upon the filtering devices fitted between the rotor and the airframe.

For that, rotor blades and fuselage are dynamically optimized, minimizing, through active feedback control, forced vibrations in each flight configuration and dynamic signature of the aircraft. A method suggests to place filters between the rotor and the fuselage, to prevent transmission of the stresses causing vibration. The best known example is a flexible element placed between the transmission assembly and the fuselage, to provide insulation.

Another method is the use of energy dissipators. A combination of filtering, through elastic components and energy dissipators, leads to a very compact resonator system.

The noise in a helicopter cabin is the sum of the rotor, engine and power train, noises, predominant being that one induced by the reduction gearbox (200 Hz to 10 kHz).

Each of the various certifying agencies for helicopters has its own unique test requirements and test criteria for transmission qualification. The manufacturer must incorporate these requisite test element into his own overall test plan. These test requirements range in load severity from normal operating conditions to 140% overload.

The U.S. Army, U.S. Navy, FAA and CAA, and the helicopter industry, all have different test requirements, ranging from extreme overstress conditions to testing that only accelerates the mission spectrum slightly. The prorated bearing load in the FAA tiedown test is higher than the prorated load for the mission spectrum shown in figure 8, resulting in a bearing life reduction factor greater than three for the Sikorsky S-76 Helicopter. Consequently, we must consider the effect of the test life reduction factor during the design phase to preclude premature bearing failure. Figure 8 also shows that the CAA test requirement is 140% of input power for 10 million cycles. This test specifies the highest load conditions required by the various certifying agencies and is conducted for a total time equivalent to 10 million cycles on the slowest operating component, Ref. 1. Figure 9 shows the severe 200 - hour ovestress test conducted on the Army BLACK HAWK transmission system.

In the coupling between the helicopter propulsion system and its structure, it is possible to avoid certain resonance problems by controlling the dynamic bahaviour of the simple components making up the main gearbox. Two types of experiments are run on main gearbox illustrating the r.p.m. and torque effects, Ref. 2. Variations with r.p.m. of noise, vibration or stress levels are measured, figure 10, to locate certain dynamic and acoustic problems during operation. Effect of torque on the dynamic behaviour of main gearbox is obtained through accelerometric measurements, figure 11.

Because of benefical strength and noise attributes, high contact ratio gears are gaining wider acceptance in transmission designs. Most of the present day spur gearing operate with a contact ratio between 1.2 and 1.6 which means that these designs have a point in the mesh cycle, or along the line of action, where one tooth must take the entire load. High contact ratio gears is any gear mesh that has at least two tooth pairs in contact at all times, i.e. contact ratio of 2.0 or more. Because the transmitted load is always shared by at least two pairs of teeth, in this configuration, the individual tooth loading is considerably less for high contact ratio gearing than for present low contact ratio designs.

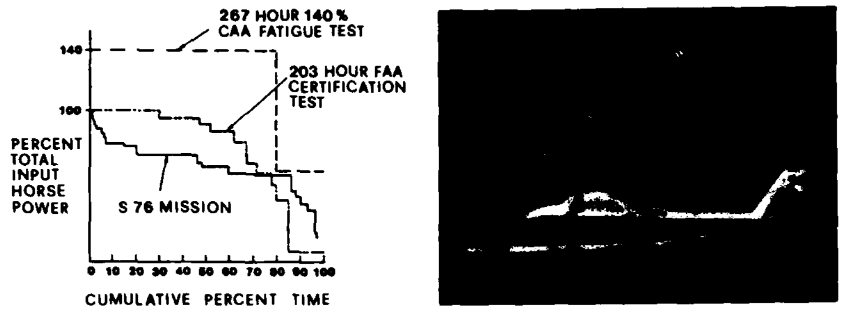


Fig. 8 Horsepower histogram for the Sikorsky S-76 helicopter

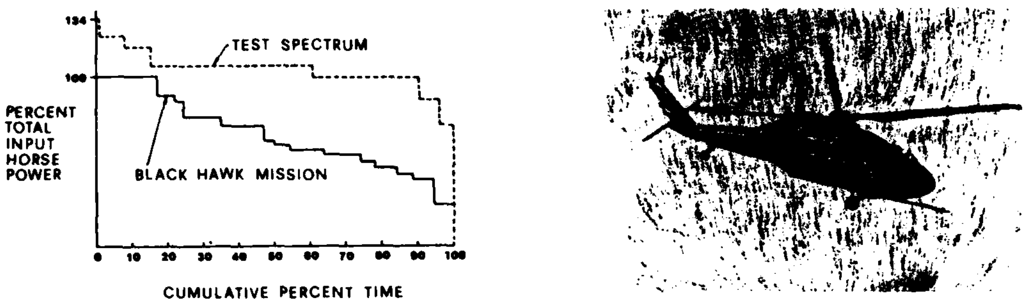


Fig. 9 - Horsepower histogram for the Black Hawk helicopter

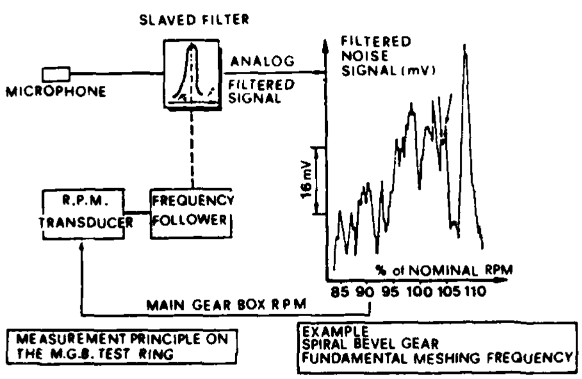


Fig. 10 - Effect of rpm on gear box noise

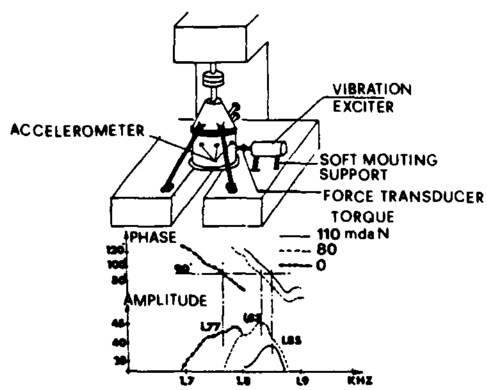


Fig. 11 - Effect of torque on the dynamic behaviour of main gear box

Recent technology developments offer step increases in life and reliability: in engines - improved blade attachment, reduced number of components, improved materials and bearing locations; in transmissions - parallel shaft gear layouts, improved materials, better lubricants, more efficient seals and fine filtration; in rotor systems - composite blades and rotor head; in hydraulic systems - improved manufacturing techniques; in electrics and avionics equipment - improved sensor and connector technology and developments in amplifier and integrated circuit technology. There are components with finite lives due to increased stress levels to save weight, operational load exceedances, maintenance induced damage or due to random shortcomings in quality control or house keeping. Therefore effective means of continuous assessment or defect development (i.e. health monitoring) and life consumed (usage monitoring) will always required.
Plans are well advanced for the introduction of an on-board computer for monitoring the major areas and so assessing the true usage of life limited components. The computer will perform: engine power monitoring; engine usage monitoring; transmission exceedance

and usage monitoring; rotor usage monitoring.
For the Westland/Agusta EH101 Helicopter, the computer will perform control and display management, navigation, communication management as well as health and usage monitoring.

To provide an engine of excellent operational capability, to replace the T58, General Electric has placed a number of design and developmental requirements in cruise fuel consumption, maintainability, long life, safety and reliability. The engine configuration became the T700 - GE - 700, the successfull competitor for the UTTAS (Utility Tactical Transport Aircraft System) and the EH 101, now the Sikorsky Black Hawk. The T700 and its commercial derivative, the CT7, have been selected for 12 aircraft models including two turboprops, proving itself to be the helicopter powerplant of the '80s. The various engine models have been subjected to accelerated endurance and mission testing as summarized on figure 12, with consequent design changes that proved the value of this type of maturity program. Finally, after reliability measures in million flight hours, the excellent engine reliability resulted in a high degree of availability as demonstrated, figure 13, by the percentage of engines that have never been removed from service, Ref. 3.

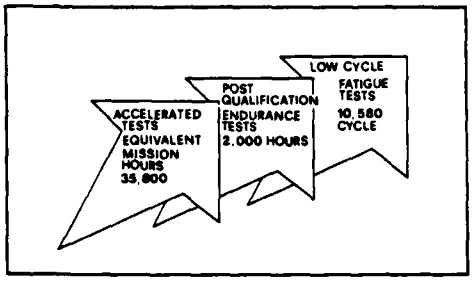


Fig. 12 - Special durability testing

ENGINE AGE GROUP OPERATING HOURS	NUMBER OF ENGINES	ENGINES NEVER REMOVED NUMBER	PERCENT
0-1,000	1 116	868	78
1,000-1,500	49	21	43
1,500-2,000	12	5	42
2,000-2,500	6	2	33
Engines Over 1,000 Hours	67	28	42

Fig. 13 - Engine T700 service experience

In terms of reliability and maintenance, *new main rotor hubs represent significant* progress. For example, the Spheriflex head concept, Ref. 4, by Aerospatiale Helicopter Division, developed from Starflex for the AS 332 Super-Puma, appears to be a flexible concept from the technological and operational point of view.
The starflex rotor head made of composite materials, fitted to production AS 350 and AS 355, figure 14, has proved to be a particularly successful solution with respect to fail-safe properties, but with disadvantages in increased size and drag. The Spheriflex rotor head can be considered as a development of the Starflex, and retains its basic features: laminated elastomer thrust bearing and visco-elastic damper. To reduce the size of the Starflex, the blades are attached directly to the thrust bearing by means of a yoke shaped blade root, closer to the hub center, eliminating the flexible arms of the Starflex star. The viscoelastic dampers are then positioned laterally between the hub and the blade, figure 14.
Since the rotor head will have no more flexible arms, the material used for its manufacture can be chosen freely. The hub can then be made integral with the rotor shaft, figure 15, there by eliminating all problems likely to occur at the hub/mast connection (fretting corrosion, loss bolt torque loading, etc.). The resulting hub-mast is made of metal, but the fail-safe properties of a composite hub are recovered with a composite winding of the upper plate, giving redundant resistance to centrifugal forces and to shear loads due to blade flapping.
By replacing the splines at the base of the mast with a large diameter bolted joint it could be possible to make the hub/mast from titanium, and even from graphite composite material, figure 16.
In particular, the AS 332 Spheriflex, figure 17, consists of four main components:
- the hub integral with the mast; four fittings between the cutouts are used for attaching the viscoelastic dampers, figure 18;
- the spherical thrust bearing, using the laminated elastomer tecnology, allowing the blade to move in three ways (flapping, drag, incidence) and carry the centrifugal forces from blades to the hub;
- the sleeve, used to attach the blade to the thrust bearing and keep it the same distance from the hub center;
- the visco-elastic damper, connecting the hub to the sleeve, providing stiffness and damping of drag movements through shear effect of two layers of visco-elastic elastomer.

The considerable reduction in the number of assemblies and components gives a much higher level of safety and reliability.

Fig. 14 - Starflex-spheriflex evolution

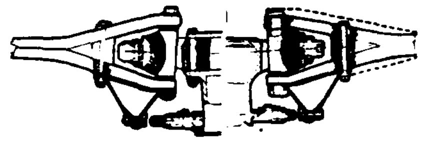


Fig. 15 - Integration of hub in mast

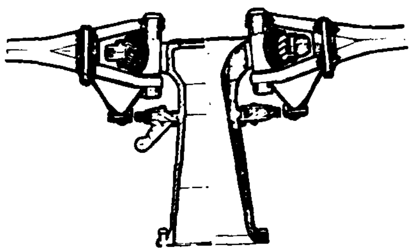


Fig. 16 - Mast hub material: steel to titanium or graphite

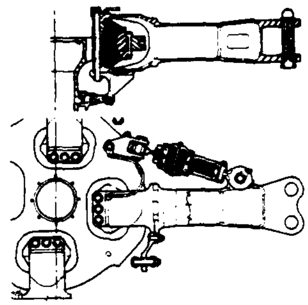


Fig. 17 - AS 332 spheriflex rotor head

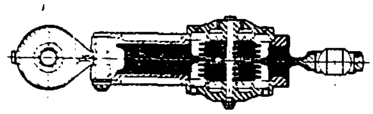


Fig. 18 - Visco-elastic damper

POWER TRANSMISSION AND THEIR RELIABILITY AND SURVIVABILITY

Drive system input speeds increased over 20,000 rpm, always higher power to be absorbed, requirements for reduced weights, reduced noise, increased survivability, improved dependability, increased safaty and lower life-cycle costs, have imposed a significant change in the design and technology of helicopter transmissions.
A typical drive train configuration for the single-rotor machine is shown in figure 19 with the main variations occurring in the location and configuration of the engines. The OH-58 has a single main rotor transmission which represents current design practice in light helicopters, figure 20. There are four reduction stages between the engine and main rotor shaft. The engine output speed of 35,350 rpm is reduced in two stages of helical gears to 6,060 rpm at the input of the main gearbox. The helical gears provide an offset between the engine and bevel pinion axis and allow power to be extracted from the final helical gear for the tail rotor. The first-stage gearing in the transmission is spiral bevel (19/71 reduction) and provides a speed of 1,622 rpm to the sun gear of a fixed-ring planetary unit. The planetary unit provides the final reduction of 4.67 : 1 and gives a speed of 347.5 rpm to the planet carrier and main shaft.

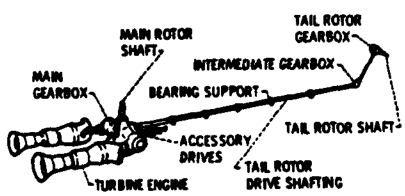


Fig. 19 - Typical transmission systems in single-rotor helicopter

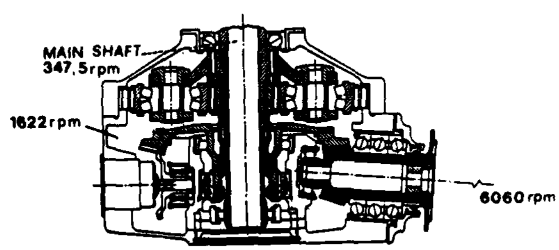


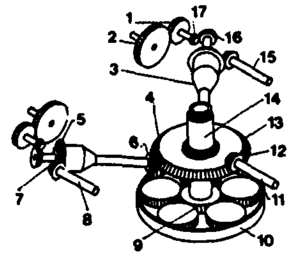
Fig. 20 - OH-58 main transmission

The UH-60 main transmission has five separate, interchangeable modules, figure 21. They are the main module, two engine input modules and two accessory modules.
The CH-47, figures 22 and 23, has transmission typical of tandem rotor design. The CH-47 has engine and combining transmissions in addition to forward and aft main rotor transmissions. Each engine gearbox changes the direction of axis from the power plant to the

combining transmission and reduces the speed.
Typically, seals on the input and output shafts are spring loaded lip, elastomeric types. For high speed and more critical requirements spring-loaded carbon type seals may be used.
Trends for future developments that are expected will be in the areas of improved reliability, quieter drive trains, better materials for high temperature components, better materials for corrosion resistance and high fracture toughness.

The main rotor gerabox of a helicopter represents a significant proportion of the empty weight, typically about 10%. A major reduction in the gearbox weight can therefore be an important contribution in achieving improvements in helicopter performance. In the design phase there are numerous gearbox configurations, all providing the requisite torque, speed, life and reliability that could be adopted. It is therefore necessary to have some accurate means of predicting gearbox weights if the lightest to be selected. Equations are presented, in Ref. 5, which show that gear and gearbox weights are proportional to the torque on the gear and gearbox respectively.

The development of high hot hardness materials has permitted substantial improvements in both durability and scoring capacity. The application of three-dimensional finite-element methods to the analysis of complex spur, helical, and spiral bevel gears, Ref. 6, has progressed rapidly in the last few years. The next step in this effort will be the linking together of fully generated finite-element models of mating gear sets so that their interactions, particularly load distribution among the teeth in contact and along the instant lines of contact, can be studied.
The problems associated with assembling gears to their shafts with keys, splines, or bolted flange arrangements have prompted the use of electron beam welding to provide an integral shaft and gear configuration. The design of the CH-47D, Ref. 7, and CH-46E transmissions carries this concept even further by using advanced computer analysis and graphic techniques to permit simultaneous optimization of gear and shaft geometry while also considering the gear cutting and grinding tooling requirements. The maximum benefit will be obtained by truly integrating the entire gear-shaft-bearing system, Ref. 8, as in figure 24.



1 . GENERATOR GEAR
2 . HYDRAULIC GEAR
3 . OVERRUNNING CLUTCH
4 . MAIN GEAR
5 . INPUT GEAR
6 . MAIN PINION
7 . INPUT PINION
8 . ENGINE INPUT SHAFT
9 . SUN GEAR
10 . RING GEAR
11 . TAIL TAKE OFF SHAFT
12 . TAIL TAKE OFF PINION
13 . TAIL TAKE OFF GEAR
14 . MAIN ROTOR SHAFT
15 . ENGINE INPUT SHAFT
16 . ACCESSORY GEAR
17 . ACCESSORY PINION

Fig. 21 - UH-60 Black Hawk main transmission

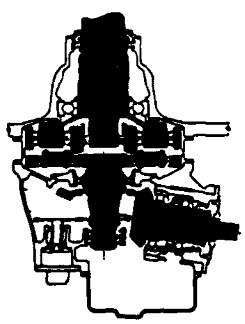


Fig. 22 - CH-47 forward transmission

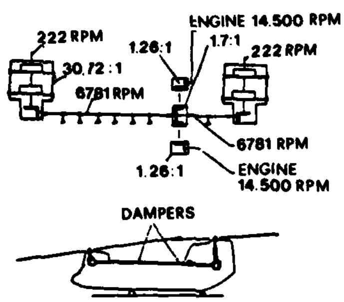


Fig. 23 - CH-47 transmission system

An oil with a higher viscosity (in the range of 10 cs at 210°F) will improve bearing life due to an increase in lubricant film thickness.

Consumable-electrode vacuum-melted AISI 52100 and 9310 steels provide satisfactory performance for gears and bearings operating at less than 300°F. Many bearings have been fabricated from consumable-electrode vacuum-melt M-50 steel which has demonstrated excellent load-carrying capacity, fatigue life, and stable operation up to and exceeding 650°F.

The simple configuration possible with an integrated gear/bearing system design, figure 24, will result in fewer individual component parts. The need of an integrated gear/bearing system design has evolved over the years, figure 25, Ref. 9.

Composite materials are more and more used as structural airframe members. The next generation of rotary-wing aircraft will use composites in the drive systems as well, non only in the static applications but also as rotating, dynamic load-carrying members. The application of fiber-reinforced-resin composites to selected areas of the helicopter drive system will produce weight and cost savings as well as improvements in reliability and safety.

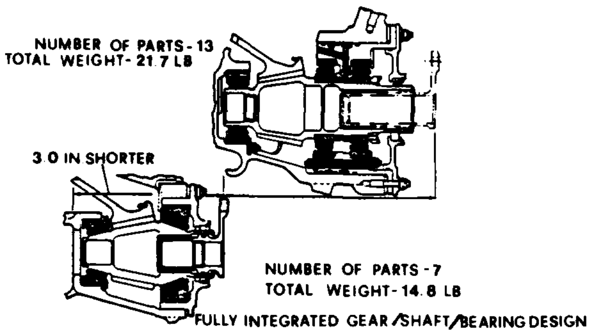


Fig. 24 - Comparison of typical non-integrated and fully integrated designs

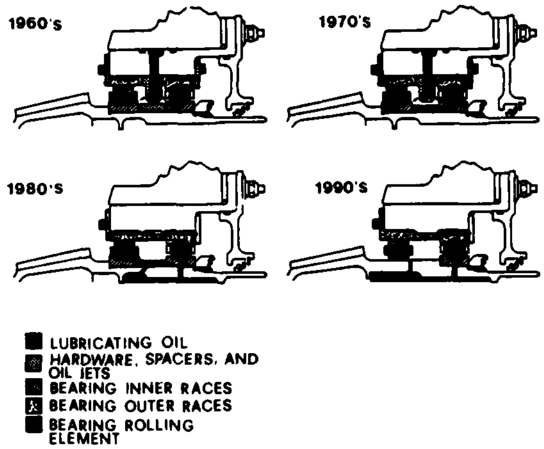


Fig. 25 - Integrated bearing and gear design

The important subject of increasing the time between overhauls (TBO) and the mean between removals (MTBR), i.e. the capacity of the transmission to perform according to the specification (reliability), is connected to a large number of factors: materials, lubricant properties, use of two lubricants (one for the engine and the other for transmission), filtration, surface topography and roughness, elasto-hydro-dynamics, chemical-mechanical interaction, tooth geometry (conformal vs. involute), diagnostics.
Certain areas must be accurately checked when designing and when manufacturing a transmission, because anything forgotten in that field can use dramatic reduction of mean time between removals (MTBR) in service. In this regard, the following areas require accurate thinking:

- possibile stress concentration on teeth as consequence of gearshaft support deformation, gear and support structure thermal expansion, and tolerance of relative position of mating gears;
- lubrication effectiveness at the different load and speed rating;
- quality of lube oil;
- control and corrective actions on the production, to avoid unknown factors leading to premature and random failures;
- choice of the most adequate type of bearing geometry, to correct interference and position of contact patterns between races and balls or rollers;
- correction of tooth geometry and adequate lubrication for splines, avoiding relative movement on splines specifically designed for being fixed;
- alignment on outer and inner rings of freewheels, and no external load to be applied on freewheels;
- natural frequencies of the transmission, sufficiently far way from possible excitations.

A convenient shape and dimension of the structure has to be chosen in order to avoid any natural frequency to be excited by meshing frequency. It is better the holography method using laser light for detecting the natural frequencies of a given structure, other than the calculation. It is important to act on the way the loads are transmitted to the structure, introducing dampers that would absorb part of the total energy introduced by the vibration source and applying extra-masses in convenient points.
Regarding fail safety, the designer will put his best skill for covering the first failure by a doubling element, or, if this procedure is unpractical, by assuring that the element will not fail during its service life. The result of a study, Ref. 10, carried out on epicyclic stages applied on helicopter gearboxes, for increasing the fail safety of his design is shown on figure 26.

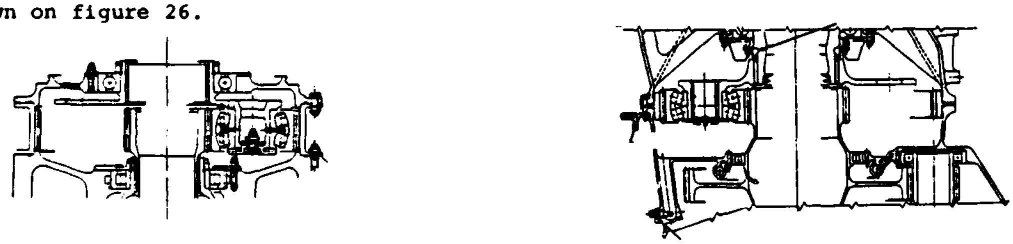


Fig. 26 - Two helicopter bearings. The first is the result of an optimized fail safety engine

The working temperature in transmission gearboxes shows a continuous trend to increase, mainly for the need of weight minimization. In fact: gears are operated at the highest possible stresses, thus increasing the heat to be dissipated; the oil temperature is increased to the highest possible level with the consequent possibility of reducing the dimensions of the heat exchangers and the quantity of oil to be kept in oil tanks. The high working temperature thus affects the metal structure weakening the transmission elements, and reduces the effectiveness of lubricating oil, increasing the probability of serious damages of the parts. The designer must therefore seek those design choices that allow for a reduction of the risks due to high running temperature.
Transmission efficiency measurements and correlations with physical characteristics of the lubricant are carried out with special equipment, like the NASA 500 hp helicopter transmission test stand shown in figure 27. Advanced experimental machine has been recently proposed, Ref. 11, for measuring lubricant film thickness and shape in a elasto-ydro-dynamic lubrication, suitable to simulate the commonly used mechanical contacts, e.g. that between sphere or rolling bearing raceways, figure 28.

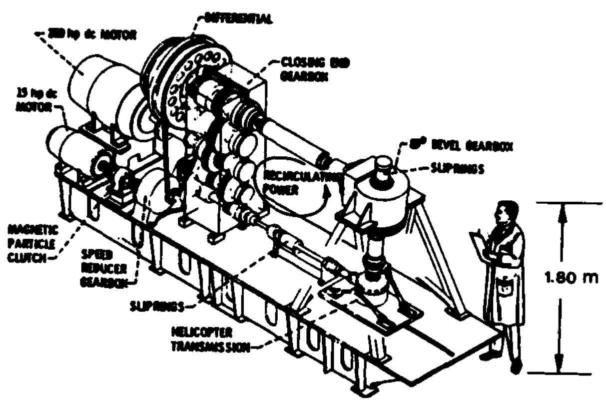


Fig. 27 - NASA 500 hp helicopter transmission test stand

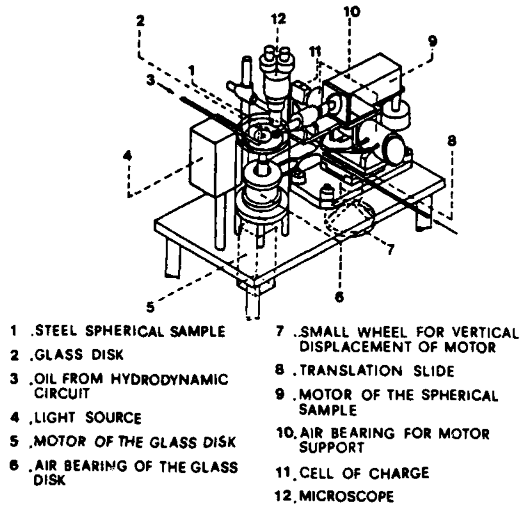


Fig. 28 - Advanced machine for lubricant film evaluation

About the evolution of the design techniques for helicopter main transmission gearboxes, we mention the technical solutions, Ref. 12, adopted by FIAT Aviazione in its cooperation with the French Firm Aerospatiale, through the design, the development and the manufacture of mechanical transmissions for the SA 321 Super Frelon, figure 29, SA 330 Puma, SA 360 single engine Dauphin, SA 365/366 twin Dauphin, figure 30, helicopters. During the years of such activity the design techniques have shown a great evolution due to changes in user's requirements.
Ref. 12 foresees the design solutions that will be valid for a gearbox to be made available in the late eighties. With reference to a hypotetical main gearbox for a medium twin engined helicopter (input power 2 x 650 hp, input speed 6000 rpm, main rotor speed 385 rpm, tail rotor drive speed 1700 rpm, driven accessories 1 or 2 hydraulic pumps, 1 a.c. generator, 2 oil pumps, 1 brake, 1 cooler fan, synthetic oil conforming to MIL-L 23699), four different main gearbox configurations have been taken into account. In figure 31, solutions A, B, C, D are respectively shown, with the following main characteristics.

Configuration	Number of stages	Number of gears	Number of bearings
A	2	7	14
B	2	7	14
C	2	7	14
D	3	11	18

All the configurations have been designed without using epyciclic trains; this being the main difference in comparison with existing gearboxes. Particularly for all the solutions the following requirements have been set up: minimum life for each bearing, 6000 h; minimum life for gears and splines, 20,000 h; gears material, AMS 6265 carburized; ring gear material, 32 CDV 13 nitrided; bearings material, 100 C6 AISI 52100 (M50); shaft material, AMS 6415 hardened & temperated.

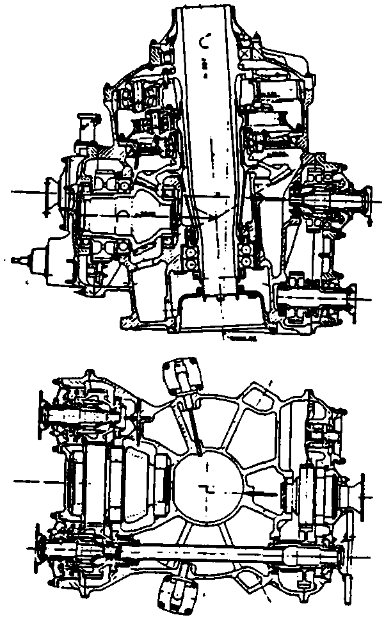
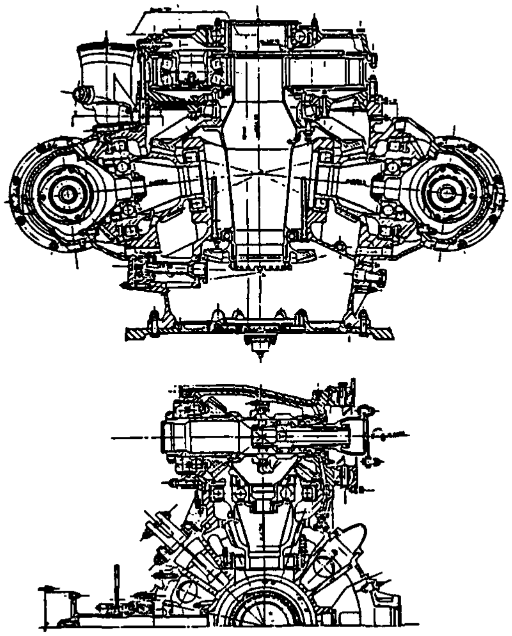


Fig. 29 - SA 321 main gearbox



SA 365 MAIN GEARBOX

Fig. 30 - SA 365 main gearbox

Taking into account weight, costs and reliability, through value analysis techiniques, the application index is reported in figure 32. The configurations that achieve the best results are C and A, with rather ligh reduction ratio in the last stage. The main advantage of the A solution is its typical installation flexibility due to a configuration using bevel gears only. The remarkable characteristics of the C configuration is the presence of an internal spur gear.
From the results of such analysis, depending upon the importance given to the appreciation functions, the conclusions are: in such range of reduction ratios, the two-stage solution is quite preferable to the three-stage solution; the delection of the epicyclic train results in remarkable advantages as weight, cost and reliability; the ring gear solution is quite preferable to the solution using external spur gears.

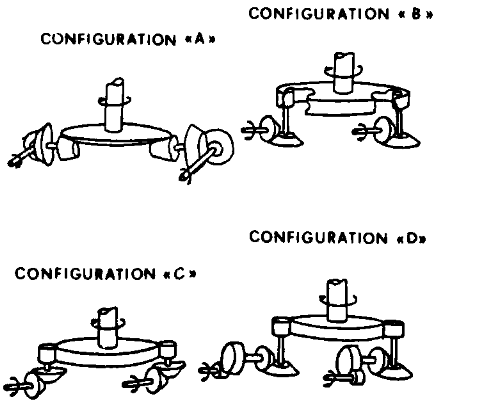


Fig. 31 - Design solutions for main gearbox configurations

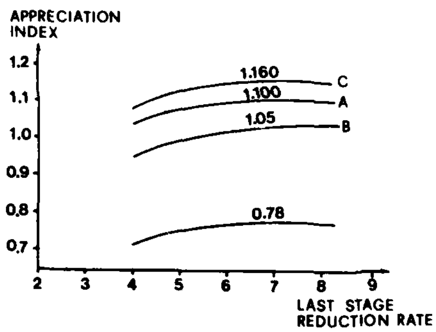


Fig. 32 - Appreciation indexes for the design solutions of figure 31

As a complete example of integrated transmission system design, to achieve low weight, high life, and maximum reliability, we present the design criteria of the A 129 helicopter drive system, Ref. 13, figure 33.
The transmission ratio of 78.2 is obtained through four reduction stages tailored in such a way as not to interfere with the ring diameter of the planetary stage, which, on the other hand, was required to provide a rational suspension system for the transmission, figure 34. The first and second stages consist of spiral bevel gears, the third stage of helical gears which drive the final gear consisting of a spur gears planetary system. The

tail rotor shaft is driven directly by the collector gear and by a spiral bevel gear set, figure 35.
The accessory gearbox can be driven both by the collector gas (in flight) and by the gear shaft of the left engine first stage (on ground) and has the possibility to drive one alternator, a hydraulic pump, a filter compressor and the lubrication oil cooling fan.

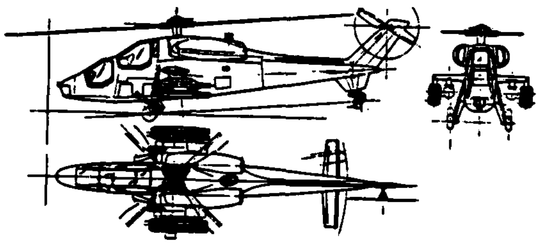
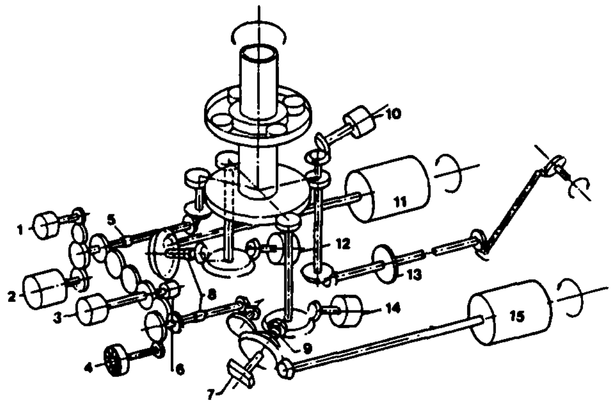


Fig. 33 - Agusta A - 129



1 .ECS COMPRESSOR
2 .GENERATOR
3 .HYDRAULIC PUMP
4 .COOLING FAN
5 .FREE WHEEL
6 .LUBE PUMP
7 .LINEAR ACTUATOR
8 .FREE WHEEL
9 .FREE WHEEL
10 .HYDRAULIC PUMP
11 .ENGINE N.2
12 .GENERATOR (KIT)
13 .ROTOP BRAKE
14 .GENERATOR (KIT)
15 .ENGINE N.1

Fig. 34 - Schematic drive system of A - 129

The power transmission scheme is completed by two engine shafts provided by flexible couplings and by a supercritical shafts system which drive the intermediate and tail-rotor gearboxes.
Respondence to the imposed requirements, in the anti-tank role, is achieved through integration of the transmission system into the helicopter, confining new technologies to those components that could not be conventionally developed.

Although the modern helicopter and its associated propulsion system is highly sophisticated, many problems still persist. In addition, new requirements are demanding still further improvements in the technology of engines and transmissions.

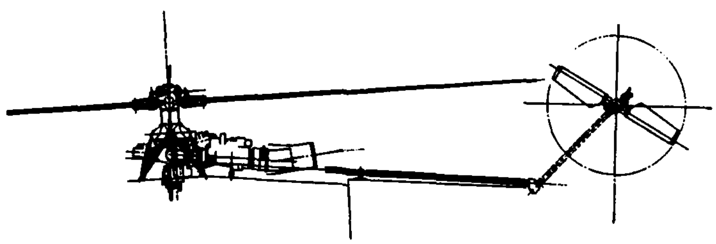


Fig. 35 - A - 129 propulsion and drive system

ENGINE AIRCRAFT STRUCTURE COMPATIBILITY

Engine aircraft structure compatibility in helicopters is a complex process, and the various components present different problems to be considered separately.

Regarding the main rotor blades, most of the fatigue testing has been done using a single mechanical exciter to provide combined flap and lag loading.
The main rotor hub is subject in service to a very complex pattern of loading, particularly if it is of a non-articulated design such as the Westland W.G.13 Lynx hingeless rotorcraft. On-off centrifugal loads, on-off torque and the oscillatory flap, lag and torque loads all have different influences on different parts of the hub.
From the other hand, it is necessary to use the test to establish an endurance limit for the failure site and then to use this in conjunction with flight stress measurements to calculate the life.
The exact failure location is unknown until the hub breaks, and this will not where strain gauge measurements are taken either on test or in flight. However, it is often in an area of stress concentration with a rapidly changing stress field.

Regarding the gearboxes, the testing method is to run them under a program of various power levels to provide a well mixed loading. The levels are normally subject to a factor of 1.4 for a single test and 1.3 for four tests. The size of the factors does not mean however that there could be a risk of the mode of failure on test being different from that in service. It is therefore important to check that the meshing patterns of the gears are of similar character to those in service and that excessive distortion of the casing is not taking place.

Regarding the structural airframe members directly connected to the main gearbox, i.e. the main gearbox attachments and the adjacent structure, attempts have been made to test these parts by using a complete airframe as specimen and loading it via a dummy gearbox and rotor head. The major difficulty of this method lies in obtaining a distribution of high frequency loading which is even approximately correct. The fact that these high frequency loads must be factored also tends to produce unrepresentative failures elsewhere in the structure, due to panel buckling etc.. For these reasons it has been found more satisfactory to do separate tests on the individual attachment points using several hydraulic jacks on each to obtain the correct stress distribution.

Until now being fully aware of the sensitivity of helicopters to fatigue, and the inability to provide alternate load paths, the helicopter structural engineer has been forced to adopt a safe fatigue life approach in determining the air-worthiness of the helicopter structure.
Referring to accident rates which are due to structural failures, for example more than 3 per 100,000 hours in a civil UK twin engined helicopetr flying, it has been suggested to improve the test methods to reduce those accident rates.
In the maintenance of helicopters it is essential that the rotor heads be properly serviced to ensure that dampers have adequate fluids and that the blades are properly tracked.
The accepted method of calculating safe fatigue lives is based on: fatigue testing of new components; strain gauge analysis of prototype helicopter operated within a specified flight envelope; predetermined flight schedule reproducing what occurs in practice. But, in this way, it is not considered any fault due to: determination in service for corrosion and mishandling; undiscovered mistakes in quality control; lightning strikes; exceedance of flight envelope due to weather conditions; vibratory stresses due to imbalance of rotors, erosion of blades, etc.; any change in operational requirements. To cover these accident parameters, we have available "early warning indicators".
The majority of accidents to helicopters involving structure have been due to fatigue of some kind, and, in comparing the civil helicopter with the civil jet aeroplane, the accident rate for helicopters appears to be worse by a factor of at least two. Again, a higher proportion of helicopter accidents are due to structural failure. For this reason, there is need for improvement.

Significant transient loading of engine structure itself and airframe structural members will be associated with rotor blade torsional and flap-lag oscillations. Moreover g forces from helicopter turns, pullups, hard landings, etc., act on the lifting and thrusting rotor and cause displacement of the rotating parts of the engine, Ref.s 14,15, 16 and 17, as consequence of accelerations as high as 10 g in some maneuvers.
Vibrations in the airframe structural members may lead to excessive level of fatigue. The analysis may be applied to the problem of determining which part of the fuselage structure is most effective in reducing the rotor induced vibrational response in the region of the pilot's seat.
The realistic rotary-wing aeroelastics problem is obviously the interblade mechanical coupling, or the coupling between rotor and fuselage, or the coupling between the rotor/ fuselage and the control system.
Prediction methods for estimating total aircraft dynamic stability and response characteristics have been developed by starting with a minimum number of degrees of freedom to describe the system. However, the capability to insulate the total vehicle response and rotor loads due to a given pilot action is not now at level to permit theoretical design of damage prevention devices of airframe members directly connected to the power transmission system.

Intake/engine/aircraft compatibility

The design of intakes for helicopters is involving a number of interdisciplinary problems; the integration with the airframe is effected to a great extent from the interaction between rotor and inlet and from the fuselage shape. The aerodynamic design of the intake must depend upon the size, type and mission, of the helicopter.
With the aim of a good engine-airframe integration, special care has to be devoted to the real intake design in order to optimize the aerodynamic and environmental characteristics. Due to the current limitations of theoretical methods for these analyses, testing on models in facilities is necessary to develope the inlet and achieve the required performance.
Significant development work has taken place over the last decade to produce a basic intake design which has total environmental protection with a minimum of engine and air-

craft performance degradation.
The acceptance test criteria now used for intake protection systems have been considered, by service use, adequate in most cases.
The helicopter is essentially a low level aircraft, which normally operates below 3,000 metres. If icing conditions are encountered, the helicopter is often unable to "climb over" the weather. Ice on the air intake and on the forward parts of the engine and the compressor blades will erode progressively the performance of the engine, and its formation will shed into the engine with the probability of damage to the compressor and/or engine surge and flame-out. The operation of gas turbine engines in high sand and dust concentrations can cause rapid erosion of the compressor blades and its casing with resultant engine performance and handling degradation. Momentum separation systems for use in icing and sandy conditions have been found to be beneficial in protecting engines from the effects of salt water contamination (for salt concentration levels well in excess of 2.10^{-7} by weight of salt in air). Specification foreign object damage testing for engine/intake combinations is generally limited to single tests with small birds and either 25 mm or 12 mm diameter hailstones. However, there is now an increasing amount of ad hoc testing carried out using items which are known to have caused engine compressor damage in service.

To assess the performance of the Agusta A 129 air intake, Ref. 18, a full scale glass fiber model was manufactured to measure the aerodynamic, dust filtration and icing characteristics. Even though static tests were considered to be sufficient to conform the intake with the engine requirements, nevertheless, the aerodynamic interaction of the fuselage upstream of the intake in special flight attitudes was analyzed via small scale wind tunnel tests. Static aerodynamic tests were regarding pressure loss and distortion, and foreign object ingestion. The same test ring was used for foreign object separation tests; a particular facility was used for dust and sand separation tests. For icing conditions, a number of antiicing systems were considered, including hot air, electrical heating, electrical pulse and pneumatic inflation boot. Hot air was selected on the basis of semplicity and reliability.
Regarding engine/airframe integration, an activity on forward flight conditions is in progress to complete the information from the static tests, with data on the effect of fuselage on pressure loss and distortion of the intake.

A side ways facing anti-iced intake in conjunction with an engine mounted separator was selected as the means of protection for the Westland/Agusta EH 101. This intake system was chosen as being the lowest risk means of achieving line of sight protection from airframe shed ice, up to five hours mission time in icing conditions and general sand, foreign object damage (FOD) and salt water spray protection.

An integral inlet separator for the General Electric T 700 engine is shown in figure 36, Ref. 19. The T 700 separator had excellent performance with ice particles less than 12 mm cubed and its performance was maintained with the side intake fitted. Air intake velocities for the side intake were so low that only very small ice particles were attrachted toward the intake at hover power conditions.

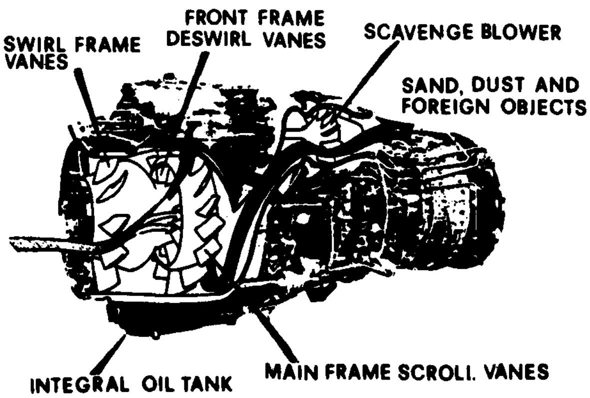


Fig. 36 - GE - T 700 inelt particle separator

AIRCRAFT COMPATIBILITY WITH INTEGRATED FLIGHT CONTROL/AVIONICS/CREW/SECONDARY POWER SYSTEM

To achieve mission success, an increasing number of sensors are necessary to supplement the crews' abilities during a day and night, good and bad weather, operation. Therefore, in order to maximise the effectiveness of such a helicopter, it is essential

that the various avionic sub-systems and sensors should function in harmony with engine, airframe and, not least, the crew.
Thus a total system approach is necessary for integration of the new generation helicopters, in order to meet the increasingly severe operational requirement.

The crew/system interface, installation and avionic system integration, must be considered from the beginning of the design of the vehicle and its guidance and control system. The reliability of the new systems must be improved considerably, and indeed, some of the more sophisticated and complex systems may have to be rejected.

The ability of a rotorcraft pilot to perform the flightpath management function is determined by the handling quality of the vehicle. Handling qualities are determined not only by the stability and control characteristics of the vehicle, but also by the displays and controls which define the pilot-vehicle interface, the environmental characteristics, and the performance requirements.

The primary task of the pilot of a two-crew rotorcraft is to stabilize the aircraft and to control the magnitude and direction of its velocity vector, that is, to perform the flightpath management function. The copilot's responsibilities include most of the other systems supervisory and control tasks.
With the need to simplify the total pilot workload, there is impetus to help with decisionmaking and to automate certain tasks. The goal of this effort is to produce a predictive methodology to aid the understanding of human supervisory control of highly interactive systems. Contract have been initiated to use concept design for incorporating artificial intelligence and smart systems techniques into LHX (next generation light rotorcraft) cockpit automation features.

Several advanced technologies are rapidly approaching realization in the rotorcraft industry. Gain in technology levels for areas such as digital/optical flight controls and automated cockpit design will provide greatly increased mission capability in future rotorcraft designs. Other advances, such as the all-composite fuselage, will decrease vehicle size, weight, and cost.
The full benefit of advanced technology is obtained by the integrated effect of, Ref. 20: advanced rotor; advanced composite structures; digital fiber optics flight control system; drag reduction; advanced engines; advanced mission equipment and integrated/automated cockpit. According the particular analysis conducted in Ref. 20, with reference to a two-place (tandem) seating arrangement and twin engines using only currently-fielded technologies as represented by the UH-60A and AH-64A helicopters, the combined benefits of these advance technologies are remarkable: 59% smaller engine (T 700 - GE - 700); 49% smaller empty weight; 52% less mission fuel; 48% smaller design gross weight. In particular, among several proposed advaced flight control systems, one of the most promising is the digital, fiber optics flight control system. Such systems have reliability, maintainability, advantages over conventional mechanical control systems. Optical systems also appear to have weight advantages over mechanical control systems, the weight advantages becoming more pronounced as the size of the aircraft increases. The design with the advanced flight control system has a 25% decrease in flight control weight, yielding a 3% reduction in design gross weight.

The maneuvre and gust alleviation active control strategy applied to the fixed wing aircrafts, Ref. 21, can be extended to the rotor wing vehicles in order to control blade flexibility. This strategy, based on the spectral data computed from the flexible blade structural mode of vibration measurements, is employed to relax, by appropriate longitudinal cyclic pitch modulation, Ref. 22, the flatwise bending moments induced by environmental disturbances. The restoring cyclic pitch commands are derived processing the output data from and electro-optical laser sensor by means of a microprocessor performing the spectral power density real time computations. The obtained results yield promising indications regarding the gust alleviation control system feasibility, and its potential usefulness particularly in high speed low level operations.

The evolution of sophisticated integrated avionics packages has caused and will continue to cause significant perturbations to avionics and airframe manufacturers' integration plans.
Future developments will tend to increase complexity and expand the boundaries of airframe integration making the use of combined cockpit airframe and avionics simulators a necessity. Therefore, these techniques must be universally adopted by airframe and avionics integrators if they are to remain in a competitive position whilst providing high integrity, viable avionics installations.

<u>Secondary power systems/airframe compatibility</u>

The secondary power system for rotorcraft generates power for uses other than primary propulsion. Secondary power systems are used to manage and control the operational activity of the helicopter and to provide a safe and comfortable environment for sensitive equipment, crew and passengers.

Power source include: propulsion engine; auxiliary power unit (APU); and stored energy such as bottled gas or fluid, chemical processes and mechanical energy.
Some of the primary functions of the typical rotorcraft secondary power system are here shown, in conjunction with their source and type of energy produced.

POWER SOURCE	ENERGY FORMS	FUNCTIONS
Propulsion engine	Electrical	Main engine starting
Auxiliary power unit	Hydraulic	Hydraulic flight control supply
	Pnaumatic	Hydraulic utility sypply
		Electrical utility supply
		Cockpit & avionics air conditioning
Stored		
Bottled gas/fluid (hydraulic accumulator)	Mechanical (direct drive)	Ice protection system
Chemical process (battery, monopropellants)		Hoisting
Mechanical		Powered landing gear

A Particular helicopter configuration, designed for a given mission criteria, will have specific functional load requirements which in turn determine secondary power systems sizing/rating. The integration of the secondary power systems to perform this function can affect the rotorcraft weight and power requirements.
Ref. 23 reports some results of a study to illustrate secondary power systems concepts and trade selection in a hypotetical advanced baseline attack helicopter with a defined mission/load profile. Helicopter and secondary power system flight load profile are shown in figure 37.

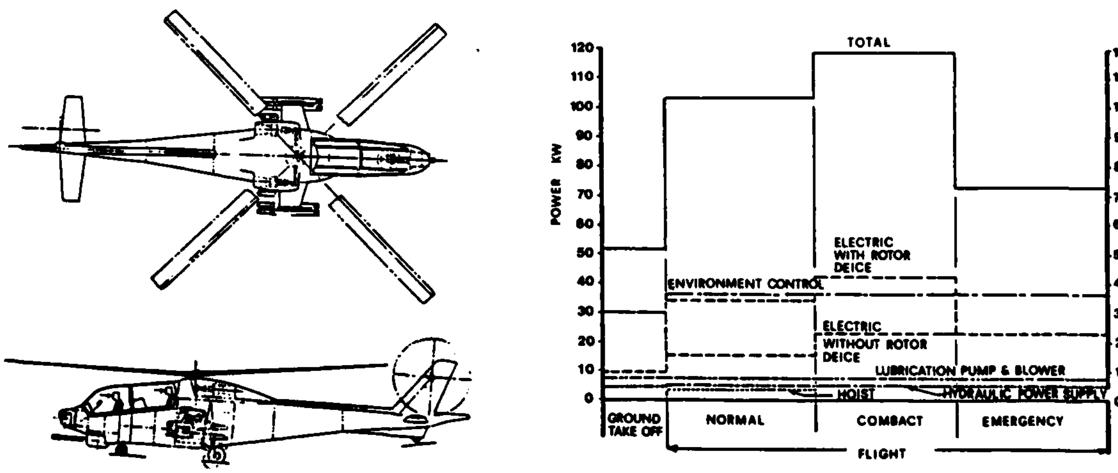


Fig. 37 - Advanced attack configuration base-line for secondary power system

COMPOSITES AND ELASTOMERS FOR POWER TRANSMISSION/AIRFRAME COMPATIBILITY

Composites

Although many of present helicopter design have composite components, in the main these are in use in non-fatigue critical areas or parts where a large ad-hoc superfactor to cover uncertainty can be accepted without compromising the design. However we are now engaging in the design of rotor components in reinforced plastic materials.
The main problems are the lack of background knowledge of strength variability in the various failure modes and the long term effects of environment on the properties. The latter is the more difficult to provide rapid answers.
Those who have experience of making composite blades may well ask why we do not just take

note of their experience and avoid the problem; the answer is that certain features of their designs are significantly different from what has been done before and might have a different susceptibility to things like moisture ingress.
These factors have to be checked out, since one of the basic principles of our composites programmes is that these components shall be shown to have a standard of airworthiness at least equal to metal parts in service.
The strategy in composite demonstrator programmes running at present has been initially to avoid the environmental issue by aiming only for a limited clearance, say 50 hours flying. We start by using a hopefully conservative estimate of variability to obtain factors which are used in the design phase. The component is then built and tested to find out where and how it fails. Structural elements representing these failure modes are then tested in larger numbers to determine the variability. Tests on actual components, if necessary after redesign, are again carried out to obtain the mean strength and confirm that failure modes have not changed if there has been any redesign. Initial lives can then be calculated from this knowledge of mean strength and variability using calculated loads. Provided that the lives are satisfactory, ground running and initial flight trials to measure loads can take place.

However, the application of fiber composite technology to rotors is an active subject. The performance pay off through the use of composites is particularly promising because of the ability to mold almost any aerodynamic shape and tailor the elastic properties expecially to take advantage of their anisotropic possibilities. Probably, the most sophisticated use is found in the Boelkow BO-105 and Aerospatiale SA 341. But the most advanced application of composites so far reported is undoubtedly the Boeing CH - 47 blade.
Tail rotors also stand to benefit from the application of composite technology. Tail rotors should have significant potential for weight reduction which is especially important since weight reduction at the tail of a single rotor helicopter can have a big impact on easing the design balance problem. By using a single continuous unidirectional fibreglass epoxy spar, running through the hub from tip to tip, all the centrifugal and bending loads are reacted by the high strength and stiffness of the unidirectional fibers. Feathering action was provided by the torsional deflection of the basic spar which has low torsional stiffness. The primary pay off here was the reduction of maintenance requirements.

Studies of the application of composites to rotor heads have suggested only 10 - 15% weight reduction relative to conventional steel and aluminum constructions; in rotor heads the local and attachment problems become all important, so whole new concept of rotor head design may be required to fully exploit the high stress potential of composites.

The most important attribute of composites applied to shafting of drive systems is the ability to tailor stiffness distributions to optimize shaft length and supports for minimum weight, by using 45° laminae to react torque and axial layers for bending.

In the transmissions, composite for gearbox housing and for large ring gears will be used. Substitutions of composites for magnesium/aluminium technology in housing and support assembly suggest up to 25% weight reduction.

Carbon fibre reinforced plastic is a material most for stiffness or aero-elastic requirements. In its cross-plied form the ratio of the elastic modulus to the density is twice that of metals. Its tensile strength for half the weight is comparable with that of dural and evidence for simple specimens shows that the fatigue stress for infinite life can be as high as 60% of the ultimate stress. Is is available in short lengths or as continuous fibre, as felt or mat or cloth and in the form of sheet of tape. Bond with other composite materials are easily made.
The value of mixing glass fibre and carbon fibre is becoming generally recognized. The selective use of carbon produces structures which are basically cheaper, easier to fabricate and more efficient than elements made entirely in that fibre. Some examples, such as propeller blades with carbon fibre spars and rotor blades with carbon fibre trailing edges are already in use. The torsional stiffness or glass fibre and honey comb rotor blade is very effectively increased by a skin of carbon fibre.

The development of the various composites has led to a better fundamental understanding of the mechanics of materials. In particular, it is becoming clear how to make substances which combine the desirable features of ease of fabrication, low density, high resistance to fatigue, ultimate strength, thoughness, etc.. An example of this mixes carbon, boron and epoxy resinn.

Advanced composite materials, such kevlar epoxy and graphite epoxy are introduced in the manufacturing of dynamic parts: main rotor blades, hubs and primary air frames structures, tail units, panel, etc..

Composite airframe structures have gone thorough three generations of evolution at Sikorsky Aircraft, Ref. 24. In the first generation, fibreglass was used to fabricate

lightly loaded parts and secondary structures. The feasibility of using molded construction to replace sheet metal detail was clearly demonstrated.
In the second generation, advanced composite materials such as kevlar epoxy and graphite epoxy were introduced. These new materials have significantly higher strength to weight ratios than fiberglass, and as a result, nominal weight as well as cost savings are achieved.
The third generation is applying composites to primary airframe structures to achieve significant improvements in weight, cost, reliability, maintainability, crash worthiness, and survivability.
The first generation of composite airframe parts exemplified by the use of non metallic materials on the Sikorsky CH-53A helicopter. The fiberglass is used in secondary structures containing complex shapes and contours.
The second generation of composite airframes is exemplified by the composite usage on the S-76, Black Hawk, and CH-53E helicopters. Composite applications on the S-76 and Black Hawk have expanded to include virtually all of the secondary structures on the aircraft including crew doors, main rotor pylons and even the cargo floors on the Black Hawk. The S-76 stabilizer, figure 38, is the first primary airframe structure made from advanced composite material certified by the FAA for a commercial aircraft. Certification of this structure includes static, fatigue, and environmental-temperature-wet qualification testing. Techniques for designing critical joints such as main transmission attachment mounth have been also developed under Army NASA research and development contracts. A transmission support structure has been built entirely from graphite epoxy material. In this design, transmission attachment loads are introduced directly into graphite epoxy beams by transmission load links. It is so demonstrated that major attachments might be accomplished in composites by simply building up the laminate thickness in lieu of having to resort to complex machine fittings as used on metal structures. Weight savings of 23% have been demonstrated.
The UH-60A Black Hawk composite rear fuselage program, as third generation at Sikorsky, has shown that composite could be applied to primary structures in a cost effective manner. Such composite rear fuselage shown in figure 39 weighs approximately 400 pounds and is predominantly of kevlar and graphite epoxy material.

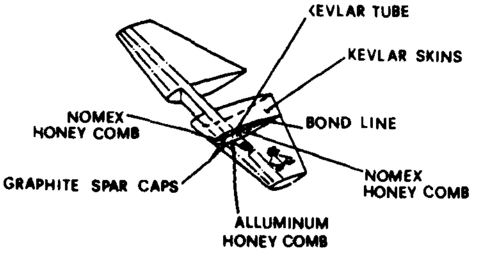


Fig. 38 - S-76 composite material stabilizer

Fig. 39 - Black Hawk composite rear fuselage

The inability of most composite systems to deform plastically and thereby alleviate stress concentrations combined with the extreme anisotropy in stiffness and strength properties and thermal expansion characteristics are particularly challenging. Therefore, effective methods of joining composite materials are key to successful and efficient structural concepts regardless of the nature of the materials involved. The weight advantages of most lightweight materials are all too frequently eroded at a joint. In composite structures the subject has special significance and poses a major challenge to the engineer. Indeed, a joint in a composite structure arises in the following levels:

- micromechanics level, the fiber-matrix interface phenomenon;
- macromechanics level, at interfaces between layers as characterized by so-called free edge problem;
- structural level, at interfaces between two or more separate components as in the conventional joints.

The successful use of composite materials for transmission/structural airframe members compatibility is depending upon the joning reliability.

Elastomeric bearings and dampers

About 20 years ago, a program has been initiated to investigate elastomeric bearings, dampers, etc., and their application in helicopter mechanisms. Since then, these bearings have been evaluated in flight on main and tail rotors in both the flapping and pitch-changes axes. Each of these elements is composed of alternating layers of elastomer and metal shims, bonded and vulcanized to major metal components in order to accomodate static and/or dynamic input of motion and loads. A major breakthrough in bearing technology for oscillatory or reciprocating motion is so provided.

Referring in particular to bearings, the result of bonding the elastomer to the shims and major metal components behaves no sliding action requirement, thus eliminating the concern for brinelling and abrasive wear.

The need for a simplified, more efficient and failsafe, helicopter rotor system is of prime importance to the future of helicopter industry.
The elastomeric-bearing rotor hub shown in figure 40 uses a single spherical bearings per blade to provide blade motion in the flap, lag and pitch directions. The elastomeric bearing consists of concentric spherical laminations of an elastomer and metal bonded together in thin alternate layers. The bearing is mounted so that the centrifugal force exerced by the blade results in compression of the elastomer, while pitch, lag, and flap motions result in shear deflection of the elastomer. Thus, the assembly can provide the required blade motion and still withstand high centrifugal loads. Use of the elastomeric bearing results in a universal hinge that gives all three degrees of freedom while eliminating lubrication, seals, and relative motion between highly loaded contacting surfaces susceptible to dirt, water, and other contaminations. All motion occurs by deformation of the elastomer, so the assembly, reliability, inspection, and maintenance requirements are greatly simplified. The inherent mechanical simplicity of the elastomeric-bearing rotor hub may be seen in the exonometric projection of the three bladed horizontal hook design, figure 41. Such elastomeric-bearing rotor hub represents a new concept in rotor hub design, and, combined with the new advanced-geometry blade, contributes to the advancement of articulated rotor system technology. The damper attachment point on the pitch axis which has been chosen for the elastomeric-bearing hub is shown in figure 42, Ref. 27. The structural substantiation of the design in figure 42 was made in accordance with MILS-8698 Structural Design Requirements, helicopter, Ref. 28. The elastomeric-bearing hub is designed to absorb 7,500 horsepower in a 60 - 40 percent distribution of power applied to the two rotors of a tandem helicopter, corresponding to the CH-47, of 44,000 pounds gross weight. Three advanced-geometry blades with a centrifugal force of 93,000 pounds at the normal rotor speed of 230 rpm, figure 23, are to be used. The predicted spring rates for the elastomeric bearings are: flap or lead-lag = 30,000 inch-pounds per radian; pitch = 9,500 inch-pounds per radian.

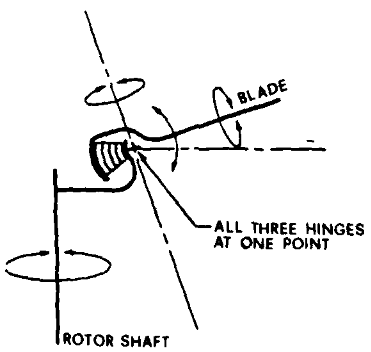


Fig. 40 - Schematic drawing of elastomeric-bearing rotor hub

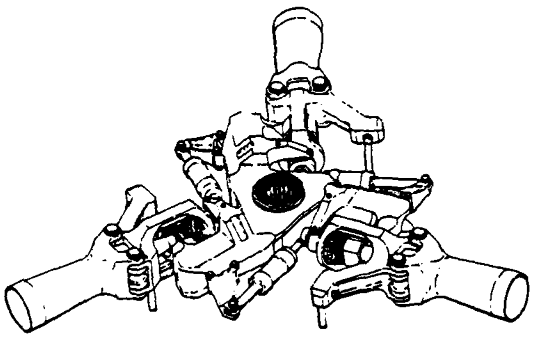


Fig. 41 - Three-bladed elastomeric-bearing rotor hub

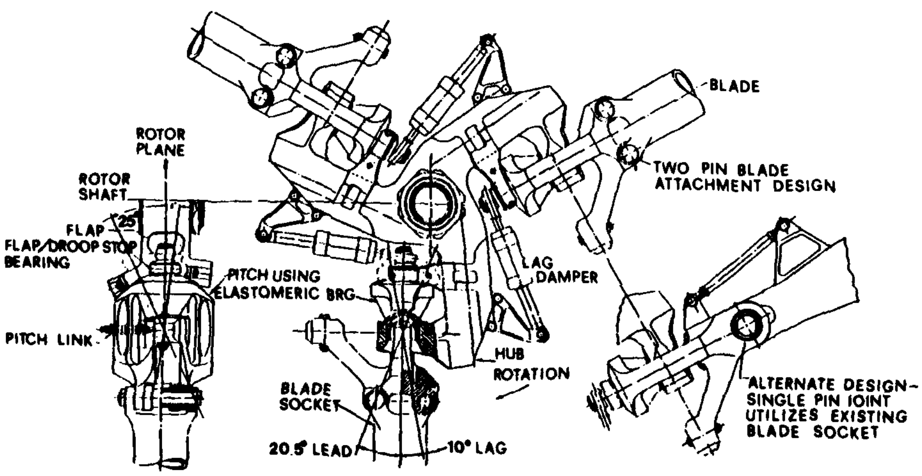
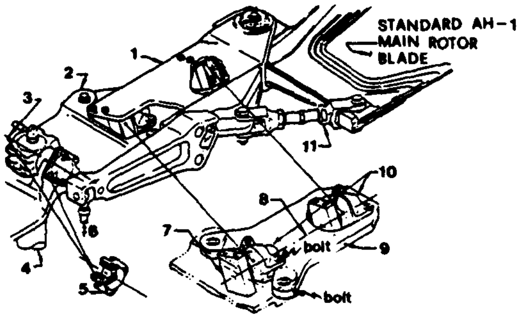


Fig. 42 - General configuration of elastomeric-bearing rotor hub

The rotor with elastomeric bearings would require less maintenance, since elastomeric bearings wear gradually and can be designed to be inspected visually with rotor fully assembled.
A program at Bell, in 1970, evaluated elastomeric bearings in the main rotor of an AH-1 G helicopter. The experimental rotor, figure 43, used two elastomeric pitch-change bearings in each grip and two elastomeric bearings in the flapping axis. The flapping bearings carry the rotor lift and drive loads and accomodate flapping motions up ± 12 deg. The pitch-change bearings carry the blade centrifugal force and transfer the blade shear and bending loads to the rotor yoke. The bearings also accomodate the collective and cyclic pitch-change motions.
Figure 43, Ref. 29, shows the yoke extension and pitch-change bearing assembly. Two bolts attach each bearing to the yoke extension; a single bot (not shown) fastens the bearing block to the inboard end of the pitch-change bearing. Two bolts through the grip and each bearing block hold the yoke extension assembly in the grip, and the yoke extension is attached to the flexure in the same manner as on the production rotor.

In figure 44, elastomeric thrust bearings for tail rotor are shown, designed for use on the blade pitch-change axis. Such bearings were designed to be used in a two-bladed seesaw tail rotor operating at 1,650 rpm. The thrust bearing was designed for a blade centrifugal force of 16,000 lb and, with any other grip bearing in series with it, must accomodate changes in collective blade pitch from -6 to +18 dey. The thrust bearings also transfers the blade bending load to the rotor yoke.
An all elastomeric-bearing grip designed to take advantage of the elastomeric bearing's favorable characteristics is shown in figure 45.



1 . GRIP
2 . STANDARD YOKE
3 . FLAPPING AXIS
4 . MAST
5 . ELASTOMERIC FLAPPING BEARING
6 . PASSO
7 . BEARING BLOCK
8 . PITCH CHANGE AXIS
9 . YOKE EXTENSION
10 . ELASTOMEPIC PITCH CHANGE BEARING
11 . DRAG BRACE

Fig. 43 - Details of rotor yoke extension and pitch-change assembly

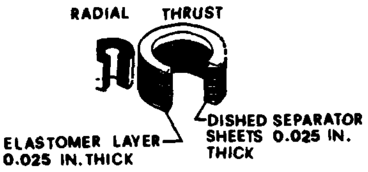


Fig. 44 - Bearings designed for tail rotor

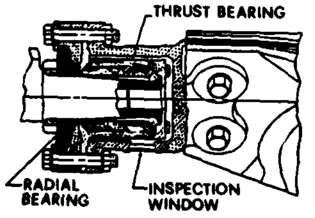


Fig. 45 - An all elastomeric bearing grip

Many elastomeric bearing have been designed and tested. The calculation of such testing results, combined with more accurate and sophisticated finite element analyses have now made designing more scientific, allowing at least preliminary sizing to be done by designers with only basic experience in the field.
Production rotorcraft applications for elastomeric bearings are primarily rotor blade support or hinge bearings, and bearings for pylon/transmission isolation systems. For fully articolated rotors, as on the Sikorsky UH-60A Black Hawk and the S-76, the elastomeric bearings must support very high axial compression load smaller radial shear loads, but still allow large oscillatory pitch, lead-lag, and flap rotation. For semi-rigid rotors, as on the Bell 412 and the 400 series, elastomeric, bearings react large centrifugal force and shear loads in compression with cyclic pitch change in torsion, but accomodate smaller angles of lead/lag or flap. Representatives of the above bearings have been subjected to qualification tests of up to 2500 hours and have been analyzed by computer programs.

Conventional fabric-lined spherical rod ends have been developed to a significant degree, Ref. 30, so that excellent performance, within wear limits, have been obtained. The most common configuration is the spherical tubular bearing as shown in figure 46. The laminations of rubber and metal form a zone of a sphere. Where little or no cocking motion is imposed and axial loads are small, the lower cost cylindrical bearing may be considered, as shown in figure 47. To increase the axial spring rate, conical bearings are frequently used, as in figure 48. Figure 49 shows one configuration using two small spherical bearings.

In addition to basic geometric configuration, other design features affect performance.

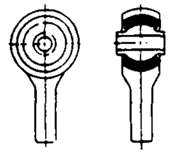


Fig. 46 - Spherical tubular rod end

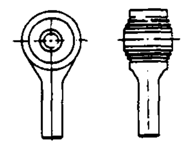


Fig. 47 - Cylindrical rod end

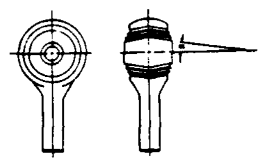


Fig. 48 - Conical rod end

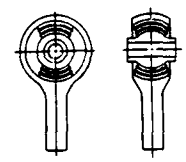


Fig. 49 - Spherical thrust rod end

Elastomeric dampers are a new generation of dampers designed to prevent helicopter and/or VTOL rotor system instability. The damper employs a highly damped viscoelastic polymer, vulcanized and bonded to metallic members which in turn are connected to the rotor system. Deformation of the viscoelastic material produces a total resisting force composed of a damping and an elastic component operating 90° out of phase due to the hysteresis inherent in the polymer. The damper service life can be in the range of 1500 to 2000 flight hours with no maintenance or lubrication required.
This kind of damper is suitable to prevent an instability phenomena, in helicopter rotor system, known as ground and air resonance from occurring while the rotor is operating on the ground or during flight. The instability is called resonance due to the fact that it occurs when a fuselage natural frequency involving horizontal or in-plane rotor hub motion, is coincident or close to the difference between the rotor speed and the rotating blade in-plane natural frequencies. Failure to adequately control this phenomena can lead to total destruction of the aircraft. Elastomeric dampers will increase the total in-plane stiffness, thus increasing the natural frequency. The metallic components, to which the viscoelastic material is bonded, are attached to the rotor head and blade cut off as shown schematically in figure 50, Ref. 31. Normal lead-lag (in-plane) motion of the blade causes an oscillatory angular (one cycle per evolution) motion of the blade cuff about the drag hinge as shown for the fully articulated system of figure 50. By geometry, this angular motion is transformed into oscillatory axial linear motion of the damper which in turn causes deformation (shearing) of the viscoelastic material between the outer and inner metallic attachment plates as shown in view A-A of figure 50. The deformation of the elastomeric material produces the dynamic forces necessary to prevent air or ground resonance of the rotor system. Figure 51 is a partially detailed view of an elastomeric damper to explain the construction. The damper inner member is attached at one end to the rotor blade cuff, while the two outer members are connected to the rotor head.
The versatility of elastomers allows the rotor designer new freedom in determining the location and configuration of the damper. The broad range of elastomers available also permits tailoring the damper characteristics for specific applications.

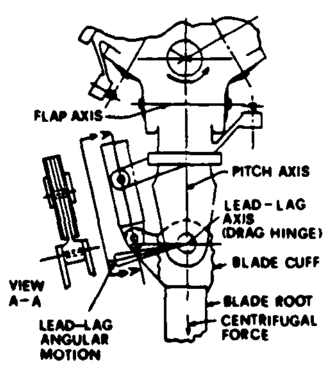


Fig. 50 - Installation of visco-elastic damper in articulated rotor system

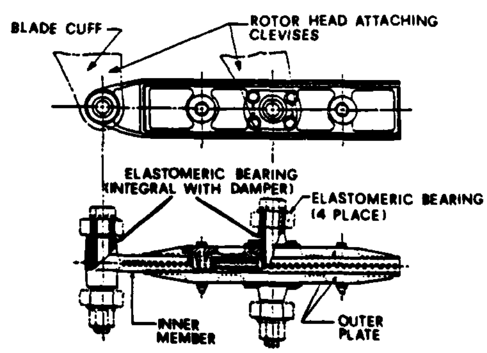


Fig. 51 - Cutway of elastomeric damper with elastomeric bearings at attachment points

A conventional articulated rotor head for a helicopter usually incorporates three sets of needle or ball bearing assemblies for each blade to provide freedom in three modes: cyçlic pitch, lead-lag, and flap. These three motions are now accomodated in one compact Lastoflex bearing by Lord Corporation.
As a resilient connection between rotor hub and blade, the spherical bearing provides for vertical and in-plane cocking to take flap and lead-lag motions.
Torsional shear flexibility accomodates cyclic pitch.

Elastomeric attachment elements make possible a reliable connection among engine-transmission and structural airframe members.

TRANSMISSION OF THRUST AND HUB MOMENTS TO THE HELICOPTER

As an example, the rotor system on the YAH-64, with fully articulated blades and elastomeric lead-lag dampers, is mounted on a static mast with a rotating drive shaft inside. The driving torque for the main rotor is transmitted through the drive shaft to the splines in the drive plate and through the attachments on the top of the hub. Two tapered angular contact roller bearings transmit the thrust and hub moments to the helicopter, figure 52. The hub has a dual function of supporting the main gear train to transmit the torque to the rotor and also to carry the rotor lift and moments into the airframe.

The rotor drive system of the Bell Model 222 helicopter is shown in figure 53. It consists of two engines, engine-to-transmission drive shafts, two overrunning clutches, a tail rotor drive of four equal-length shafts, and a tail rotor gearbox. The semirigid two-bladed main rotor hub, figure 54, uses elastomeric bearings for pitch change and for flapping. Pitch change is accomplished by twisting of the conical elastomeric bearings. Either of the two bearings can fail and the remaining bearing is sufficient to carry the centrifugal force of the main rotor blade. The main transmission combines power from the two engines to drive the main rotor, tail rotor, and hydraulic pumps, and is designed to accept full engine power from either engine with one engine inoperative. Such transmission is attached to the nodel beams by a four-bar linkage, figure 55. Elastomeric bearings in the links isolate the fuselage from noise and high-frequency vibrations. The nodal beam structure provides crew and passenger isolation from the rotor-induced vertical two-per-revolution vibration. Elastomeric mounts provide fore and aft lateral pylon restraint.
The Bell Model 222 has a bypass air filtration system for the engine installation, figure 56.

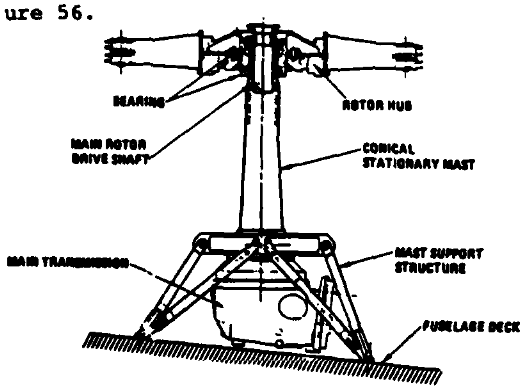


Fig. 52 - Static mast-rotor support

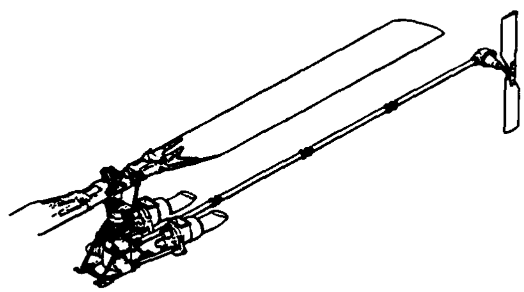


Fig. 53 - Drive system

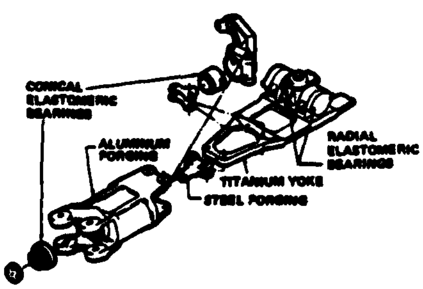


Fig. 54 - Main rotor hub

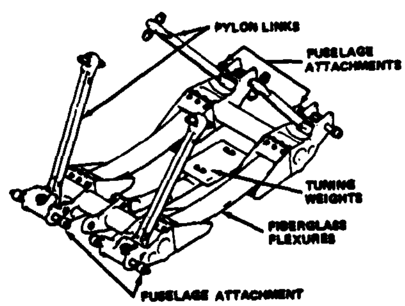


Fig. 55 - Nodal beam arrangement

The engine-structure attachment of the AS 350 light helicopter is very simple, and the engine contributes to the vibration filtering efficiency. In fact, the engine-main gearbox coupling shaft is fitted with flexible couplings. It is housed in a large diameter tube which connects the engine to the main gearbox and transfers the engine counter-torque to the latter, figure 57. The anti-vibration system is based on the use of laminated elastomeric pads which transfer the torque almost without any deflection, while allowing horizontal displacement on the main gearbox bottom, figures 58 and 59.

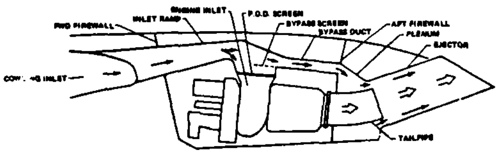


Fig. 56 - Engine air filtration system

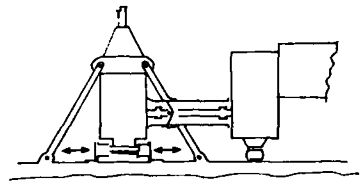


Fig. 57 - Engine/main gear box coupling

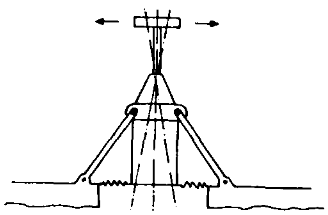


Fig. 58 - Mounting system: main gear box/rotor shaft and head

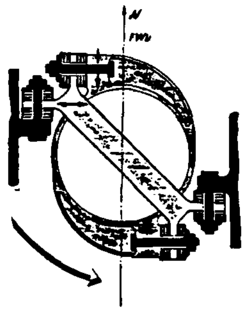


Fig. 59 - Directional mount "AS 350" type

The rotor systems research aircraft (RSRA) developed by Sikorsky Aircraft under NASA/U.S. Army sponsorship includes provisions for the installation of an active transmission isolation/rotor loads balance system over a wide rotor speed range and maneuver envelope without vibration restrictions.
The primary elements in the insolation system are hydropneumatic, servo controlled actuator units, figure 60. The unit, Ref. 32, is basically a hydraulic piston reacting against captured air chambers with a relatively low gain mechanical displacement feed back servo valve. The captured air bulk modulus provides a spring restoring force with piston displacement. Also when the piston displaces, the servo valve feeds hydraulic fluid into the piston chamber in the direction of motion, compressing the air and creating a restoring force on the piston 90 degrees out of phase with the piston displacement. The net result is that for static or transient loads on the isolator the displacement servo feature keeps the unit centered in mid-stroke, while for high frequency (N/rev) motion the unit acts as a air spring, as insufficient fluid flow through the servo occurs to create appreciable forces. The hardware associated with the active isolation system consists of the primary structural support of the main transmission and the interface hardware between the gearbox and the drive system and controls, figure 61.

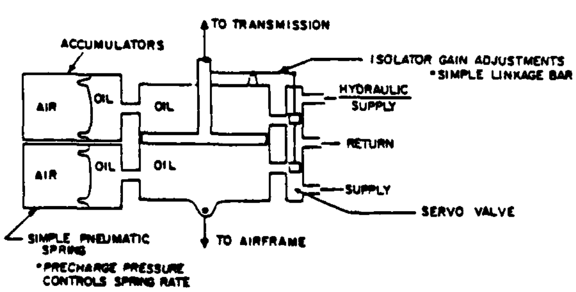


Fig. 60 - Schematic of hydropneumatic active isolator

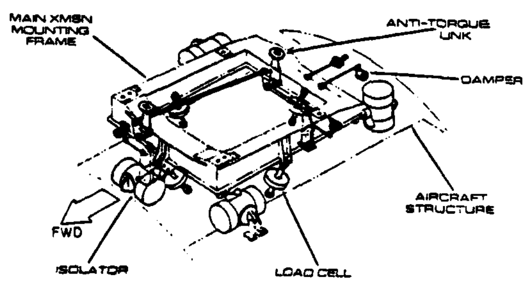


Fig. 61 - RSRA transmission active isolation rotor balance system

Rotor oscillatory aerodynamic and inertia loads cause fuselage vibrations and structural fatigue which must be eliminated or reduced.

Passive means of fuselage or cabin isolation can be realized using soft springs which include damping devices (elastomerics). But, in this application, low frequency motions are aggravated by the soft springs, and large static deflections between the rotor-system and the fuselage during transient maneuvering may occur, this being incompatible with helicopter control requirements.

However, pivoted spring-mass-systems, as using the so called kaman DAVI (dynamic anti-resonant vibration isolation) principle, enable sufficient isolation between the rotor transmission and the fuselage in the vertical and lateral translational, pitch, and roll degrees of freedom.

Further technical improvements in helicopter vibration control can be achieved by introducing active control technology, using electro-hydraulic actuators with feedback rather than passive springs. The basic principle is to control the rotor-induced oscillatory loads such that, independently from deflections between rotor and fuselage, a constant value is reached on the airframe. In this case, vibrations will not be induced on the airframe, whereas there will be vibrations on the rotor system. On the other hand, any deflection between rotor and airframe, caused by transient maneuvers, have to be compensated.

DFVLR, Ref. 33, has contributed to the solution of this problem by developing special controller logics based on optimal control theory, as in the dynamic compensator of figure 62. Here, the dynamic response of the incremental control actuator signal u and the fuselage heave deflection z for BO-105, 4/REV oscillatory rotor loads P_R are indicated in the first column. The rotor induced fuselage response z is attenuated within one rotor revolution (1 REV). For a 2.5 g transient maneuver (second column), initiated by a P_R - ramp input, the relative deflection between the rotor and the fuselage Δz is minimized within less than five rotor revolutions (5 REV).

As conclusion, it can be stated that low multi-frequency vibration levels may be achieved in the future with active control technology.

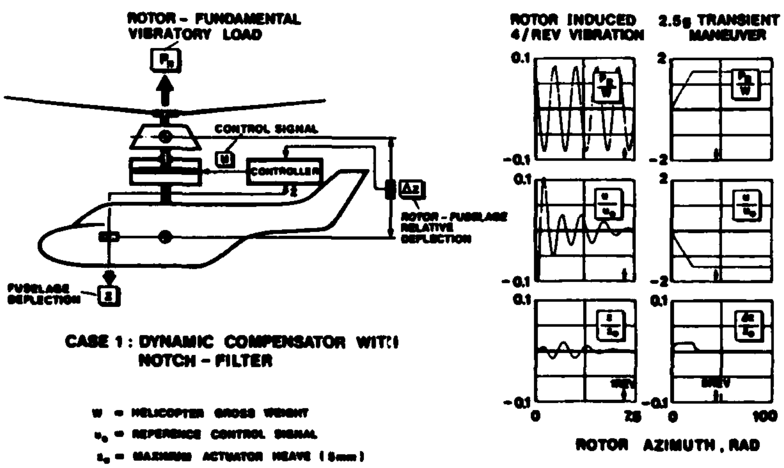


Fig. 62 - Vibration isolation by active control

ENGINE INSTALLATION ON THE AGUSTA A129

Aircraft design criteria

On the integrated battlefield of today and the future, the A129 is a modern weapon system conceived and developed by Agusta to satisfy the operative requirement of the Italian Army, primarely in the anti-tank role.

The A129, a twin-engine four-blade helicopter, figure 33, delivers unprecedent firepower quickly and with accuracy. The crew is seated in tandem with the aircraft commander/pilot seated aft and above the copilot/gunner.

To enable the A129 to fly and fight at night and during period of reduced visibility, a unique pilot night vision sensor and target acquisition and designation sight was developed and integrated to permit navigation and precision attacks under low visibility and night battlefield conditions. The A129 mission roles include anti-armor, covering force, flank security, economy of force and airmobile escort. The A129 provides;

- This chapter has been prepared in cooperation with Divisione Elicotteri Agusta. The help of Dr. Eng. F. Reina (Technical Director) and Dr. Eng. G. Virtuani being appreciated.

- performance that meets or exceeds Italian Army requirements;
- day, night and adverse weather, operations;
- multiple firepower options;
- twin-engine performance and reliability;
- improved combat survivability;
- adavanced crashworthy design features;
- high reliability, availability and maintainability;
- rapid rearm and refuel compatibility;
- self-deployability;
- air transportability;
- suitable weapon system for the attack mission;
- high degree of availability with minimum support equipment.

The need to develop an helicopter able to tactical flexibility an to survive and win on the modern battlefield induced the designers toward the use of new technologies and the continuous and iterative studies of physical and functional integration among the helicopter subsystems.
Several trade-offs have been made during the development phase, but always considering the following tasks: accomplish the mission; avoid detection; if detected, avoid being hit; if hit, continue the mission; if emergency landing is necessary, make it safely, Ref. 34.
The A129 is the result of the optimization of the responses to this requirement.

In spite of the growth, the A129 is relatively small when compared to other attack helicopters. It is about 25% lighter than the AH-1S and 50% lighter than the AH-64 in the primary mission configuration.

It is worth underlining that technology has aided significantly in weight reduction. In particular, sensible savings have been achieved in the airframe, main rotor and systems management.

Power plant

The engine installation of the A129 is a good example of a system integration, developed to satisfy these design requirements:

- minimum vulnerability;
- operativity in severe ambient conditions;
- supportability on the field.

Regarding vulnerability:

- low detectability is achieved by minimizing the IR signature from the engine bay and exhaust gases;
- low vulnerability to ballistic damage is achieved by: the choiche of a twin engine configuration; the definition of an emergency power level applicable to the engine and transmission in case of the other engine failure; a proper positioning of the two engines, set at a suitable distance each other and ballistically separated; a fire resistant engine bay; a redundant engine support system, able to withstand the loads arising in case of crash; the design of a ballistic and crash tolerant dual fuel tanks, connected to the engines by independent fuel lines, with the fuel pressure continuously monitored by the on board computer able to manage the fuel valves in case of ballistic damage.

Operativity in severe ambient is achieved by: integrating into the air intake design an inertial engine particle separator (FOD and sand); developping an engine air intake able to operate in icing conditions, Ref. 35.

Supportability on the field is achieved by: an easy accessibility to the engines for checks or removal; avoiding any need of alignment procedure or tooling when the engine is installed on the helicopter; the use of an integrated engine, where all the engine accessories, as, the oil cooling system, the electronic governor, the suction fuel system, are packed into the engine; a modular engine architecture, where the most of the engines modules are physically and functionally interchangeable; an engine health monitoring system installed into the helicopter, that continuously monitor the proper engine behaviour in flight, automatically manages the caution and warning messages in case of malfunction or limit exceedance, and record the engine usage for maintenance purpose; an instantaneous engine power check, by the crew, within the helicopter flight envelope.

The production engine for the A129, the Rolls-Royce Gen 2 MK 1004, is capable of providing the performance margins which will give the aircraft the required performance characteristics in N.O.E. flight. A large power reserve is specified to give excellent performance and high maneuvrability, combined with reliability and low cost of operation. The failure of one engine during take off, landing or hovering, is minimized by an emergency rating of 1,035 hp for up to 20 seconds.

The two turboshaft engines incorporate direct drive with an advanced digital electronic fuel control system. Essential priorities in design were safety, reliability and maintainability together with economy of operation. In order to meet these objectives and at the same time fulfil the technical demands of operating services, the two spool gas generator concept was chosen with a free power turbine to provide the power take off from a third shaft, figure 63. The two spool gas generator has low inertia rotating assemblies for fast response and operates at optimum speed for high overall pressure ratio to give low fuel consumption without the need for couplex variable geometry. Thus the design objectives are achieved with fewer compression stages, fewer parts and a simpler fuel control.
Improvements to basic Gem are introduced in the Gem 2 MK 1004 for Agusta A129, as in figure 64 and 65. Mounting flange and eye for installation, and squeeze-film bearings for reducing transmission of vibration from rotors to casing, are shown, respectively, in figures 66 and 67.

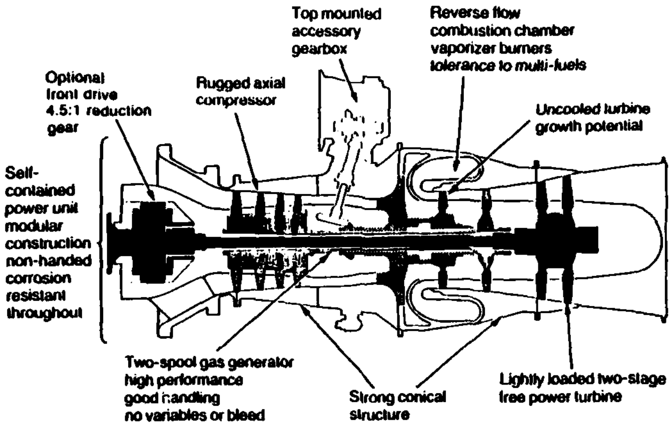


Fig. 63 - Gem 2 MK 1004 for A129 - basic design features

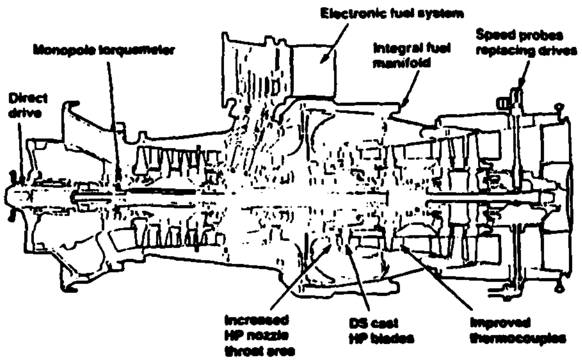


Fig. 64 - Gem - improvements to basic Gem

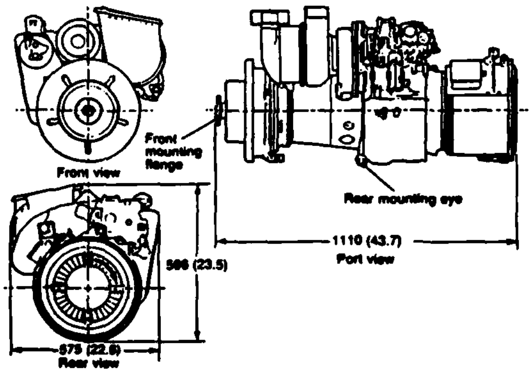


Fig. 66 - Engine installation diagram

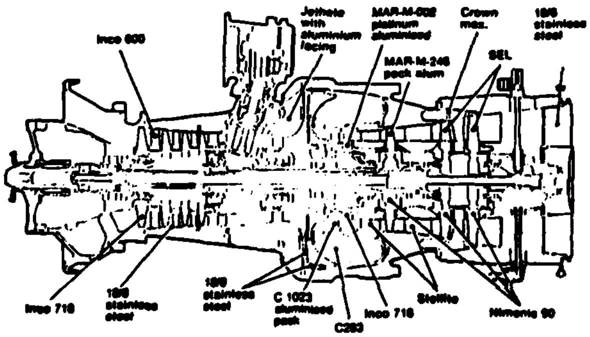


Fig. 65 - Corrosion protection

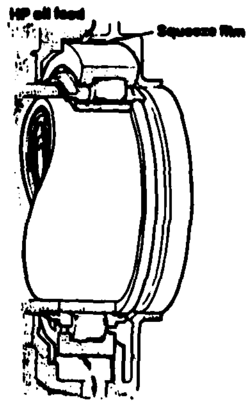


Fig. 67 - Squeeze-film bearings

Engine health monitoring system

A centralized computer supplied by Harris Corporation is monitoring the engine usage and condition, figure 68, together with a fault diagnosis capability.
The system will maintain the calculative counts of low cycle fatigue for all main components.

Ballistic tolerance of transmission and rotor systems

The transmission input comes from two separated independent drives; this being an advantage, from the point of ballistic protection. The transmission can operate 30 minutes after loss of lubrication. The lubrication reservoirs are internal to the housing.
The trade-off of providing transmission output ballistic protection affected the engines installation, forcing them to be spaced wider apart, which resulted in slightly widening the frontal view of the airframe. Even with the loss of two transmission attachments, the A129 will be capable of flying for 30 minutes.

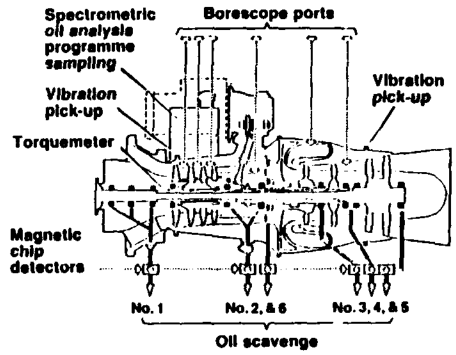


Fig. 68 - Engine health monitoring

The fully articulated four-bladed rotor system, with a single spherical elastomeric bearing (for each blade) which provides the hinge functions, is ballistically tolerant, easy to be maintained, and has a low radar return (flight controls inside the mast).
The ballistically tolerant tail rotor is two-bladed and provides directional control in wind gusts up to 45 knots from any direction.
The main and tail rotor blades are maded in composite material.

Flight control systems

The project hypotesis of the A129 helicopter have been: mission continuity after a first failure; safe landing after the second failure; chance of the ballistic damage; operating ambient temperature of 323 K.
These conditions have brought to helicopter configuration such as: pilot and copilot tandem seats; non-powered mechanical mode, used by pilot in emergency; "fly by wire" mode, used by copilot in emergency; tail rotor servo "fly by wire" in normal mode and non-powered in emergency mode.

From these assumptions, it is easy to note the flight control system's new aspects. At the same time it appears clear the inherent high technical risks.

The exposed pitch change links to the main rotor blades are ballistically tolerant. The pilot's primary control of the main rotor is a mechanical hydraulically boosted system.

The tail rotor servoactuator is intended to be normally operative in "fly by wire" mode using the normal body, with the capability to fly in a emergency case in "fly by wire" (backup body): it has also the possibility of the pure mechanical mode (no "fly by wire", or both hydraulics off), Ref. 36.
Pilot's inputs, figure 69, are given to the pedals assembly and read by three rotary variable differential transformer (RVDT), the electrical output of which is the input in the "fly by wire" loop.
At the same time, the pedal displacements are transmitted to the tail servo by means of a typical "pulley and cables" assembly.
The tail rotor servo is installed inside the right angle gearbox, coaxial with the tail pitch link.
A crash wortly/survivability feature of the pilot's controls is the attachment of the cyclic stick and the control rod to the bottom of the seat.
During a crash situation, if the pilot's seat is stroked downwards, the cyclic stick also descends, preventing injury in case of pilot body forward motion.

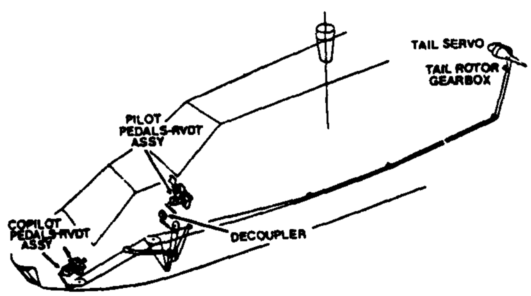


Fig. 69 - Tail rotor servoactuator

The copilot/gunner station is also equipped with a "fly by wire" system to control main and tail rotors and an override mechanical connection to couple pilot and copilot con-

trols. If the injured pilot falls forward on the cyclic control, the copilot can decouple the pilot's controls and operate the controls by his "fly by wire" system.
A stability and control augmentation system is installed in the pitch, roll, and yaw axes to reduce pilot work load and provide command augmentation to the autopilot. The system will damp any short period disturbances through attitude and rate feedbacks, and increases the pilot's control/agility at all times by adding a stabilizing factor in all axes, Ref. 37.

<u>Areas of significant progress</u>

The areas in which a significant progress has been made are (in parenthesis the weight gained vs. a conventional helicopter):

- rotor; single elastomeric bearing, composite ballistically tolerant, low noise, (15%);
- transmission; no external oil lines, 30 minutes dry running capabilities, modular engine direct drive input, (10%);
- flight controls; dual, triple redundant, "fly by wire", ballistically tolerant actuators and controls, protected, (20%);
- shafts-main rotor; ballistically tolerant, protect flight controls, (10%);
- shafts-tail rotor; ballistically tolerant, self centering and aligning, (10%);
- fuselage; use of composite materials (70% of netted area), (21%);
- A/C systems management; integrated multiplex system, (25%).

<u>Air intake system</u>

The intake design is based on the inertial separator scheme: the scavenge by-pass flow is designed to give high separation efficiency and is entrained by an exhaust ejector thus ensuring high reliability and low cost, Ref. 35. The design has been conducted with the main objective of a good intake/engine compatibility, but also looking for the best aircraft/intake integration by means of a careful aerodynamic design.
The integration with the airframe is affected to a great extent from the interaction between rotor and inlet and from the fuselage shape. So, configuration changes were applied during the development phase on the basis of the test results.
This evolution concerned mainly the internal design in order to improve the performance in terms of pressure loss and distortion or F.O.D., visualization tests showed a region of flow separation at the bend and suggested the fairing of the gimbal shround and an entry lip change. This action resulted in a lower Δp at about the same distortion level. These modifications naturally had an effect on the dust filtration characteristics and on the response in icing environment. A vortex pack can be accomodated to enhance its dust filtration efficiency, as it is required for desert operation.

An experimental program has been conducted both on the intake alone (full scale component testing) and on models of the complete aircraft, during the development ad qualification phases, Ref. 38.

A full scale glass fiber model was manufactured to measure the aerodynamic, dust filtration and icing characteristics.
This type of intake resulted rather insensitive to the affects of forward speed in presence of the fuselage boundary layer diverter. However, the aerodynamic interaction of the fuselage upstream of the intake in special flight attitudes was analyzed via small scale wind tunnel tests.
The pressure distortion and swirl angle were well below the limits stated in the engine manual. Also the circumferential variation in total pressure and flow angle did not exceed the values representing dangerous mechanical excitation for the compressor blades.

The same test rig was used for foreingobject separation tests. Debris was introduced into the inlet by three methods: using a chute; by leaving debris lying in the inlet before start up; by throwing items by hand.
Tests were conducted at a scavenge ratio of 50% and at an engine flow equivalent to a power of 680 shp. Different kinds of metal, rubber and other, objects were used; the intake was found to be virtually 100% efficient at separating foreign objects. Some modification to the initial geometry derived from tests conduced by leaving a representative of each debris type on the botton surface of the intake about 6 inches from the entry plane.
Two methods were used dust and sand separation tests: iso-kinetic sampling with cyclone separator and Coulter Counter Analysis; Knollenburg 230X occulted laser sampler analysis (for counting particles in the undisturbed airstream).
The iso-kinetic system removes a particle from the airstream for subsequent analysis by Coulter Counter.
Airflow rates in the engine and scavenge ducts were measured by turbine flow meters. Pitot static traverses were carried out in the planes selected for iso-kinetic sampling.
Several configurations were tested; including changes in the intake entry geometry and modifications to the duct shape in order to increase flow curvature without unacceptable increases in pressure loss or distortion. Besides geometry effects, other parameters were

varied to analyse the corresponding variation in filtration efficiency.
Scavenge effect showed that both filtration efficiency and life improvement factor (reduction in total erosivity for a particular inlet dust type) increase progressively with scavenge over the range tested. Variations in engine mass flow (+/-10%), at constant scavenge, demonstrate insignificant changes in separator performance.

For tests in icing conditions, a number of antiicing systems were considered, including: not air; electrical heating; electrical pulse; pneumatic (infiltration boot). Hot air was selected on the basis of simplicity and reliability. The system comprises a P3 driven ejector entraining ambient air which ensures that the composite intake cannot overheat if the systems is selected at high ambient temperature. The engine intake is also hot air de-iced but uses undiluted P3 air.
In the test arrangement, supercooled water droplets are created by the spraymast and are then carried in the airflow within the approach duct to the model intake. The approach duct changes in cross-sectional shape from circular at the spraymast to a shape that matches the model intake at the discharge plane. Hot air is supplied to the model heated lip and to the engine inlet casing from a facility supply via a flow measuring section and valve.
In order to achieve the equivalent corrected flows, the tests were ram at 5,000 ft. pressure altitude, at the icing condition appropriate to the test.
Icing tests to - 20°C were successful, with no ice forming on surfaces where shedding would lead to engine ingestion. Engine bleed air heating was found to be necessary on the splitter, drive shaft cover and at the engine interface.

Wind tunnel models, with the simulation of suction and discharge mass flows of the engine installation, have permitted the study of the interference on forward flight conditions, to complete the information obtained from the static tests.

The chosen intake configuration has been installed on the A129 prototype for flight testing, Ref. 39.

Safety capability of engine bay in the event of fire

The probability of firing to the engine bay is related to a ballistic damage and accidental causes related to the fuel system or the engine oil system or both.

The fuel system is of the sucking type. Then, a failure to the depressurized lines cannot result in a serious risk of firing. The risk is, however, present for the high pressurized area of the system, because of a partial rupture of a tubing allowing a leakage of fuel. A kerosene nebulization, with development of high local temperature, would take place.
The concentrate flame is considered as applicable to a limited number of areas, such as engine mount, engine fittings and bulkheads.

Less critical are the risks related to the failure of parts of the lubricating oil system for two aspect: combustion temperature higher (≈ 550°C) than the fuel; very limited amount of oil, corresponding to flame of 2 - 3 minutes.

To verify the safety of engine bay in the event of fire, the aerodynamic, engine (weight, torque), inertial and gyroscopic loads and factor "g", have to be simulated, applying then the concentrate flame to the various critical parts (engine mount, support bulkheads, joints, floor bulkheads, hinges).

The engine compartment structure, figure 70, in its whole globality is able to contain a spread flame which may develope in any aerodynamic flight condition, without permitting that the flame propagation may reach the other parts of the helicopter. The typical flame considered is the one consequent to a loss of fuel and/or oil inside the compartment, with the influence of inside and autside conditions, i.e.: compartment volume and ventilation; engine installation; drain lines; ventilation by external airflow; accessory air inlet.

A method for testing all the bay was established, considering two different conditions: concentrate and diffuse flame, the first as development of fire at 2,000°F (simulated via a Lennox burner) in a limited area of the installation, the second by a specific amount of fuel enriched with fresh air because of rupture or disconnection.
A full engine installation and the corresponding fuselage area, equivalent in geometry and material to the A129 helicpter, have been manufactured; a steel mock-up has been used in place of the engine. The typical load and vibration conditions of the engine instalation has been reproduced.
The engine ejector flow is generated by an external fan. A second fan is producing the external air flow, figure 71. Exhaust, by-pass intake flows and aerodynamic loads are simulated through other fans.
Local flame tests allow to verify, using Lennox burner, the resistance under operating load of some particular parts of surfaces (support, engine mounting, seals, fire walls, etc.).

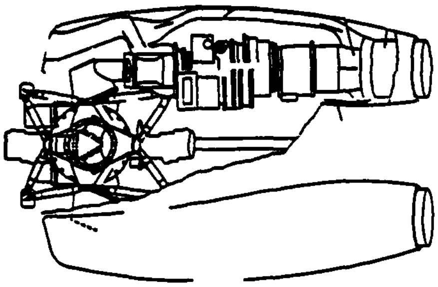


Fig. 70 - Engine compartment structure

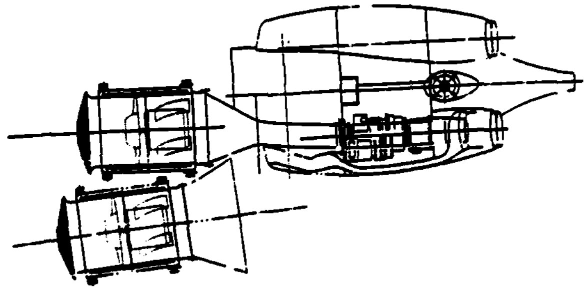


Fig. 71 - Fire testing of engine bay

Laboratory testing for reducing the operating risk

A complete evaluation of the flight control system has been made, investigating in details the aspects:

- pilot's capability to control the helicopter without using hydraulic power in carrying out a safe landing;
- capability to control the helicopter in "fly by wire" mode, within a normal flight mission and expecially during a long return flight;
- transient during the change from normal control mode into "fly by wire" or pure mechanical modes;
- behaviour of the hydraulic system at the maximum temperature limits;
- consequences of simulated failures.

For that, a metallic structure reproducing the helicopter geometry has been realized.
The same hydraulic system of the helicopter was installed, respecting the pipelines length. Sufficient information about the helicopter controllability is provided by a simulation hardware.
The operativity of the flight control rig is fully controlled by a computer, whose functions are:

- test feasibility control;
- acquisition of input data from the flight control operated by the pilot;
- computation of the helicopter attitude, speed and position, solving a dynamic simulation program;
- computation of the output parameters in respect to flight loads on controls.

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Impact of IPS and IRS Configurations on Engine Installation Design

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INTRODUCTION

Helicopters operate in a variety of roles many of which pose severe problems to the engine due to ingestion of sand, ice, salt and other foreign objects. It is possible to design a compromise engine intake which combats all these hazards to a limited degree. Such an intake is often required by military operators who have to maintain a capability to fly anywhere, at anytime. However, other operators do have specific, unchanging types of operation for which specialised intake designs are a better solution It is often the requirement of the first type of operation which causes an engine to be developed. Since the customer may at this stage be uncertain of the airframe in which the engine is to be installed, the engine manufacture is encouraged to develop an engine which will withstand the worst airframe and he will embody an integrated IPS on the engine although an airframe mounted IPS may be a better solution.

Other users with specialised operations are presented with an engine which goes some way, but not far enough, to meeting their particular requirements. Embodying a filter which does meet this requirement results in a non-optimum solution since the performance and weight penalty of the integrated IPS cannot be removed. On the other hand some operators may require much less protection than that given by an integrated IPS and again the solution is not optimum.

It is the thesis of this paper that designers and customers for helicopters/engines should look at the complete picture and accept that both airframe and engine mounted IPS have their place and that both helicopter and engine designs should maintain the ability to fit different types of intake in different environments. Figure 1 illustrates such a concept.

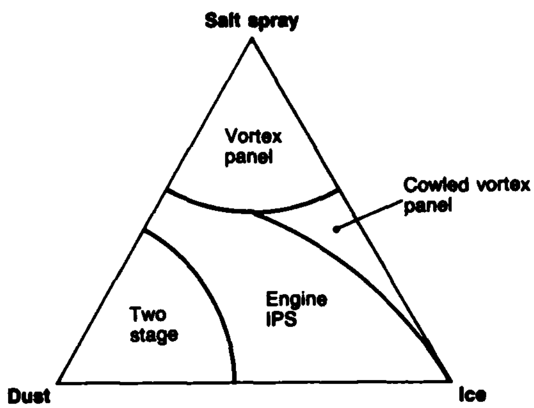


Fig.1. The effect of environment on optimum IPS concept

DESIGN METHODS

Intake Characteristics

Matching the protection required by an engine with that provided by engine and/or aircraft mounted separators is of course a first consideration. Most of the problems have already been satisfactorily addressed as the following tables shows:

	Parameter Definition	Definition of Engine Tolerance	Comments
Pressure Loss	Area/Mass Mean	Easily calculable Cycle Effect	Satisfactory.
Pressure Distortion	DC, Nc etc.	Predictions from Compressor Test	Satisfactory.
Icing	Visual Test Evidence + Analysis	No ice allowed on surfaces leading to engine or size limit	Satisfactory.
Salt Spray	Concentration	Calculable Power Loss from correlations	No interaction with engine design.
Foreign Objects	% No. Removed	Damage above Defined Size	Satisfactory
Sand/Dust	Erosivity and Concentration	Prediction of Power Loss From Material Properties, Engine Geometry, Impact Velocities.	Engine design interacts with IPS - Model Specs usually ignore these parameters.

TABLE 1. INTAKE DESIGN PARAMETERS

Accepting that this table fairly summarises the position, it can be concluded that the area of least clarity is the definition of sand/dust removal and its link with engine life.

In order to improve this link and allow a co-ordinated design approach, Rolls-Royce have adopted a design procedure for calculating from IPS performance data a single parameter known as Life Improvement Factor (LIF) and a further procedure for the calculation of the required LIF for a particular engine to achieve a desired erosion life.

IPS Life Improvement Factor (LIF)

The majority of recently published data on IPS dust separation performance expresses efficiency in terms of dust concentration reduction. A few authors use dust weight removal efficiency which is related to concentration reduction by the relationship.

$$\eta c = \eta m \cdot (1+S)$$

where $\eta m = 1 - \dfrac{\text{Dust Mass in Engine}}{\text{Dust Mass Fed}}$

$$\eta c = 1 - \frac{\text{Engine Dust Concentration}}{\text{Fed Dust Concentration}}$$

$$s = \frac{\text{Scavenge Air Flow}}{\text{Engine Air Flow}}$$

This would be an adequate definition if it were not for two important factors:-

Firstly published data from a series of engine erosion tests performed with dusts of different size have confirmed that life is a strong function of dust spectrum (Fig.2. Ref.1.). Secondly most momentum separators preferentially separate certain dust sizes. These two factors taken together make the expression of IPS performance in terms of concentration reduction misleading and the need for a parameter based on erositivity vital.

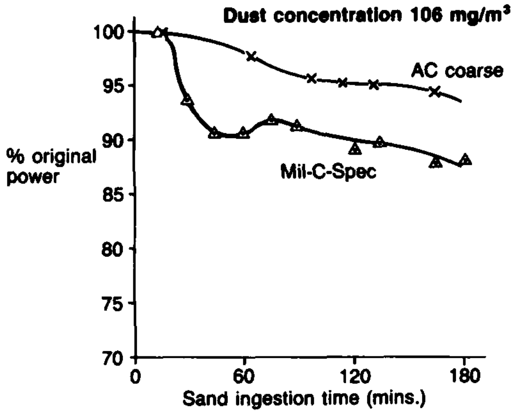


Fig.2. Effect of dust spectrum on engine life

The procedure adopted by R-R is as follows:-

1) Firstly it is necessary to establish the relationship between the particle size distribution entering and leaving an IPS. (spectrum shift). Any change in distribution results from the combined effects of:-

a) Momentum Separation
b) Particle Bounce
c) Particle Fragmentation
d) Aglomeration

In practice the first two effects dominate with the third (fragmentation) being a significant though secondary process.

Measurement of spectrum shift is best performed by a number of tests using narrow cuts of dust covering the size range of 15µm-1,000µm . Fig.3 shows some test results together with a trajectory prediction calculation including a fragmentation correction.

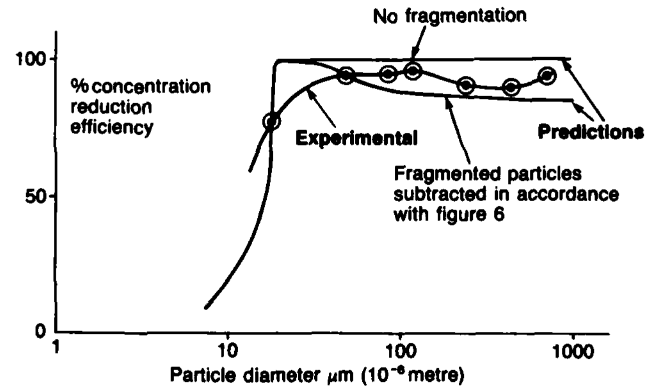


Fig.3. Effect of dust size and fragmentation on separation efficiency

2) The second stage is to integrate the efficiently Vs size with erosivity Vs size over the size range of the dust of interest.

eg Total Erosivity Entering IPS $= \int$ dust weight fed ϵ ds.

Life Improvement Factor is then:-

$$\text{LIF} = \frac{\text{Total Erosivity Entering IPS}}{\text{Total Erosivity Entering Compressor}} = \frac{\int \text{dust weight fed.} \epsilon \ \text{ds}}{\int \text{dust weight penetrated} \epsilon \ \text{ds}}$$

where

ϵ = dust erositivity

ds = size increment

This calculation must of course be performed with an erosivity Vs size characteristic that is relevant to the impact velocity of the critical component. Fig.4. (ref.2.) shows the variation in the erosivity of crushed quartz with size and impact velocity.
If the above procedure is used to calculate the LIF of a number of IPS geometries a characteristic of concentration based efficiency versus LIF can be constructed as shown in figure 5. This characteristic illustrates the large increase in LIF that has resulted from the dust spectrum change through this IPS.

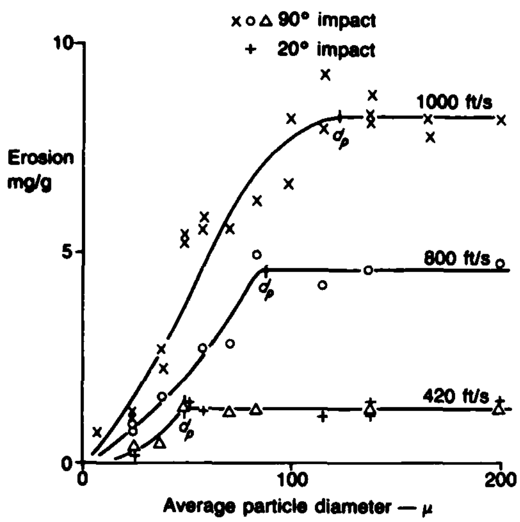


Fig.4. Influence of particle size and impact velocity on erosion (from ref.2.)

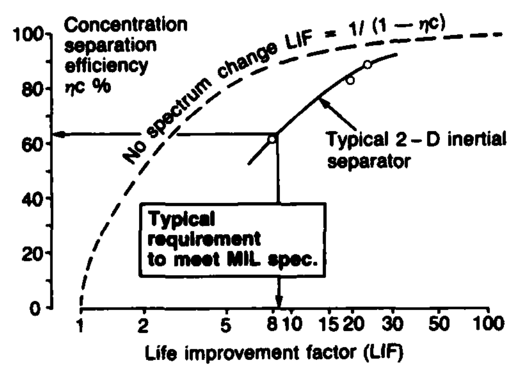


Fig.5. Effect of dust spectrum change through separator on efficiency vs LIF characteristic

Required LIF of Engine

Determination of the required LIF needs either data from controlled component erosion tests - (ideally tests with the dust of interest plus data on the effect of particle size on life) or predictions of these characteristics from trajectory and erosion predictions. These should be backed up by correlations from tests on engines of similar geometry. Fig.6 shows an example of data from a number of erosion tests of the kind which can be used as a base for scaling the predicted erosion life of projected engines.

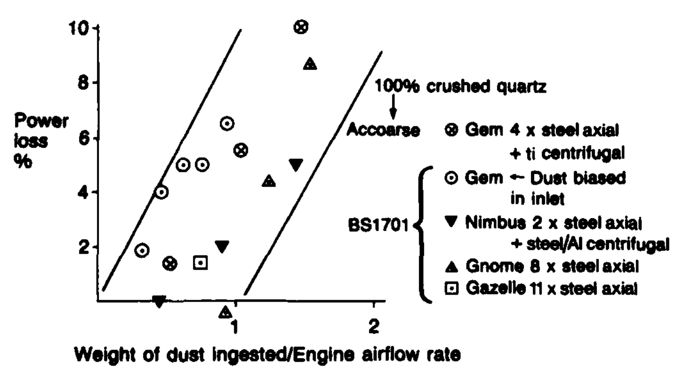


Fig.6. Rate of power loss for various engine types

Scaling

The following factors should be considered when scaling erosion data to a new engine.

1) Compressor Tip Speed
2) Blade Material
3) Blade Chord
4) Blade Leading Edge Shape and Thickness
5) Number of stages before centrifugal compressor
6) Material of Centrifugal Compressor

Most of the data required to perform this scaling are available in the literature. Data on the effect of chord and leading edge shape is not available however and specific tests might need to be performed.

Critical Component(s)

All of the helicopter engine types for which R-R has responsibility suffer the majority of their power loss through compressor erosion. This is true for designs with either steel or Titanium blading. However it is not impossible to conceive a geometry which resulted in the majority of the power loss being caused either by erosion of the back stages of a multi stage compressor or of components downstream of the combustion chamber. In such a case the effect of dust fragmentation through the first stages of compression will strongly reduce the importance of dust spectrum (Fig.7.), ref.2. For such geometries a design method based on dust concentration might suffice.

Experience at Rolls-Royce with several quite different helicopter engine types has shown that the reasons for eventual failure of the engine to accelerate or self sustain can be unconnected with those causing initial power loss. Since it is usually the initial loss of say 10% power that causes an engine to be rejected, it is important to differentiate between the various causes.

Based on the strip examination, selective repair and synthesis of eroded engines returned from service and of engines deliberately eroded to destruction, the following allocations can be made:-

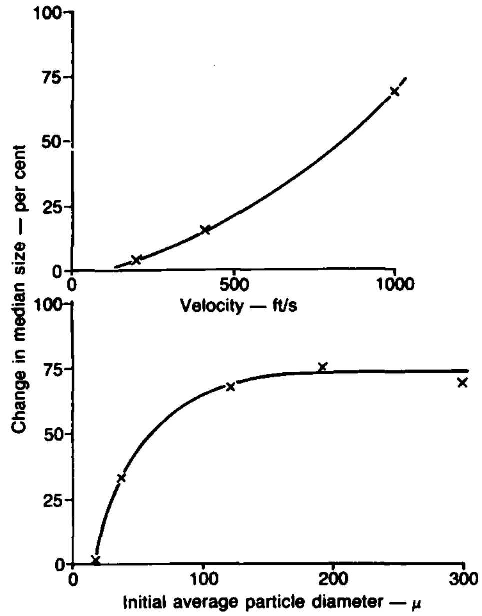


Fig.7. Effects of impact velocity and initial particle size on degree fragmentation of quartz(from ref.2.)

ERODED COMPONENT	RESULT
1. Compressor first rotating blade	a) Immediate power loss from leading edge burring. b) Progressive loss of surge margin from throat area increase caused by chord loss.
2. Compressor back stage tip erosion	Progressive loss of surge margin and power.
3. Impellor vane flank areas	Sudden large power loss when vane pierced.
4. Impellor tip backplate	Potential impellor failure from rim cracking. (This has never occurred before vane flank wear has made the engine unrunable.
5. Turbine nozzle vane	Increase in throat area causing progressive loss of power at a given power turbine inlet temperature.
6. Turbine shroud	Progressive power loss from tip clearance increase. Sudden large power loss if shrouds are pierced.

Further information can be found in Ref.3.

Overall Environmental Protection

Referring to Fig.1., the important factor is to ensure that the engine and IPS are not overdesigned and therefore over penalised for the role. The most important requirements for European theatre operations are:-

1) Foreign Object Separation > 98%.

2) Complete Invulnerability to icing, sleet and snow.

3) Erosion life of > 50 hours in MILC Spec dust at 50mg/m^3.

4) Erosion life of 25 hours in AC coarse dust at 50mg/m^3.

5) Salt spray concentration reduction of > 5:1.

Designs aimed at improving AC Coarse dust life above 25 hours or, for large engines, achieving a 5:1 reduction in salt concentration, are unlikely to be competitive with alternative configurations such as vortex tube panels.

Engine Position.

Recent trends in helicopter power train design have concentrated on front drive installations positioned in semi pods to reduce the chance of battle damage occurring simultaneously to both engines. A geometry of this kind results in a very three dimensional aircraft intake since the duct either has to contain the bevel box and shaft or has to swan neck around it. This choice is largely determined by the width of the fuselage.

In the case of a wide fuselage and semi buried engine the aircraft duct presents a strongly non symmetrical air and contaminant distribution to the engine face. The choice of an engine mounted axisymmetric separator in these circumstances is less than ideal and from a performance viewpoint a 2-dimensional separator of the type described in a later section would impose a lower pressure loss for the same separation performance.

Engines with rear drive have considerable attractions both for the efficiency of the engines turbomachinery and for the installation designer. A rear drive configuration having engines in pods either side of the rotor gearbox offers the prospect of complete separation of the engines environment from that of the airframe. The IPS in this case could be an axisymmetric engine mounted design. However rather than confine the entire IPS to the engine a better compromise is likely to result from a design using the engine bay walls to form a scavenge passage ducting the scavenge flow to an exhaust ejector. A design of this sort would benefit from the complete pod being the responsibility of the engine designer.

INTAKE TYPES

Having talked thus far of the advantages of integration it is time to consider some options and assess their performance.

Rolls Royce have since the early 1960's been engaged in the design of both engine and airframe mounted separators for both civil and military use world wide. This work has been supported by exploratory development and research aimed at offering the customer the widest possible range of choice for installing and protecting the engine from his particular operational hazard. The most recent and diverse examples are an aircraft mounted IPS project design (Fig.8.)and the engine mounted RTM322 IPS (Fig.9).

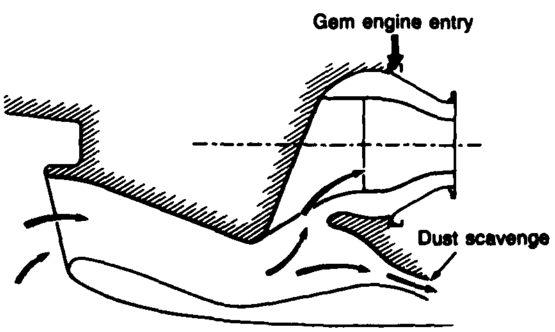


Fig.8. 2-D separator project study

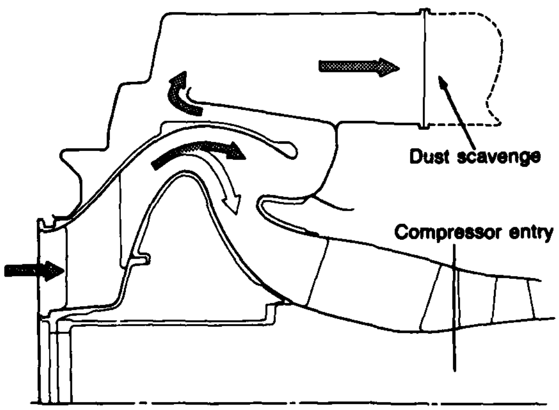


Fig.9. RTM322 IPS

Rolls-Royce research has concentrated on improving and extending the performance of three types of intake namely:

a) Vortex tube panels.

b) 2 dimensional scavenged bend.

c) Axisymmetric Separators.

Each of these has specific advantages over the others and together with appropriate engine core design offer a flexible approach if considered properly at the design stage.

Vortex Tube Panels

These were one of the first types of filter employed on helicopters and inevitably suffered from some early poor designs which have given them the reputation in some quarters for high drag and poor installed performance. In fact - if properly used they can be about the most effective way of reducing salt spray fouling and can offer a very convenient means for zero staging an existing IPS in order to raise its performance for severe desert conditions.

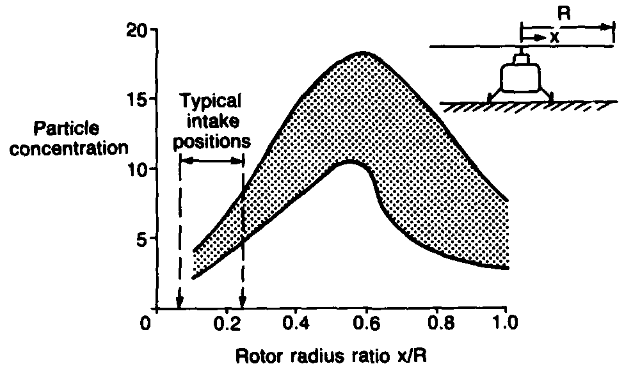
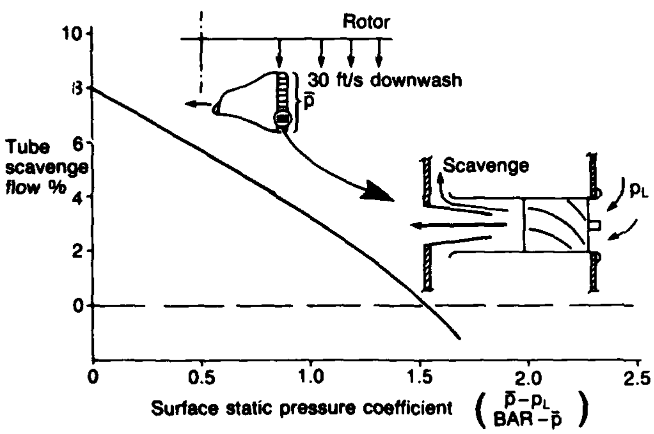


Fig.10. Effect of rotor radius ratio on rotor induced dust concentration (from ref.3.)



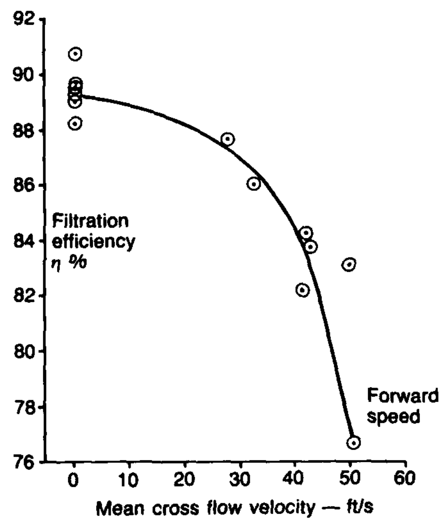


Fig.11. Effect of vortex tube panel surface pressure variations on local tube scavenge flow

Fig.12. Effect of cross flow on vortex tube panel efficiency

The main points are:-

1) Ensure that the position of the panels does not raise the entry contaminant concentration significantly. - Piasicki data (Fig.10. Ref.4.) shows how rapidly sand concentration increases with rotor radius. Getting this wrong can easily more than halve the potential engine life improvement of a vortex panel.

2) Ensure that rotor down wash/forward speed does not induce scavenge reversals. Fig.11 shows how little panel surface pressure variation is required to reverse scavenge flow directions. Fig.12. shows the effect of crossflow on panel performance.

3) Ensure minimal scavenge chamber static pressure variation and thus even scavenging.

4) Ensure that the majority of tubes face down or sideways to reduce the ingestion of large particulate.

5) Do not install panels downstream from potential oil vents.

Observing these rules will prevent the worst pitfalls but even better performance can be achieved if the panel can be cowled as shown in Fig.13. This arrangement has been tested experimentally by Rolls-Royce to investigate the feasibility of salt spray and icing protection for Naval helicopters. The results were encouraging and Fig.14 shows that with an exhaust induced scavenge flow of 35% indefinite flight in icing can be achieved without intake anti icing. The reason for this is that the exhaust induced cross flow ensures the vast majority of icing droplets accrete on the leeward edge of the vortex tubes forming small ice cowls which periodically shed to the exhaust scavenge zone. Salt spray separation is also good since the first surface struck by droplets leads to the scavenge chamber.

Dust separation performance of the cowled design was found to be equal to that of an uncowled panel with the prospect of no installation penalty from rotor or forward speed effects.

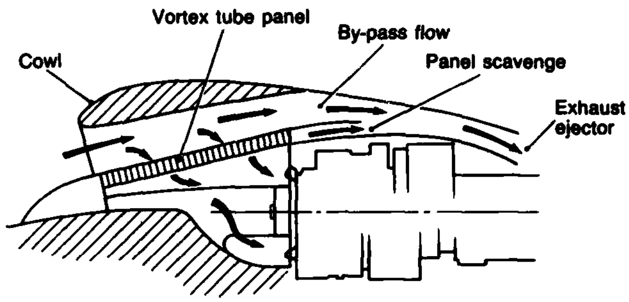


Fig.13. Cowled vortex tube panel

2D Scavenged Bend

One of the first protected helicopter engine installations that Rolls-Royce was involved with was a 2-dimensional scavenged bend employed as an aircraft mounted momentum separator for the Wessex (S58). Our experience with that installation spurred us on to conduct a series of exploratory tests on a range of separators suitable for buried engine installations.

Figure 15 illustrates some of the options investigated. In the main it was found that pressure loss rather than distortion governed the separation performance achievable. Figure 16 shows the trends that were obtained. In general it was found that a life improvement in dust of about 8:1 on ACC and over 30:1 on MILC Spec could be achieved for a 7lb/s model for a total power penalty of 2½% including scavenge system.

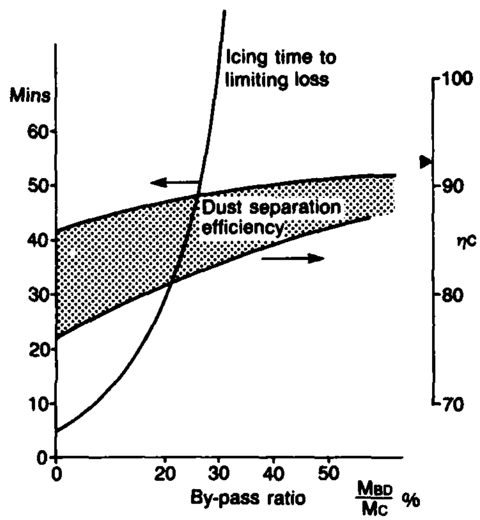


Fig.14. Performance of cowled vortex tube panel

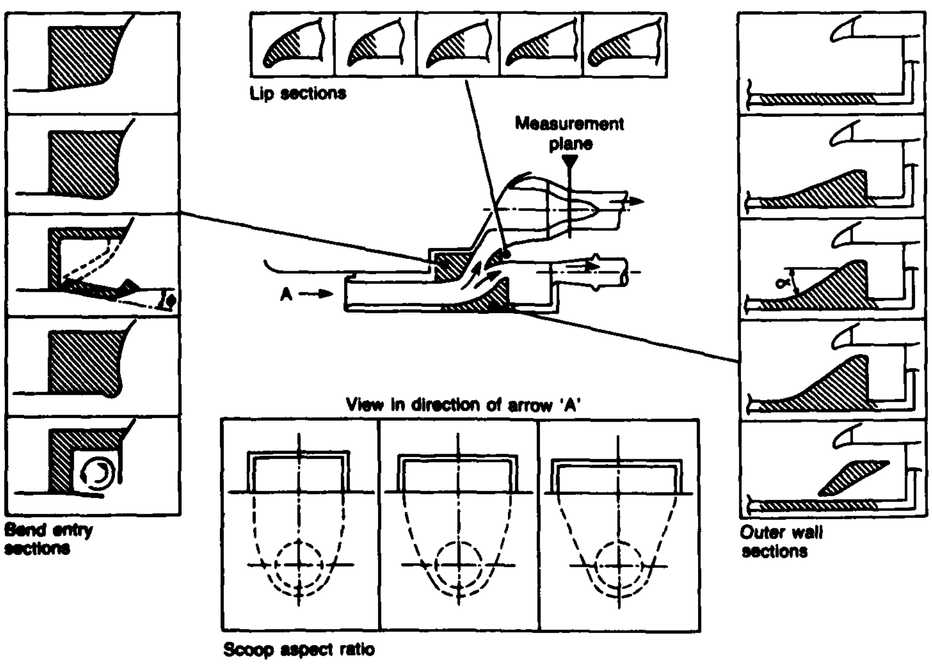


Fig.15. 2-D scavenged bend test variables

Designs with fixed geometries were found to be the best overall and a typical particle size against efficiency characteristic is shown in figure 17. It is interesting to note that no drop in efficiency was measured for particle sizes above 100 – a common feature of most of the geometries tested. The reason for this excellent characteristic is likely to be the high scavenge fraction (50%) which reduces the percentage of particles ingested from rebounds coupled with the absence of high velocity impacts on the wall leading to the engine. Fig.7. showed that a significant proportion of impacted dust will fracture at velocities above 150ft/s. If this is allowed to occur on the engine side of the duct the proportion that can be momentum separated is severely reduced. This phenomenon is likely to have accounted for the disappointing performance of the moving flap designs.

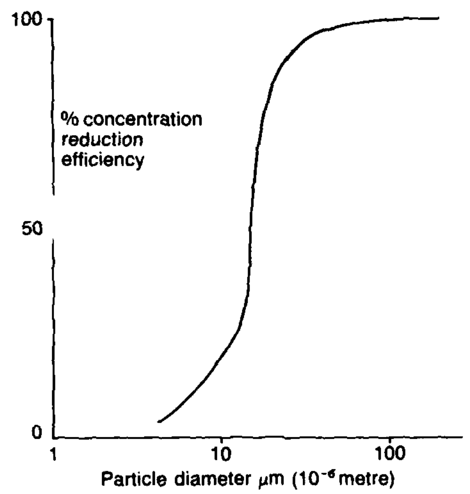


Fig.17. 2-D separator characteristics

Tests using the range of foreign objects shown in figure 18 produced separation efficiencies of 98% for 1,300 objects fed. These objects were either thrown in or left lying in the inlet during a simulated engine start.

Icing tests were performed on an electrically heated model and showed that a power of 5Kw would be necessary to completely anti ice a 6lb/s model. However, shedding tests later showed that only the duct area outlined on figure 19 need be anti iced which reduced the anti icing load to 3.8Kw.

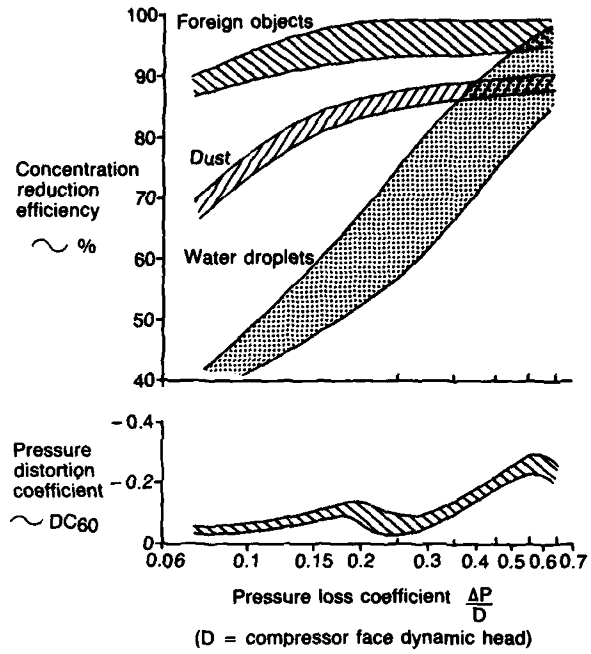
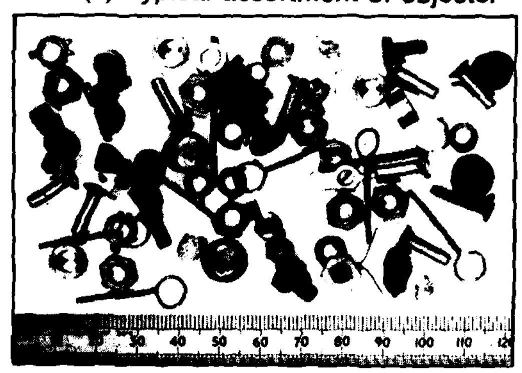


Fig.16. 2-D separator performance trends

(a) Typical assortment of objects.



(b) Table showing numbers of objects used.

Item	Description	No.	Item	Description	No.
	Plain nuts	30		Spacer rings (A1)	20
	Lock nuts	15		Pipe blanks	15
	Bolts & screws	30		Rivets (A1)	30
	Assorted washers	30		Stones	20
	Lock wire	20		Plastic tube	10
	Wire inserts	20		Expanded polystyrene	20
	Split pins	10	Total objects per test		270

Fig.18. Tested range of foreign objects

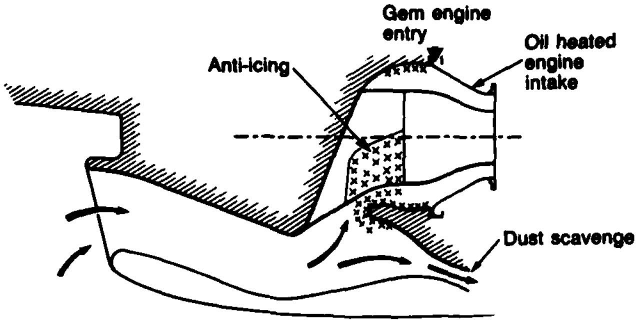


Fig.19. Area requiring anti-icing on 2-D separator

Axisymmetric Separators

Many helicopter designers now use buried engines. In these installations, the intake duct is usually curved and is amenable to incorporation into a 2-D aircraft mounted separator. The intake is axisymmetric at engine entry and it is practical at that plane to use an engine mounted axisymmetric separator. This does enable the engine manufacturer to develop an IPS which can be fitted to several airframes. Most engine manufacturers have developed variants and Rolls-Royce are no exception. Experiments have been done with types employing swirl as well as those using meridianal curvature to separate the debris.

Designs employing swirl have enjoyed some popularity - probably because they offer a useful improvement in small particle separation efficiency. However if larger particles and foreign objects are a major issue then Rolls-Royce experience is that swirl vanes are a disadvantage because of their tendency to block with foliage and to bounce some particles directly into the engine. There is also evidence to show that swirl vanes fragment some particles to sizes that cannot be easily separated.

With these disadvantages it is not surprising that many designers are now turning their attention to swirless concepts. Rolls-Royce have employed such a design on the RTM322 (fig.9.) which employs a combination of ballistic focusing and aerodynamic separation to achieve the performance given below.

Foreign Object Separation (see fig.18 for debris)	>98%*
MILC Spec. Dust Erosivity Reduction	22:1
AC coarse Dust Erosivity Reduction	8:1
Pressure Loss	1.4%
Pressure Distortion	Reduces Entry Distortion
Icing	Exceeds Current Regulations

*100% separation required for birds and other large objects not shown in figure 18.

Whichever concept is chosen it is important that a reliable scavenge system is employed. By dispensing with high pressure loss features such as swirl vanes, an ejector can be employed rather than a vulnerable rotating pump which will erode very quickly. Integrating the ejector exhaust with the engine exhaust offers a system with exceptionally low power consumption.

Two Stage Separators

An aircraft mounted 2-D momentum separator is likely to prove satisfactory in most temperate regions and world wide in average civil use. However military operations in either high dust or salt spray environments are certain to require enhanced separation performance. In the case of a high salt spray environment the best option would be to employ a vortex tube panel filter in place of the momentum separator. However enhanced dust separation performance can be achieved by the addition of a vortex tube panel as shown in figure 20.

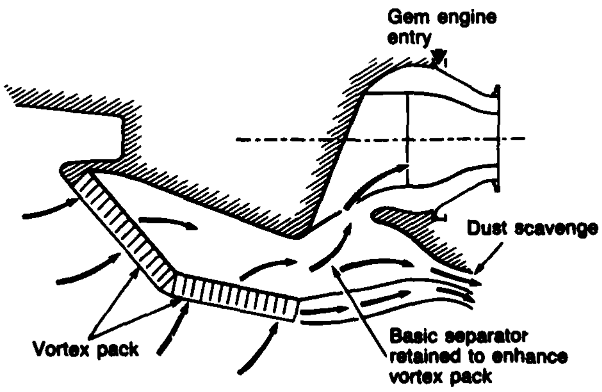


Fig.20. Enhanced 2-D separator for extreme dust environment

In this configuration the momentum separator scavenge flow would be reduced from 50% to 15% and a bleed air ejector employed to scavenge the vortex tubes. Overall performance can be estimated by combining the particle size Vs efficiency characteristics of the two filters and a figure of 96% can be predicted giving an erosivity reduction of over 20:1 on ACC dust.

Power loss for such an arrangement is less than might be thought because the entry loss of the momentum separator is eliminated. The performance of a project system was as shown below:-

	Power Loss		
	2 stage	2-D Bend	Un-Protected Intake
Loss to engine entry	3%	1½%	1%
Exhaust Ejector	1½%	1½%	1%
Bleed Air Ejector	1½%		
Total	6%	3%	2%
Erosion Life Improvement (AC coarse Dust)	20	8	1

Assuming a life improvement factor of 20:1 and an average dust concentration of 50mg/m^3 the power Vs running hours plot would be as shown in figure 21. As can be seen a helicopter with this intake would have a superior performance to an unprotected aircraft after 50 hours and to a single stage separator after 350 hours.

Performance at speed could be enhanced by employing a flap in the vortex tube panel to prevent outflow from the side panel. Figure 22 shows the predicted effect of such a device.

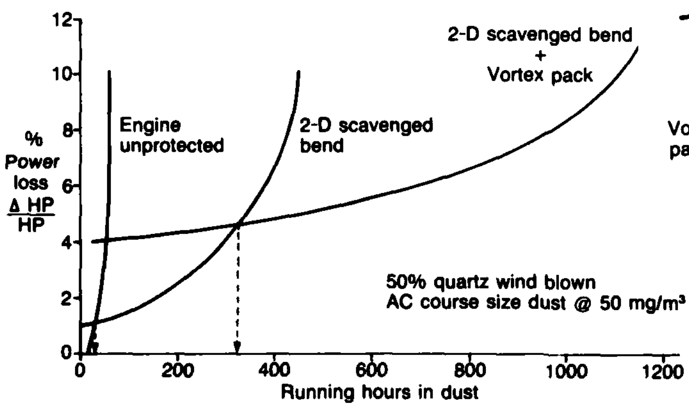


Fig.21. Effect of intake on power loss in dust

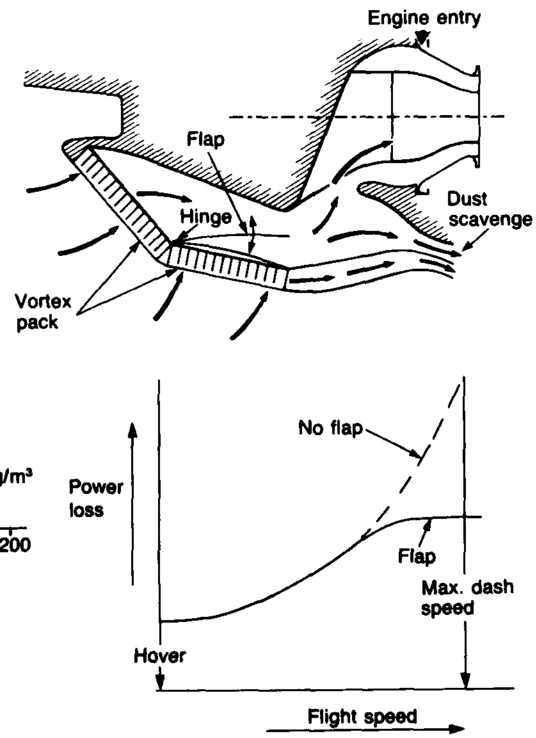


Fig.22. Effect of flap on vortex panel power loss

Entry Effects

The internal geometry of separators can be strongly dependant on the entry flow field which can cause large variations in the trajectories of particles at entry with changes in downwash. Ref.5 describes the considerable work required to simulate downwash for a separator having the separation bend close to the inlet. Despite the potential high efficiencies achievable by utilising the entry flow field, it is on balance probably best to isolate the separation process from external effects which could be a strong function of wind direction and changes in fuselage geometry.

INFRA-RED SUPPRESSION

A significant infra-red homing threat to helicopters has been present for some 15 years with the advent of light mass portable launchers - notably the Russian SA7. Missiles of this era employed an uncooled lead sulphide detector operating in the 1.5-3μm band and used a rotating sectored disc to modulate the signal and generate an aiming error signal. Figure 23 shows that the wave band of this type received most of its signal from high temperature parts of the helicopter - principally the exhaust pipe.

Suppressors were soon devised to minimise this threat to prevent line of sight onto hot parts from the ground or screens cooled by air induced by the forward speed of the aircraft. These early devices were later replaced by ejector cooled exhaust screens enabling protection to be maintained in hover. As the sensitivity of the reticle systems were increased, jammers were introduced which introduced additional modulation into the missiles aiming system thus causing aiming error.

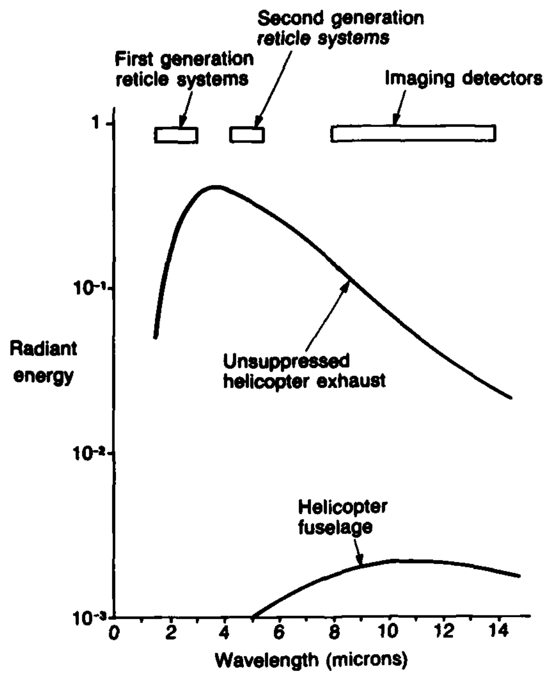


Fig.23. Spectral radiant energy of a black body at various temperatures

Integration of Suppressor and Airframe

Future IR threats will tend to be imaging systems not amenable to jamming. They will undoubtedly be sensitive to small temperature variations over the fuselage even though some of the sensitivity will be lost by the ambiguities of cluttered backgrounds. The figure 24 thermal image of a helicopter shows up the engine bay, exhaust and aircraft gearbox as the main contributors. You will also notice that hot areas extend from both oil cooler and engine exhausts. These effects are not unique to the helicopter but are nevertheless strongly influenced by airframe geometry as can be seen from figure 25. This figure shows thermal images of a variety of quite different engine/airframe geometries which are also shown in silhouette.

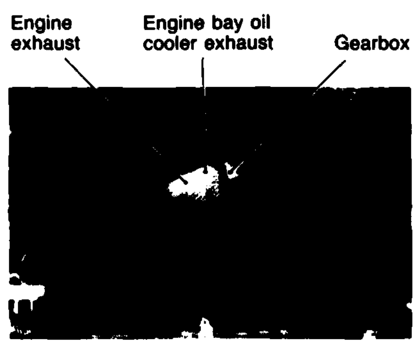


Fig.24. Thermal image of a helicopter

The points to note are:-

1) The very conspicuous tailbooms on geometries with overboom exhausts.

2) Very conspicuous bay cowlings.

3) A very sharp fall in bay cowling temperature where IPS scavenge air scrubs the cowl surface.

These points underline the importance of not only engine and aircraft air system design but also engine position in achieving a low thermal signature.

The last point on engine position brings to mind a Boeing Vertol paper presented at the 31st Forum of the American Helicopter Society (Ref.6.). This paper favoured from a number of viewpoints - not especially IR, a vertically mounted front drive engine exhausting rearwards and upwards close to the ground. This arrangement (shown simplified in figure 26) now appears to have the added attraction of avoiding extensive fuselage heating.

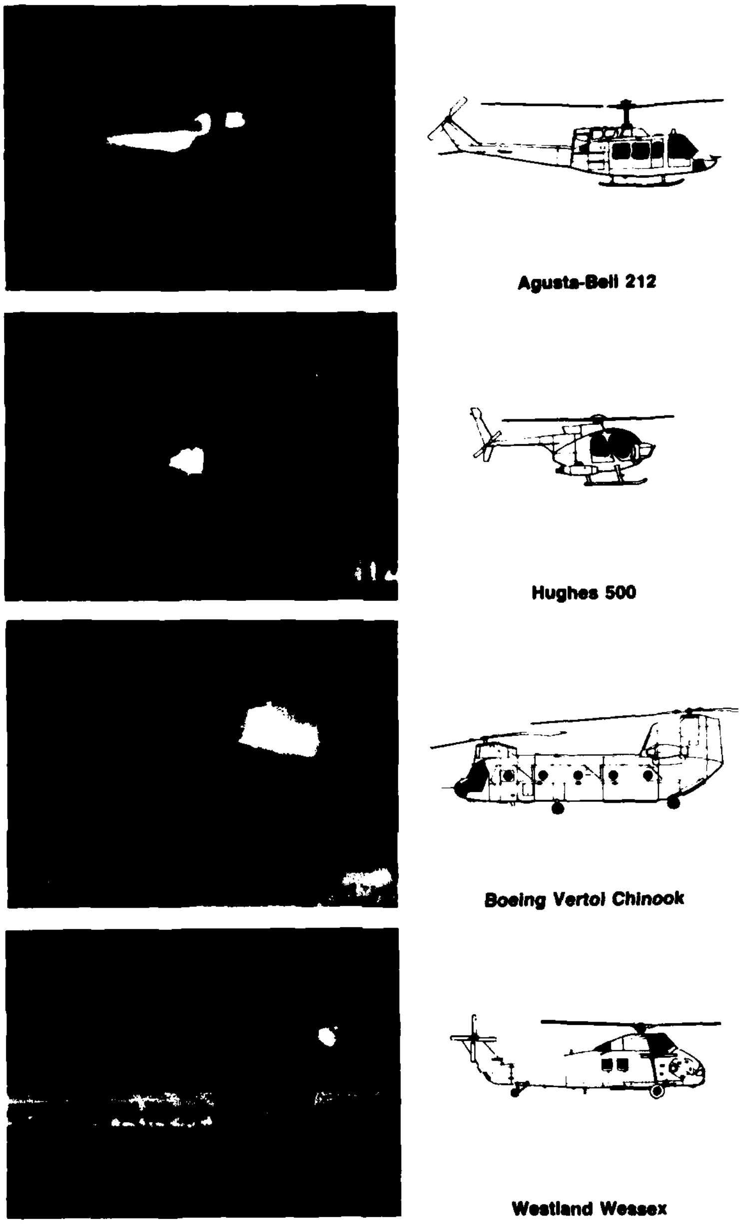


Fig.25. The effect of engine position on thermal image

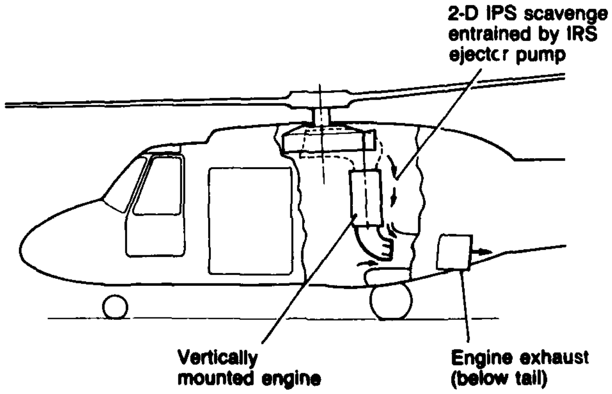


Fig.26. Vertically mounted front drive engine avoids exhaust tail boom heating
(from ref.6.)

CONCLUSIONS

1) Engine design should be coordinated with both engine and airframe mounted separator performance characteristics to ensure that optimised designs can be offered to customers with very different operational requirements.

2) A design method has been outlined which would allow the erosion reduction characteristics of engine and airframe IPS designs to be taken into account during engine design.

3) IPS design optimised for European operations can be efficiently adapted for extreme climate conditions.

4) IRS designs can be integrated with IPS scavenge requirements to produced a low signature engine installation.

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ENGINE-AIRFRAME INTEGRATION FOR ROTORCRAFT: COMPONENT SPECIFICATIONS AND QUALIFICATION

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SUMMARY

This paper is intended to provide both past history and current principles associated with engine-airframe integration for rotorcraft as related to component specification requirements and their individual qualification assurance. Component specification and qualification requirements have tended to be generic and standardized without consideration of actual usage, location, exposure, etc. As such, overall requirements have had problems of being overly restrictive in some cases and not exacting enough in others. The current trend in tailoring for specific applications as well as insuring system performance requirements by conducting early component/bench level tests will be discussed along with lessons learned from past efforts. Components involved in the overall engine-airframe integration effort such as pumps, fans, starters, valves, sensors, IR suppressors, inlet particle separators, fuel cells, etc., will be addressed.

INTRODUCTION

Rotorcraft and its associated engine-airframe integration specification process was initially derived from the fixed wing base. As rotorcraft specifics have been addressed, the specification evolutionary process has also been occurring simultaneously. For the discussion at the component level, it is also necessary to understand how the system level requirements have likewise led to specific application and operational requirements at a component level.

Although a definition of what is meant by a component could be provided, the individual case exceptions would end up addressing individual parts, assemblies, or subsystems. This would be necessary to either be able to address basic requirements or methods of verification. As such, a variety of component levels will be covered as they are individual pieces or portions of the engine-airframe system. Specifications and qualification at the component level also follow the "as necessary" approach. For these reasons, examples will be used to best illustrate the myriad of considerations that must be addressed in formulating component specifications and qualification requirements.

SPECIFICATIONS

The process in which a specification is evolved leads first from the basic requirements for the rotorcraft. The specification then is developed as a document to support acquisition, describe the essential technical requirements for purchased materiel and includes procedures for insuring the specification requirements are verified as being met. A similar document that may also evolve or be used is that of a standard. A standard will establish engineering and technical requirements for processes, procedures and practices. A standard may also establish requirements for selection, application and design criteria for materiel. Ultimately, component specifications and standards will lead and/or be part of the component techncial data package.

The types of specifications vary according to both the application and usage. The need can be based on standardization where a specific standard component is needed to meet interface requirements or match standard parts in the inventory. A performance type specification, on the other hand, would only define the essential overall requirements a component must meet. Another type of document utilized extensively in the engine-airframe integration design process is that of an Interface Control Document. Due to the multiplicity of detail requirements and interfaces of the engine and its association with the airframe, the document insures both the engine manufacturer and the airframe manufacturer have a clear definition of each primes' responsibility and defined design requirements.

The specifications then provide several specific needs which can address all aspects of rotorcraft as well as specific engine-airframe integration. These cover aspects, such as, standardization, performance, environmental, usage, reliability, interfaces, compatibility, design requirements, maintainability, human factors, quality and qualification. The manner and method this complexity of criteria is finally defined into a specification has never reached perfection and continues to be studied, evaluated, and changed today.

SPECIFICATION EVALUATION

The Signal Corps, United States Army specification in 1908 for the first heavier-than-air flying machine from Wilbur and Orville Wright was one page. It met the criteria of today's terminology of a performance specification as it defined basic end item performance requirements: Must carry two persons, range 125 miles, and airspeed of at least 40 miles per hour. These requirements had evolved from both what a review of technology at that time was expected to be able to achieve and the mission requirement for the Signal Corps.

This process along with the desire to promote standardization, reduce development risk, maximize competition, and ensure compliance, has been the basic method employed since the Wright Brothers' time. However, current public perception of the specification process today would say: We overspecify, the specification process lacks discipline, specifications are too restrictive, specifications are driving up costs, specification tiering is out of control and many specifications are outdated. Although these statements were taken from public newspapers and periodicals, those individuals directly associated in the specification process could think of examples of each of these "perceptions".

One clear example of the current problem is reflected in the availability of U.S. Department of Defense specifications and standards. The current listing has over 78,000 documents in which 46,730 are defined as active. In citing one standard on human factors could result in referencing 7582 other documents through the fourth tier. A method being utilized today to keep this proliferation of document compliance from overwhelming the design process is through "tailoring". By this process any reference to an existing specification must be tailored to the requirements for the specific application.

With initiatives to improve the basic specification design requirements process, another initiative has been put into place to insure the test and evaluation process also is considered. A Continuous Evaluation (CE) concept has been developed to challenge the requirements to insure those specifically related to the end item needs are utilized.

ENGINE-AIRFRAME SPECIFICATIONS

With a background on generic specifications the need to address directly related engine-airframe components is a little easier. At this time, only those component specifications utilized for an initial Request for Proposal (RFP) will be addressed. A part of the RFP may require that the prime contractor develop a specification as part of the contract scope to be used as an end item procurement specification. This same process may have been utilized before during a Research and Development (R&D) effort. This R&D effort may be used as a judge of what the current state-of-the-art can meet and at the same time develop specification criteria. Additionally, component specifications may have been developed as standards due to the repeated need for the same type of component. The following address examples of each of these and the success and problems with each.

During the 1960's, survivable post-crash fires were identified as a major safety hazard for rotorcraft. The overall fuel system design was reviewed with investigation into component design of such items as the fuel cells, self-sealing hoses, and breakaway valves. The fuel cell received the most attention as full scale crash tests were needed to verify the many materials and designs. A bladder type construction method was found to provide the necessary capability and a fuel cell general component specification was developed based on these demonstrated capabilities. Although, the success of this specification has not been questioned based on the success of preventing survivable post-crash fires, the substantiation criteria has still prevented other concepts to be addressed that do not meet the criteria that was based on the single concept of a semi-rigid bladder. As such, an R&D process to achieve the end performance was successful, but the specification remains restrictive to those technologies that were available at that time.

In the simpler method of providing standardization of a component, a specification was written to provide a standard series of fire bottles to be utilized in an aircraft fire extinguishing system. Once again the specification has been satisfactory, but specific rotorcraft designs have required multiple deviations to the standard to be more efficient in the specific application. These have caused some non-standardization in given cases, but have met conditions of being overall cost efficient. Examples of changes have been the exact size of bottle, the total quantity of extinguishing agent for a given size bottle, the number of squibs per bottle and the pressure gauge location.

ENGINE-AIRFRAME COMPONENT DESIGN

The vast majority of detailed component specification requirements are developed as part of the overall design/development process. Engine-airframe characteristics require attention to basic overall rotorcraft environment and usage as well as induced environment and usage. With the dynamics of engine-drive system-rotor system in a rotorcraft, the importance and criticality of this process is essential to providing basic rotorcraft operation and safety. U.S. Army statistics show that the top causes of all rotorcraft mission aborts are caused by those three elements in the rotorcraft propulsive system: engine, drive system, and rotor system. Elements key in insuring the overall engine-airframe integration design are addressed in the following paragraphs.

With the basic safety related aspect associated with the overall engine-airframe integration, several series of the overall design process address this issue. A Failure Modes, Effects, Criticality Analysis (FMECA) is performed which looks at the detailed component design to address what type of failure modes could occur, what the effect of each failure mode is, and if that effect is critical to flight operation and safety. This type of indepth analysis of components could lead to redesigns, redundancy considerations, and/or requirement to add detection/diagnostic means for the pilot/maintenance crew. A broader scale effort is determined under a System Hazard Analysis which also may cause the same type of impact on the component design/development as the FMECA. This "what if" review provides for detailed analysis of the overall engine-airframe system without having to address the issue after a part has failed.

The criticality nature of the engine-airframe integration is also deeply involved in the survivability/vulnerability assessments of military rotorcraft. Component design may be directly influenced due to crashworthy, ballistic damage, signature suppression, (noise, infra-red, smoke emission), and electromagnetic threat requirements.

Environmental considerations represent one of the most diverse areas of consideration in engine-airframe integration. Temperature extremes have to consider cold ambient extremes to well beyond hot ambient extremes in many areas due to the additional heat generated by operation. With engine/APU, fans, cooling airflow requirements, components are exposed to all natural and induced forms of sand and dust, water, salt fog, icing, humidity and assorted materials due to ground proximity operation. These effect both external and internal airflow components as well as adds to contamination of fluids (fuel, oil, hydraulic).

Rotorcraft operation and the dynamics involved in the engine-airframe system add additional special requirements in areas of vibration, critical speeds and frequencies, torsional stability, and low cycle and high cycle fatigue mechanisms. These items require special attention to an integrated approach to design as a change in one area to provide an improvement can have an adverse effect elsewhere.

COMPONENT QUALIFICATION

Component qualification represents a specification verification process. This process is two-fold during the development phase as both the component is verified to see if it meets the specification requirements but in some cases problems shown in development may also show where the specification can be improved. With the additional following system verification, problems shown there also may end up causing revision of the component specification or qualification requirements.

Without consideration for all the factors that may affect component qualification during the development phase, the basic process is one of demonstration of compliance with specific design requirements. Typically these requirements can be grouped in categories of functional, structural, endurance and environmental. Methods of substantiation also can be categorized as analysis, testing or similarity. Qualification could be one of these single methods or a combination.

Analyses are usually employed in the overall design process to prove both design details for a given component as well as the overall system. The analysis method is either verified by past efforts and/or by tests conducted in conjunction with the development. Where these can be employed to meet a specific qualification requirement will be subject to individual evaluation at that time. In most cases they are considered supplemental to testing in lieu of a replacement, but past experience may allow their usage in this manner.

Qualification by similarity is intended to take advantage of previous qualification of components. This could be for a single requirement or for all requirements for a given component. Similarity can be based on an identical component used in another application with the same operational and environmental requirements at one extreme to another where either the component is similar in design and/or its operational and environmental requirements are similar. This procedure is obviously based on the circumstances of an individual design or an individual manufacturer.

Qualification by testing is the most common method for the initial qualification of a component. Additional sources of supply are more likely to utilize the other means to become a qualified source of the component.

Functional tests for qualification may be the same tests required for acceptance of production units or may be an expanded basis with the acceptance tests being a chosen reduction to insure quality. For engine-airframe components these may include such tests as structural/leakage verification (proof pressure), accuracy/tolerance of an output parameter, or performance measurement (torque, flow rate, voltage or pressure).

Structural tests for qualification will vary according to application but may be static strength, burst pressure, or fatigue type tests. The fatigue tests represent one of the most important tests and requires continuing evaluation. Although the level of the loads can be established for operational environment, the frequency of occurrence can vary based on how the rotorcraft is subsequently operated. Variances must be monitored to insure retirement lifes associated with fatigue remain valid.

Endurance tests are one of the most complex tests for many of the engine-airframe components as the operational spectrum, transients, and effects of other influences such as temperature and vibration must be considered for a valid test set-up. Ability to simulate those conditions in a realistic manner complicates the test even more. Endurance can be related to motors, transmissions, and starters that have a normal operational sequence but also to static hardware such as an exhaust system which must be able to withstand engine exhaust temperatures, external cooling temperatures, as well as vibrating loads to demonstrate its endurance capability.

Environmental tests cover the overall ambient environment including sand, dust, salt fog, rain, snow, solar radiation, and fungus as well as operational environment conditions such as temperature and vibration. As indicated previously, ability to withstand singular environmental conditions may not be satisfactory as a combination may be more representative in actual operation.

COMPONENT QUALIFICATION PROCEDURES

One of the recurring problems in component qualification is the ability to address changes during the development cycle and validation of prototype hardware as equivalent to production. Components that have satisfactorily performed during the component level qualification and rotorcraft development phase are sometimes prone to problems in going to production manufacturing methods. Parts made from "hog-outs" that go to a production forging or casting, tolerances relaxed to insure production acceptance rates, multiple Class II (minor not affecting form, fit or function) changes, production manufacturing process changes from the original qualification version, etc. have all led to lessons learned statistics. As such, detailed configuration control must be established for the qualification test specimen to insure that all variances from that point are properly addressed and verified.

Test plans and reports are additionally important in assuring a component test is both clearly defined and documented. Later assessments of changes or problems can be expedited by having a good audit trail. For safety related parts, specifically including fatigue test samples, it is also good practice to retain the test sample as part of the data record. Any future problems in these areas can immediately be checked for any variances which have occurred from the qualification specimen. Temperature and vibration measurements are extremely sensitive to the exact location and details of the recording instrumentation such that precise records are necessary for any later correlation efforts.

The number of specimens utilized for qualification testing is once again subject to specific application and circumstances. The notable exception has been in the required number of fatigue test specimens. The U.S. rotorcraft community, both Government and industry, has reviewed the fatigue test methodology on several occasions with the established minimum number of specimens required being six.

ENGINE-AIRFRAME COMPONENTS-LESSONS LEARNED

The U.S. Army has been involved directly in the specification and qualification of rotorcraft since the 1960's. Prior cognizance had been done in conjunction with the other services. However, within this time period, two major rotorcraft have been designed and fielded (UH-60A BLACK HAWK and AH-64A APACHE) as well as major and minor modifications to all the other series of rotorcraft in the U.S. Army inventory. With this development period and operational experience, there have been numerous lessons learned including both recurrence of old problems as well as new problems/successes based on advanced technology and usage. The following reiterate some examples for various components included in the overall engine-airframe integration spectrum.

Air Induction System: Except for aerodynamic model testing, most of the basic air induction system components are tested as a system. However, the rotorcraft environment in ground proximity has required many rotorcraft air induction system designs to include provisions for Foreign Object Damage (FOD) protection and/or sand and dust protection. The screens, filter media, and/or air particle separators used in these applications do require component level qualification. Since screens by themselves do not address the sand and dust problem and filter media applications have shown particle migration, blockage problems, and high associated pressure drops, only the air particle separator will be discussed. An additional requirement of the air induction system in the increased rate of all weather applications is an anti-icing system. Component level requirements for the anti-icing system for rotorcraft will also be discussed.

Air Particle Separators: Since the basic criteria of an air induction system must be retained in the air particle separator design, the engine airflow, pressure drop and distortion requirements must be retained. Verification of these at a component level is dependent on the adequacy of the air induction system simulation. Pressure drops and distortion for an individual component test without good simulation of the total effect of the rest of the air induction system can cause invalid conclusions of the total system effect. It is even more important in the baseline determination of sand and dust separation efficiency. Design requirements need to account for a variety of sand and dust to insure separation efficiency is retained for both coarse and fine spectrums. Separation of the coarse particles may increase the life of the compressor only to find that the fine sand has degraded the hot end of the engine by blocking cooling passages and eroding seals. The testing for efficiency is even more critical to assure the intent of the specification is met for exposure to all spectrums of sand and dust. Due to particle bounce and interaction, it has been shown that two dimensional models do not provide accurate results. For the same type reasons, a particle separator tested without benefit of the air induction system influence likewise does not provide accurate results. The method of feeding the sand and dust must also provide for concentration levels that are realistic and provide for the diffusion resulting in a rotor induced sand and dust cloud. A test set-up using spray nozzles directing the sand and dust to the separator will give different results from one where the sand and dust is diffused in the airflow and allowed to be pulled into the particle separator. Finally, the separator efficiency is only valid for as long as the sand and dust overboard system maintains its design airflow. The design of this system, whether a blower/fan or ejector system, must be able to withstand the sand and dust effects and, as such, will require an endurance test to determine its capability in the sand and dust environment. Designs with bypasses to the particle separator also need to assure the negative impact of pilot actions to gain increased power for take-off, positive seals to prevent leakage, and lack of sand/FOD retention which would be "dumped" into the engine when the bypass utilized. Particle separator testing for FOD also requires accurate simulation. Items thrown into an air induction system may be separated because of their initial velocity. FOD from items left in the air induction system or in near proximity while on the ground, will not have the same separation effect during engine starting or with low relative velocity when ingested in flight.

Anti-Icing System: If a rotorcraft inadvertently gets into icing conditions beyond its capability, the last thing the pilot wants is to lose engine power also. Additionally, due to the associated pressure drop/temperature drops through the air induction system, the engine air induction system can be in icing conditions without the rest of the aircraft being affected. This latter condition requires specific design criteria for an ice detection system or pilot operating procedures. The anti-icing design capability of the air induction system will need to be better than the rest of the rotorcraft to insure the first condition is considered. Icing tests in artificial and natural icing is not where the air induction capability should be demonstrated. Icing tunnel testing covering the design conditions of temperature, liquid water content and droplet size will attest to the systems' capability. For bleed air heated systems where temperature (and thus anti-icing) capability is directly related to engine power, it must be assured that ground operation and low power descents will not cause a problem. Where FOD screens are involved, it is also necessary to insure that less severe icing conditions are checked as typically the protection lies in rapidly icing the screen and using bypass airflow. For marginal icing conditions, the moisture thru the screen can cause icing on air induction parts that are not anti-iced. Snow is a special case in which component testing has not been successful. The ability to produce a snowflake has not been duplicated adequately and may require system verification as snow migration through a particle separator into a plenum type system can result in subsequent snow/ice ingestion.

Exhaust System: In the past, the exhaust system has been simply a tailpipe designed to discharge engine exhaust gasses away from the rotorcraft with a minimum of back pressure to the engine. The current military requirements for an infra-red (IR) suppression system for the engine exhaust system has complicated both the design and qualification effort. Requirements to cool and shield hot parts as well as cool the exhaust plume have required a variety of detail aerodynamic and thermodynamic techniques to achieve the overall signature reduction. Whereas, the "beef" of a previous exhaust system usually provided for ease of redesign and limited testing to insure its endurance capability, current IR suppressors require extensive development and qualification testing. One of the basic problems in the past is that the exhaust system would be vibration tested and then it would be temperature cycled. In order to provide any component level results which validate the endurance capability, these have been required to be combined. Since cooling has to be uniformly controlled, the element of engine exhaust swirl angle must also be considered. As such, a current method to insure all these aspects are accounted for has required a test rig to include the respective engine operated for a representative mission spectrum with a rotorcraft induced vibration environment. Problems with this approach will be discussed later under discussion of accelerated mission endurance testing.

Starting System: An engine starting system provides for specific component level testing, but successful operation is very dependent on system level verification. Specifications for electric, hydraulic, and pneumatic starters have existed for some time and specific standardized tests have been in place. Starter control valves/devices are also fairly well standardized, but the compatibility tests between the starter and its control valve/device is very important to insure successful system operation. The starter-engine interface requirements must be clearly defined to cover transients, engagement loads, cut-out speeds, torque-speed characteristics over the ambient temperature range, motoring requirements (including water-wash cycles), starter and engine ignition duty cycles, and over-hung moments. The components for the starter energy source will be covered under their general system: electrical, hydraulic, or pneumatic.

Fuel System: Fuel system components typically make up the greatest number of individual components associated with the engine-airframe integration. In recent years, design requirements for survivability and vulnerability have both provided additional capability as well as develop new type problems for component level design/qualification. The following reiterates some of these more recent impacts on component level issues.

Fuel cells were redesigned to include crashworthiness capability to resist puncture, tears, abrasives as well as a final test of a 65 foot drop test filled with water. Ability to meet this criteria required a tank construction method to insure cross layering of plies along with stringers to carry loads around corners, fittings, attachments, etc. Ability to meet this rigid drop test criteria has typically required multiple attempts and stringent manufacturing process instructions to retain production consistency. As a manual layup procedure has been utilized, several problems have developed due to an individual workers' failure to follow detailed instructions.

Fuel cells have also utilized a self sealing material to provide ballistic sealing capability. Abrasion in the installed condition, difficulty during installation into the aircraft causing "forced" methods, and final production or overhaul finishing/repair processes have caused penetration of the protective layer causing activation of the sealer. Liner installation with "wrinkles" removed thru a finishing process have caused the more common production problem of damaging the protective liner.

Fuel cell ullage area protection from explosion has also generated both improved capability and new component problems. Use of reticulated foam inside fuel cells has required development and qualification of the foam. Aging, internal abrasive action, installation damage, fuel compatibility, etc. have led to some cases of internally induced fuel contamination. Later usage of inert gas systems have required another component development process with design requirements to retain inert conditions in the ullage area for initial start-up, full rotorcraft operational interval and for long term endurance to meet rotorcraft operational and reliability requirements.

Fuel lines have been designed to include mechanical protection from damage/chafing and self-sealing for ballistic protection. Problems with sealant activation and/or basic hose material reaction to long term fuel exposure has led to some problems with hoses swelling and thus closing off flow area. Detail manufacturing process controls have been required to retain proper hose characteristics. Additional care has been required to ensure hose fitting application, in the detail hose make-up, does not damage the hoses' liner/sealant protection.

Fuel pumps as boost pumps in back-up operation and fuel transfer operations have not changed in design requirements considerably except for system design requirements to keep ignition sources out of the fuel cells, thus, remote mounted pumps have required some changes in functional characteristics. The impact of improved survivability/vulnerability requirements for a suction fuel system has had major impacts on engine fuel pump designs and overall fuel system design. The suction system has required more attention to component designs to reduce pressure drops, eliminate potential turbulence/bubble generation in high vapor to liquid ratio conditions and the requirement to insure seal designs account for a negative pressure condition as well as positive pressure. This latter condition has caused air introduction to the fuel system due to "lifting off" of some standard pressure type seals requiring vacuum checks in addition to pressure checks during the qualification/acceptance process.

Fuel valves/fittings have required redesign features to consider frangible connections/attachments and self-sealing breakaway features to address crashworthiness requirements. High pressure drops through breakaway valves and premature breakage of frangible fittings and breakaway parts has caused improved redesigns to meet the crashworthiness intent as well as operational requirements.

Fuel quantity probes, fuel drains, fuel fill ports/devices, etc. also have all been impacted in the overall consideration of crashworthiness design features.

Engine Mounting System: Engine mounting system requirements have added additional detailed design process to address the overall engine vibration environment, redundancy for vulnerability considerations, and crashworthiness design features. Detailed component level design has been affected primarily in the ability to maintain mount integrity as related to need to control the vibration level for extended periods.

Drive System: The drive system has remained as the integral basis of the rotorcraft propulsion system. As such, as a system it has received extensive R & D effort to improve efficiencies, endurance, weight reduction, durability, transient load capability, maintainability, cost reduction, etc. One of the more definitized military requirements has been the vulnerability improvement to allow operation at cruise power conditions with loss of oil and/or with grease in lieu of oil.

Qualification test procedures to test transmissions/gearboxes, shafts, couplings, bearings, clutches, oil coolers, etc. have typically required multiple levels of tests from component level to combined system bench tests. The interrelationship between engine-drive system dynamics and difficulty in accurate simulation during bench tests has required system level tests as part of a Ground Test Vehicle or Rig. This will be discussed in more detail in another paper on system tests.

Whereas most components are qualified, based on successful completion of a given set up tests, transmissions/gearboxes are developed similarly to engines in that pre-qualification endurance type tests are also utilized. These tests consist of overload tests to verify detail gear patterns and capability as well as endurance and loss of lubrication tests. Results from these tests are measured in producing failures and from the detailed teardown analysis performed after completion of a given test. These specific and long term endurance tests provide the assurance in meeting the design requirements and ability to successfully pass the formal qualification tests.

Pneumatic System: The engine and drive system provide the power base for many other aircraft subsystems but the pneumatic system is usually closely related to the engine installation as engine bleed air is used either as a primary or as a backup pressured air source. Any air source for the pneumatic system when associated with rotorcraft operation will be contaminated. Filtration systems and particle separators only work on a percentage basis such that contamination due to sand and dust, moisture, salt fog, and environmental pollutants in general must be considered for both design and test. Most environmental specifications/standards considered these general pollutants to be an "external" test. A valve can be hermetically sealed on the outside, but internal airflow can cause problems.

The following provide a myriad of contaminated air problems determined after the fact. An engine specification required contaminated bleed air tests for its components, but the bleed air shut-off valve provided as part of the installation did not. A valve successfully passed its humidity test, but during the weekend shutdown from the end of the test to the final teardown check resulted in the closure flapper seizing to its sealing ring due to moisture left in a storage bag. A valve passed its sand and dust test, but failed to properly operate in its application due to closure of the internal sensing passage with accumulated dust. A bleed air shut-off valve passed all of its environmental tests but failed to function properly in application due to contamination associated with the water wash solvent agent.

Pneumatic components are also very likely to pass all of their prescribed component qualification tests and fail to operate in the system. Dynamic interaction, opening/closing rates, balance pressures, surge effects, transient pressure spikes, etc. have shown up later indicating the need to do a system level test. A check valve that has not failed in previous usages develops a rash of failures due to system dynamic pressure surges causing pounding of the check valve in opening/closing. A shut-off valve operated on differential pressure fails to close based on a combination of two engine power levels finding a balance point where positive closure is not obtained.

Hydraulic System: Hydraulic systems may or may not be closely tied with the overall engine-airframe integration. Problems are similar to those in pneumatic except additional problems directly related to the higher pressure system may be encountered. Hydraulic systems also add another aspect of environmental problems associated with cold temperatures. Properly functioning valves in normal temperatures may become non-functional at extreme cold conditions.

Examples of some cold temperature problems associated with engine-airframe integration are included as follows. A pump-starter relied on a valve to change from pump mode to starter mode. Cold temperature operation caused accumulator depletion going through the transition from pump to motor due to the slow actuation at cold temperatures. A hydraulically operated friction clutch using APU oil allowed successful APU starts when the most viscous oil was used in the APU for testing. Later operation with a less viscous oil at cold temperatures led to faster engagement of the clutch before the APU could sustain the higher load associated with cold temperatures. This subsequent faster engagement at cold temperature caused a failed start attempt.

Lubrication System: Most newer rotorcraft systems have integrated the oil system totally within the engine and transmission/gearboxes such that fewer components exist. Remote oil coolers, fans, and bypass valves are typically those components involved in a remote system. Problems associated with these are usually no different from other fluid coolers, fans or valves.

Diagnostics/Sensors/Electrical/Electronic: For all the mechanical systems associated with engine-airframe components, an important aspect is also tied to diagnostics and sensors associated with operation monitoring. The diagnostics and sensors are important for general rotorcraft operation and safety as well as maintenance indicators. An important issue from the design for when diagnostics and sensors are needed is tied to the initial Failure Modes, Effects, and Criticality Analysis and System Hazard Analysis. Failure modes where catastrophic failure can occur or pilot activated modes where problems can be caused should be addressed. The electronic cockpit and computer based logic will allow more exacting trending, data base and ability to monitor limits for the pilot, but these initial design steps will still be involved.

One design problem with diagnostics/sensors has been what is actually sensed and whether it defines a true indication. Some cases have measured an electrical signal to cause an event in lieu of verification that the event occurred. For example, the power signal to open a valve is used to indicate an action occurred whereas a malfunction in the valve may not allow it to open completely or at all. Another has been failure to consider inadvertent operation of a system from its intended functions. For example, an engine air induction anti-icing system may function for extended periods at cold ambients, but may cause real problems if turned on inadvertently on a hot day. The diagnostic system can alert the pilot under these circumstances.

Sensors utilized for these functions sometimes are considered "standard" and are not provided the same level of qualification as the system they are monitoring. For safety; however, sensor failures can cause a flight abort just the same as a system problem.

Improvements in sensor capability to provide valid indications has been significantly increased in the area of chip detectors. What started as magnetic plugs to electric chip detectors has now evolved to a family of "smart" discriminating chip detectors. Features for burning off fuzz, counting chips, measuring size of chips, etc. are now in production with continuing improvements in development. Vibration, noise, and frequency response type sensors are also being developed which will determine changes from a standard to alert of pilot of a specific problem or changed condition.

Electrical/electronic aspects of all sensor monitoring and component functional operation is also integral to the component design and even more important to the final system operation. With the advent of increased electronics in the overall engine-airframe integration, more detailed component design will evolve also.

The most significant nuisance and real world operating problems with electrical and electronics associated with engine-airframe integration has been associated with humidity, moisture, rain, and salt fog. Wheatstone bridge fully electronic pressure sensors provide great reliability as a component but fail due to ambient pressure sensing to the atmosphere allowing moisture to affect the resistant balance of the circuit. Conformal coatings to protect electronics are usually not perfect so any imperfection will be vulnerable to moisture. Electrical connectors all are designed to seal out the elements, but moisture can penetrate any damage or defect in the sealing surface. Electronic modules which are easy to replace will be used as a standard troubleshooting method and will solve the problem sometimes by just breaking and remaking the connection.

Component electrical/electronics are also very affected by power quality and/or transients. Switching transients from going from ground power to on board power have tested/failed many components. Susceptibility to electromagnetic fields through lack of shielding/isolating will also cause component failure or overall system malfunction.

General Component Tests: Some component tests have shown general repeated problems. Examples of problems with sand and dust, humidity, rain, and salt fog have indicated assumptions of being in an enclosed area not requiring a test or internal airflow components only be tested for external influences. Other similar problems are also caused by assumptions that do not provide proper testing. A series of examples are included as follows:

Cold temperature tests have a series of repeated problems. The most basic problem is that the component being tested is not cold soaked long enough to uniformly reach the test temperature. Another problem is a "dry" cold temperature test where a "humid" cold temperature test could cause problems due to frosting/freezing in actual operation. The most basic problem, however, has been a cold temperature test associated only with capability to withstand cold temperatures with limited operation. A functional check to whatever the component is expected to perform in service should be performed to include verification of required performance parameters, such as response time, or full authority of operation.

Hot temperature tests have the problem of reflecting actual experienced conditions. Requirements for high external ambient temperatures do not account for such conditions as internal engine compartment operating temperatures, re-radiated effects of other heated components/surfaces, solar radiation, and full range of airflow conditions. Engine compartment conditions of soakback after shut-down also can exceed normal operating temperatures resulting in problems.

Vibration is one of the most discussed environments and hardest to duplicate for a representative rotorcraft environment. Past efforts to do a resonance search and dwell on critical resonances has been changed in many cases to attempts to match dwells at actual installed vibration conditions. Use of actual installed vibration conditions is subject to knowing what they are over an entire flight spectrum at a point in time when the component test is conducted. Further rotorcraft changes and different installation possibilities will require a mixture of vibration testing to be able to address both a general rotorcraft vibration environment as well as the specific application. Special mission spectrums and different mission equipment usage can also impact previous determined vibration environments which could adversely affect specific vibration conditions.

ENGINE-AIRFRAME INTEGRATION - COMPONENT INTERFACE

For an engine designed to the latest technology to operate to its full potential, it is essential to insure its interface is defined in exacting terms and likewise the airframe interface. The U.S. Army requires the engine manufacturer and prime rotorcraft manufacturer to have an Engine-Airframe Interface Agreement which outlines the procedures which they choose to share information, transmit questions, resolve problems, etc. An end item required as part of this agreement is an Interface Control Document (ICD).

An ICD will define all details of the engine-airframe interface from exact configuration details to general performance related requirements. This document will provide design details from the engine specification and the airframe specification as they relate to this interface. As such, component designs directly associated with the engine can be determined from this document.

However, exacting documents may not solve all problems. In some cases, the requirement may not be correct, may not be all encompassing, or may not address an important characteristic. The following gives examples of several of these cases. The ICD defined the maximum allowable torque load from the starter, but the starter shear section to insure that value was too weak in fatigue. The water wash duty cycle was not defined, and thus not considered, for the starter motoring duty cycle. The engine fuel pump vapor to liquid capability was defined for steady state conditions with no dynamics/transient operation limits defined which are ever present in an actual suction fuel system. Engine component temperature limits were not exactly defined and did not consider high temperature fuel pump effects when operating at high vapor to liquid ratios.

ENGINE-AIRFRAME INSTALLATION TESTS

As components have been tested individually and in subsystems/systems, there are still a level of component tests which are not part of the aircraft system tests. These are usually conducted for the purpose of simulating overall installation as well as providing a method to do some Accelerated Mission Endurance Tests. The engine and transmission/gearbox test cell/bench tests provide endurance testing of these items with limited interface effects. Test rigs to provide better installation simulation and associated components provide this new level of effort short of aircraft tests. An engine-airframe installation test is simplistic to set up as the engine provides the power source. A load absorber will then allow the engine to be tested to some accelerated mission endurance test cycle. Attachment of air induction system, exhaust system, starter, valves, engine mounts, etc., allows each item to experience this particular engine cycle. The rotorcraft vibration environment would add one more dimension into the simulation effectiveness of this type test.

A test like this was discussed relative to the exhaust system along with a mention of problems. A rotorcraft is a very dynamic vehicle including all its structural flexibility. A test setup with actuators to produce a shake table vibration environment does not simulate the equivalent environment with the resulting transmissive vibration being different from an actual rotorcraft environment. Providing a realistic vibration environment consistent with engine mission power cycle must be also considered to provide a realistic test. This lack of true simulation can cause problems/failures in a test rig that would never be experienced in an actual rotorcraft environment.

NEW INITIATIVES

New initiatives are usually revised or enhanced methods of the process in which items are currently being handled New initiatives being introduced for rotorcraft and other Army systems are like that. However, the improved methods are still real gains. Some of these new initiatives as they relate to engine-airframe components are discussed as follows:

Environmental Stress Screening (ESS): Like the problem noted in other component qualification areas, a combination of environmental conditions tested together is needed to evaluate the actual endurance capability. ESS is that type of testing for electronic systems, assemblies or parts. A normal "burn-in" power on acceptance test is utilized for electronic components, but ESS looks at the influence of repetitive changes due to temperature, random vibration spectrums, or other applicable conditions. The purpose of ESS is to force-fail infant and latent defects at an appropriate point in the evaluation of the product so that the end product reliability is improved beyond current standard methods of verification.

Alternate Sources, Competitive Procurement: Competitive procurement has been a standard procedure for Government procurement since before the beginning of the "heavier than air" era. Spare parts procurement still has been limited to approved and controlled sources. Current mandates for fully competitive technical data packages has required a recovery process in obtaining and developing the necessary information. The key to rotorcraft component competitive procurement still has to be based on the premise of the original qualification. That is, alternate sources and/or fully competitive sources must be substantiated by analysis, test, experience, and quality to the basic qualified source. Assurance on the Government aid as well as aid from industry is required to maintain the safety and quality of current rotorcraft.

Flight Safety Parts Program: The U.S. Army is adopting a flight safety parts program to be adopted as an enhanced program to improve overall aircraft quality and safety. A key feature of this program is that it is a cooperative Army-industry effort to provide life cycle management of aircraft parts whose integrity is essential to flight safety.

The U.S. Army has had a Critical Parts Program since the early seventies which increased the quality procedures for selected critical parts. Although a high level of quality procedures has been and is continued for aviation parts, these selected critical parts were controlled to even higher standards. For these parts, specific critical characteristics were defined which were essential for part integrity and subsequent flight safety. The limitation of this program is that it only insured quality integrity of the part during manufacturing.

For this program, a Flight Safety Part has been defined as any part, assembly or installation whose failure, malfunction or absence could cause loss or serious damage to the aircraft and/or serious injury or death to the occupants. For rotorcraft, a large majority of these come directly from the engine-drive system-rotor system combinations and are thus directly involved with the engine-airframe components.

For these identified parts, the critical characteristics need to be defined. A critical characteristic is defined as any feature throughout the life cycle of a Flight Safety Part, which if non-conforming, missing, or degraded could cause failure of the Flight Safety Part. Critical characteristics include dimension, tolerance, finish, material or assembly, manufacturing and inspection processes. Life cycle covers manufacturing, operations, field maintenance and overhaul.

Implementation of this program has several key elements to provide controls for these parts. Examples include: (1) Critical characteristics are identified by the designers and annotated in the drawings. (2) The planning documents, by which the parts are manufactured and quality inspected, are approved at a high level inter-disciplinary board to insure proper controls are in place to maintain the critical characteristics. Once approved, the procedures are "frozen" and cannot be changed, varied, or waived. Only a formal change, again approved by the board, can constitute any change in procedure. (3) All critical characteristics of the Flight Safety Parts are to be inspected 100% by qualified inspectors for every part manufactured. (4) No deviations from the drawing specification for critical characteristics are permitted. Any change of the critical characteristics would require resubstantiation of the part before a change could be considered. (5) Manuals, including overhaul requirements, are to be revised to properly annotate the critical characteristics. No repair or overhaul shall be permitted to deviate from the drawing specification for critical characteristics. (6) Flight Safety Parts will be tracked for their entire life cycle. All records regarding initial fabrication and inspection are retained.

Along with this implementation, a method to insure that the parts are performing adequately in the real world aircraft environment is being put in place. This program will selectively test/inspect new and fielded parts. The purpose of this program is to: (1) confirm the validity of requirements used during initial design and qualification of Flight Safety Parts; (2) monitor the effects of usage on parts to demonstrate that replacement and overhaul intervals are adequate and safe relative to actual utilization; (3) continually assess new components to insure that minor design and manufacturing changes do not affect Flight Safety Parts in a detrimental manner; (4) confirm degraded mode limits or effects due to wear, corrosion, fretting, damage, etc.; (5) insure that repair procedures do not degrade the critical characteristics; and (6) determine any previously unknown/or known degraded condition impact on flight safety parts.

Once parts are chosen for the surveillance test, they will be subjected to appropriate non-destructive inspections, analytical teardowr analyses, acceptance test procedures, and/or static/fatigue type tests. The fatigue tests provide detailed evidence on how the parts are functioning, in a real usage spectrum environment and the validity of their associated retirement lives. These efforts will be supplemented with instrumented aircraft to establish actual and changing aircraft usage.

CONCLUSION

Changes in procedures in handling components for rotorcraft engine-airframe integration are evolving along with all other design and test philosophies and capability. With computer aided design techniques becoming standard and computer simulation techniques and capabilities continually being enhanced, the analytical base will continue to perform a stronger role. Component testing, likewise, will improve as better simulation techniques are applied. Better solutions at the component level will improve the final system integration. As long as the rotorcraft system remains in concept as we know it today, these challenges will be the goal in improving the engine-airframe integration.

INFLUENCE OF ENGINE VARIABLES ON FUTURE HELICOPTER SYSTEMS

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SUMMARY

A helicopter system defined for the year 2000 must make numerous assumptions about future technology and system requirements. In the case of propulsion system options, the required 5- to 15-year development cycle (from preliminary engine cycle studies through engine development to production) complicates the process of optimizing the engine/rotorcraft system. This paper illustrates a well-disciplined approach to engine definition studies to yield engine candidates that best use emerging technologies. An advanced, single-engine helicopter requiring a 2000-shp powerplant was used as a baseline vehicle. Both simple and heat-recovery cycles were evaluated, using trade factors to establish engine effects on helicopter direct operating cost (DOC), the optimization parameter assumed for purposes of illustration.

Year-2000 engine performance was calculated; it showed maximum improvements in SFC and specific power of 30 and 65 percent, respectively. Mission fuel burn was reduced by 21.1 percent for the simple cycle and 48.0 percent for the heat-recovery cycle. DOC improvements ranged from 7.1 to 11.8 percent. The simple and heat-recovery cycles are approximately equal at the lower fuel price; however, the heat-recovery cycle has a clear advantage in DOC as the fuel price increases. In addition, the advancements in technologies were evaluated in terms of their benefit to DOC reduction; ceramics showed the greatest benefit.

1. INTRODUCTION

It is generally predicted that, during the next 5 to 20 years, engine fuel consumption, weight, size, and cost will have an increasing impact on advanced helicopter system configurations and associated operating costs. At the same time, helicopter mission requirements, fuel prices, and the availability of emerging engine cycle and component technologies will continue to be primary "drivers" for engine selection. Thus, for both the engine and airframe manufacturers, the challenge of determining optimum engine cycles for changes in technology, and defining the relative benefits of emerging engine improvements will become increasingly important to ensure helicopter system marketability.

The challenge is to effectively manage the process illustrated in Figure 1. Within the constraints of airframe and engine definition and development time frames, engines must be configured to best respond to helicopter mission and operating-cost requirements by incorporation of emerging component, material, and cycle technologies. To illustrate this management approach, this paper describes the selection of advanced engine cycles based on future requirements and technologies. In particular, the study assumes direct operating cost (DOC) as the primary system "driver" and provides a sample engine optimization to highlight the selection process.

2. BASELINE ROTORCRAFT/ENGINE SYSTEM, GROUND RULES, AND PROCEDURES

First, a baseline year-2000 rotorcraft and missions were defined, along with the evaluation procedures. In addition, a current technology baseline engine was defined as a reference against which to evaluate improvements derived from candidate advanced propulsion engine systems.

2.1 Baseline Rotorcraft

The baseline rotorcraft for this study was configured by incorporating projected year-2000 airframe technologies in a current rotorcraft from the same size class. The configured helicopter is a derivative of an existing rotorcraft (Figure 2). Major technology projections -- including weight reductions in the airframe and associated subsystems, figure-of-merit improvements in the main rotor, and improved aerodynamics -- were applied to the existing rotorcraft.

Key rotorcraft improvements result from advancements in material technology, specifically the use of composites in the airframe. Projected weight savings range from 1 percent for the main rotor blades to 70 percent for the rotor systems controls. This results in an overall weight savings of 18 percent.

The critical sizing point for the derivative rotorcraft was set at an altitude of 4000 ft and hot day conditions (95F [35C]). Power setting is at 95 percent of intermediate rating; the rate of climb is 500 ft/min (152 m/min). Two baseline rotorcraft were sized, based on a year-1985 2000-shp engine and two selected missions (discussed in

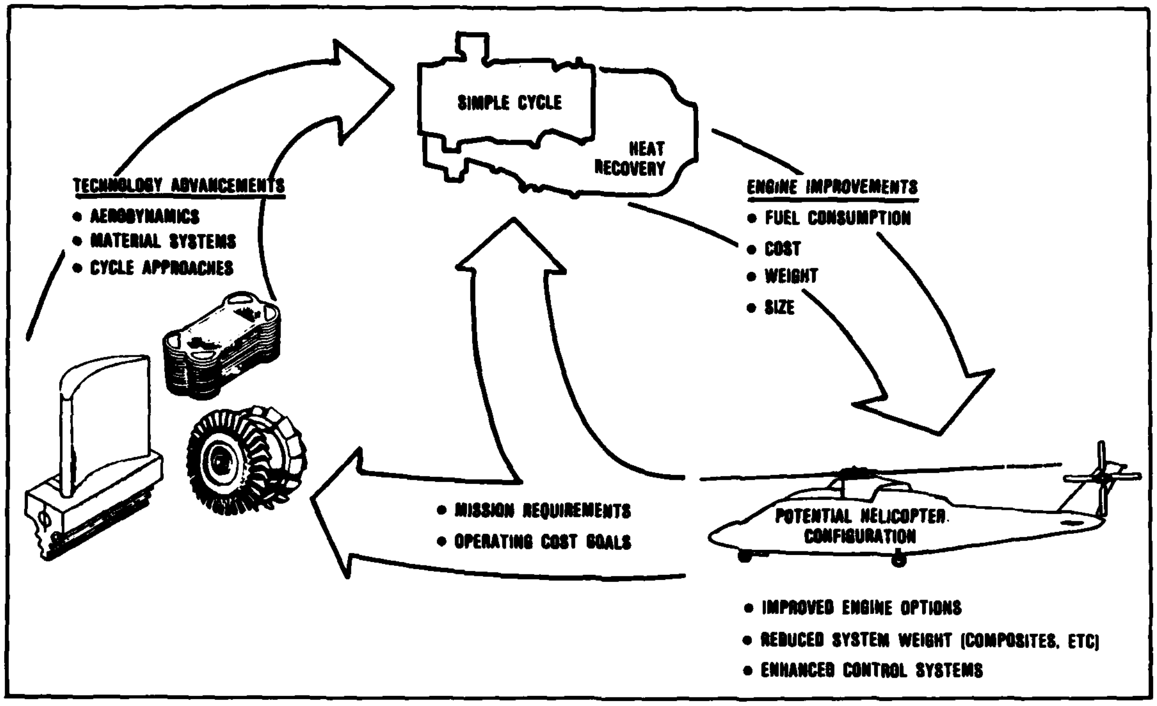

Figure 1. Engine/Aircraft Optimization Process.

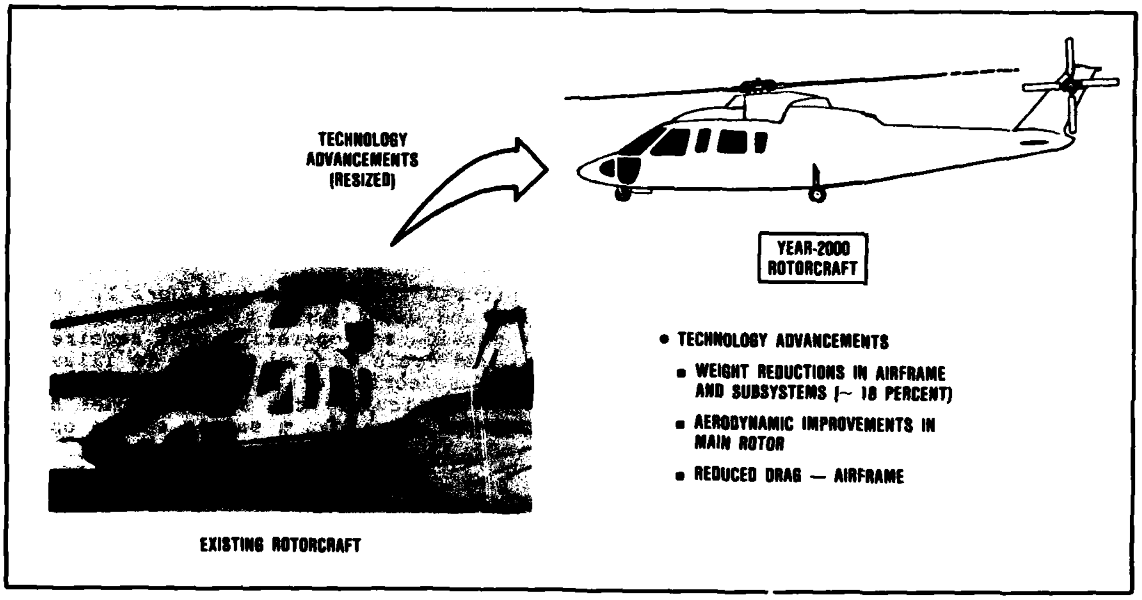

Figure 2. Baseline Rotorcraft.

paragraph 2.2). Those characteristics of the resultant vehicle that meet the mission and sizing criteria for Mission I are shown in Table 1. The rotorcraft sized for Mission II is similar; the most noticeable difference is a fuel weight increase of 491 lb (223 kg). This results in an increase in TOGW of 630 lb (286 kg). Rotor diameter increases 1.4 ft (0.43 m) due to the increased weight.

2.2 Baseline Missions for Rotorcraft

A preliminary survey of military and civil rotorcraft markets showed that existing missions fall into three typical categories: heavy lift, inspection/surveillance/scout, and ferry. The heavy lift mission is typified by a heavy load being moved a short distance, followed by a ferry flight. This results in numerous cycles between full and part power. Inspection/surveillance/scout and ferry missions are characterized by extended periods at reduced power settings for low-speed and cruise operation. The engine operates well below full-power rating during the major portion of such missions.

Table 1. Baseline Rotorcraft Data - Mission I.

Rotors	Main Rotor	Tail Rotor
Diameter, ft (m)	41.63 (12.69)	8.00 (2.44)
Number of blades	4	4
Solidity	0.075	0.172
Ct/sigma	0.080	0.070
Disk loading, lb/ft^2, (kg/m^2)	6.77 (33.05)	11.28 (56.05)
Tip speed, ft/sec (m/sec)	700.0 (213.4)	700.0 (213.4)
Drive system rating, shp (kW)	1930.2 (1439.0)	214.5 (160.0)
Weight, lb (kg)	524.4 (237.9)	121.0 (54.9)
Dimensions		
Fuselage length, ft (m)	42.2 (12.9)	
Fuselage width, ft (m)	7.0 (2.1)	

Weights	lb (kg)		lb (kg)
Propulsion	1851.9 (840.0)	Payload	3676.0 (1667.4)
Empty	4179.6 (1895.8)	Structure	1319.7 (598.6)
Fuel	1151.1 (522.1)	Gross	9216.5 (4180.5)

On the basis of recommendations by helicopter manufacturers, two representative missions that typify military and civil ferry applications were defined. The baseline missions consist of five cruise legs, separated by periods of hover at each destination point (Figure 3). The missions differ only in length; Mission II is twice as long as Mission I. The total distance is 130.4 nm (241.5 km) for Mission I and 260.8 nm (483.0 km) for Mission II. Hover conditions are the same for both missions. With the baseline engine, cruise power settings ranged near 40 percent, whereas hover required approximately 45 percent power. The low mission power requirements emphasize the importance of part-power SFC.

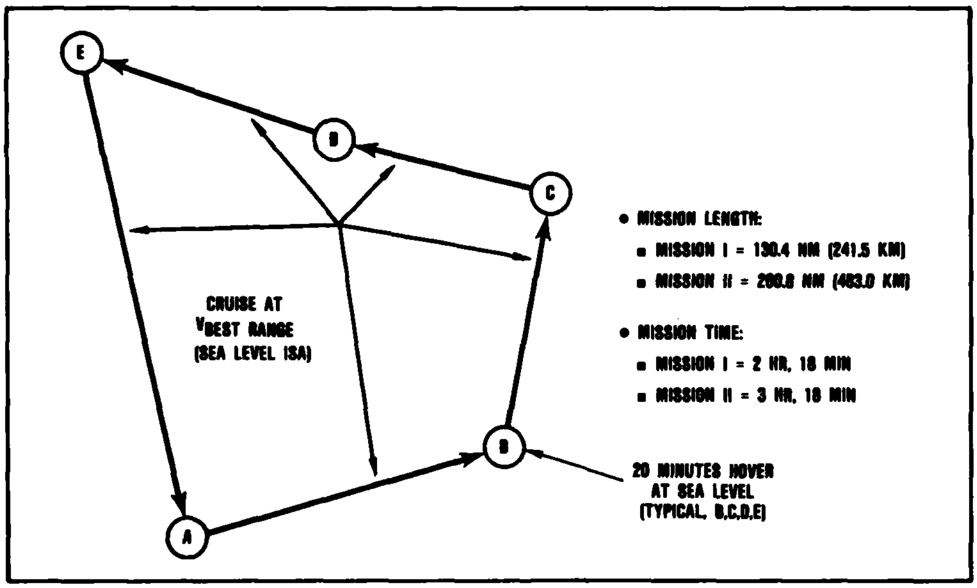

Figure 3. Baseline Mission(s).

2.3 Baseline Engine

To establish the baseline engine for the study, engines under development at Garrett Turbine Engine Company (GTEC) were surveyed and evaluated. Performance and operating parameters of these engines were adjusted to define the representative baseline engine for 1985 engine-demonstrated levels of component technology. The baseline engine, therefore, is considered representative of technologies that have been demonstrated in an engine but are not necessarily yet in full production.

At sea level, static, standard-day conditions, the engine was sized at 2000 shp (1491 kW) at the intermediate rated power (IRP) setting. The baseline engine (shown schematically in Figure 4) is a two-spool design that uses a five-stage axial plus single-stage centrifugal compressor with a total pressure ratio of 17:1 and an inlet corrected flow of 11.9 lb/sec (5.4 kg/sec). Two compressor configurations, an axial-centrifugal and a two-stage centrifugal, were investigated for the baseline engine. The two configurations had been compared over a range of inlet flows in a previous helicopter engine study. The study showed that below approximately 10 lb/sec, the two-stage centrifugal compressor had a lower life cycle cost (LCC), while at higher flows the axial centrifugal compressor had a lower LCC. Specifically, the lower cost of the two-

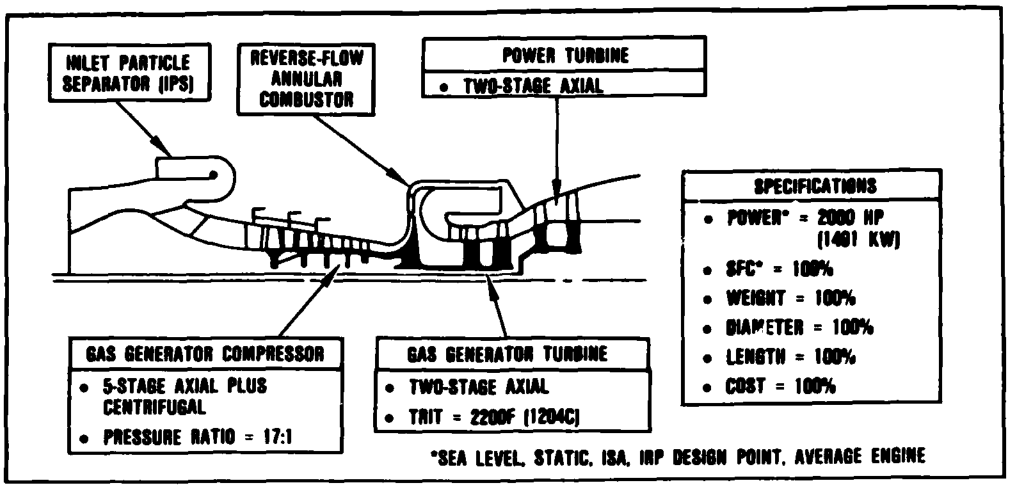

Figure 4. 1985 Technology Baseline Engine.

stage centrifugal is offset by the better performance of the axial-centrifugal. The comparison was also studied extensively for a helicopter engine currently under development in the 1200 to 1800 shp range; the results were similar. Therefore, with the baseline engine inlet flow greater than 10 lb/sec, the axial-centrifugal compressor was chosen. A reverse-flow annular combustor placed around a two-stage axial gas generator (GG) turbine results in a compact GG spool. Turbine rotor inlet temperature (TRIT) is set at 2200F (1204C). Shaft power is provided by a two-stage axial power turbine with a front-drive arrangement. The engine also has an inlet particle separator (IPS) with a mechanical blower system. Engine weight, diameter, length, and cost are set at 100 percent for reference.

2.4 Direct Operating Cost/Trade Factors

Helicopter DOC was used to evaluate the advanced, year-2000 propulsion systems. Major system-related DOC drivers are (1) manufacturing cost, (2) fuel cost, (3) maintenance cost, and (4) overhead. Individual DOC contributors were evaluated based on input from airframer manufacturers and past GTEC experience. Calculated DOCs were based on study assumptions and fixed/variable costs shown in Table 2, including the fuel prices selected for this study. The low and high fuel prices, $1 and $2 per gallon ($0.264 and $0.528 per liter), were selected to show the effect of increasing fuel costs. All costs are stated in 1985 dollars.

Table 2. Basis For DOC Computations.

Assumptions		Fixed Costs	Variable Costs
Number of engines (Excluding spares)	1500	Loan interest rate	Engine aircraft
Spares	4 percent	Imputed interest rate	Fuel*
Potential aircraft	1500	Depreciation schedule	Airframe maintenance
Annual use	2500 hr	Insurance schedule	Engine maintenance
Service life	7 years	Tax rate	Crew expenses
Takeoff gross weight (Structural limit)	9217 lb** (4180 kg)	Crew wages	
Payload	3676 lb (1667 kg)	Hangar rent	
		Miscellaneous	

*Based on $1 and $2/gal ($0.264 and $0.528/liter)

**Mission I

A detailed DOC analysis of the baseline rotorcraft/engine system was conducted for both missions. Additionally, a sensitivity analysis was performed to determine trade factors for key engine parameters in terms of their impact on DOC. These trade factors were used to relate rotorcraft system costs to changes in engine parameters. Five key engine-related DOC drivers were identified: engine SFC, weight, diameter, length, and engine acquisition cost. The resultant trade factors (Figure 5) show a change in per-trip DOC for each 1 percent change in a given engine parameter.

These trade factors take into account the synergistic effects on the engine and rotorcraft system for engine changes. As such, the change in DOC values reflect the nominal engine changes (such as engine SFC) and attendant changes to the rotorcraft system (fuel weight, tankage size, helicopter size, etc.) that result for the baseline missions. As shown, SFC and cost were determined to be the two key DOC drivers.

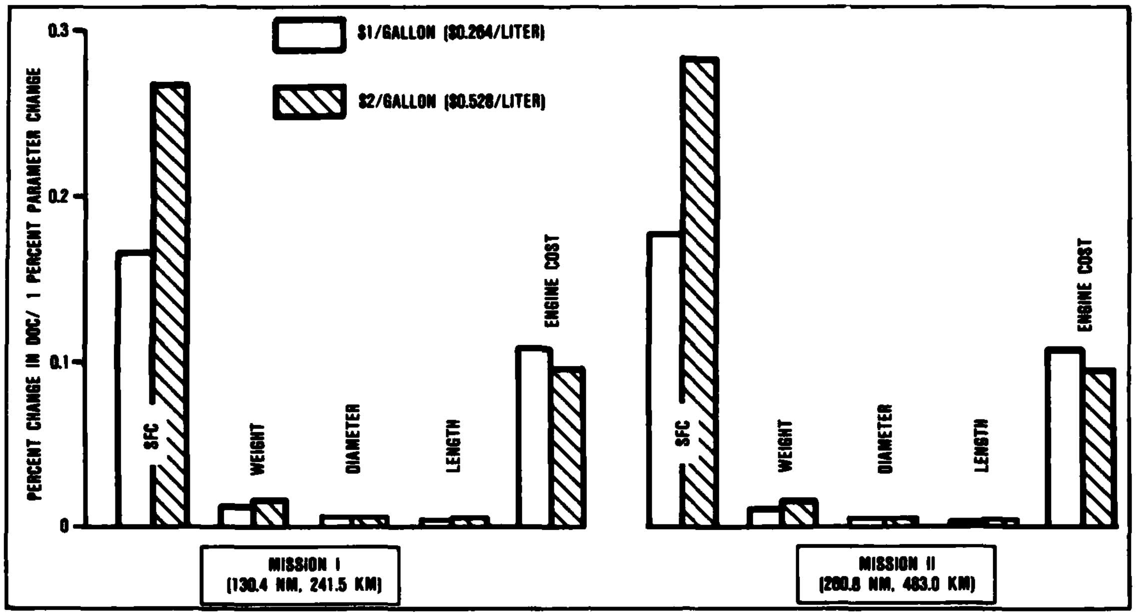

Figure 5. Trade Factors.

3. TECHNOLOGY PROJECTIONS

In configuring potential rotorcraft engines for the year 2000, expected levels of technology were projected that could be available with proper effort and funding support. Inherent in these projections was the assumption that the technologies will be ready for full-scale engine development by the year 2000. Technologies were grouped in four major areas: materials, aerodynamics/thermodynamics, heat recovery, and mechanical improvements. These technologies influence the cycle study in terms of component efficiency levels, turbine inlet temperature limits, and cooling flow requirements, as well as turbine stage count and hub speed limits.

3.1 Material Improvements

Material improvements were identified for both compressor section (cold) parts and for the turbine (hot) end. Materials for cold parts (presented in Table 3) primarily allow reductions in engine weight and cost; material improvements for hot-end parts allow enhanced engine performance by permitting increased cycle temperature and component stress capabilities.

Table 3. Year-2000 "Cold" Material Projections.

	High-Temperature, Powder Metallurgy Aluminum Alloy	Cast Titanium Alloy	Polymeric Composites	Titanium Metal Matrix Composit
Application	Vanes, blades, rotors	Vanes, blades, rotors	Gearboxes	Shafts
Cost factor*	0.7	0.8	0.9 - 1.2	3
Weight factor*	0.6	1	0.7	0.5

*1.0 is 1985 metallic engine part

Four materials offer cost and/or weight advantages for the cold section:

- o Powder-metallurgy aluminum can potentially reduce cost and weight relative to materials now used. By the year 2000, aluminum alloys will be able to operate at temperatures up to 850F (454C), which will allow application of aluminum in engines with higher compressor discharge pressures and temperatures. Current aluminum alloys have been successfully tested to over 600F (316C). Development of a suitable coating to prevent erosion of aluminum parts exposed to the flowpath will be required.

- Cast titanium alloys (with the same strength as forged alloys) offer the possibility of 20 percent cost reduction through the elimination of expensive machining operations.
- Polymeric composites, which consist of a fiber-reinforced resin with constituents yet to be identified, are predicted to reduce weight by as much as 30 percent, relative to present aluminum gearboxes.
- Metal-matrix composites for shafting may be a required technology for year-2000 turboshaft engines in this size class. A 50 percent reduction in weight, combined with required high-temperature strength and stiffness, achieves the required shaft critical-speed margins needed for the higher spool speeds and small bore sizes of future engines.

For the hot end, specifically the turbine and combustor, both metallics and nonmetallics were considered (Table 4). Three materials were identified:

- Super single crystal (SSC) for application in advanced metallic turbine blades and vanes increases the temperature capability of present single-crystal (SC) materials by approximately 100F (56C). SSC will allow higher stress and loading levels and, for a cooled turbine, could reduce cooling flow requirements at a given temperature.
- Nickel aluminide (Ni_3Al) offers the potential of improved strength-to-weight ratio by reducing weight approximately 15 percent relative to currently used Astroloy and Waspaloy turbine disk materials. This leads to higher turbine hub speed capabilities, resulting in reduced turbine stage count and/or higher efficiency.
- Ceramic components were also envisioned for hot-end application in year-2000 engines. Ceramics have the potential for use in turbine vanes, blades, and rotors as well as combustors and transition liners. The key improvement available from ceramics is the increased temperature capability that will allow higher turbine inlet temperatures without cooled turbine blades. For ceramics, the maximum material temperature limit is projected to be 2800F (1538C). Current reverse flow combustors have pattern factors* (PF) in the 0.15 to 0.20 range. Assuming an improved combustor pattern factor (PF $\leq$0.12, as estimated for the year 2000), engine cycle temperatures up to 2600F (1427C) were projected for compatibility with an uncooled turbine rotor blade. Significant weight savings will also be possible with ceramics. The reduced density of ceramics will reduce weight up to 60 percent on some components. To meet the increased temperature requirements, recuperators represent another potential ceramics application.

Table 4. Projected Properties for Hot-End Materials.

Application	Metallics		Nonmetallics
	Super SC	Ni_3Al	Ceramics
	Turbine blades/vanes	Disks	Turbine vanes, blades, rotors, combustors, transition liners, recuperators
Improvement in material temperature limit, F (C)	100 (56)	-	800 (444)
Strength ratio*	1	1	0.5 - 2.0
Cost factor*	3	1	1
Weight factor*	1	0.85	0.4

*1.0 is for 1985 metallic engine part.

*Pattern factor is an indicator of the combustor discharge temperature profile. Pattern factor is defined as the difference between the maximum exit temperature and the average exit temperature divided by the combustor temperature rise. Low numbers indicate a more uniform profile with less dramatic hot spots.

3.2 Aerodynamics

Improvements in component aerodynamic performance are also foreseen. In conjunction with advancements in material technology, aerodynamic advancements were projected by the year 2000 for compressors, turbines, and combustors in terms of increased component efficiencies, reduced losses, and higher aerodynamic loading capabilities.

Compressor performance predictions were made for three typical configurations: single-stage centrifugal, two-stage centrifugal, and axial-centrifugal. Year-2000 efficiency predictions, shown in Figure 6, are based on configuration, size, and pressure ratio. Size and pressure ratio variations were presented in terms of compressor exit flow as corrected for exit temperature and pressure conditions. Nearly a 4 percent increase in polytropic efficiency is expected by the year 2000 for an axial-centrifugal compressor, comparable to the 1985 baseline. This improvement is based on reasonable stage loading, 3 percent tip clearance, and a 0.3 exit Mach number. These improvements in efficiency are primarily attributable to the anticipated development and refinement of 3-D viscous analytical codes. Additional payoffs result from improved clearance control, reduced diffuser vane inlet losses, and reduced impeller shock and secondary losses.

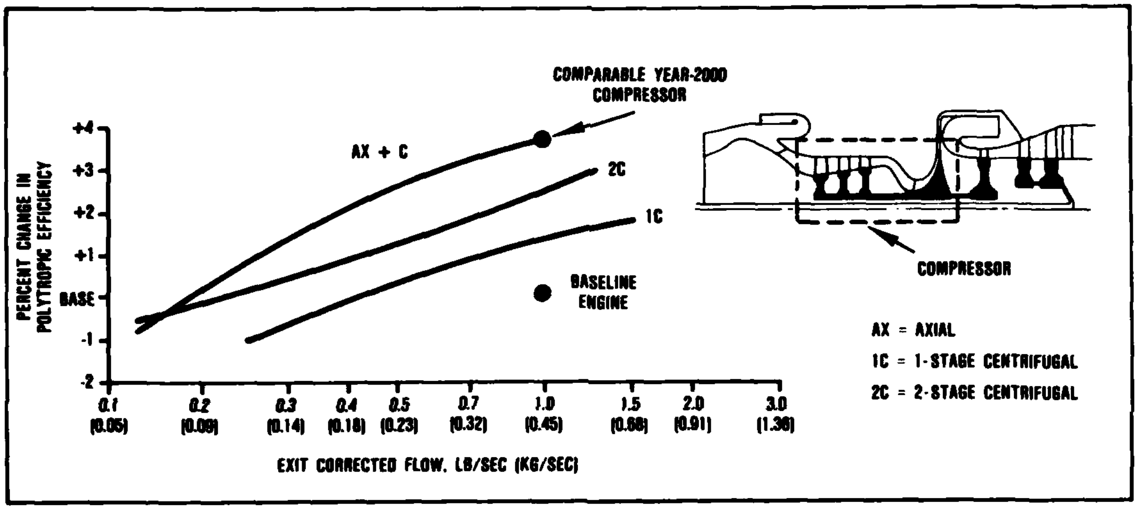

Figure 6. Compressor Efficiency Projections.

Combustor performance improvements are expected in several areas. Combustor pattern factors will be reduced to the 0.10 to 0.12 range from present values ranging from 0.15 to 0.20. Additionally, diffuser technology will be improved to maintain present pressure drop levels at increased inlet Mach numbers. Year-2000 combustors will also have higher heat release rates, reduced size, and improved durability.

Gas generator (GG) turbine performance projections were based on flow size and stage mean work coefficient for a two-stage axial configuration (Figure 7). Relating turbine performance (efficiency) as a function of the work coefficient brings rotational speed and blade radius into the cycle analysis. For the GG turbine, an efficiency improvement of approximately 2.5 percent is predicted relative to the 1985 baseline turbine. Turbine aerodynamic improvements will result from reduced rotor tip losses and minimized vane and blade interaction losses.

Power turbine efficiency improvements (Figure 8) are expected to be similar (approximately 2 to 3 percent). With shrouded blades, smaller efficiency improvements will be possible by reducing clearance losses; however, improvements from vane and blade interaction are expected to be slightly higher, compared to the GG turbine.

3.3 Heat Recovery

Heat recovery devices (recuperators and rotary regenerators) are used to recapture waste energy from the gas turbine exhaust. The recaptured heat energy is used to preheat the combustor inlet air, thereby reducing the energy (or fuel flow) required to achieve the desired turbine inlet temperature. Other studies on similar sized helicopter engines have shown that recuperated engines have slight advantages over rotary regenerated engines; therefore, this study addressed recuperated engines only. Of particular interest in this study was the counterflow plate-fin recuperator design. A typical airflow path is depicted in Figure 9. Improvements in heat recovery (recuperator) technology is projected to be made in materials and processes. As a result of these improvements in technology, costs will be reduced and temperature capability will be increased, relative to today's heat recovery engines. Heat recovery engines would then be viable candidates for future helicopters.

Material improvements are expected in both metallics and nonmetallics. As shown in Table 5, a nitride-dispersion-strengthened 300 stainless steel has the potential for

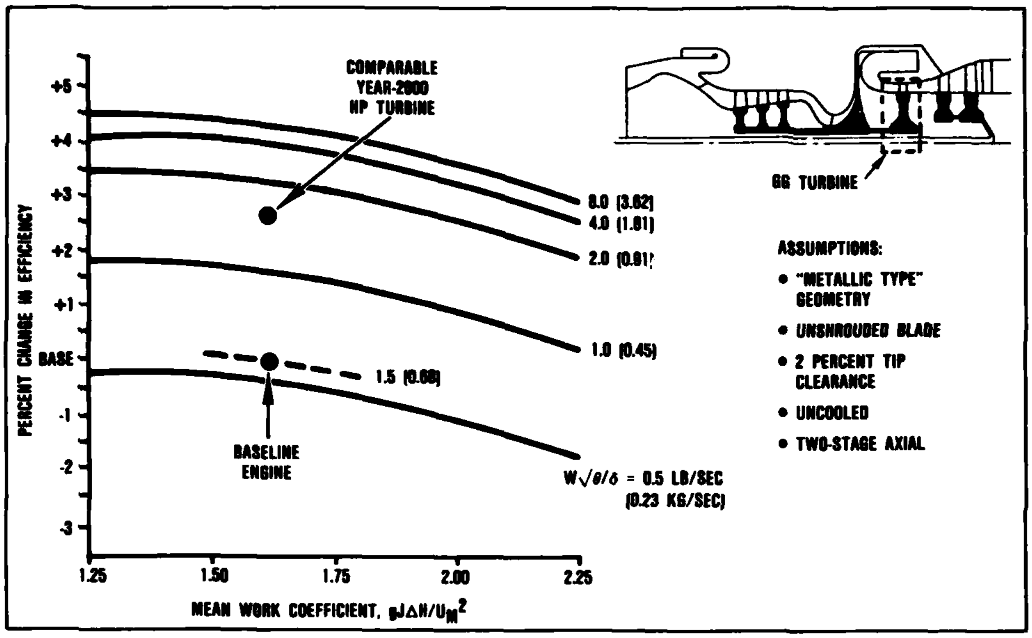

Figure 7. HP Turbine Projections.

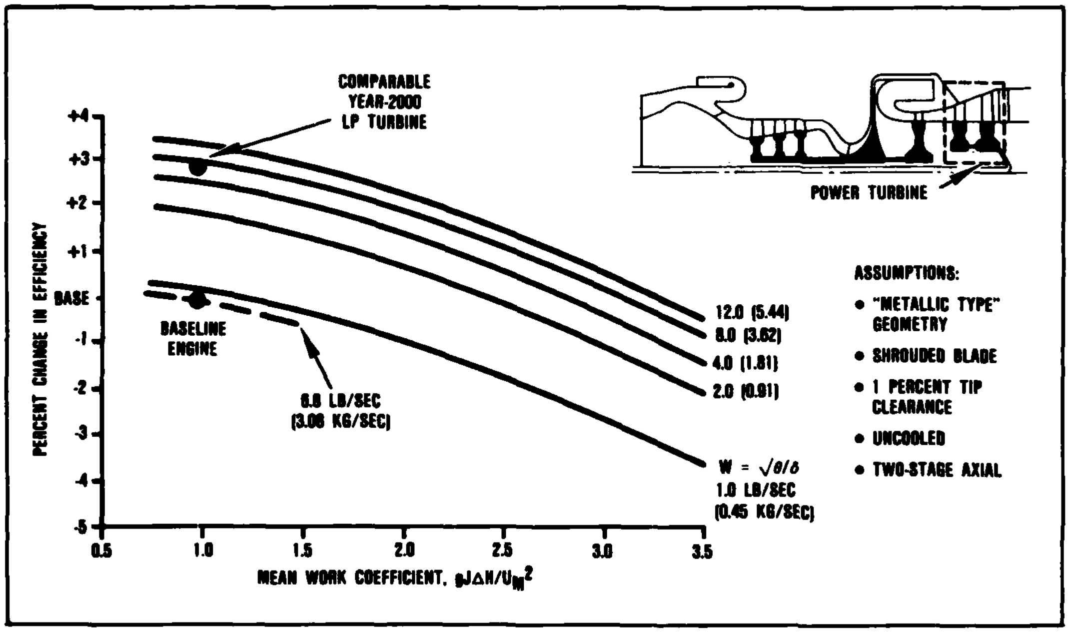

Figure 8. LP Turbine Efficiency Projections.

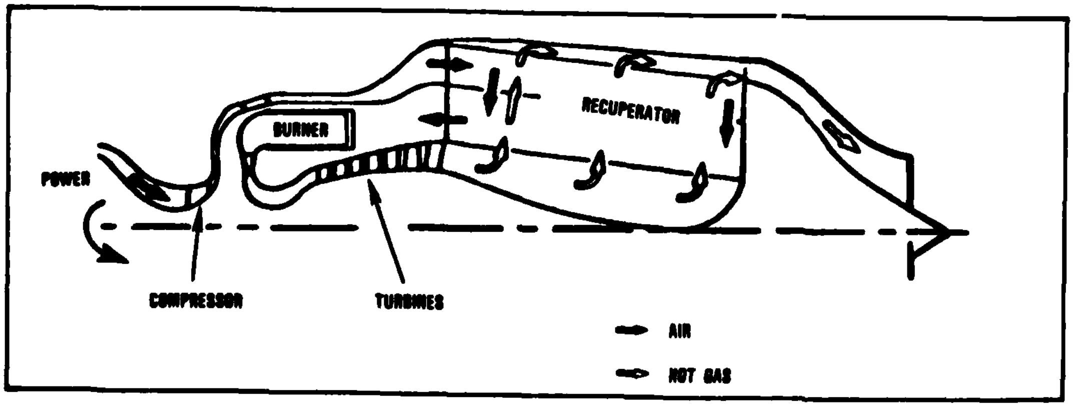

Figure 9. Typical Counterflow Recuperated Engine Flow Path.

Table 5. Recuperator Technology Projections.

Metals	
1985	**Year 2000**
Chromium-molybdenum steel	Nitride-dispersion-strengthened 300 stainless steel
Fin density: 37 fins per in. (2.54 cm)	Fin density: 37 fins per in. (2.54 cm)
Maximum temperature is 1500F (816C) for short times	Maximum temperature is 1800F (982C) to 2000F (1093C)

Ceramics	
1985	**Year 2000**
Experimental plate-fin HXs	Operational plate-fin units
Plain fins:	Offset fins:
30 fins per in. (2.54 cm) 0.015 in. (0.04 cm) thickness	35 fins per in. (2.54 cm) 0.010 in. (0.03 cm) thickness
	Maximum temperature is 2600F (1427C) to 2800F (1538C)

increasing recuperator operating temperatures from a present limit of 1500F (816C) to between 1800F (982C) and 2000F (1093C). Ceramics could further improve recuperator temperature capabilities. Inlet temperatures of approximately 2600F (1437C) will be possible. Furthermore, the 3 to 1 density advantage of ceramics is predicted to reduce overall recuperator weight by approximately 50 percent. At present, with ceramics, only experimental heat exchanger modules have been evaluated but, by the year 2000, ceramic heat exchangers are projected to be operational. Much of the potential success of ceramic recuperators lies in developing an efficient low-cost manufacturing process. Ceramic plate-fin counterflow recuperators were assumed in the study.

3.4 Mechanical Technology

With a given cycle and configuration, mechanical design constraints are dependent on projected material and aerodynamic capabilities. Life requirements were assumed constant between 1985 and year-2000 engines. Based on these assumptions, the use of empirical correlations (extended to cover year-2000 technology) can provide mechanical inputs to the cycle study in terms of turbine hub speed and AN^2 limits, TRIT constraints, and cooling flow requirements. Turbine hub speed and AN^2 are considered to be indicators of disk and blade stress, respectively.

4. ENGINE CONFIGURATION AND CYCLE EVALUATION

Traditionally, the typical parametric cycle study examines a range of key cycle variables such as turbine inlet temperature and compressor pressure ratio. Moreover, a number of simplifying assumptions are typically made. The idealized assumptions lead toward engine selection at the highest practical compressor pressure ratios and TRITs, as shown in Figure 10.

Unfortunately, an idealized study does not address the effects of compressor and turbine flow size, stage counts, materials/mechanical constraints, spool speeds, and cooling flows based on the number of cooled stages and cooling air temperature. Incorporation of these mechanical limitations in the cycle study forces the results into a number of distinct families such as those shown in Figure 11. It also eliminates many unrealistic cycle/engine configurations.

The cycle studies for the rotorcraft parametrically considered a range of potential configuration combinations in terms of TRIT; cycle pressure ratio (CPR); component types, materials, and associated efficiencies; cooling flows; pressure drops; and leakages as projected for year-2000 capabilities. Both conventional and heat recovery cycles were investigated.

4.1 Conventional Simple-Cycle Engines

Performance

Advanced simple-cycle engines were evaluated over CPRs ranging from 20 to 26:1 and TRITs from 2200F (1204C) to 2800F (1538C). Configuration options included evaluation of multistage axial centrifugal and two-stage centrifugal compressors, as well as single-stage or multistage axial turbines. Both advanced metallic and ceramic turbines were considered. The inclusion of mechanical constraints, as previously discussed, tends to group the performance results into distinct engine families. The various families can

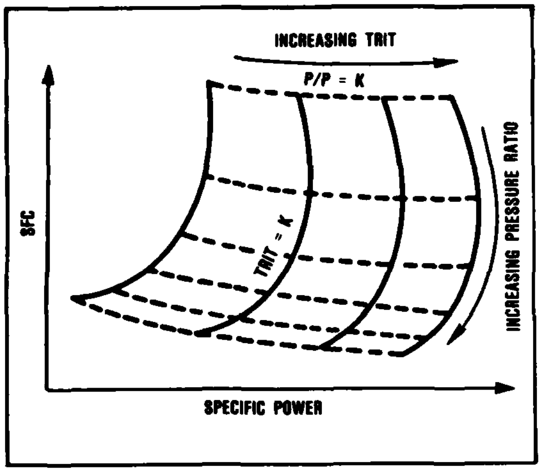

Figure 10. Idealized Cycle Study Results with Constant Efficiencies.

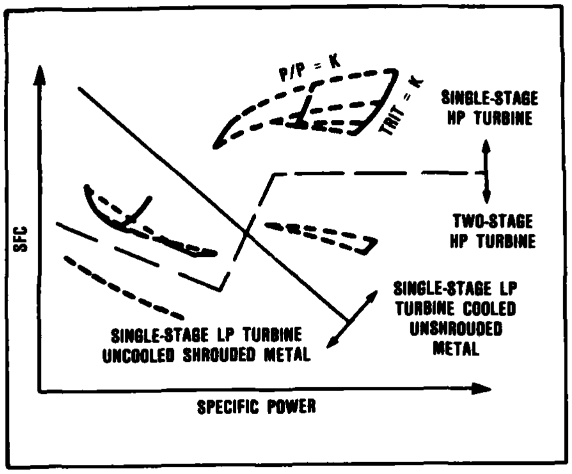

Figure 11. Cycle Study Results with Mechanical Limits.

be identified according to the turbine configuration as described in Figure 12. The cycle study performance results are displayed in terms of percent SFC and specific power relative to the baseline engine.

As shown in Figure 13, the advanced metallic engines optimize, in terms of SFC, at a TRIT of 2200F (1204C) and at a pressure ratio of 20 to 22:1. Relative to the 1985 baseline engine, an SFC reduction of approximately 12 percent is possible with this advanced cycle configured with a single-stage, cooled GG turbine and two-stage, uncooled power turbine. The higher temperature and pressure cycles result in increased turbine cooling flow requirements, which penalizes SFC.

One means of greatly reducing the need for costly cooling air is through the use of ceramics for the turbine blades and vanes. As shown in Figure 14, the elimination of turbine blade cooling flow allows higher temperature/pressure cycles to achieve better SFC and specific power. Other constraints apply, however. Both the 2800F (1538C) TRIT and 26:1 CPR cycles were eliminated from further consideration because of material temperature and turbine blade height limits, respectively. The best SFC occurs at a TRIT of 2400F (1427C) and a CPR of 24:1. This is nearly 20 percent lower than the baseline engine. Compared to the best year-2000 metallic engine, SFC is improved by 7.5 percent. Specific power is also improved by ceramics, relative to advanced metallics, by 10 to 15 percent.

Weight, Size, Cost

Engine size and weight were estimated with the Weight Analysis of Turbine Engine - Small (WATE-S) computer program, originally written by GTEC for NASA (Report No. CR-168049). The WATE program uses various mechanical, aerodynamic, material, and cycle inputs to calculate stresses, and the size and number of components to establish a power

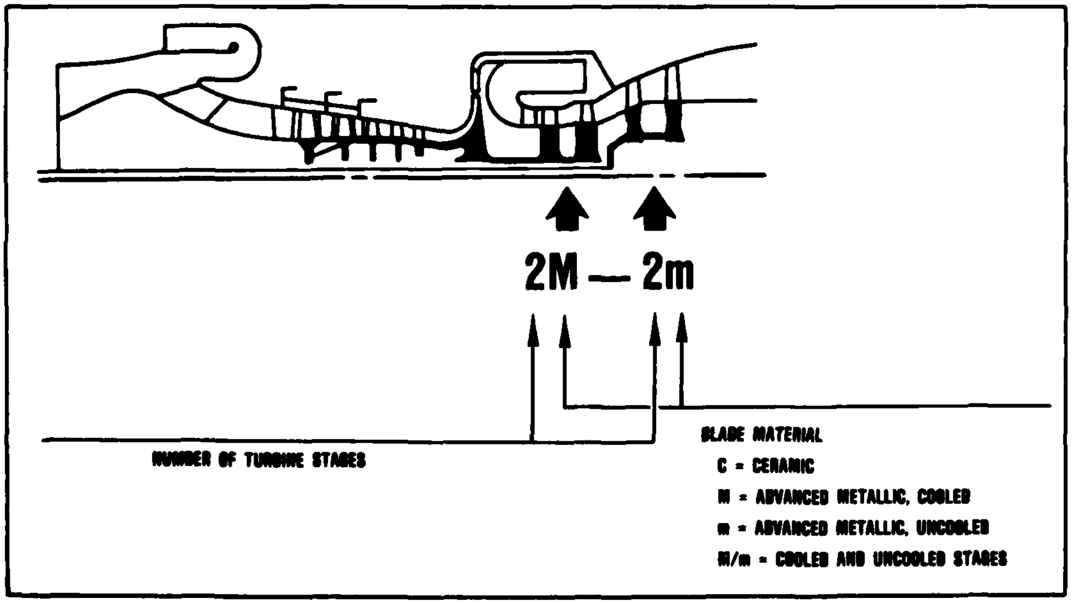

Figure 12. Legend for Turbine Configurations.

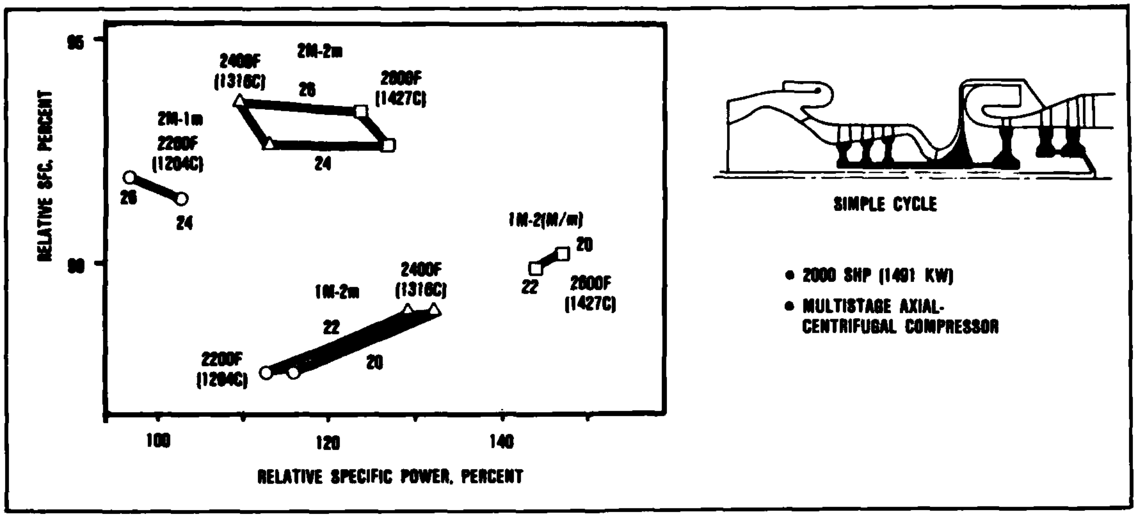

Figure 13. Performance of Simple Cycle with Advanced Metallics.

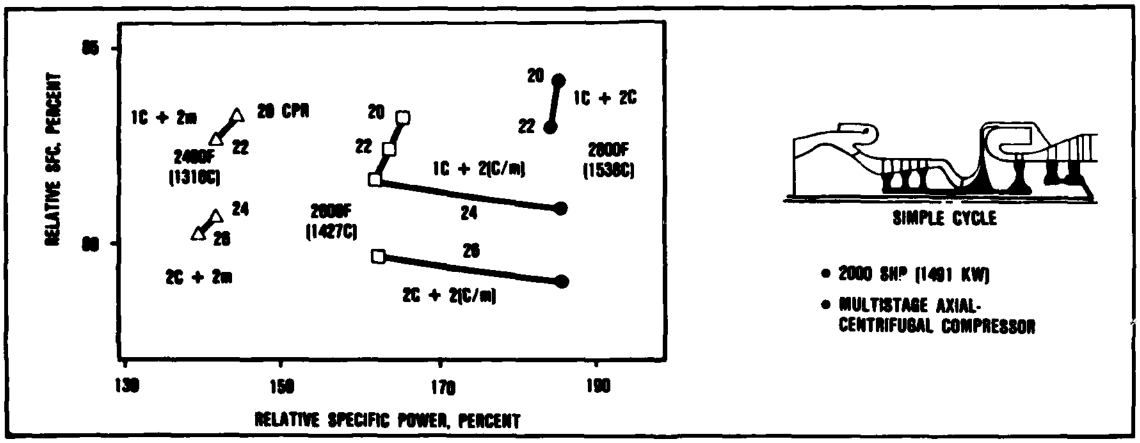

Figure 14. Performance of Simple Cycle with Ceramics.

section size and weight buildup. Additional components such as the accessory gearbox, accessories, recuperator, and associated ducting were estimated separately (based on historical data) to provide total engine size and weight.

The cost estimates for the candidate engines of this study were based on the cost of the baseline engine and on a buildup of component costs. Each engine cost was therefore estimated separately, based on technologies employed, number and type of components, TRIT, and size. The costs were projected to year 2000 and were expressed in year-1985 dollars.

Ceramics allow a significant performance advantage over advanced metallics and provide a substantial improvement relative to the 1985 baseline. However, weight, size, and cost must also be considered. A comparison of engine weight (Figure 15) indicates that the engines with ceramic turbines are lighter, by 10 to 15 percent, than those with metallics. The figure also shows that lower pressure ratio and higher TRITs tend to minimize weight in the range investigated. The cycles previously identified for superior performance do not fare as well in terms of weight. Similar trends can be found in terms of size (Figure 16). The engines with ceramic turbines are again superior to the metallic configurations, although the difference is less dramatic. Again, lower CPR and higher TRIT tends to reduce size; however, these variations are less significant.

A cost comparison is made in Figure 17. The difference between metallics and ceramics is more dramatic in terms of cost than either size or weight, particularly at higher pressure ratios. The ceramic engines are approximately 20 to 25 percent less costly. A metallic turbine cycle (2200F [1204C], 20:1) results in the least costly of the advanced metallic configurations; however, the ceramic turbine cycle at 2600F [1427C], 20:1, results in the lowest cost engine. Cost for both ceramic and metallic cycles increased with higher pressure ratios due to increases in compressor and turbine stage counts as well as the additional airflow required. In general, cost tends to decrease with increasing TRIT due to decreasing engine size. However, for a given size class, costs associated with changes in material can override the size factor. This occurs at approximately 2200F (1204C) with the metallic cycles. These engines are low in cost due to the use of DS turbine blades, which are much less costly than the SSC blades required at higher temperatures.

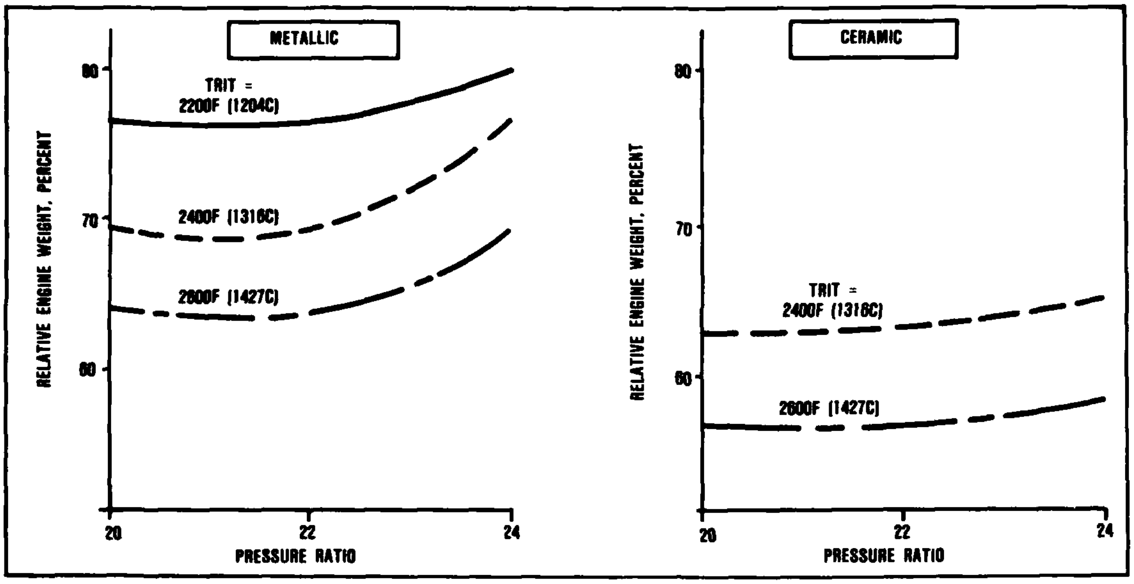

Figure 15. Engine Weight Comparison (Metallic versus Ceramic).

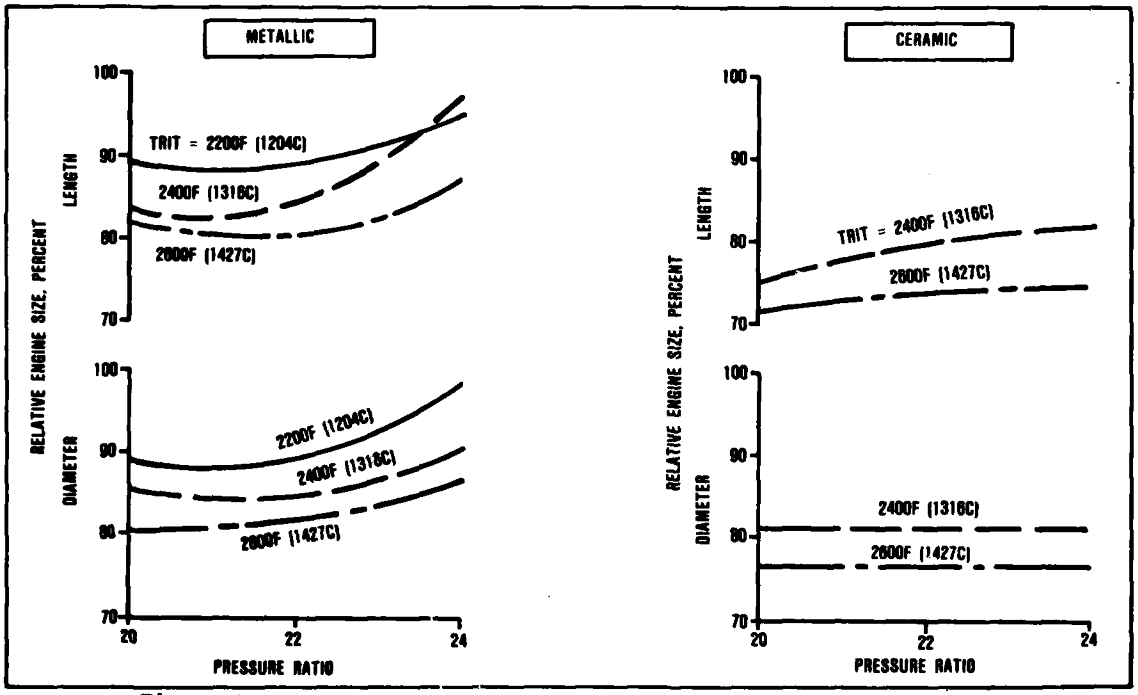

Figure 16. Engine Size Comparison (Metallic versus Ceramic).

Direct Operating Cost

Using the foregoing engine analysis, the trade factors described in paragraph 2.4 were used to define helicopter DOC for the advanced engine/helicopter systems. A comparison of DOC values (Figure 18) between the advanced metallic and ceramic engines demonstrates the expected advantage of ceramics. Ceramics can reduce DOC over 8 percent, whereas the best metallic engine can achieve only a 5 percent DOC reduction. Due to cooling penalties, the metallic engines optimize at relatively modest cycle conditions. Ceramics, on the other hand, take advantage of the benefits from higher TRIT. Over the range investigated, a fairly flat response to cycle pressure ratio is indicated by the ceramic engines, in terms of DOC. What did have a significant impact on the absolute DOC payoff achievable with the advanced cycles was fuel price. Similarly, the importance of SFC is clearly increased. The DOC results for Mission II (Figure 19) are similar to the Mission I results. They differ only in dollar magnitude, but have approximately the same percent improvement.

Because of the increase in specific power of the ceramic engine, the two-stage centrifugal compressor design looked equally attractive in terms of DOC, as shown in Figure 20. Trends in centrifugal and axial-centrifugal compressor technology need to be monitored with respect to the influence of size on optimum performance. System require-

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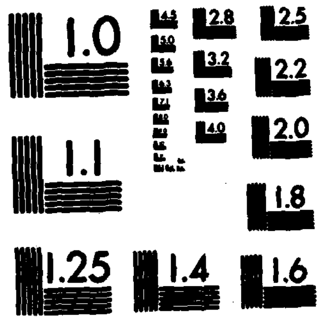

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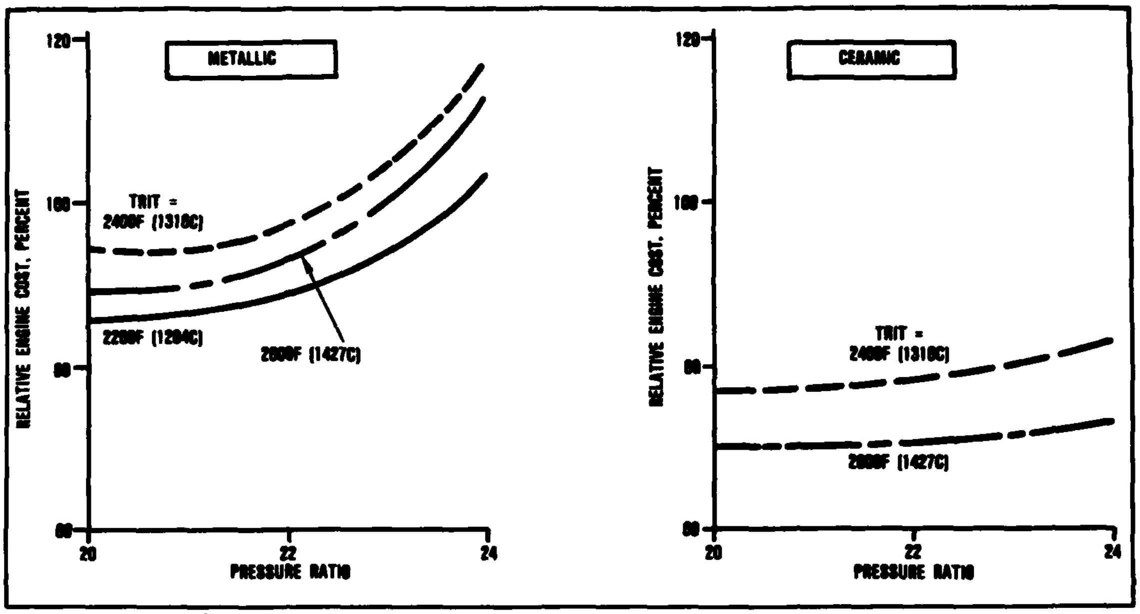

Figure 17. Engine Cost Comparison (Metallic versus Ceramic).

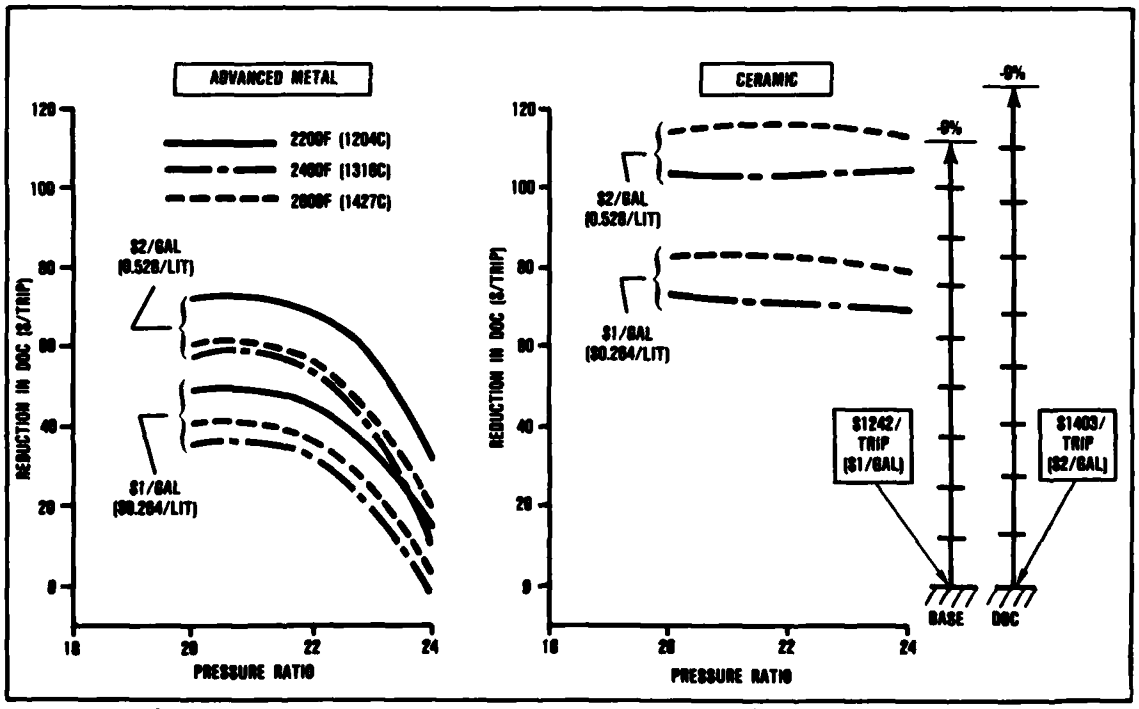

Figure 18. Mission I DOC Comparison (Metallic versus Ceramic).

ments such as FOD resistance, erosion, maintainability, cost, etc., need to be thoroughly considered in any compressor configuration selection.

Based on initial DOC results, a conventional cycle with ceramic turbines was selected for further evaluation. The selected cycle is close to the optimum engine and is representative of engines possible from improvements in technology. This cycle was later used to calibrate the results from the trade factor analysis. The selected engine is configured with a multistage axial-centrifugal compressor with a pressure ratio of 24. The compressor is driven by a single-stage uncooled ceramic axial GG turbine with TRIT set at 2600F (1427C). The combustor is also ceramic and is a reverse-flow annular design. The power turbine consists of two uncooled stages and is ceramic, as necessary, to remain uncooled.

4.2 Heat Recovery Cycle Engines

Performance

A wide range of configurations and cycle parameters were considered for the heat recovery (recuperated) cycle study. The base recuperated configuration evaluated con-

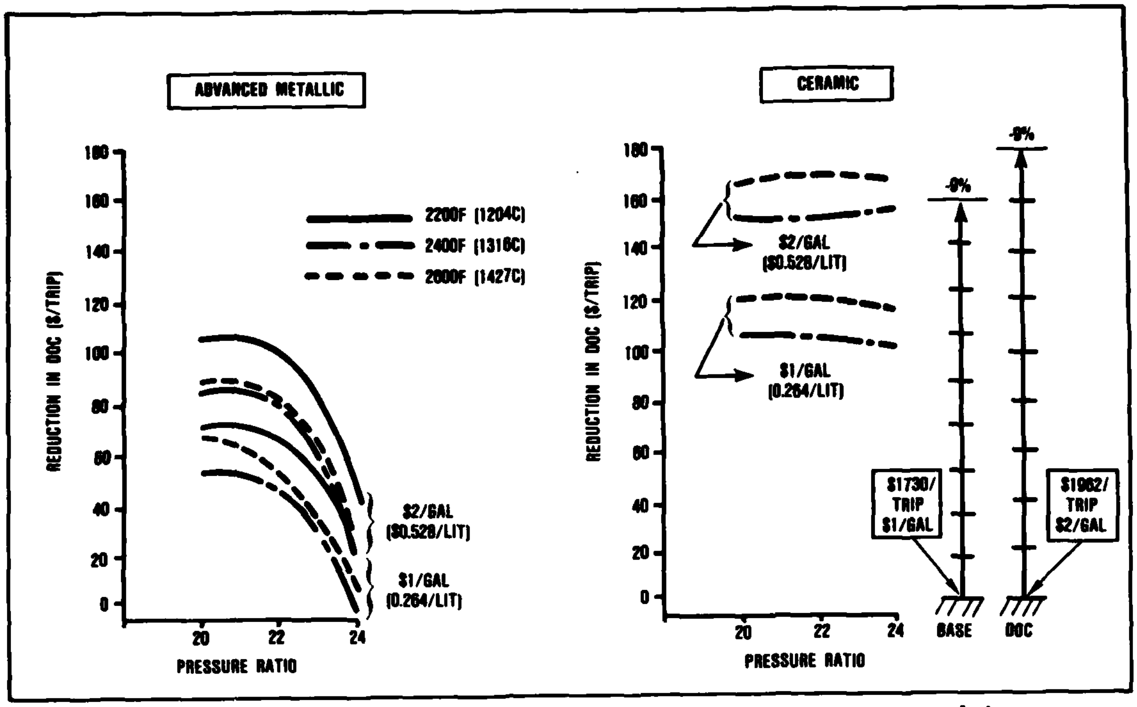

Figure 19. Mission II DOC Comparison (Metallic versus Ceramic).

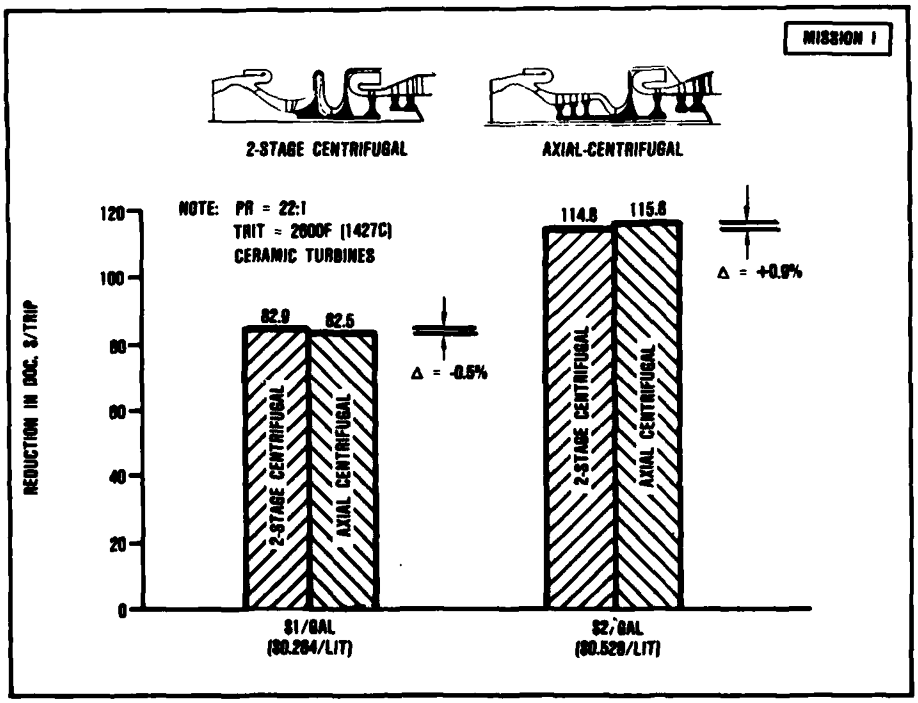

Figure 20. DOC Comparison (Axial Centrifugal versus Two-Stage Centrifugal Compressor).

sists of a single-stage centrifugal compressor, reverse-flow combustor, single-stage GG turbine, and multistage power turbine. Ceramics are used in the combustor and turbine blades and vanes. In addition to the base configuration, a number of component and material options were considered for the compressor and turbines. For the compressor, axial-centrifugal configurations were compared with the single centrifugal design. Variable diffusers were included on all configurations. Pressure ratios were evaluated from 6:1 to 12:1. For the GG turbine, advanced metallics were examined and compared with ceramics. TRITs ranging from 2200 to 2800F (1204 to 1538C) were investigated. The power turbine was also configured with axial stages and with both advanced metallic and ceramic turbines. Variable vanes were assumed for all the configurations; this maintains high recuperator inlet temperatures at part load, which maximizes the waste heat recoverable by the recuperator and provides better part-power SFC. A range of effectiveness from 0.6 to 0.85 and pressure drops from 6 to 10 percent were evaluated for the fixed-boundary recuperator.

For the ceramic turbine configuration, SFC improvements of over 30 percent were found to be possible (Figure 21) relative to 1985 baseline. Increases in specific power of up to 70 percent are also possible. The optimum compressor pressure ratio, which depends on both TRIT and effectiveness, fell between 8:1 and 10:1. As expected, SFC improved substantially with increasing effectiveness (by nearly 15 percent with effectiveness varying from 0.6 to 0.85).

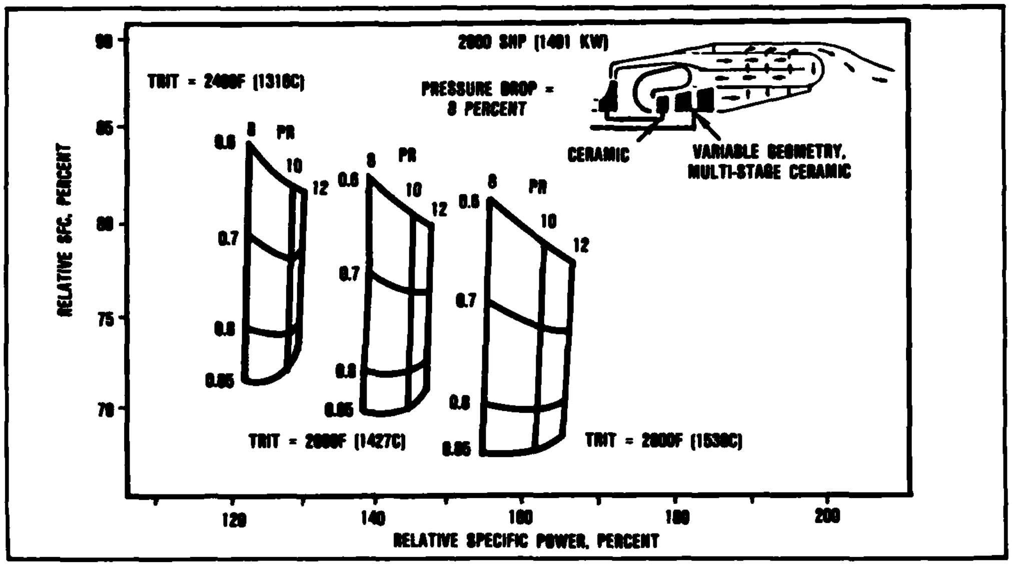

Figure 21. *Performance of Recuperated Cycle with Ceramics.*

The cycles shown assumed uncooled ceramic turbines; with a material temperature limit of 2800F (1538C), the practical TRIT limit of ceramics is considered to be 2600F (1427C) in light of expected year-2000 combustor pattern factors. The 2800F (1538C) cycles would, therefore, require some cooling, which reduces their performance. The highest effectiveness and temperature (0.85 and 2600F [1427C]) clearly resulted in the best SFC and specific power (at a CPR between 8:1 and 10:1).

One of the penalties incurred with a recuperated cycle is the pressure drop in the recuperator and the associated manifolding and ducting. To determine the impact on engine performance, a range of pressure drops (total incurred in the hot and cold streams) from 6 to 10 percent was investigated (Figure 22). Varying the recuperator pressure drop was found to have little impact on the optimum cycle pressure ratio, but results in approximately a 0.5 percent increase in SFC per percent increase in pressure drop. Therefore, the recuperated cycles inherently need high effectiveness and low pressure drop in terms of performance, which unfortunately (as discussed later) have an unfavorable impact on recuperator size, weight, and cost.

Advanced metallic and ceramic engines are compared (Figure 23) at an effectiveness of 0.8, with ceramics demonstrating an 8 to 10 percent SFC advantage and a substantial increase in specific power.

__Weight, Size, and Cost__

The impact of effectiveness and pressure drop ($\Delta P/P$) on recuperator weight, size, and cost can be found in Figures 24 and 25. As shown, weight, size, and cost increase with effectiveness and with decreasing pressure drop. Increasing the effectiveness from 0.6 to 0.85, for example, increases weight and cost by approximately 130 and 60 percent, respectively. A reduced pressure drop, in particular, has a large impact on recuperator length, which (in turn) increases overall weight and cost. Recuperator size and weight (as well as performance and cost) were estimated by the AiResearch Manufacturing Company of California, a sister division of The Garrett Corporation.

Total engine weight, size, and cost were compared for ceramic and metallic cycles. A comparison of metallic and ceramic engine weight (Figure 26) reveals that ceramics provide a substantial weight advantage (approximately 10 to 15 percent for the ceramic cycles over the advanced metallic cycles). Higher TRITs and lower pressure ratios result in reduced engine weight. A size comparison, as shown in Figure 27, reveals similar trends. The ceramic engines also have a size advantage over metallics. Size, particularly length, is reduced at higher TRITs and lower pressure ratios due to the

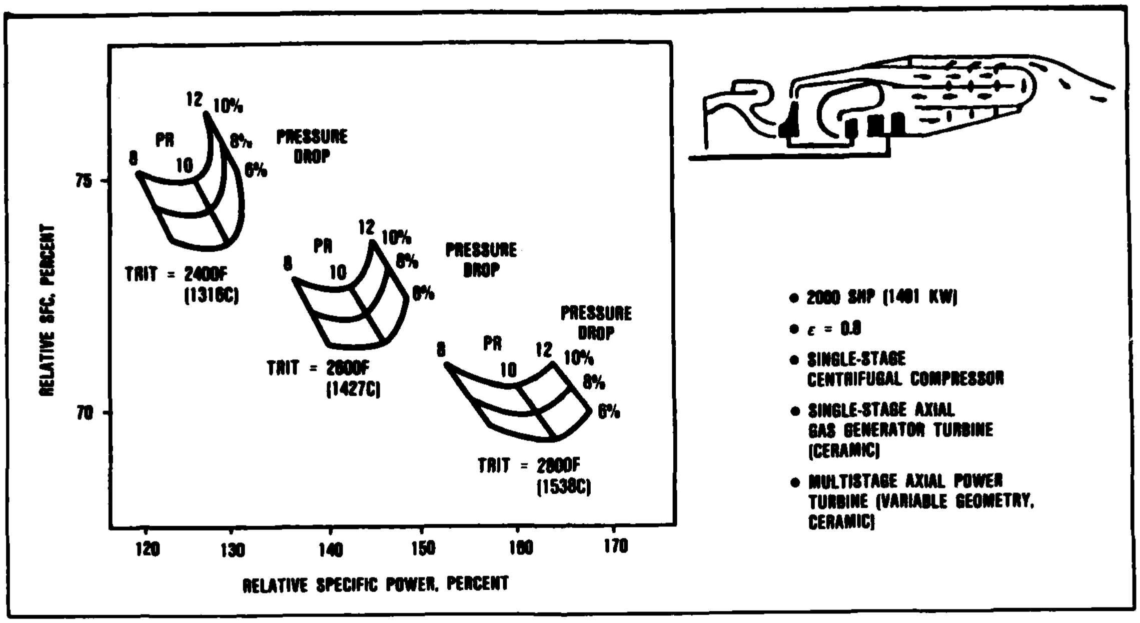

Figure 22. Performance Impact of Recuperator Pressure Drop.

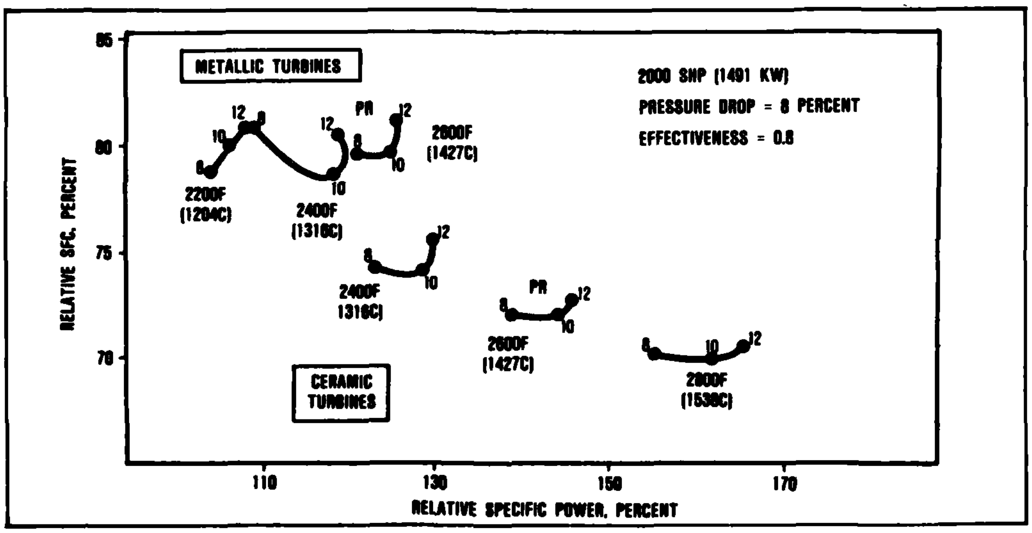

Figure 23. Recuperated Performance Comparison (Metallic versus Ceramic).

smaller airflow requirements. Finally, a cost comparison (Figure 28) shows that significant savings (nearly 15 percent) are possible with ceramics, relative to metallics. In terms of TRIT and pressure ratio, the cost trends were similar to those shown for weight and size.

Direct Operating Cost

Although high recuperator effectiveness and low pressure drop are highly desirable in terms of engine performance, a penalty is paid in terms of size, weight, and cost. As with the simple-cycle engines, the trade factors described in paragraph 2.4 were used to determine aircraft DOC.

Fuel price was shown to have a strong influence on the optimum cycle effectiveness (Figure 29). Based on the 2600F (1427C) TRIT and 10:1 CPR cycle, effectiveness optimizes at approximately 0.8 at $1/gallon ($0.264/liter) with both missions. At $2/gallon ($0.528/liter), effectiveness optimizes at just above 0.85. Since the $1/gallon ($0.264/liter) curves are fairly flat above an effectiveness of 0.8, a selection at 0.85 would satisfy both the low and high fuel price studies. Pressure drop, on the other hand, has little impact on DOC (for the range evaluated) at either fuel price or mission. Clearly, the need for pressure drops of less than 8 percent is not indicated.

The advanced metallic and ceramic engines are compared in terms of DOC reduction in Figures 30 and 31 for Missions I and II, respectively. As expected, the ceramic engines show a DOC advantage over metallics (approximately twice the reduction in dollars per trip), independent of fuel price or mission. Furthermore, cycles optimize at a pressure ratio of approximately 10:1, independent of fuel price or mission. A TRIT of 2600F

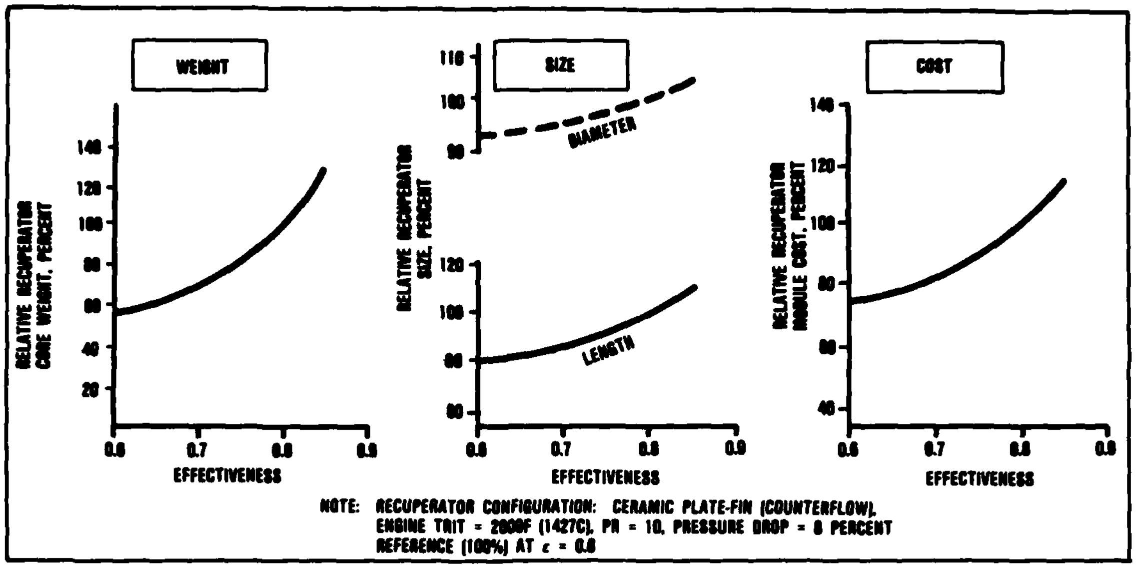

Figure 24. Impact of Effectiveness on Recuperator Weight, Size, and Cost.

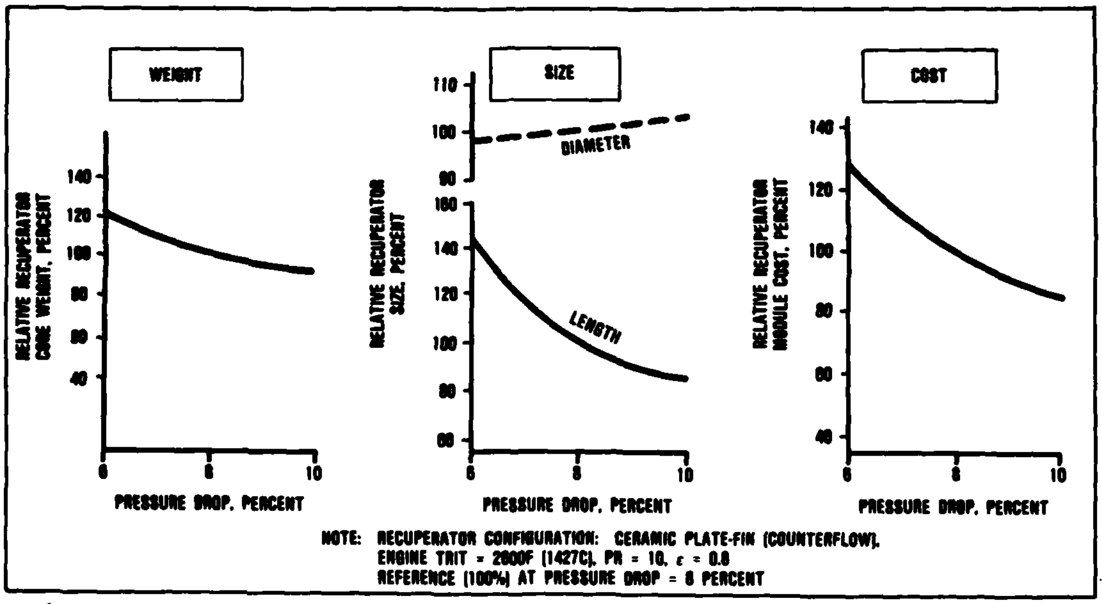

Figure 25. Impact of Pressure Drop on Recuperator Weight, Size, and Cost.

(1427C) at the 10:1 CPR provides the greatest DOC reduction. Mission length and fuel price do substantially affect the magnitude of potential DOC savings, with changes in DOC (at the optimum PR point) ranging from a decrease of approximately $10 per trip (-0.8 percent) to $140 per trip (-7.1 percent).

Also considered was the impact of compressor configuration on DOC. Two configurations, an axial-centrifugal and the base single-stage centrifugal compressor, were compared. Only small differences in DOC were noted between the two configurations, with the axial-centrifugal showing a slight advantage (0.5 percent at a pressure ratio of 10). Based on the small DOC advantage demonstrated, the added complexity of the axial-centrifugal compressor was not considered warranted.

Based on the DOC results, a recuperated cycle with ceramic turbines was selected for further evaluation. The selected engine is configured with the base single-stage centrifugal compressor with a variable diffuser and a pressure ratio of 10:1. A single-stage, uncooled axial ceramic turbine (TRIT = 2600F [1427C]) that drives the compressor is tucked beneath a ceramic reverse-flow combustor, which results in a very compact gas generator. The uncooled power turbine is a three-stage design with variable stators. The recuperator, which is ceramic, uses a counterflow plate-fin design with an effectiveness of 0.85 and a pressure drop of 8 percent.

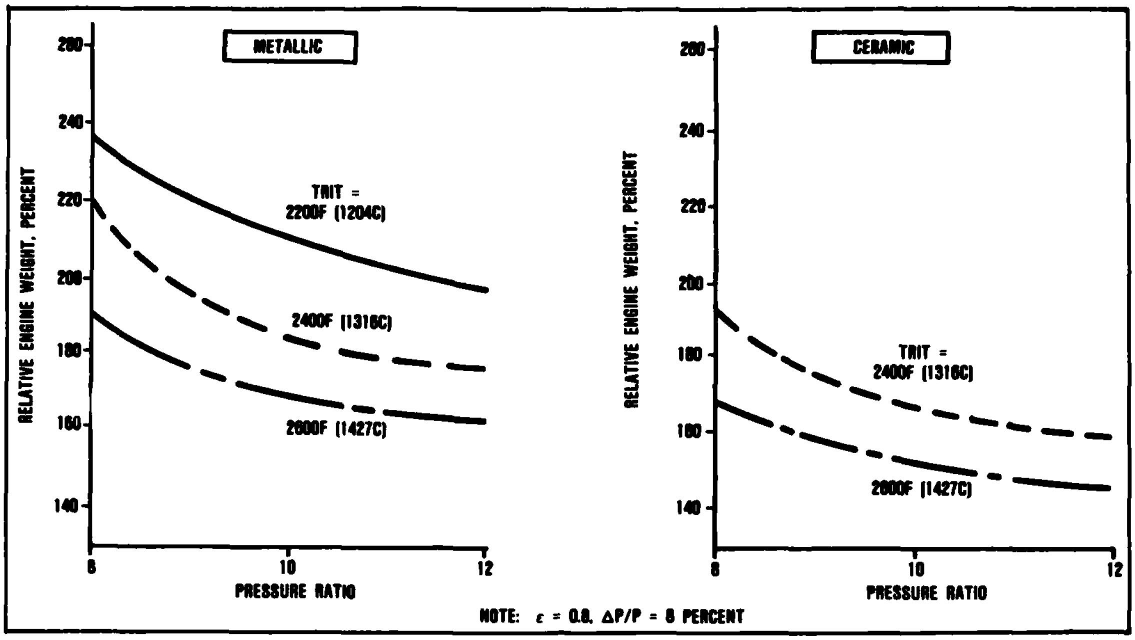

Figure 26. Engine Weight Comparison - Recuperated (Metallic versus Ceramic).

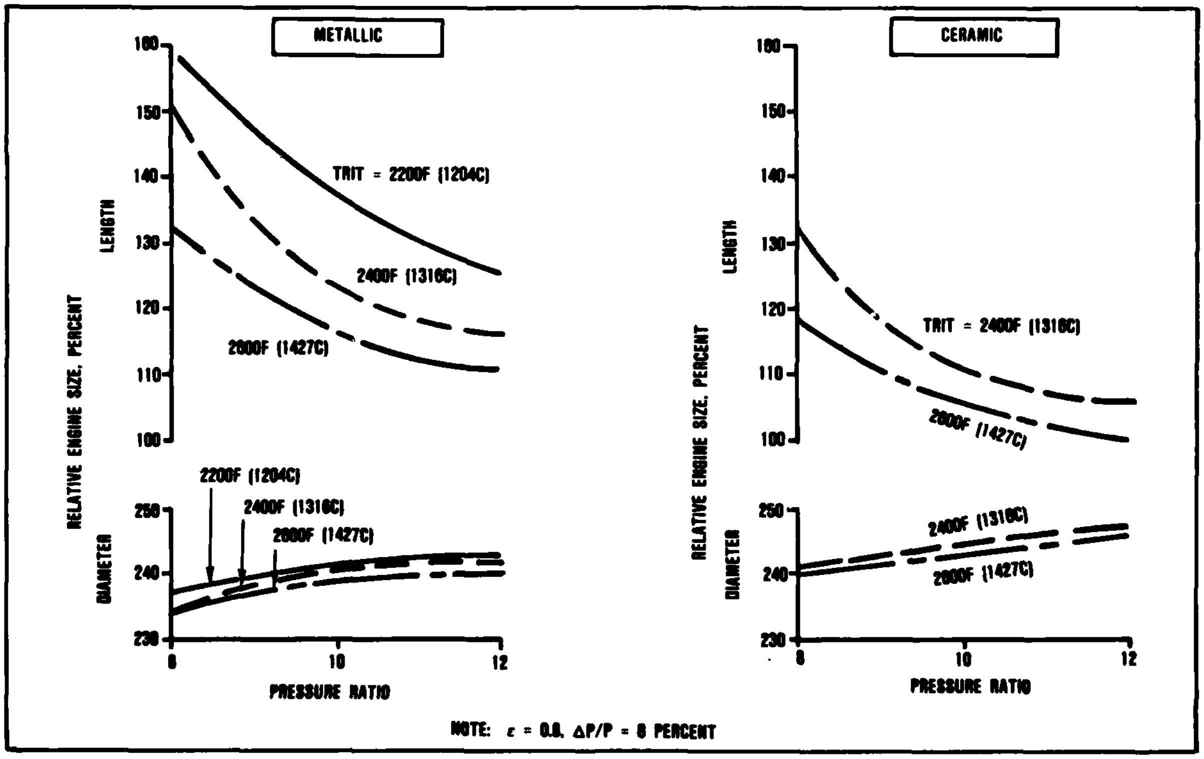

Figure 27. Engine Size Comparison - Recuperated (Metallic versus Ceramic).

5. SYSTEM PERFORMANCE EVALUATION

The selected cycles, one simple and one recuperated (as previously described), were evaluated in greater detail. The trade factors provided a good means of evaluating a large number of engines; however, additional analysis was required to fully account for part-power performance characteristic changes. A matrix of off-design performance conditions and power settings was generated to support the system analysis. A comparison of sea-level, static load lines (Figure 32) indicates that the recuperated engine has part-power SFC superior to both the advanced simple cycle and the baseline engine. The inclusion of the variable diffuser and variable power turbine on the recuperated cycle pays off significantly in part-power SFC, despite a small design point penalty.

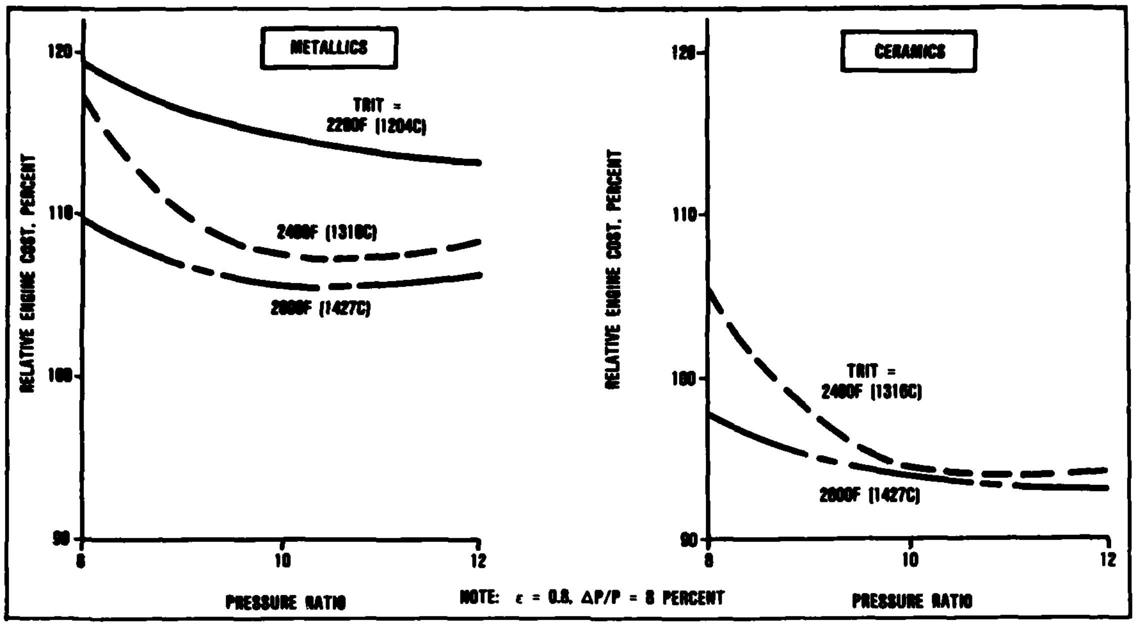

Figure 28. Engine Cost Comparison - Recuperated (Metallic versus Ceramic).

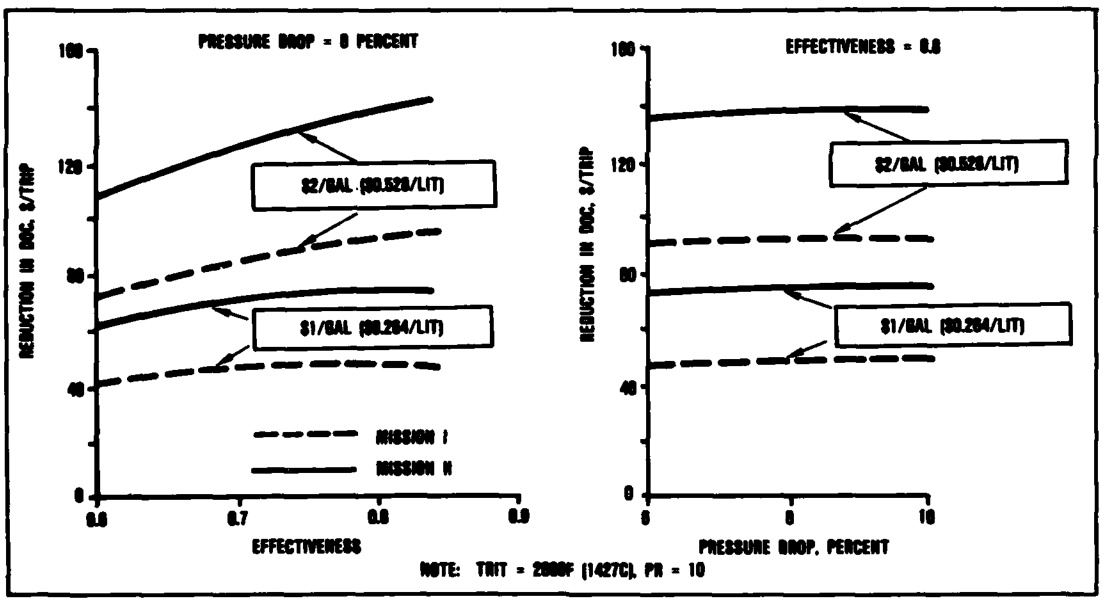

Figure 29. Impact of Effectiveness and Pressure Drop on DOC.

The resulting rotorcraft changes were summarized in Figure 33 (Mission I). As shown, the key change was the reduction in mission fuel burn, 48 percent for the recuperated cycle and 21.1 percent for the simple cycle. Takeoff gross weight has also been reduced by 3.1 and 1.4 percent for the simple and recuperated cycles, respectively. Mission II results are similar.

Figure 34 shows the recuperated cycle to be superior in DOC to the simple cycle at the high fuel price for both missions. Despite the negative impact of weight, size, and cost incurred with the recuperated engine, the SFC benefit is large enough to override these aspects, if fuel price is high enough. Relative to the baseline engine, the reduction in DOC for simple and recuperated cycles are approximately equal at the low fuel price. The year-2000 recuperated cycle has a distinct advantage at the high fuel price.

Technology Benefit Analysis

To estimate the benefits derived from each of the selected technologies, each technology was isolated using an approach which involved removing one technology from the year-2000 engines, setting new cycle limits as necessary, and generating new engine SFC, weight, diameter, length, and cost data. Finally, trade factors were applied to the new engine parameters to determine the new DOCs. Comparing the resultant DOC values to the

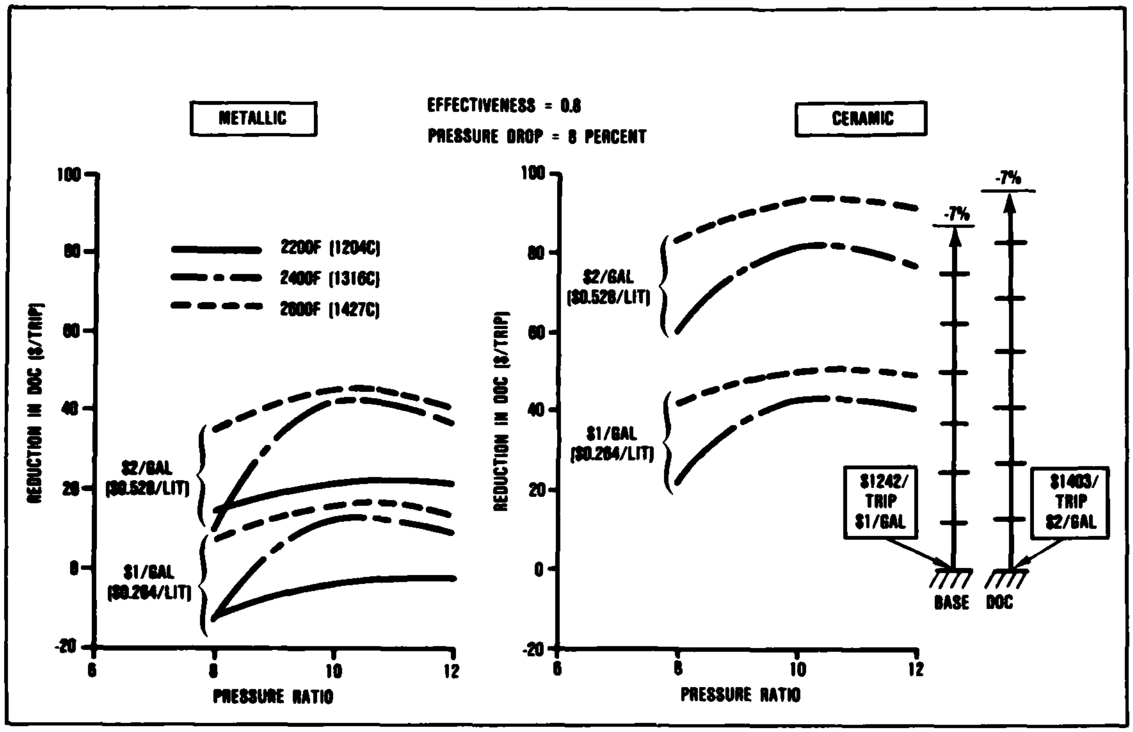

Figure 30. Mission I DOC Comparison (Metallic versus Ceramic).

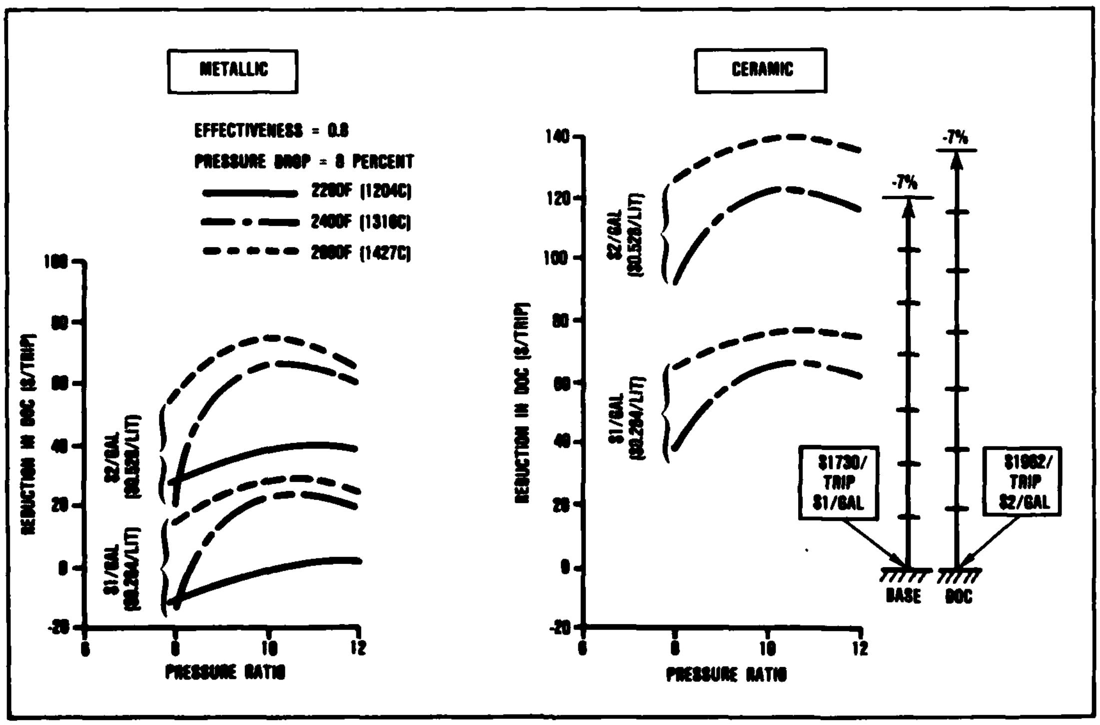

Figure 31. Mission II DOC Comparison (Metallic versus Ceramic).

year-2000 DOC (year-2000 engine with all the selected technologies) identifies the improvement derived from that technology. The selected technologies are not independent from one another and are therefore not additive.

These technologies, which are identified below (unranked), have shown some benefit in terms of SFC, weight, size, and/or cost. Several of these technologies, such as metal matrix shafts, are difficult to quantify in terms of DOC, but their use is considered beneficial or necessary to meet engine design goals.

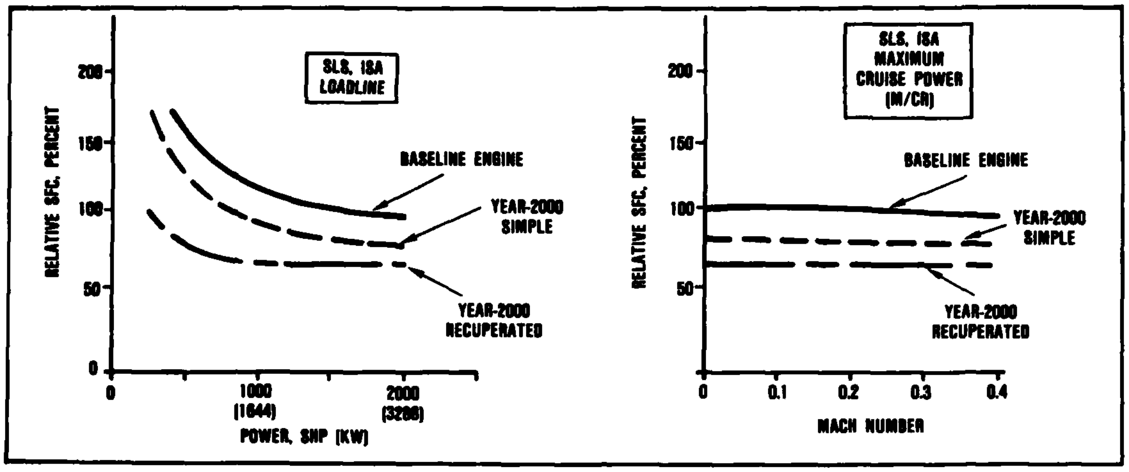

Figure 32. Off-Design Performance Comparison.

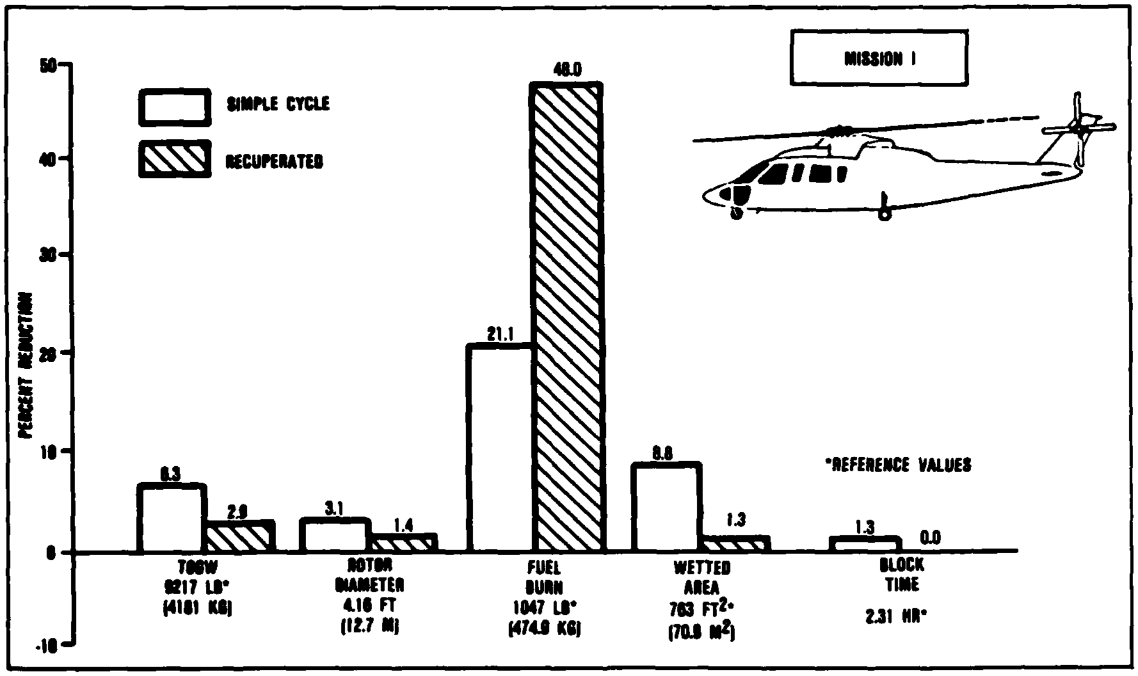

Figure 33. Comparison of Rotorcraft Changes (Simple versus Recuperated Engines).

- Component performance (aero)
  - Compressor efficiency
  - Turbine efficiency
  - Combustor ΔP/P
- Materials
  - Ceramics (for turbines, combustors, recuperators)
  - Ni_3Al disk (turbines)
  - Aluminum powder metal alloy (compressors)
  - Cast titanium (compressors)
- Combustor
  - Low pattern factor
  - High heat release rate
- System technologies
  - Metal matrix shafts
  - Noncontact face seals/brush seals
  - High-temperature lubricants

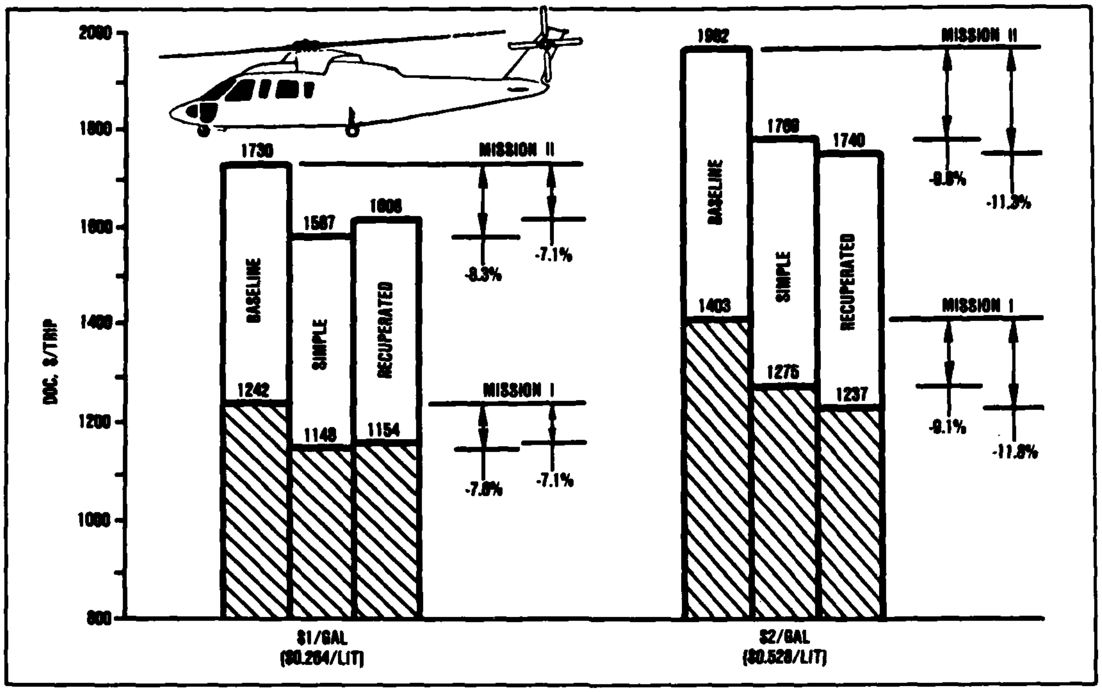

Figure 34. DOC Comparison (Detailed Analysis) Simple versus Recuperated Engines.

Hot-end materials were found to have the greatest DOC impact (Figure 35) for the simple-cycle technologies. For example, ceramics for application in turbine airfoils contribute over half of the overall DOC improvement projected for the simple-cycle engine. Also, the use of ceramics in combustors showed a noticeable DOC benefit. Compressor and turbine efficiency improvements also show significant DOC benefits, contributing up to one-fourth of the total, dependent upon fuel price. As previously stated, the selected technologies are not independent from one another and are therefore not additive.

The same technologies, with the addition of the recuperator technology, were evaluated for the selected recuperated engine. As with the simple cycle, ceramics provide the greatest payoff in DOC. In turbine applications, half of the overall DOC improvement can be attributed to ceramics (Figure 36). Ceramics in recuperator applications

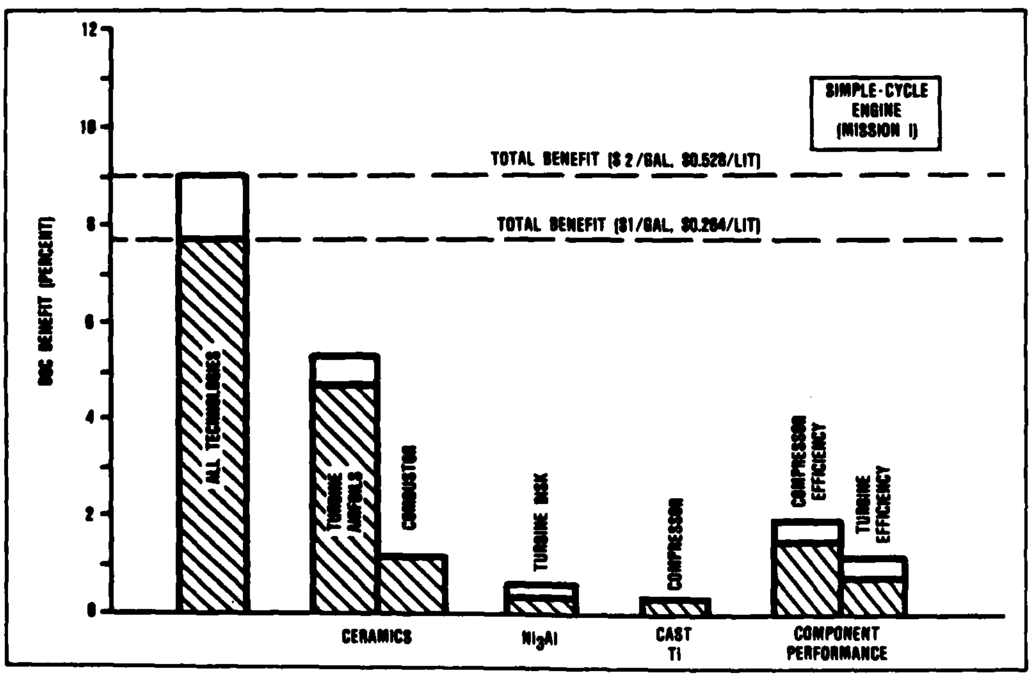

Figure 35. Projected DOC Benefits (Percent) for Isolated Technologies (Simple Cycle).

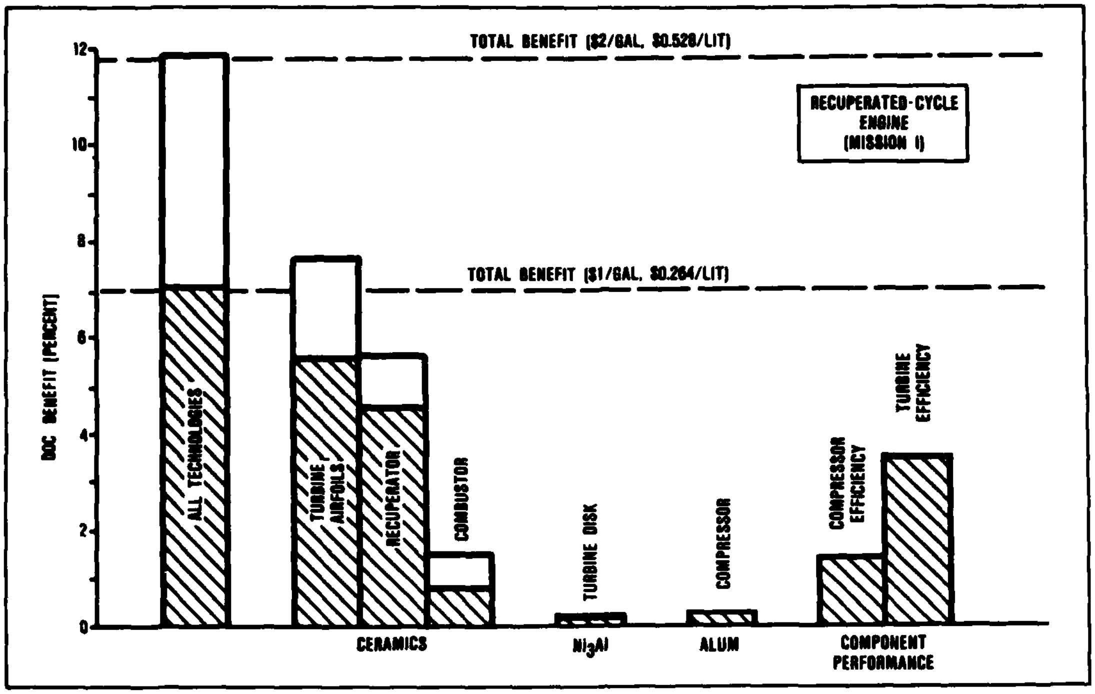

Figure 36. Projected DOC Benefits (Percent) for Isolated Technologies (Recuperated Cycle).

also provide a substantial payoff, contributing nearly half of the overall improvement. Ceramics in combustor applications also demonstrate a noticeable influence on DOC. Compressor aerodynamic improvements show benefits similar to those demonstrated with simple-cycle engines. Turbine aerodynamic improvements, however, show a much greater benefit. The expected improvements in turbine variable stator technology (assumed for all the recuperated cycles) has a significant impact on turbine stage count and performance.

CONCLUSIONS

Based on the assumptions made and the results of this study, the following conclusions were reached:

- Materials and aerodynamic technologies set future engine configuration and performance. Improvements in materials technology allow higher turbine rotor inlet temperatures, which permit better performance and smaller engines at a given power level. They also allow higher stresses and speeds, which result in smaller, lighter engines with fewer stages.

- Of the engine parameters, SFC has the largest influence on helicopter DOC. This is followed by engine cost. The influence of SFC ranges from one and one-half to three times greater than the influence of engine cost, depending on the mission and the fuel price. The influence of engine weight is less than one-fifth the influence of engine cost. The effect of engine length and diameter are negligible on DOC.

- From a thermodynamic standpoint, increasing the pressure ratio and the temperature increases cycle efficiency. However, engines with the highest pressure ratio and turbine inlet temperature did not result in the lowest DOC when stage count and aerodynamic loading are taken into account. Therefore, stage count and aerodynamic loading are important in determining the optimum engine cycles.

- Using technology projections for the engine, DOC savings for future engines can be quantified. This study identified DOC savings ranging from 7.1 to 11.8 percent, depending on the engine cycle, mission, and fuel price.

- Both simple and recuperated cycles are possible candidates for future helicopters at $1/gallon ($0.264/liter), the lower fuel price. At $2/gallon ($0.528/liter) the recuperated engine has a clear advantage in DOC.

- The relative importance of the engine technologies has been identified, with advancements in ceramic materials having the highest payoff.

HELICOPTER AIR INTAKE PROTECTION SYSTEMS

by

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Summary : Air Intake Protection System

The primary function of an engine air intake is to provide air supply with minimum pressure drop and distortion. Static or dynamic air intakes are selected according to both the type of engine air intake and the helicopter missions envisaged.

The other function of an engine air intake is to protect the engine against foreign object ingestion, sand erosion and the various atmospheric agents such as rain, snow, ice. As a general rule, the air intake protection systems are all the more penalizing as regards performance as they are efficient. The best tradeoff has therefore to be found between both funtions.

This document first deals with the dimensioning and test criteria for each function of an air intake and then gives the various development stages of the SA 366 G1 Dauphin air intake as an example.

In our opinion, the sand protection system should be fitted as an optional item, since it results in a too penalizing installation loss ; its efficiency should exceed 92 % with A.C. coarse sand and, to our knowledge, only the vortex systems can achieve such a performance level. Protection against ingestion of foreign objects can be combined with ice protection by the installation of FOD screens dimensioned for such a purpose ; this passive solution presents the advantage of being very little vulnerable for various cases of failure and providing the lowest weight increase and installation loss. Snow protection does not always require active systems but imposes special constraints for the air intake design, mainly for lateral air intake engines.

1 OPTIMIZING AIR INTAKE DESIGN

The ongoing effort to obtain higher airspeeds and lower fuel consumption has led AEROSPATIALE to optimize helicopter air intake design in order to obtain the best performance tradeoff between hover and forward flight.

1.1 Air Intake Design Criteria

An air intake design must meet not only engine-specific requirements (operating characteristics, performance and protection) but also helicopter aerodynamic requirements. These include :

1.1.1 Installation power losses

The installation losses are defined as the differences between the powers noted on test bench (ideal air intake) and those noted in flight. They are attributable to the air intake duct pressure drop, to the pressure value at the intake point, to excessive temperature or pressure distortion in the compressor intake plane, as well as to possible reentry of air from aircraft warm air sources : the oil cooler airstream, or engine exhaust gases recycled through the main rotor downwash or directly from the tail pipe in hover flight

1.1.2 Fuel consumption penalties associated with the above engine power losses.

1.1.3 Engine operating problems (compressor stalling, surge).

Excessive pressure or temperature distortion at the engine air intake can lead to operating problems resulting in momentary power loss and possible engine overheating. The sudden onset of these conditions may affect flight safety, especially in low-altitude hover. Proper engine operation must therefore be ensured with a sufficient safety margin to prevent the occurence of such problems.

1.1.4 Increased aircraft drag

A poorly optimized air intake configuration with a large cross sectional area and a poorly lip design may produce cowling boundary layer separations responsible for increasing aircraft drag by up to 10 %.

1.1.5 Protection against ingestion of foreign objects and atmospheric moisture (rain, snow or ice).

1.2 Air Intake Design

Two broad categories of air intakes may be defined. With a "static" air intake, the plane of the intake duct is parallel with the general airstream flow direction. "Dynamic" or "ram" air intakes, on the other hand, face into the airstream.
Air intake operation differs in hover and in forward flight. In hover, the airflow in the intake duct is generated by engine suction pressure throughout the surrounding space. In forward flight, the helicopter airspeed is largely predominant over the type of airflow set up in the air intake.

1.2.1 Static Air Intakes (Figure 1)

Static air intakes have a number of advantages, including simple design, light weight, relative immunity to ingestion of foreign objects, and reduced aircraft drag resulting from better integration in the fuselage profile. When properly designed and optimized, a static air intake is capable of recovering up to 50 % of the ram air pressure in forward flight. However, the pressure distortion index in the compressor plane and the hot air intake hazard are higher than for a ram air intake.

The optimum configuration is thus defined by a large curvature radius on the upstream edge of the intake to prevent airstream separation and to recover a fraction of the dynamic pressure at high speeds. Provided the overall aircraft design avoids hot air reentry, this configuration represents an excellent compromise from the standpoint of aircraft performance.

1.2.2 Ram Air Intakes (Figure 2)

Ram air intakes are capable of recovering the dynamic pressure in forward flight, and ensure very low pressure distortion in the compressor intake plane. Optimum design is determined by :

. the forward position of the air intakes relative to the cowlings, which often requires long intake ducts ; the inherent duct pressure drop is then compensated in hover by the absence of any recycled hot exhaust gases, and in forward flight by the recovered dynamic air pressure and non-absorption of the boundary layer.

. the air intake lip profile, which must be sufficiently thick and shaped to prevent airstream separation in hover or in forward flight.

2 AIR INTAKE PROTECTION

Helicopters may be required to operate under a wide variety of conditions, including maneuvering on unprepared terrain, low-altitude or IMC flight in the lower layers of the atmosphere. As a result, the air intakes must be protected against foreign objects and the effects of atmospheric moisture (rain, snow or ice). However, the protection system adopted must not significantly affect engine performance.

2.1 Air Intake Protection Requirements

2.1.1 Foreign Objects

The downwash from a helicopter rotor in low-level hover may raise up quantities of matter liable to be ingested by the engines (twigs, straw, dead leaves, snow or sand). Moreover, the engine must continue to operate following hail or bird strikes.

Impact weights and velocities for civil certification are specified by FAR 33 § 77 or by BCAR C4-6 § 19 & 20 (JAR E). The JAR E requirements are generally used in Europe for military aircraft.

2.1.2 Atmospheric Moisture

The principal forms of atmospheric moisture are the following :

. rain, for which engine ingestion must be limited to no more than 4 % of the engine intake airflow mass (FAR 33 § 77 and BCAR C4-6 § 20 (JAR E)).

. falling and blowing snow, for which acceptable compliance margins have recently been specified to increase the safety factor of the demonstration margins. In addition to verifying air intake operation at various temperatures in different types of snow, a 90-minute flight is required in extreme precipitation conditions (visibility less than 400 meters without fog, i.e. approximately 1 g/m³) including 5 minutes of hover in recirculating snow and one hour of cruising flight : the remaining time corresponds to taxiing or ground standby operations.

. ice : icing atmospheric conditions are conventionally defined by FAR 25 Appendix C or FAR 29 Appendix C, and completed by requirements concerning ground operation in icing fog (FAR 29 § 1093b2). Another definition of icing atmosphere was recently proposed for helicopters in the 0-10000 ft altitude range, in Advisory Circular 29-2 § 386.

2.2 Air Intake Protection on AEROSPATIALE Helicopters

2.2.1 Standard Version

The objective of the standard air intake protection system installed in the basic versions of AEROSPATIALE helicopters is to provide a simple system ensuring protection against foreign objects with the exception of sand. This solution meets certification requirements and ensures satisfactory protection in routine operation.

Sand protection is limited to specific operating conditions in desert regions, requiring the use of sand filters.
These filters are not really required on all production aircraft since they involve weight, complexity and performance penalties.

The system used consists of a braided stainless steel screen at the entry of the engine air intake duct (Figure 3). On certain aircraft configurations, this screen may be equipped with a bypass provision.

2.2.2 Sand Protection

Specific sand protection is required for operation in desert regions (generally in tropical climates). The air intake is then equipped with a high performance filter system compatible with the airflow volume of a helicopter turboshaft engine. The barrier filters used on the first AEROSPATIALE helicopters were quickly supplanted by inertia type particle separators : barrier filters can quickly become clogged, causing the bypass valve to open and thus supplying the engine with unfiltered air, whereas inertia separators are capable of operating for unlimited time periods.

The system adopted is based on a small vortex element (Figure 4) with a mass flowrate on the order of 4-5 g/sec. This concept allows a filter system to be developed in a short time for a specific type of aircraft (Figure 5) with suitable efficiency (approximately 92 % with AC Coarse sand of the type raised by the rotor bl ast). Sand is removed by a suction system inside the filter unit. On the first models, the suction pressure was generated by a Venturi nozzle supplied with engine bleed air ; this function is now generally ensured by an electric fan to avoid further penalizing engine performance by adding the effects of bleed airflow to the losses inherent in particle separators.

The sand filter also protects the air intake against ingestion of foreign matter, icing, rain or sudden intake of a large quantity of snow.

2.3 Operating Principle

2.3.1 Foreign Objects

The screen prevents ingress of foreign objects larger than the mesh dimension (generally 4.75 mm). Possible entry of foreign objects through the bypass is minimized by locating it aft so that the fraction of the engine airflow through the bypass is nil in normal operation.

Air intake screen strength is substantiated by hailstone and bird impact tests in compliance with certification requirements. With regard to bird strikes, it may be noted that the protection provided exceeds certification requirements, as the residual screen deformation after impact does not result in a drop in performance as tolerated by the regulations. This eliminates the need for direct engine ingestion tests and thus avoids penalizing engine design to meet these requirements. However, the screen mounts and stiffeners, if any, must be designed to ensure progressive energy absorption without perforation of the screen on impact.

2.3.2 Rain

Apart from the engine flameout limit, the sensitivity of a powerplant system to flight in falling rain depends on the extent to which the air intake collects fuselage runoff water. The first requirement is therefore to prevent the design of the air intake and its environment from favoring water drainage towards the engine. The intake screen has a favorable effect on concentrated water masses : it has been demonstrated that the engine flameout threshold on sudden ingestion of a given quantity of water may be doubled by the presence of a protective screen.

In certain cases, drip channels may be used to drain fuselage runoff water away from the air intake.

2.3.3 Snow

Two types of problems may arise with helicopter air intake operation when flying in falling snow :

- clumps of snow may become detached form the fuselage and enter the air intake, resulting in engine surge of flameout,
- snow may accumulate and turn into ice inside the airframe or engine portions of the air intake duct ; if a piece of ice then becomes detached it may result in engine surge, flameout or compressor damage.

The frontal screen protects then engine against masses of snow by breaking them up on impact, resulting in less sudden ingestion. The effectiveness of this system may be enhanced by placing a second screen 10 to 20 mm behind the outer one in critical areas. As regards the second hazard the external screen is not fully protective since snow (ice crystals) does not adhere to it ; however, installing a screen as close as possible to the compressor inlet prevents the ingress of ice fragments that may have formed in the air intake duct.

2.3.4 Icing

In icing atmosphere, the air intake screen provides natural engine protection. The air drawn into the engine in icing conditions is laden with supercooled water droplets which remain liquid at temperatures below the freezing point. As they pass through the screen, a fraction of these droplets are trapped by the wire mesh, and tend to "grow" on the wires, gradually reducing the flow cross section. Depending on the local air velocity and water concentration, the droplet diameter and the temperature, the screen openings may be completely obstructed or may continue to allow air to flow. Two catchment mechanisms may be observed on the screen.

. High Speed Flight (> 70 knots)

The flow tube supplying the air intake is concentrated, resulting in the passage of a large volume of air (and therefore of water) through a small screen surface area. Under these conditions, the portion of the screen affected quickly becomes clogged and the engine continues to be supplied through the "clean" portion of the screen. The airstream is deflected to pass through this zone, separating out the water droplets before they strike the screen. As a result, certain areas of the screen do not freeze up (Figure 6). This "ice shield" forms quickly, so that few supercooled water droplets enter the air intake duct.

. Low Speed Flight (< 30 knots)

The flow lines are more widely spaced, and screen obstruction is not only slower but more uniform (Figure 7). If the airflow per unit area is not too high, the screen may remain porous, and the ice buildup on the wire mesh may cause sufficient deflection to prevent complete obstruction. Proper engine airflow may be ensured by suitable screen dimensions or by installing a bypass duct.

The natural protection offered by ice buildup occurs more slowly than at high speeds, so that greater amounts of ice penetrate into the air intake duct. The engines must be capable of ingesting the resulting ice quantities without malfunctioning.

2.4 Effects on Engine Operation

The presence of an air intake screen results in a pressure drop of about 1 mbar in hover flight and, with a ram air intake, about 6 mbar in forward flight. This represents a power loss of between 0.2 and 1 %. The screen also limits pressure distortions in the air intake, and thus has a favorable effect on the installed engine surge margin.

This system thus provides adaptable and effective engine protection against foreign matter and icing, with only very minor penalties on engine performance.

3 EXAMPLE OF AIR INTAKE DEVELOPMENT FOR FLIGHT IN SNOW AND ICING CONDITIONS

The following example illustrates the development of an air intake system for the SA 366G1 helicopter to ensure satisfactory air intake operation in icing conditions and falling snow.

3.1 Initial Specification of the SA 366G1 Air Intake System

The SA 366G1 is equipped with a Lycoming LTS-101 engine having an annular air intake located aft of the engine reduction gear module. The general aircraft architecture and the hover and cruising flight performance requirements (fuel consumption) dictated the choice of a ram air intake. The engine itself is supplied through a scroll specified by the engine manufacturer. Icing protection is provided by a screen and bypass at the mouth of the air intake duct, and enhanced by heating the edge of the engine intake (Figure 8).

3.2 Design Development for Flight in Falling Snow

This design was found to be unsatisfactory for extended operation in snow, and especially in recirculated snow. Ice buildup occurred inside the engine intake scroll, and detachment of ice fragments at unpredictable intervals resulted in engine surge or even compressor damage. This problem is critical with the LTS-101 engine, which is particularly susceptible to ice ingestion (5 g can result in engine surge phenomena).

3.2.1 Analysis of the Problem

Snowflakes (ice crystals) entering the air intake strike the engine intake scroll wall as a result of the substantial airstream deflection inside the duct. The high temperature environment in the engine bay maintains the scroll wall at a temperature above the air temperature. At slightly below freezing temperatures encountered when flying in snow, the snowflakes begin to melt on contact with the scroll wall. However, the wall temperature is not high enough to completely melt the snow, which tends to adhere to the wall and freeze again. The resulting ice accumulation may exceed 100 g.

This analysis was substantiated by testing with and without simulated heating of the scroll due to the proximity of the hot engine casing. Snow buildup and ice formation occured inside the scroll when the thermal environment was simulated but was not observed in the absence of this simulation.

3.2.2 Solution Adopted

Various solutions were considered in this case :

- heating the scroll wall
- insulating the scroll wall
- installing a particle separator.

These options were not implemented, either because of development problems or because of ineffectiveness on view of the high engine sensitivity to ice ingestion, or because of installation problems raised by the powerplant architecture.

The concept finally adopted consists in retaining any ice catchment in the air intake as close as possible to the engine. A sufficient retention volume was required for this purpose upstream from the barrier, and with a large enough surface area to ensure proper engine air supply even when partially obstructed by ice. The scroll was therefore replaced by a plenum chamber from which air is drawn into the engine through an annular screen (Figure 9). Ice buildup may occur in the plenum, but ice fragments can only reach the engine in very small volumes that do not interfere with normal operation. An anti-icing system provided on the wall downstream from the screen, but with a low installed power (< 1 W/cm^2). The anti-icing system was required to prevent moisture from freezing again on the conical panel downstream from the screen as a result of the heavy ice catchment on the aft face of the plenum.

This solution which ensures more effective engine protection, involves higher power losses than the initial configuration.

3.2.3 Development Procedure

In order to accelerate development work and to avoid a long search for suitable snow and temperature conditions, tests were conducted with artificial snow in a refrigerated chamber.

The systems available for producing artificial snow actually generate ice crystals, forming powdery snow without snowflakes. The most satisfactory snow quality was obtained at temperatures below -10° when snow formed on contact with an obstacle. However, since critical icing conditions for the air intake are often encountered at temperatures between 0 and -2°C, the snow cannon could not accurately simulate air intake icing buildup. Moreover, this method does not allow accurate concentration estimates to be made. It was thus necessary to produce snow at low temperatures and store it for subsequent use at the test temperature. This required certain precautions, as any contact with air at temperatures above freezing transformed the snow and prevented satisfactory testing.

The snow was therefore sprayed into the air intake by a distribution system equipped with a blower (Figure 10). Given the small intake rates necessary, the best results were obtained simply by brushing the snow by hand.

This method makes it possible to conduct development tests quickly and easily in a controlled environment chamber, and can be applied to most helicopter air intake systems. The results always correlated very well with tests conducted in natural conditions. This procedure may therefore be used to determine the critical operating conditions (temperature, engine rpm, etc.) of an air intake and to limit the need for actual cold-weather testing even if the latter cannot be entirely eliminated.

3.3 Design Development for Flight in Icing Conditions

Shakedown testing of air intake behavior in icing conditions showed that the front screen had to be optimized for low speed flight, as critical ice buildup was observed in the screen bypass area (Figure 11). Under these conditions, uniform screen obstruction due to icing resulted in a high-volume airflow via the bypass and thus contributed to icing in this zone.

3.3.1 Analysis of the Problem

The screen and bypass were optimized following a theoretical analysis of the bypass in icing conditions. The velocity field in the bypass area and the supercooled water droplet paths were determined by a hodographic method. It was found that the width of the bypass, and thus the air velocity through it, had a considerable effect on the dimensions of the impact zone and the droplet incidence angle on the duct wall. This preliminary study made it possible to optimize the bypass dimensions for the subsequent development tests.

3.3.2 Development Testing

The optimized design was tested at the NRC facility in Ottawa, where it was observed that ice catchment in the bypass was stabilized at a very thin layer ; this confirmed the preceding analysis results.

The air intake system was then submitted to tests in the icing wind tunnel at the CEPR facility in Saclay where various characteristic points in the flight envelope were simulated :

. Altitude :	sea level, 1700 m, 3900 m
. Temperature :	-2°C, -5°C, -10°C, -20°C, -30°C
. Airspeed :	100 km/h, 130 km/h, 250 km/h
. Water concentration :	up to 2.4 g/m^3.

The test findings confirmed that the airspeed-dependent screen catchment mechanism (cf § 2.3.4) provided adequate engine protection, and that a stable configuration was reached after 10-20 minutes. The resulting system provides effective icing protection of unlimited duration.

3.4 Development Summary

The development work on the SA 366G1 air intake system showed that satisfactory engine operation could be ensured in severe snow and icing conditions. This result was obtained after numerous development tests (Figure 12) during which it became clear that the qualities of the engine itself and the engine air intake configuration were of considerable importance.

It now appears indispensible to supplement existing helicopter turbine engine qualification procedures by performing the following tests of parameters which significantly affect aircraft behavior and performance :

. engine susceptibility to rapid ingestion of specified masses of water, snow and ice,
. determination of the most critical engine ratings,
. engine operation in snowy atmosphere.

These tests should complete existing requirements concerning continous water ingestion and icing.

4 CONCLUSION

Selecting the type of air intake on a helicopter turbine engine should be based on a tradeoff between the aircraft-mounted engine performance, the total aircraft drag and protection of engine against external agents (foreign objects, ice, snow, rain).

Ram air inlet is well adapted to front air intake engines. On the other hand, a static air inlet best suits lateral air intake engines. This corresponds to the architecture of advanced-design turbine engines (through shaft, forward reduction gear) mounted aft of main gearbox. After extensive analysis, this static air intake provides good supply to engine both in hover flight and at high airspeed.

The system for protection of air intake against foreign objects is very efficiently and simply provided by a specially designed protective screen. In addition, installing a screen inside the air intake as close to the engine as possible allows providing effective protection of engine when flying under snow, when the air intake and engine configurations are especially sensitive to such a condition of operation.

The sand protection corresponding to special operating conditions can be achieved by a specific device consisting of vortex elements. This performance-penalizing system provides effective protection. This penalty has already been partially mitigated through the use of a fan extractor system.

Therefore, the air intake screen is still the most versatile and reliable system which is the best adapted to most helicopters and operating conditions.

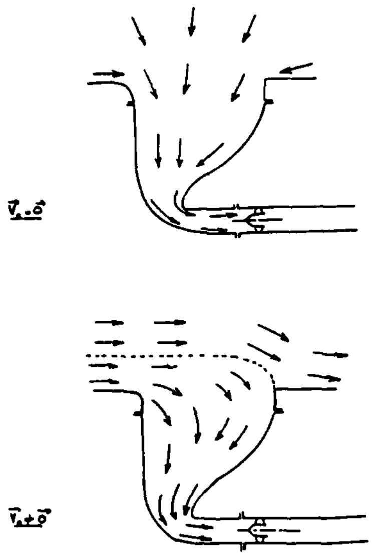


Figure 1 Schematic view of static air intake

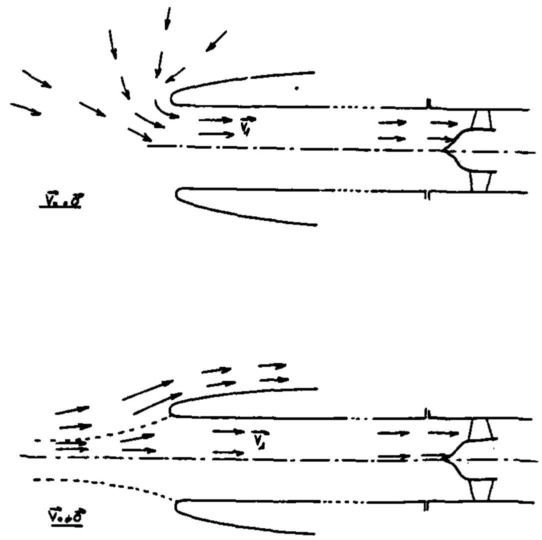


Figure 2 Schematic view of a ram air intake

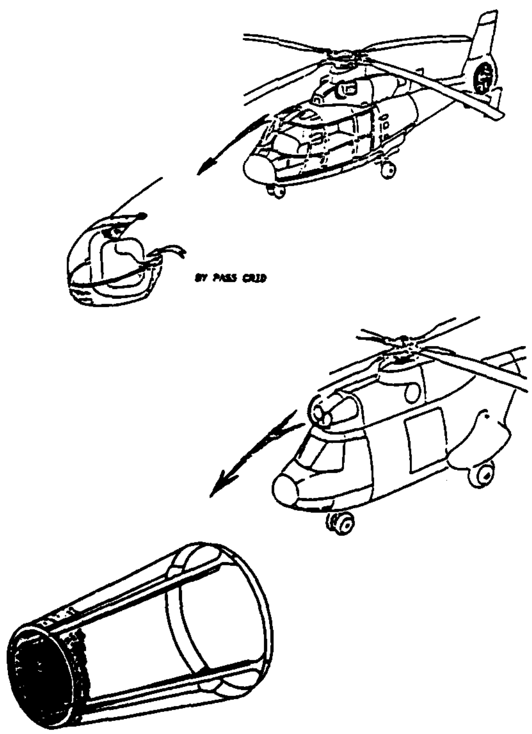
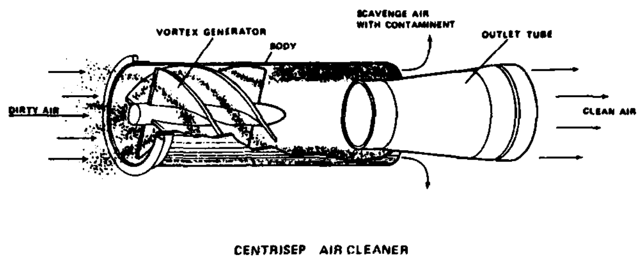


Figure 3 Typical air intake screens



F. gure 4 Vortex separator operating principle

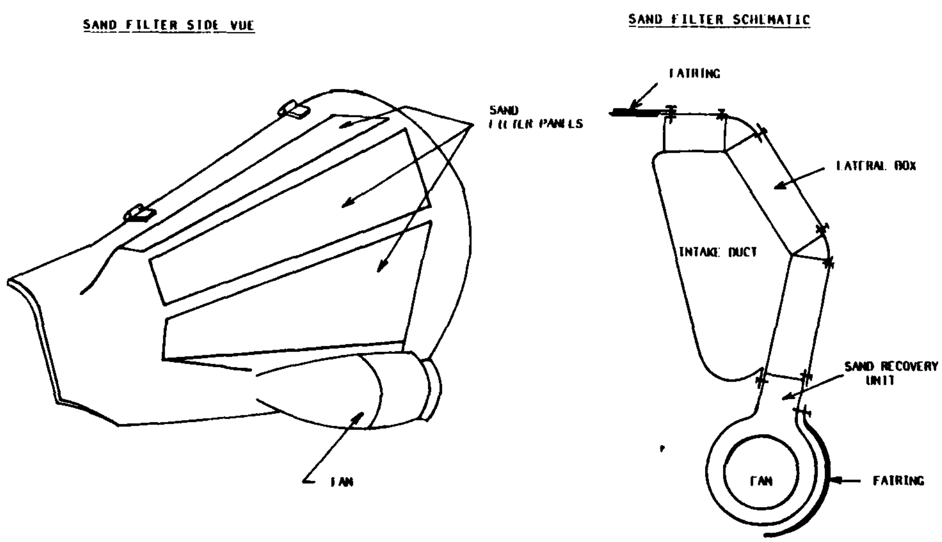


Figure 5 Schematic view of a vortex type sand filter system

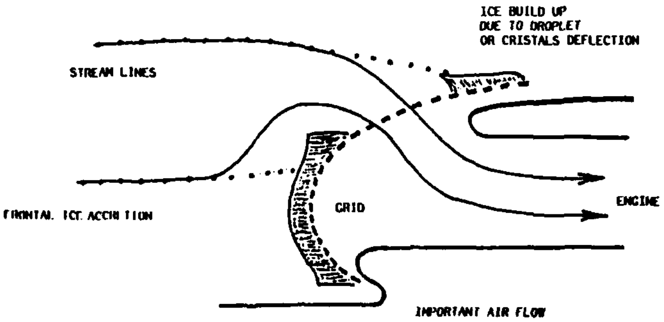


Figure 6   Air intake screen operation at high speeds

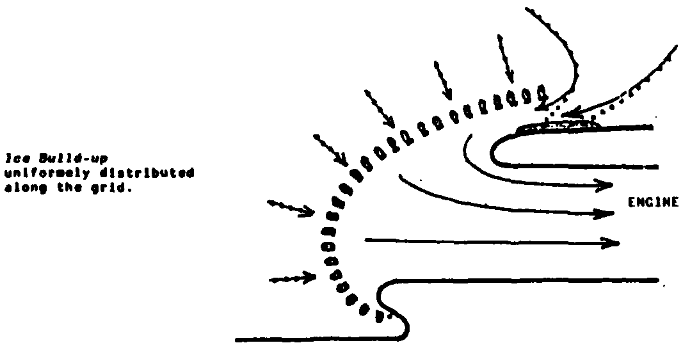


Figure 7   Air intake screen operation at low speeds

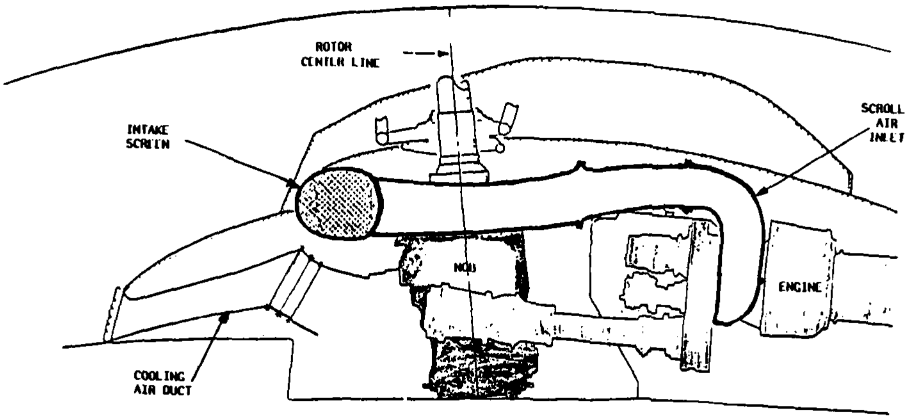


Figure 8 SA 366G1 air intake design : scroll version

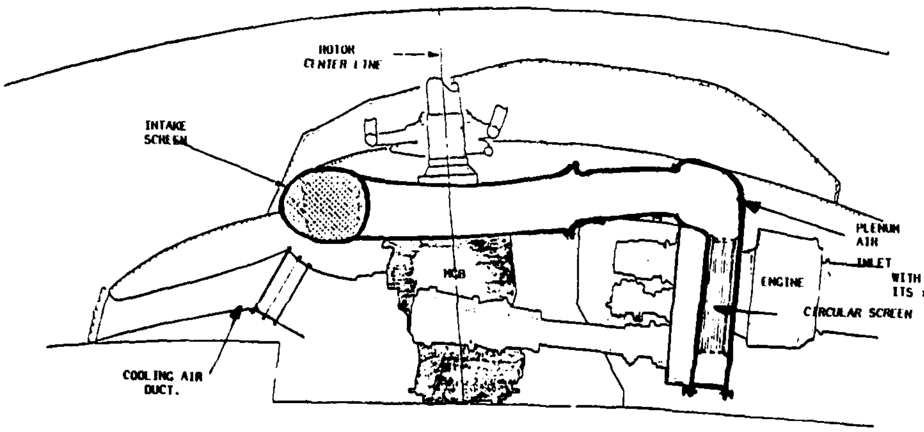


Figure 9 SA 366G1 air intake design : plenum version

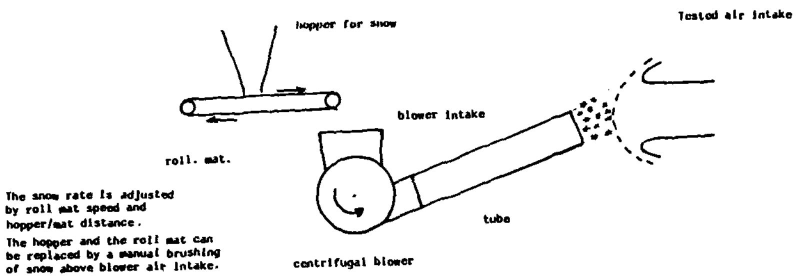


Figure 10 Snow spraying system

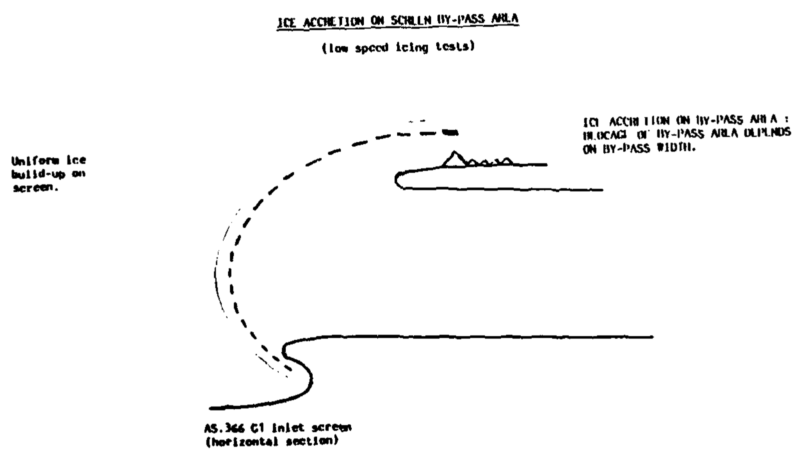


Figure 11 Catchment in the bypass area of the SA 366G1 air intake screen

AS.366 G1

ICING AND SNOW TESTS FOR AIR INLET

DEVELOPMENT AND CERTIFICATION

ICING TEST :

SIMULATION TESTS IN WIND TUNNEL : (CEPr SACLAY) WITH ENGINE AND HELICOPTER INTAKE :

23 H30 INCLUDING 8 H 00 FOR FINAL SOLUTION

SPRAY-RIG TESTS (N.R.C OTTAWA) ON AIRCRAFT : 16H 20

INCLUDING 9H 50 FOR FINAL SOLUTION

SNOW TEST :

BENCH TESTS (ENGINE AND HELICOPTER INTAKE)

30 H INCLUDING 10 H FOR FINAL SOLUTION

HELICOPTER TESTS WITH ARTIFICAL MEANS.

23 H INCLUDING 11 H FOR FINAL SOLUTION

HELICOPTER TESTS IN NATURAL SNOW

22 H INCLUDING 14 H FOR FINAL SOLUTION

FIGURE 12

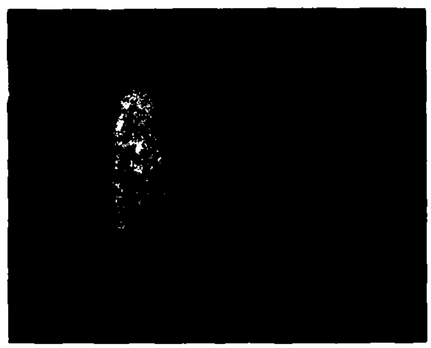
AIR INTAKE PROTECTION TESTS ON HELICOPTER AS.366 G

ORIGINAL CONFIGURATION

BIRD INGESTION TEST :

ICING TESTS IN CEP_R (SACLAY) :

15 HOURS 30



SNOW TESTS :

NO ACCEPTABLE
RESULTS

FIGURE 12

AIR INTAKE PROTECTION TESTS ON HELICOPTER AS.366 G

DEVELOPPEMENT TESTS :

MOCKUP
(COL DU LAUTARET) :
1 MONTH

TESTS ON AIRCRAFT
(COL DU MIDI)
8 DAYS

CLIMATIC CHAMBER
TESTS
ON MOCKUP
(BOURGES)
3 WEEKS

FIGURE 12

AIR INTAKE PROTECTION TESTS ON HELICOPTER AS.366 G

DEVELOPPEMENT TESTS (SUITE)

CLIMATIC CHAMBER
(EGLIN AFB)
1 MONTH

NATURAL SNOW
WITH AIRCRAFT
2 WEEKS

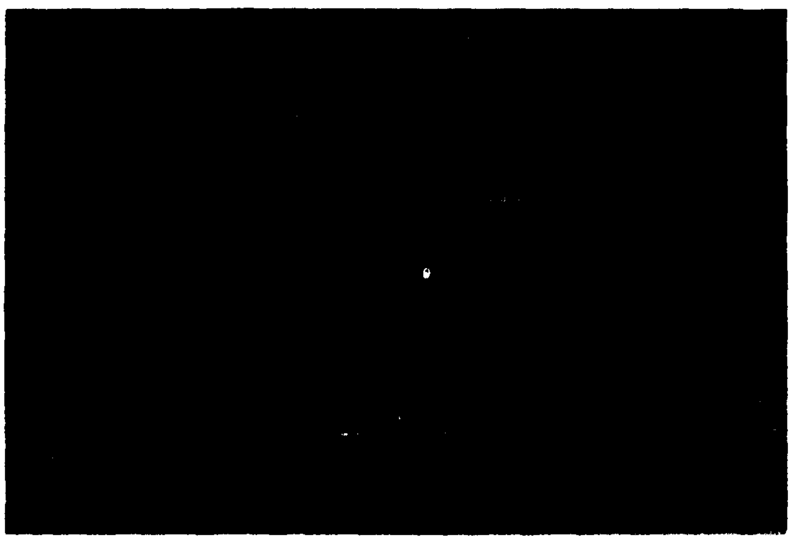


FIGURE 12

AIR INTAKE PROTECTION TESTS ON HELICOPTER AS.366 G

CERTIFICATION TESTS

ICING SPRAY RIG :
(OTTAWA)
16 HOURS 20

WIND TUNNEL ICING TESTS :
(SACLAY)
8 HOURS

SATISFYING RESULTS

FLIGHT IN
NATURAL SNOW
1 MONTH
12 HOURS 10 IN
FLIGHT

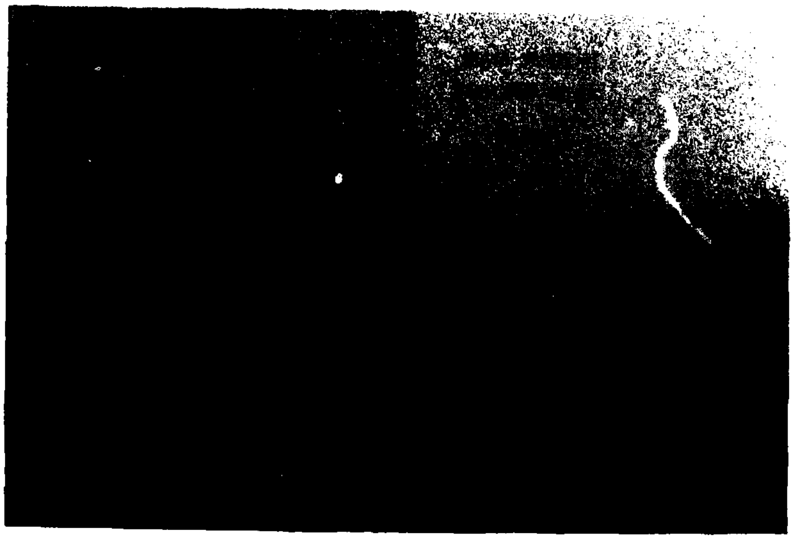


FIGURE 12

ENGINE-AIRFRAME INTEGRATION FOR ROTORCRAFT: SYSTEM SPECIFICATIONS AND QUALIFICATION

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SUMMARY

This paper is intended to provide both past history and current principles associated with engine-airframe integration for rotorcraft as related to system specification requirements and the resulting qualification assurance. System Specification requirements have evolved through periods of few details and lack of definitive requirements to the opposite extreme of exacting details and extremely specific requirements. The current trend of performance related specifications is intended to enhance the freedom of design while meeting the essential mission related requirements. This has required a revised qualification program effort to insure essential requirements are being met throughout the development effort. Lessons learned in this evaluation as well as current methods will be discussed for overall engine-airframe integration system requirements. In particular, system qualification will be addressed for such areas as adverse environment requirements for icing, cold/hot temperatures and sand/dust; mission related requirements for engine response, IR signature reduction, and crashworthiness, and basic engine integration requirements for torsional stability, vibration and cooling.

INTRODUCTION

For a rotorcraft, its engine-drive system-rotor system integration is the premise of its ability to fly. The system specification has thus had to insure integration of this dynamic combination addresses specific details to insure rotorcraft overall performance and safety. The dynamic nature of the system also had to address both ground and flight operation as well as overall interrelationship with the airframe structural dynamics. The ability to address this complexity in system specification requirements has depended upon the overall development and qualification assurance program to assure compliance. Thus any discussion of the system specification process must also address the airworthiness qualification specification requirements also. To address the overall topic, examples will be used to cover successes and problems with the system specifications and their associated qualification assurance effort.

SYSTEM SPECIFICATIONS

Prior to developing a system specification for a rotorcraft, a requirements document is defined by the user. This process may be lengthy to include research and development of concept exploration, trade-off analysis of options, method that given mission requirements are considered in conjunction with a total military concept, and considered technology available during a development effort. Procedures also must account for risk assessments and the budgetary process. When formulated, this requirements document becomes the baseline for finalization of system specification requirements. It is understood this requirements document will be revised as needs arise which will require consideration of methods to meet these. Where the development/production cycle is in time will determine whether the option is a system specification change with an accompanying engineering change proposal or a major modification to the previous rotorcraft definition.

Actual development of a system specification is typically a parallel effort with the requirements document as many of the "building blocks" require detailed revision. Many of these are the component level specifications or basic design specifications and standards. With current requirements for "tailoring" these to the specific application, detailed review and evaluation must be conducted to insure requirements are appropriate for the application.

The system specification is usually considered in two parts: the first part being the basic requirements, and the second part being the qualification assurance portion. As initially released for a request for a proposal, these overall specifications will meet the general criteria of a "performance type" specification. That is, it gives the end result requirements that must be met, both in design and test but does not give the "how to" method in meeting them. A rotorcraft system specification submitted in response to the request for proposal will then provide details of how the requirements are to be met and the procedures defined as to how qualification assurance will be demonstrated.

Before a system specification is released in a request for proposal its review process will include the developer, the user and even the industry. The industry review of a draft system specification provides a means to identify where areas are considered overly restrictive because of a new technology, where feasibility of meeting a requirement may be high risk, and where recommended improvements could be made.

Final resolution of all inputs may compromise some or may reject some, but the method allows the best product to be developed for the future rotorcraft definition.

ENGINE-AIRFRAME SPECIFICATION

As part of the overall system specification, the engine-airframe integration requirements must be addressed to cover the total system. Whereas, an engine-airframe interface control document (ICD) becomes a mechanism to clearly define overall integration requirements, the system specification is the baseline for the rotorcraft portion. It is within the system specification where requirements are defined for the overall interrelationship as well as details to include specific component specifications and requirements. Typical areas covered in the system specification are addressed as follows:

General Design: This area covers general system characteristics. It may include maintainability, survivability/vulnerability, reliability, system safety, etc. Also overall compatibility requirements for engine/airframe/drive system for dynamic (torsional stability/vibration/critical speeds and frequencies), environmental (sand and dust, temperatures, rain, snow/ice, salt fog, humidity, fuel/oil, etc.), and structural (static and fatigue). Also general requirements for drains, engine mounts, cowling/nacelle construction, fire protection subystem (fire walls, warning system, and extinguishing system), and servicing (fuel, oil, water wash, hydraulic fluid, pressurized air, etc.).

Air Induction System: This area covers the overall air induction design to include airframe integration. Integration includes requirements for non-turbulent air flow as well as protection from exhaust gas/armament gas reingestion, temperature distortion effects, protection from foreign object damage, ice protection and sand and dust protection. Specific requirements for these such as distortion limits, anti-icing capability, and particle separator efficiency are addressed as applicable.

Exhaust System: This area covers the basic exhaust system design to include specific requirements for infrared suppression. Capability for thermal cycling in an adverse vibration environment also typically requires specific life requirements. Protection of other airframe hardware, rotor effects and hazards to personnel also are addressed due to the extreme temperatures associated with the exhaust gasses.

Cooling System: Due to the nature of the thermodynamic cycles associated with propulsion systems, overall cooling requirements are addressed. Considerations for hot day environment along with operating conditions of high/low power, ground conditions with wind and soakback conditions after shutdown. Specific temperature limits for known materials, components, or safety hazards may also be defined.

Fuel System: The fuel system must cover all required operational and mission requirements. Fuel management, fuel supply, fuel capacity, center of gravity considerations, fuel dump, ground refueling, aerial refueling, extended range provisions/requirements, draining/sampling, etc. will cover the overall engine/airframe interrelationships.

Starting System: This area must cover the system requirements to meet the engine characteristics as well as operational requirements; such as, number of starts within given time, temperature/altitude requirements, ground/inflight requirements, fuel/oil effects, multiple engine starts and overall start sequence/time. Detail starter/power source requirements to meet the operational requirements are generally covered in component level specifications.

Propulsion System Controls: Propulsion system controls must cover the broad area of all cockpit controls and associated diagnostics as well as other required controls and diagnostics that may be used for maintenance troubleshooting or general operation. The total system characteristic must be addressed to include accuracy, tolerances, hysteresis, feedback, dynamics, etc. The human factors design elements must also be addressed in the overall man-machine interaction/interface.

Drive System The drive system has typically been one of the more detailed systems as related to the rotorcraft application. Overall details of system torques, speeds, endurance factors, transient capability, vibration, torsional stability, and overall dynamics are addressed. With the overall lack of component level test capability, specific requirements for lubrication, gears, bearings, shafts, couplings, diagnostics/inspection, etc. are usually included.

Lubrication System: With the trend to self contained lubrication systems for engines and transmissions/gearboxes, lubrication system requirements typically only cover general requirements. Heat exchangers and associated fans may still be remotely located and have specific requirements. Filtration system parameters are also typical with the trend to finer filtration systems.

Engines/APU's: Although the engine/APU's are defined by their own specifications, the integration of these must be addressed in the System Specification. Each of the above systems have a direct/indirect integration with the engine/APU's operation in the system. The Engine/Airframe Interface Agreement and its associated Interface Control Document (ICD) are the means to address the programatic procedures to insure the proper integration is developed as well as a joint document clearly defining the integration requirements.

ENGINE-AIRFRAME SYSTEM DESIGN

The steps leading to the overall engine-airframe system design are somewhat clearer than at the component level as the overall rotorcraft system requirements will provide the basic criteria. The System Specification has been developed from this process and defines both general and specific criteria for the total engine-airframe integration. Specific system design will evolve from the conceptual stages, thru preliminary design, and then detailed design. Parametric analyses and trade-off studies will be part of this initial stage that leads to the rotorcraft design concept to meet the overall mission requirements. It is from this stage that engine-airframe system design evolves.

The engine in a new or major modification rotorcraft program may be newly developed, a major modification itself or an existing series. In all cases, the lead time for engines has required earlier starts for development such that at the time of a rotorcraft System Specification development, the engine specification is already in existence. The conceptual design stages have usually sized the engine and defined certain characteristics such as output shaft speed, weight, growth potential, transient capability, etc. Thus, the engine specification already includes some effects of the engine-airframe integration process before the rotorcraft System Specification exists.

SYSTEM QUALIFICATION

The preliminary design process of analytical techniques, supporting laboratory tests, component tests, and rig/subsystem tests will lead to the actual development rotorcraft. It is from this stage that the system qualification assurance process will start. It is recognized that there is no clear line or change at this time as additional design, support tests, and component qualification tests will be ongoing, but it is a point where the overall system qualification process begins.

For this paper, the system qualification will only be considered as part of ground and flight test vehicle efforts. As explained in the companion paper covering components, many bench/rig tests are system oriented. They may be a duplication of the system as installed, but usually still have several items simulated and leave certain unknowns that require further rotorcraft system testing. It is this level of system qualification that is addressed.

System qualification is usually not a one-time effort. Requirements for functional operation is over an entire flight envelope as well as all environmental and usage conditions and it is also coupled with the reliability/endurance requirements. As such, both specific tests and general overall experience are used in the final assessment.

For engine-airframe integration system qualification, the specific requirements are addressed in ground test vehicle (non-flyable) tests, propulsion interface surveys, specific environmental/usage tests, and overall propulsion demonstrations. Each of these are defined in more detail as they are used in the overall engine-airframe integration process.

Ground Test Vehicle (GTV): The GTV is typically defined as an airframe complete with propulsion system, hydraulic system, rotor system and flight control system installations. A feature which causes some problems in direct vibration/dynamic simulation is the structural provisions included for the tie down and the subsequent hard point tie down to the ground. The overall engine/drive system/rotor system operation still facilitates considerable development/qualification data for the engine-airframe integration. To insure safety and efficiency of the GTV operation, specific engine/drive system/rotor system bench/component tests are required to be completed prior to GTV operation. GTV operation primarily consists of long term endurance effort against a spectrum of powers, shut downs/start ups, rotor speeds, and flight control positions. General endurance results, operation verification and general engine-airframe integration results come from this effort.

Propulsion Interface Surveys: These surveys are utilized primarily as a development process of all the engine-airframe ground/flight test requirements. They are scheduled in a program to allow redesigns to be included in the formal demonstration test efforts. Specific tests include engine/airframe compatibility, engine/APU vibration, propulsion temperatures, air induction system, exhaust system, and related subsystem operations; such as, pneumatic, hydraulic and fuel. Since these require rotorcraft ground/flight operation, there are several prerequisites to first flight requiring completion of certain analyses, lab tests, component tests and GTV tests. The propulsion interface survey results are considered supplemental to the qualification of the total system. These surveys usually require less test conditions than the formal qualification and do not require total success to be considered complete. However, if they are totally successful with no changes required, they may be used to satisfy the applicable portions of the formal demonstrations required for qualification.

Specific Environmental/Usage Tests: Several system qualification tests not directly related to the engine-airframe integration, but requiring successful performance also are conducted. Examples are Infrared Radiation Measurement Tests, Rotorcraft Icing Tests, Structural Demonstration, Adverse Environment Tests (such as Climatic Hangar Test or remote site testing), Handling Qualities Demonstration, Height Velocity Demonstration, and Electromagnetic Compatibility.

Propulsion System Demonstrations: The propulsion system demonstrations are the formal qualification of the overall engine-airframe integration with associated systems. Engine/airframe compatibility tests are conducted for propulsion control system, starting and torsional stability. Additional tests cover the lubrication and fuel system, drive systems, rotors/propellers, propulsion system vibration, propulsion system temperature, air induction system, exhaust system, as well as related systems such as fire detection/extinguishing, drainage, fuel, pneumatic and hydraulic.

ENGINE-AIRFRAME SYSTEM QUALIFICATION - LESSONS LEARNED

Problems related to qualification are sometimes related to something missed in the design process but just as likely are something missed in a prior qualification effort at the component or bench/rig area. As such, problems discovered in a formal rotorcraft ground/flight test program may have been better solved at a different level if improved test methods had been in place. Another problem has just been due to the lag in the component qualification effort such that problems disclosed in actual component level testing are not solved before the rotorcraft test program is in place. Details of these type problems were covered in the comparison paper on the component process. Program schedules with the ability to allow component level discovered problems to be fixed prior to formal demonstrations should be the goal.

The engine-airframe integration process is not much different than many other system problems. That is, all influences from direct/indirect related functions must be addressed. For example, the key to the success of a hydraulic starting system may lie with an adequate electrical system for valve actuation. These interrelationships along with an understanding of the total "system" is necessary for a total integrated engine-airframe system. The following address "lessons learned" from previous rotorcraft development/qualification programs related to these overall system considerations.

Air Induction System: The basis of an air induction system is to provide smooth airflow with minimal losses to the engine inlet. The basic design for these conditions is fairly straight forward with improvements from current computer aided design systems. However, as a rotorcraft system maintaining this minimal distortion condition under all circumstances has been a source of many lessons learned. In addition, protection for the engine from foreign object damage (FOD) and adverse elements associated with rotorcraft operation has also been a definitive problem. Finally, some general system integration problems have introduced specific examples of air induction problems.

Basic performance of the air induction system must address both steady state and dynamic conditions to consider both pressure and temperature. Location of these instrumentation rakes, type of sensors, sensitivity/tolerance of instrumentation, single point measurements versus averaged conditions, response rate of sensors, etc., are a considerable discussion point between the airframe manufacturer and engine manufacturer. The need for repeat tests due to lack of proper instrumentation has justified the need to insure this is a priority in the test planning stage.

The engine specification and subsequent ICD define the acceptable distortion characteristics. The basis of a definition is typically a formula based on measurements associated with a defined measurement location. The engine qualification is then based on a test with a selected distortion generating screen. This has thus verified a singular set of distortion patterns which are steady state pressure distortions. Temperature distortions verification is not as well substantiated as a multitude of different test methods or analyses are utilized. Thus, the engine distortion qualification is based on a fixed set of steady state test/analytical conditions for conditions of no power loss effects and no adverse functional performance effects. For military applications, another type test may be conducted to simulate gun gas temperature transients. Once again, the definition and tests are defined for some specificed condition.

Actual operation and testing in the rotorcraft provide a myriad of steady state and transient distortion conditions which don't match the defined engine definitions. The first problem is building a measurement rake that doesn't induce further problems and to measure at the same plane location specified in the engine specification. The measurement rake must also have the sensitivity to pick up transient conditions that may be more adverse than the steady state conditions. After all that is determined, then testing for the most adverse conditions is necessary. With so many possibilities, the actual problem distortion condition probably won't be picked up during formal air induction tests, and those that are detected usually are not the cause of a serious problem.

The following examples illus rate the type problems that have occurred which match the above conditions. A buried inlet, irregular inlet and/or back-up structure may not allow a rake installation in the same plane as conducted with the engine. A nose gearbox/cross-shaft installation may induce a dynamic condition if no cross-shaft cover is utilized (flow field around whirling shaft or from vortices induced by shaft coupling geometry). Seals associated with gearboxes/adjacent compartments may allow leakage allowing high temperature air to be ingested as a singular area. Air induction system/rotor system/exhaust system interaction may induce exhaust reingestion under certain ground/flight conditions. Anti-icing system may discharge high temperature air directly into the engine. Airframe shapes/equipment upstream of the air induction system may induce turbulent conditions. Gun/rocket gas reingestion may cause high temperature and/or pressure spikes beyond the capability of the pressure/temperature sensors.

The need for an anti-icing system for the overall air induction system to protect from blockage and potential ice caused FOD requires an overall component level qualification and rotorcraft icing test to qualify its required capability. System related problems associated with the anti-icing system may also have to be evaluated. Several problems have been caused by off season hot environment operations with inadvertent or failure causing default to the anti-icing mode. High temperature inlet conditions have affected engine operation as well as caused failure of air induction hardware/anti-icing system (e.g. electrical anti-icing mat burn-out and high surface temperatures causing seal damage, debonding, burn through, etc.). Energy requirements for anti-icing may also cause problems due to high demand in association with other high demands causing energy source saturation/deficiencies (e.g. electrical requirements beyond system capacity or bleed air performance effects on engine power). Excessive engine power degradation due to anti-icing system operation has caused severe rotorcraft transient conditions due to on/off cycles and safety related in the case of inadvertent/failure condition of a bleed air system on a hot day. A unique condition due to the smaller size of many rotorcraft air induction systems has been the use of a ice detector in the inlet. Mechanical failure due to fatigue/damage may generate a FOD source or failure in a heater circuit/normal action of ice shedding may lead to ice FOD.

The need to protect the engine from FOD, as well as sand and dust erosion, has required particle separators in many rotorcraft applications. However, general air induction system design must still consider all potential FOD sources, both from ground proximity and rotorcraft hardware related. Places where objects can be caught/retained/left either in the air induction system or in the proximity must be considered. Maintenance actions near/in an air induction system could generate a problem due to left tools and hardware. Statistics have shown inspections for FOD damage causing an increase in FOD damage. Left over pieces of lockwire, nuts, washers, cotter pins, etc. have been a major source of maintenance related FOD. Ice buildup on airframe hardware in front of an air induction system has also caused FOD due to ice shedding with a perfectly normal anti-icing system. FOD has also been generated with a perfectly normal anti-icing system due to activation of it after the ice had built up.

Inflight/ground operation has not been the only problem source for air induction systems. Maintenance actions have caused structural damage due to conveniency of hand holds or foot step points. Air induction system covers that may be properly placed and removed, but with inadequate sealing may have allowed water (and subsequent ice) or sand/dust buildup which is ingested during normal start-up. Screens utilized for FOD protection have been sources for FOD due to leaving lockwire in the vicinity when taking it off and on and securing device failures with subsequent ingestion due to improper fastening.

Exhaust System: Exhaust system design used to entail a system to discharge exhaust gasses away from the airframe with a minimum of pressure loss. This was usually a fairly high speed structured flow which was directed away from the aircraft to preclude any problems with rotorcraft aerodynamics effects or local heating problems. With the advent of IR suppression requirements and noise abatement, exhaust flows are both cooled, require additional ambient air mixing, and a resultant lower velocity discharge. This diffused exhaust flow now has shown considerable influence in the overall rotorcraft system design to require assurance that all influences are properly taken into account.

In association with the air induction system, exhaust flow interaction with rotor influences must be addressed to insure exhaust reingestion is precluded. Conditions of hover, rearward/sideward flight, in ground effect maneuvering, takeoff/landings all associated with wind effects must be explored in detail in the rotorcraft test program. Those same conditions may also contribute to reingestion in cooling air inlets for other subsystems, local skin surface heating, and exhaust effects on rotor/tail rotor performance.

With the introduction of IR suppressors, problems with all of these areas have been encountered requiring considerable rotorcraft testing. Specific examples are noted as follows: (1) Air induction system reingestion during flared landings and ground operation with tail into wind; (2) Local tailboom skin surface heating causing IR hot spot, tailboom structural degradation and tail rotor drive shaft bearing deterioration; (3) Oil cooler inlet reingestion causing overheating condition; (4) Tailboom/rotorcraft vibration increase due to interaction of exhaust flow with tail rotor and (5) Downwash causing exhaust gasses effects on ground personnel.

As an effect on engine performance, instrumentation problems with verifying back pressure conditions coupled with exhaust gas swirl angle effects is as much of a problem as the air induction system. The engine manufacturer tests the engine during performance checks with a "referee" exhaust pipe. The delta effects of the aircraft exhaust system must be compared against this effect to adjust the installed performance. Ability to measure at a given plane must be a closely coordinated effort between the airframe manufacturer and the engine manufacturer to insure accuracy and agreement in method. Numerous disagreements have prevailed around test data usefulness all driven by the instrumentation type, location, sensitivity, etc. It is essential that the same level of planning is accomplished like that required with the air induction system.

Overall endurance testing of the exhaust system has been addressed as a component rig test, and during a Ground Test Vehicle (GTV) test, but flight test vehicle tests are still the final basis to account for the vibration and thermal cycling environment. As changes evolve due to the many items addressed above, flight test may be the only mechanism left to determine the final configuration qualification. This would also be a candidate for continued evaluation improvements through a "fleet leader" rotorcraft program due to the difficulty of determining long term effects during a qualification test program. As considerations for variable nozzle and cooling air supply evolve to cover the total airspeed range of advanced rotorcraft, endurance verification will be even more important to insure continued proper operation.

Like the air induction system, consideration of problems associated with ground storage/maintenance must be considered in the final system design. Problems due to sand/dust buildup in cooling passages or in general capture areas, ability to drain water/oil/fuel, and general maintenance must be addressed.

Cooling System: The basic problem with cooling any engine and its associated equipment is to define the requirement. The basic requirement is to insure proper and long life operation of all items over the total operational environment of the rotorcraft. As a method to give some bounds for this requirement the specification will usually define a hot ambient temperature versus altitude requirement, a solar radiation requirement, and selected component temperature requirements. The airframe manufacturer and engine manufacturer will then define more specific component temperature limitations and their measurement location. This becomes the first basic problem, i.e., inadequate definition of temperature and measurement locations as these will define the basic test plan for measurements. Component problems associated with high temperatures where no measurement was previously required is a common problem in each development/qualification program. Many cases in this regard are based on an assumption that if air temperature at location "A" is allright, surface temperature of component "B" will be allright. These assumptions were usually made by the same personnel who did the initial cooling analyses that assumed initial cooling air temperature equal to ambient without consideration of internal heating effects. Variances in airflow and radiation effects in a compartment have shown these type assumptions to cause problems later.

Assuming the limit temperatures can be defined and measurement locations defined, the next problem is getting instrumentation that will have the proper response and will accurately measure the required temperature at the measurement location. An engine mounted accessory may have an air temperature limit defined with a multitude of options to measure it at various locations around the accessory with varying results. It should be determined what is the critical point/area that established the temperature limit and to define the measurement location accordingly. Enough examples of an agreed to specification with multiple interpretations of the measurement location affecting compliance can be defined for all programs.

After defining the limit temperature and its measurement location the next process is to insure the actual test has the measurement at that location and the location verified both before and after the test if it is possible to be varied. The standard problem in this area is a thermocouple used to measure air temperature at a defined location. Each time the engine cowling is opened for maintenance and/or instrumentation trouble shooting, the location can vary - and has. When a configuration change requires a repeat of a cooling test, it is also important to have detailed recorded locations, including pictures, to insure a one-to-one comparison is being done.

Final problem is just the instrumentation itself. A temperature sensor to measure air temperature at a given point that is heated due to radiation from adjacent equipment, does not provide the proper temperature. Likewise, instrumentation bundles installed in an airflow path to a measurement location can cause an erroneous measurement of the actual environment at that location.

When properly instrumented, the only key to proper testing has been to test the most critical conditions. Ground tests in all wind conditions; "soakback" tests after shutdown; in ground effect with hover, sideward/rearward flight conditions, out of ground effect, maximum power/torque, etc. Tests must be conducted with enough time for the temperatures to stabilize. Stabilization definition utilized has normally been no more than 2°F per minute change with data taken for 5 minutes after temperatures have stabilized.

After tests are completed the only problem is correcting the data to the specification conditions. Problems obtaining full power/torque due to ambient conditions tested is the initial problem, but test means can usually be devised to cause higher engine operating temperatures and/or a correction for heat rejection at higher powers. For correcting temperature values a one-to-one correction method has been shown to be valid for most cases to correct to the specification temperature at a given altitude.

Besides just a problem in defining measuring and testing, cooling system problems have also been noted due to the means that cooling airflow is induced. In many systems cooling airflow utilizes compartment and/or fuselage skin as the boundaries for its flow path. Ground operations with access doors, cowling doors, hatches, etc. open can cause total disruption of the airflow and in some cases cause excessive temperatures to be encountered causing problems in both short term and long term. Additionally, maintenance operations may cause an abnormal operating condition in which a cooling problem could result that was not seen during normal test conditions. For example, a maintenance operation with the rotor blades off caused engine operation at an abnormal idle condition disrupting normal ejector cooling airflow.

Endurance testing of a cooling system is not a usual problem, but degraded mode of operation must be addressed in the qualification. Use of fans or potential for blockage of coolers/cooling inlets must be addressed. Ground debris including paper, grass, leaves, etc. can be assumed normal blockage material in a rotorcraft application. Testing for blockage should be done to verify detection systems and/or redundant systems. Degraded operations due to fan erosion from sand/dust must also be considered.

Fuel System: Fuel system testing is predominantly done at the component level in conjunction with rig tests. System qualification is then the final functional verification of all operation requirements. As discussed in the companion paper on components, fuel system operation has been complicated with requirements for crashworthiness and survivability/vulnerability. The primary operational effect has been the suction fuel system and inert gas ullage volume systems. Specific system test requirements/problems associated with these requirements are discussed below.

For designs where a pressure fuel system is involved, details of pressure drops, flow disturbances, line purging, system leaks, etc. have not been major issues. As long as the pump operated, fuel would get to the engine with only operational problems being negative "g" conditions. If a system had leaks, it was apparent due to the fuel spillage. With a suction system, any interruption, flow disturbances, leak, maintenance action, check valve leak over-night, lack of an air purge, etc., could potentially lead to an inflight engine shut-down. Detail system tests must insure these characteristics are clearly defined and means to overcome problems are identified to insure the suction fuel system safety and vulnerability improvements do not cause other operational and safety related problems. Prime pumps for purging the lines with manual procedures and/or an automatic means must insure air is purged from the system. This must include all direct feed lines, cross feed lines and transfer lines. Maintenance suction pressure tests must be conducted in addition to positive pressure checks due to different sealing arrangements effects associated with either a negative or a positive pressure condition. Backup boost pumps should be included if the suction pump cannot handle the dynamic conditions of hot day, high vapor to liquid conditions. Partial valve positioning that used to be overcome in a pressure system may be a primary turbulence producer in a suction system operation. Pressure drop through a breakaway valve/coupling may be overcome in a pressure system but be a major problem in a suction system. Line runs with turns, bends, downslopes, low points, etc. may be overcome in a pressure system, but cause problems in a suction system. Overall requirement is careful design and attention to details followed thru with expanded testing to verify suction system performance.

Venting for a regular fuel system must account for altitude changes along with rate of fuel usage, but when coupled with an inerting system will have to consider retention of an inert gas-air mixture to prevent combustion in a fuel tank ullage area. The ability of the inerting system to manufacture inert gas to maintain the mixture, the ability to maintain an inert mixture during shutdown and/or recover on start-up and no adverse effects on fuel system operation is imperative in the design phase. Verification of actual conditions during fuel system operation will thus be a qualification requirement also.

Overall fuel system qualification will also need to determine all other functional capability including refueling operations (inflight and ground), fuel management, fuel dump/drain/sampling, fuel gage and low fuel warning system accuracy, etc. for both the basic system as well as any auxiliary system. Most general problem in this area has been fuel system and low warning system accuracy under all normal flight conditions. With multiple fuels and fuel system attitude conditions, accuracy has been difficult to insure for all conditions.

Starting System: The basic starter and associated support systems were tested at the component level. System testing verifies their overall interrelationship to provide successful engine starts for the specification required envelope. Key to successful starts ties to the engine required characteristics associated with the starting system power system characteristics. Cold ambient conditions, hot ambient conditions, and high altitude conditions typically cover the worst aspects of both.

Cold temperature is undoubtably the worst problem in starting system testing. Getting an aircraft cold soaked at the specification temperature usually requires a Climatic Hangar Test facility or a remote site operational test. Cold soaked is always a problem in definition in that some skin temperatures will reach ambient hours before the internal engine oil or battery reaches that condition. Some method of verification or a long soak period is necessary to insure a proper test. However, this is also the most severe condition for both the starting system and the engine start characteristics. The variables of type of fuel, oil, hydraulic fluid add to problems of conducting an all encompassing test.

The weakest link in all these besides viscous oil/hydraulic fluid and lack of vaporization of fuel is the electric power supply. Battery internal resistance is so high that available electrical power has been the nemesis of many non-start conditions. Even in APU assisted, hydraulic, and pneumatic systems, a battery is still involved to actuate a valve, power instrumentation/ignition, or provide electrical power for automatic monitoring/control in electronic associated fuel control system. Testing with a full-up, new battery will not demonstrate what the effectiveness of a typical battery will provide. Systems to switch dual batteries from parallel to series and internal battery heaters have attempted to improve its capability, but its sensitivity to cold temperature has not been removed. If it is assumed the battery will be removed in extreme cold conditions to maintain its warmth/charge, testing must be conducted with this condition to determine the next limiting parameter in the system.

Start procedures are usually the final problem in overall system qualification. Any different procedure must be carefully documented to preclude an individual test pilot procedure that resulted in successful starts being different from the actual published procedure.

Propulsion System Controls: Most propulsion system control qualification is qualitative in nature by verification of proper operation and human factor evaluations. Exception to this is the quantitative requirements for control forces and hysteresis measurement and instrumentation system accuracies. Control forces and hysteresis measurements are difficult to cover for an operational flight environment and are sometimes done only as ground check. Actual flight stresses on the rotorcraft, however, may cause angular or linear displacements which may adversely affect both measurements. Means to verify these under flight conditions should be pursued. Instrumentation system accuracies are also assumed to be directly related to component accuracies with no full up system test conducted. Once again, the overall system may have much greater error than a sum of component accuracies and should be verified for such.

Because the propulsion system controls are mostly qualitative, a big problem in qualification is lack of documentation of operation. Flight test pilots will modify procedures or accept some limited operation without either relating it to a System Specification requirement or a given operational procedure. The result is usually a much later "discovery" of a problem that just wasn't documented previously. Test procedures should clearly emphasize those areas where documentation is needed to clearly substantiate system operation.

Drive System: No specific drive system testing is required beyond component/rig/GTV efforts except for general system operation and endurance. Once again, although a specific quantitative requirement does not exist, a means to document operational problems and hardware endurance capability should be defined in the test procedures. Besides direct hardware problems, the drive system oil/grease usage, filter history, and diagnostic capability must be documented to insure changes can be considered as required. Again the drive system meets the need for a "lead the fleet" program to determine long term operational requirements with the ability to introduce improvements as early as possible in the production cycle.

Lubrication System: System tests are keyed to cooling, oil supply, diagnostics, and general operational requirements. These are generally conducted in conjunction with other qualification and do not constitute a separate test effort.

Engine/APU: During rotorcraft development it is expected that a continual engine-airframe interface between prime contractor must be retained to evaluate, solve and implement changes that are required. The basic system qualification process will require this same process as each specific test will require some engine-airframe integration issue resolution.

General: As covered in the specification process, this area is the all encompassing one for general system operation as well as some related system requirements. In many cases, this is the only time verification is possible as component level did not allow the overall system interaction. These include the overall adverse environment requirements, the overall mission related requirements, and the basic engine integration requirements. Samples of these general tests are discussed in the following paragraphs.

Vibration has been an expected condition involving rotorcraft. With the multiple shaft balances/isolations, critical speeds/frequencies for both engine generated and rotorcraft generated sources, the total dynamic environment is expected to include adverse vibration. Component and structural part lives can be directly related to this adverse environment such that modern rotorcraft have consistently reduced acceptable values for vibration conditions. Again, the basic problem is like the temperature issue: need to define the characteristics, measurement locations, and accuracies, sensitivities and response of the instrumentation. Engine vibration may be defined in strain measurements as well as accelerometer measurements for specific frequencies over given power conditions and for both steady state and transient flight conditions. Each element of this type of definition needs a strong engine-airframe interface to both insure agreement as well as be valid indications of necessary vibration conditions. To discuss lessons learned from vibration testing would be another paper due to the many issues. However, they can easily be summarized from the above generalization. If detailed effort can be put into the definition, the measurement means and locations, and the instrumentation, the basic improvement in the process can be achieved. To describe just a few problems, the following examples are submitted: (1) Accelerometer mounted on test bracket that measured bracket response, not the component that it was associated; (2) Signal response was not conducive to all dominant frequencies; (3) Definition for steady flight conditions only, not inclusive for maneuvers; (4) Number of accelerometers used dictated by number of recorder channels in lieu of the specified locations; (5) Two engine rotorcraft assumption that one engine was all that was necessary for full instrumentation; (6) Lack of proper calibration of instrumentation; (7) Lack of instrumentation checks during testings to insure data was being taken along with valid readings; (8) Three axis requirement covered with the assumed worst axis; and (9) Data compression system utilized in data analysis suppressed the peak readings.

Torsional stability testing is in the same category of vibration as it is directly associated with a rotorcraft and has a long history of problems in definition and verification. This one also requires a direct engine-airframe manufacturer interaction as a change on either side can change the overall stability. Analytical methods have improved, but have always been limited to the modeling method utilized and the parameter values provided. The ability to model with fairly accurate parameters is usually associated with an improved basis as further test data is available. Its prime usage can highlight potential problems including specific flight conditions for testing. Adequate phase/gain margins are usually part of the basic definition problem along with the ability to test at all conditions. The actual test is the next problem in that most test efforts have required the test pilot to manually excite the system near its resonance condition. The lack of a "calibrated arm" has caused this procedure to be reviewed for improvement. A system employing a fuel interrupt device to induce a specific frequency has been utilized as one method to overcome this problem.

Critical speed testing of individual shafts is always subject to the simulation of the support conditions. A bench rig is usually a solid support base which does not react in the same manner as airframe support. As such, critical speed test verification on a rotorcraft may indicate a substantial shift of the critical speed of the shaft system. In many cases, shaft balance procedures have had to be verified as a system in lieu of piece parts to account for similar system efforts.

Overall environmental capability of the engine-airframe integrated system is verified in many individual tests and specific system tests, but a general overview must also be considered. Some specific areas are discussed below.

General overall electrical/electronic system capability must be demonstrated to meet the requirements for electromagnetic interference/compatibility. With the increase to more electrical/electronic control features for engine operation this area becomes paramount for rotorcraft safety. Also, the influence of moisture (rain/humidity), fuel/oil/hydraulic fluid, and dirt on electrical/electronic systems has been shown to be a major problem to system operation when adequate protection for electrical connectors and electrical/electronic components is not taken into account. Troubleshooting is a problem in some of these cases as the problem is only when moisture is present and disappears immediately upon drying out. Along with the above, electrical systems suffer from chafing, maintenance damage, inadequate pin engagement, poor grounding, etc. which make this a major problem to be properly documented during system tests to correct/improve the specific problem areas.

Moisture causes another basic problem in all systems including the engine-airframe area; i.e., corrosion. Detail design consideration along with verification during overall system evaluation is necessary. One problem with test aircraft has been they have been kept clean and sheltered such that an operational test may be necessary to disclose any potential corrosion problems. One test that can be done is that associated with verifying proper drainage. The engine/APU, drive system and associated accessory compartments should be verified for adequate drainage as well as areas on components; such as, castings, to insure no standing water. Engine/APU compartment drainage requirements must also insure flow rate requirements are met to prevent pooling of fuel/oil in case of a problem. Protection from water where freezing can cause problems also must be considered. For example, engine control cables not totally sealed from moisture have "frozen" on cold mornings after a rain.

Sand/dust is a generic problem for military aircraft which operate from unprepared sites. Contamination and erosion effects must be addressed as basis for general operation of the overall system. Blockage of coolers; contamination of bearing surfaces, mechanical switches, relays, circuit breakers causing sticking, wear and general inoperability; and erosion of rotating parts have all been well documented problems.

With a vibration environment along with operational cyclic conditions, fatigue and adverse wear will be generally show up in the long run. Specific component fatigue lives are determined based on their basic strength and usage, but many other general structural parts may show signs of the same fatigue mechanisms. These relate to the rotorcraft environment and no specific testing except for total system experience is defined to uncover these.

Besides specific operational requirements, maintainability requirements must also be verified. Except for Line Replacement Unit (LRU) and major component replacement times no specific requirements are defined. Servicing requirements; however, do require specific interface conditions and operational requirements. A specific verification procedure should be defined that validates each daily/periodic maintenance and servicing requirement. This will prove ability to inspect and service. Problems disclosed later with incompatible or problems with ground support equipment interfaces with servicing requirements, inability or lack of good indication of a sight gage, problems with servicing procedure, etc. can be addressed during qualification in lieu of after fielding.

Mission requirements will have basic performance related affects for an engine installation, but the most dominant requirement in newer aircraft has been the engine/airframe control system interaction in regard to transient response. With Nap-of-the-Earth and basic rotorcraft maneuvering capability, the engine-airframe system must be responsive to the operation demands. Specific maneuvers and pilot utilization of a more responsive aircraft has caused large transient rotor speed excursions. Ability to correct this has required intensive system integration effort to modify engine fuel control schedules, add additional anticipation sensors to the engine control, and tighten up airframe control hysteresis. Test techniques have also evolved to insure the type of transient maneuvers are properly addressed in an overall engine-airframe compatibility test.

One additional general system test has been the overall engine/APU fire protection system. Fire wall and proper sealing of penetrations/openings has normally been verified by analysis, similarity and during component level test. Fire detection and extinguishing capability have required both component and system level testing. As a system test, the fire detection system is usually verified for absence of false warning along with simulated fire detection. Fire extinguishing system testing has required verification through concentration of agent versus time over all fire zones to show compliance. This concentration test has had the most problems due to the turbulent airflow condition through an engine compartment causing variation in test to test as well as location. Sensor location as well as multiple bottle firings are important to insure adequate coverage.

CONCLUSION

The evaluation of rotorcraft design has required changes in the specification process to account for new design features and requirements. However, the basic success of these efforts is still dependent on a good qualification process. The "system" has to be qualified in total to insure all physical and functional interfaces are coordinated. The importance in planning this effort in detail is essential for getting documented results to correct/improve/validate the system. With the trend toward "fully integrated" design concepts this theme for a system approach has to be carried forth as the primary objective.

SELECTIVE BIBLIOGRAPHY

This bibliography with Abstracts has been prepared to support AGARD Lecture Series No. 148 by the Scientific and Technical Information Branch of the US National Aeronautics and Space Administration, Washington, D.C., in consultation with the Lecture Series Director, V. E. Edwards, Directorate for Engineering, U. S. Army Aviation Systems Command, St. Louis, Missouri.

UTTL: Ka-32 multimission helicopter designed to operate in harsh climate Aviation Week and Space Technology (ISSN 0005-2175), vol. 123, Sept. 2, 1985, p. 69.

ABS: The Soviet Ka-32 utility helicopter has been designed for both land-based transport operations and a variety of shipboard missions, all of which, as in the case of Arctic icebreaker escort duties, have led to the design of a configuration well suited for harsh environments. The Ka-32 is equipped for all-weather day/night flying, using an automatic flight control system and a radar that facilitates instrument approaches to remote landing sites. Two 2225-hp turboshaft engines are employed. The cabin has provisions for up to 16 passengers or 4000 kg of cargo, but can also undertake airlifts of externally carried loads. The two rotors of the helicopter contrarotate and incorporate thermoelectric deicing systems. 85/09/02 85A46597

UTTL: Rotor dynamical instability; Proceedings of the Applied Mechanics, Bioengineering, and Fluids Engineering Conference, Houston, TX, June 20-22, 1983

AUTH: A/ADAMS, M. L., JR. Conference sponsored by the American Society of Mechanical Engineers. New York, American Society of Mechanical Engineers (Applied Mechanics Symposia Series. AMD Volume 55), 1983, 105 p.

ABS: A cross-section of recent engineering research on rotor dynamical instability is presented. The physical nature of rotor instability mechanisms is inventoried and the forces induced on a whirling rotor by a labyrinth seal are investigated. Attention is given to the influence of unbalance on the stability characteristics of flexible rotor bearing systems. Design criteria are defined for improving the stability of centrifugal compressors, and perturbation tests of bearing/seal configurations for evaluating the dynamic coefficients are described. For individual items see A84-13227 to A84-13231 83/00/00 84A13226

UTTL: Engine/drive/airframe compatibility: A way of life

AUTH: A/ALBRECHT, C. CORP: Boeing Vertol Co., Philadelphia, Pa. In AGARD Helicopter Propulsion Systems 7 p (SEE N82-17203 08-07)

ABS: Engine/drive/airframe compatibility in development stages or during production of a new design were studied. Various compatibility encounters on Boeing Vertol helicopters, analyses, solutions as effected by current technology and engine/airframe manufacturer relationships and dynamic interface items are discussed. 81/09/00 82N17220

UTTL: Engine-airframe transient compatibility - Analysis and test

AUTH: A/ALWANG, J. R. In: Specialists' Meeting on Helicopter Propulsion Systems, Williamsburg, Va., November 6-8, 1979, Technical Papers. (A81-17501 05-01) Washington, D.C., American Helicopter Society, 1980. 9 p.

ABS: This paper discusses dynamic analysis and test requirements related to engine-airframe compatibility in two areas: drive system torsional stability and transient response. Prediction of stability characteristics using linear techniques and correlation with experimental data are reviewed. Control testing, leading to determination of worst-case conditions, is presented with an aircraft flight test procedure for providing an accurate evaluation of propulsion system stability. The importance of developing a nonlinear component-by-component engine simulation for optimizing control functional configurations and transient schedules is emphasized. Use of a dynamic simulation based on steady state matching deck performance for analyzing specific interface problems, including transient loss in compressor surge margin, is demonstrated. 80/00/00 81A17517

UTTL: Heavy-lift helicopter engine control system

AUTH: A/ALWANG, J. R.; B/MCLAIN, R. D. American Helicopter Society, Annual National V/STOL Forum, 30th, Washington, D.C., May 7-9, 1974, 12 p.

ABS: The Heavy-Lift Helicopter is planned as a component of a U.S. Army logistics system with the primary mission of picking up and delivering a 22.5-ton external payload. The control aspects of the XT701-AD-700 engines for the Heavy-Lift Helicopter are described. Control system requirements as related to the air/vehicle interface are discussed with emphasis on the engine control fly-by-wire concept, mission objectives, and system response. A description of the major engine control components is given, and particular airframe integration efforts are discussed. A brief review of the dynamic analysis and representative results are presented.

RPT#: AHS PREPRINT 870 74/05/00 74A36605

UTTL: Infrared suppressor effect on T63 turboshaft engine performance
AUTH: A/BAILEY, E. E.; B/CIVINSKAS, K. C.; C/WALKER, C. L. CORP: National Aeronautics and Space Administration. Lewis Research Center, Cleveland, Ohio.
ABS: Tests were conducted to determine if there are performance penalties associated with the installation of infrared (IR) suppressors on the T63-A-700 turboshaft engine. The testing was done in a sea-level, static test cell. The same engine (A-E402808 B) was run with the standard OH-58 aircraft exhaust stacks and with the ejector-type IR suppressors in order to make a valid comparison. Repeatability of the test results for the two configurations was verified by rerunning the conditions over a period of days. Test results showed no measurable difference in performance between the standard exhaust stacks and the IR suppressors.
RPT#: NASA-TM-78970 E-9730 AVRADCOM-TR-78-38(PL) 78/09/00 79N11043

UTTL: Dynamic analysis of multi-degree-of-freedom systems using phasing matrices
AUTH: A/BIELAWA, R. L. CORP: United Aircraft Corp., East Hartford, Conn. CSS: (Research Labs.) In NASA. Ames Res. Center Rotorcraft Dyn. p 35-43 (SEE N74-34489 24-02)
ABS: A mathematical technique is presented for improved analysis of a wide class of dynamic and aeroelastic systems characterized by several degrees-of-freedom. The technique enables greater utilization of the usual eigensolution obtained from the system dynamic equations by systematizing the identification of destabilizing and/or stiffening forces. Included, as illustrative examples of the use of the technique, are analyses of a helicopter rotor blade for bending-torsion divergence and flutter and for pitch-lag/flap instability. 74/00/00 74N34493

UTTL: Adaptive fuel control feasibility investigation
AUTH: A/BOLTON, R. L. In: American Helicopter Society, Annual Forum, 38th, Anaheim, CA, May 4-7, 1982, Proceedings. (A82-40505 20-01) Washington, DC, American Helicopter Society, 1982, p. 150-155.
ABS: Responsive, stable power control has been an important design objective since the advent of the helicopter. Many helicopters flying today, however, suffer from less than desirable power response because of the difficulty in designing a system which is rapid in rotor speed control over the entire operating envelope. An optimization regarding the response of engine power for all flight regimes would be particularly important in the case of a combat helicopter. A description is presented of an on-going program of investigation of adaptive fuel control concepts utilizing existing full authority electronic systems capability. The concepts developed in this program are being analyzed by use of a full dynamic computer simulation of the engine/fuel control/airframe. It was found that fuel consumption can be reduced as much as 10% by variation of rotor speed for very specific cruise conditions, i.e., high speed, high altitude, and heavy load. 82/00/00 82A40519

UTTL: Engine/airframe/drive train dynamic interface documentation
AUTH: A/BOWES, M. A. CORP: Kaman Aerospace Corp., Bloomfield, Conn.
ABS: An historical review of Kaman Aerospace Corporation helicopter development effort was conducted. Information pertaining to instances of engine/airframe/drive train incompatibility was extracted from the resulting review data. The details of each incompatibility problem and its solution are presented and discussed. These problems were found to be associated with normally occurring rotor and drive system vibratory excitations and resonance amplified rotor and drive system vibratory excitations. It is concluded that the type of incompatibility problems encountered in helicopter development programs can be expected in the future. For the most part, these problems, and their solutions, are an intrinsic part of the development process, and they will not be eliminated through the application of increasingly more sophisticated and complex analytical design tools. It is recommended that research in this area by directed towards improved dynamic testing methods which will enable more rapid, efficient determination of the exact nature of future incompatibility problems, once they are encountered.
RPT#: AD-A058197 R-1536 USARTL-TR-78-14 78/06/00 79N12094

UTTL: A summary of rotorcraft handling qualities research at NASA Ames Research Center
AUTH: A/CHEN, R. T. CORP: National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. In NASA. Langley Research Center NASA Aircraft Controls Research, 1983 p 51-68 (SEE N84-20567 11-08)
ABS: The objectives of the rotorcraft handling qualities research program at Ames Research Center are twofold: (1) to develop basic handling qualities design

criteria to permit cost effective design decisions to be made for helicopters, and (2) to obtain basic handling qualities data for certification of new rotorcraft configurations. The research on the helicopter handling qualities criteria has focused primarily on military nap-of-the-earth (NOE) terrain flying missions, which are flown in day visual meteorological conditions (VMC) and instrument meteorological conditions (IMC), or at night. The Army has recently placed a great deal of emphasis on terrain flying tactics in order to survive and effectively complete the missions in modern and future combat environments. Unfortunately, the existing Military Specification MIL-H 8501A which is a 1961 update of a 1951 document, does not address the handling qualities requirements for terrain flying. The research effort is therefore aimed at filling the void and is being conducted jointly with the Army Aeromechanics Laboratory at Ames. The research on rotorcraft airworthiness standards with respect to flying qualities requirements was conducted to collaboration with the Federal Aviation Administration (FAA). 84/03/00 84N20571

UTTL: Flap-lag-torsion aeroelastic stability of circulation-controlled rotors in hover
AUTH: A/CHOPRA, I.; B/JOHNSON, W. CORP: National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif.; Army Aviation Research and Development Command, Moffett Field, Calif. In: American Helicopter Society, Annual National Forum, 34th, Washington, D.C., May 15-17, 1978, Proceedings. (A79-18126 05-01) Washington, D.C., American Helicopter Society, 1978. 15 p.
ABS: The results of a theoretical investigation of the flap-lag-torsion stability of circulation controlled rotors in hover are presented. Stability boundaries are presented as a function of thrust and lag frequency, at several levels of blowing coefficient, for flap frequencies of 1.1/rev and 1.8/ rev. The effects of several parameters on the blade flap-lag stability are examined, including structural damping, structural coupling, pitch-lag and pitch-flap coupling, and the blade feathering motion. The trailing edge blowing can have a major impact on the blade aeroelastic stability, which should be considered in the rotor design. The implications of these results for the current CCR and X-Wing rotorcraft designs are considered.
RPT#: AHS 78-64 78/00/00 79A18185

UTTL: Gas turbine sand and dust effects and protection methods
AUTH: A/CONNORS, H. D.; B/MURPHY, J. P. SOCIETY OF AUTOMOTIVE ENGINEERS, COMBINED NATIONAL FARM, CONSTRUCTION AND INDUSTRIAL MACHINERY AND POWERPLANT MEETINGS, MILWAUKEE, WIS., SEP. 14-17, 1970.
RPT#: SAE PAPER 700705 70/09/00 70A42671

UTTL: Design and development of the Model 412 helicopter
AUTH: A/CRESAP, W. L.; B/MYERS, A. W. In: American Helicopter Society, Annual Forum, 36th, Washington, DC, May 13-15, 1980, Proceedings. (A81-40136 18-01) Washington, DC, American Helicopter Society, 1980. 12 p.
ABS: Using multibladed rotor technology developed with the Model 654 rotor, a four-bladed, soft-inplane rotor system has been developed for the Model 212 helicopter. The new rotor was designed to permit an increase in the allowable gross weight of the 212 and to eliminate airspeed restrictions caused by vibrations and rotor loads. The four-bladed system was chosen in lieu of a new two-bladed rotor/nodal beam combination because fewer fuselage modifications were required and retrofit was easier. In eight months of developmental flying, the Model 412, as the new configuration is designated, has accumulated over 200 flight hours, including high altitude and cold weather tests. Measurements during the preliminary flight load survey to speeds of 162 knots and maneuvers to 2.3 g indicate the design objectives have been achieved. Predicted weight and performance improvements have been verified, and the helicopter is ready for FAA certification testing.
RPT#: AHS 80-56 80/00/00 81A40181

UTTL: Flag-lag-torsional dynamics or extensional and inextensional rotor blades in hover and in forward flight
AUTH: A/CRESPODASILVA, M. R. M. CORP: Cincinnati Univ., Ohio. CSS: (Dept. of Aerospace Engineering and Applied Mechanics.)
ABS: The differential equations describing the flap-lag-torsional motion of a flexible rotor blade including third-order nonlinearities were derived for hover and forward flight. Making use of the two boundary conditions, those equations were reduced to a set of three integro partial differential equations written in terms of the flexural deflections and the torsional variable.
RPT#: NASA-CR-165078 81/12/00 82N15013

UTTL: Problems of engine response during transient maneuvers
AUTH: A/DINI, D. CORP: Pisa Univ. (Italy). CSS: (Ist. di Macchine.) In AGARD Helicopter Propulsion Systems 14 p (SEE N82-17203 08-07)
ABS: Helicopter transition flight regime and extreme flight maneuvers which determine abrupt variation of the rotor drag torque during each revolution, and the corresponding effects on the engine are discussed. Repeated surging and the attendant transient torsional loads from the engine cause damage to the airframe components. The angular rotor acceleration, cyclically variable, influences the turbine gas producer rotor speed, introducing flow distortions and aeromechanical effects in all the engine. It is found that periodic and inertial blade loading has serious consequences to discs and shaft to which the blades are attached. The higher harmonics of the excitation over the discs provokes relevant flexural modes of forced vibration. Operation in transient rating with pitch increase or decrease has the greatest effect on the helicopter flight performance, owing to the power application capability and the fuel control system adaptability. It is shown that to obtain sufficient engine/airframe dynamic compatibility, aircraft developments have to incorporate technical advances, which include harmonic integrated controls for propulsion and flight systems. 81/09/00 82N17221

UTTL: Miniaturized multisensor probes based on coated wire ion selective electrodes
AUTH: A/FREISER, H. CORP: Arizona Univ., Tucson. CSS: (Dept. of Chemistry.)
ABS: Research concerned with the development of selective, sensitive, and rugged methods of analysis for organic species in sea water and general oceanographic environments as well as those that are associated with CW agents, that would be capable of application in general field use has been carried out. A two-pronged attack involving (a) multi-sensor ion probes based on coated wire electrode technology and (b) a newly developed liquid chromatographic technique for neutral species has been employed.
RPT#: AD-A123415 TR-1 83/01/11 83N24837

UTTL: Engine/transmission/airframe advanced integration techniques
AUTH: A/HIMKA, T.; B/SEMPLE, R. D. CORP: Boeing Vertol Co., Philadelphia, Pa.
ABS: Innovative engine/transmission/airframe integrated design concepts were developed to provide total airflow and power management for a utility transport helicopter which meets projected requirements of future Army aircraft. These requirements include engine compartment cooling, drive train and transmission oil cooling, engine oil cooling, exhaust plume and hot metal infrared (IR) signature suppression, and engine inlet foreign particle protection.
RPT#: AD-A012236 D210-10900-1 USAAMRDL-TR-75-16 75/05/00 76N11094

UTTL: The vibratory airloading of helicopter rotors
AUTH: A/HOOPER, W. E. (European Rotorcraft and Powered Lift Aircraft Forum, 9th, Stresa, Italy, Sept. 13-15, 1983) Vertica (ISSN 0360-5450), vol. 8, no. 2, 1984, p. 73-92.
ABS: A survey has been made of all major wind tunnel and full-scale flight tests conducted over the last 29 years to examine the nature of the vibratory aerodynamic loading which causes helicopter vibration. Using computer generated surface plots, the present paper compares the airload distributions for rotors which have from 2 to 6 blades by presenting the data in identical plotting formats which allow comparisons of the effects of major parameters including blade number, blade/vortex proximities, propulsive force and forward speed. By harmonically analyzing the data, it has been possible to show striking similarities between the vibration-causing higher harmonics of airloading on different rotor designs and in widely varying flight conditions. By selectively eliminating the lower harmonics, the predominant modes of vibratory forcing, to which all helicopter blades are subjected, are revealed. 84/00/00 84A46273

UTTL: Elementary applications of a rotorcraft dynamic stability analysis
AUTH: A/JOHNSON, W. CORP: National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif.; Army Air Mobility Research and Development Lab., Moffett Field, Calif.
ABS: A number of applications of a rotorcraft aeroelastic analysis are presented to verify that the analysis encompasses the classical solutions of rotor dynamics, and to examine the influence of certain features of the model. Results are given for the following topics: flapping frequency response to pitch control; forward flight flapping stability; pitch/flap flutter and divergence; ground resonance instability; and the flight dynamics of several representative helicopters.
RPT#: NASA-TM-X-73161 A-6717 76/06/00 76N33129

UTTL: Development of particle separator engine air induction system for the OH-58A light observation helicopter.
AUTH: A/KAWA, M. M. IN- ENVIRONMENTAL EFFECTS ON AIRCRAFT AND PROPULSION SYSTEMS, NAVAL AIR PROPULSION TEST CENTER, ANNUAL NATIONAL CONFERENCE, 9TH, BORDENTOWN, N.J., OCT. 7-9, 1969, PROCEEDINGS. P. 5-1 TO 5-35. /A70-10678 01-28/ 69/00/00 70A10681

UTTL: Operational experience with T53 engines in U.S. Army helicopters.
AUTH: A/KLEINER, M. L.; B/SABOE, M. S. AMERICAN HELICOPTER SOCIETY, JOURNAL, VOL. 14, P. 1-11. /AMERICAN HELICOPTER SOCIETY, ANNUAL NATIONAL FORUM, 24TH, WASHINGTON, D.C., MAY 8-10, 1968./
RPT#: AHS PAPER 215 69/01/00 69A18867

UTTL: Characteristics of the occurrence and elimination of a stall in an axial-flow compressor in the presence of a rotating irregularity at the inlet
AUTH: A/KOZAREV, L. A.; B/FEDOROV, R. M. Aviatsionnaia Tekhnika (ISSN 0579-2975), no. 1, 1983, p. 33-37. In Russian.
ABS: The characteristics of a two-stage axial-flow compressor with a large hub are investigated experimentally over a wide range of air flow rates including both stable and unstable operating conditions. A special rotating stall simulator is used to study the conditions existing at the inlet of the high-pressure stage of a two-shaft engine in the case of a stall in the low-pressure stage. It is shown that the effect of a rotating irregularity at the compressor inlet on the preformance characteristics of the high-pressure stage is largely determined by the rotation frequency of the irregularity zone and that of the compressor rotor. 83/00/00 83A42132

UTTL: A formulation of rotor-airframe coupling for design analysis of vibrations of helicopter airframes
AUTH: A/KVATERNIK, R. G.; B/WALTON, W. C., JR. CORP: National Aeronautics and Space Administration. Langley Research Center, Hampton, Va.
ABS: A linear formulation of rotor airframe coupling intended for vibration analysis in airframe structural design is presented. The airframe is represented by a finite element analysis model; the rotor is represented by a general set of linear differential equations with periodic coefficients; and the connections between the rotor and airframe are specified through general linear equations of constraint. Coupling equations are applied to the rotor and airframe equations to produce one set of linear differential equations governing vibrations of the combined rotor airframe system. These equations are solved by the harmonic balance method for the system steady state vibrations. A feature of the solution process is the representation of the airframe in terms of forced responses calculated at the rotor harmonics of interest. A method based on matrix partitioning is worked out for quick recalculations of vibrations in design studies when only relatively few airframe members are varied. All relations are presented in forms suitable for direct computer implementation.
RPT#: NASA-RP-1089 L-14243 NAS 1.61:1089 82/06/00 84N33832

UTTL: A stability and control prediction method for helicopters and stoppable rotor aircraft. Volume 1 - Engineer's manual Final report
AUTH: A/LIVINGSTON, C. L. CORP: Bell Helicopter Co., Fort Worth, Tex.
RPT#: AD-707881 AFFDL-TR-69-123-VOL-1 70/02/00 70N39729

UTTL: Turbine engine dynamic compatibility with helicopter airframes
AUTH: A/MARD, K. C.; B/VON HARDENBERG, P. W. CORP: United Aircraft Corp., Stratford, Conn. CSS: (SIKORSKY AIRCRAFT DIV.) IN NRL THE SHOCK AND VIBRATION BULL. 39, PART 3 JAN. 1969 P 17-30 /SEE N70-13621 03-32/ 69/01/00 70N13623

UTTL: Propeller blade vibrations
AUTH: A/MARYNIAK, J.; B/MIERZEJEWSKI, W.; C/KRUTUL, J. Mechanika Teoretyczna i Stosowana, vol. 11, no. 3, 1973, p. 229-243. In Polish.
ABS: Calculated modes and frequencies of natural vibrations for helicopter tail-rotor blades are compared with experimental results. Measured values of mass distribution, moments of inertia, and stiffness are used as the input data for calculations. The rotor blade is represented by eleven segments and is treated as a discrete system. Six mathematical models of the blade are analyzed with allowance for rotational inertia of individual blade segments. Eigenmodes and frequencies of flexural, torsional, and torsional-flexural vibrations calculated for the six models are compared with resonance measurements. 73/00/00 73A45245

UTTL: Oil-air mist lubrication for helicopter gearing
AUTH: A/MCGROGAN, F. CORP: Sikorsky Aircraft, Stratford, Conn.
ABS: The applicability of a once-through oil mist system to the lubrication of helicopter spur gears was investigated and compared to conventional jet spray lubrication. In the mist lubrication mode, cooling air was supplied at 366K (200 F) to the out of mesh location of the gear sets. The mist air was also supplied at 366K (200 F) to the radial position mist nozzle at a constant rate of 0.0632 mol/s (3 SCFM) per nozzle. The lubricant contained in the mist air varied between 32 - 44 cc/hour. In the recirculating jet spray mode, the flow rate was varied between 1893 - 2650 cc/hour. Visual inspection revealed the jet spray mode produced a superior surface finish on the gear teeth but a thermal energy survey showed a 15 - 20% increase in heat generated. The gear tooth condition in the mist lubrication mode system could be improved if the cooling air and lubricant/air flow ratio were increased. The test gearbox and the procedure used are described.
RPT#: NASA-CR-135081 SER-50959 76/12/00 78N25080

UTTL: The Model 412 multi-bladed rotor system
AUTH: A/MEYERS, A. W.; B/PHILLIPS, N. B.; C/SNYDER, D. E. American Helicopter Society, National Specialists Meeting on Rotor System Design, Philadephia, PA, Oct. 22-24, 1980, Paper. 9 p.
ABS: A multibladed soft-inplane main rotor system is being certified for the Bell Model 412 utility helicopter. This rotor system is a third-generation design and is the result of technology developed in earlier multibladed soft-inplane rotor programs. The new rotor was designed to easily retrofit existing helicopters in the field. The design of a fiberglass blade was also included in the program. Torsional loads encountered during the development program were higher than anticipated because of the influence of live twist on rotor response and control loads. Increased trailing edge reflex was used to significantly affect both rotor response and control loads. Bell's technical predictions for the rotor have been confirmed in the development and FAA certification phases of the program. 80/10/00 82A26376

UTTL: Pilot command interfaces for discrete control of automated nap-of-earth flight
AUTH: A/MOUNTFORD, S. J.; B/PENNER, R.; C/BURSCH, P. IN: Digital Avionics Systems Conference, 6th, Baltimore, MD, December 3-6, 1984, Proceedings (A85-17801 06-01). New York, American Institute of Aeronautics and Astronautics, 1984, p. 386-392.
ABS: It is pointed out that workload levels are excessive in many crewstations. This applies, in particular, to the two-man crew of an attack helicopter performing nap-of-the-earth (NOE) flight. For a number of reasons, it would be desirable to reduce the crew size to that of a single operator. Preliminary workload studies conducted by an American aerospace company on single pilot mission scenarios of scout-attack helicopters indicate that multitask performance during continuous flight control can only be achieved by automating some tasks. The automation of some features of flight control could be one of the prime candidates for implementation. An important human engineering design issue is related to the design of the control interface to allow the pilot to reenter the flight control loop to update flight objectives. The present investigation represents a preliminary attempt to develop some command language interface concepts for integration with an automatic flight control system.
RPT#: AIAA PAPER 84-2621 84/00/00 85A17859

UTTL: The Shock and Vibration Digest, volume 16, no. 7
AUTH: A/NAGLE-ESHLEMAN, J. CORP: Shock and Vibration Information Center (Defense), Washington, D. C.
RPT#: AD-A143958 84/07/00 84N31679

UTTL: Engine/airframe/drive train dynamic interface documentation
AUTH: A/NEEDHAM, J. F.; B/BANERJEE, D. CORP: Hughes Helicopters, Culver City, Calif.
ABS: This report pertains to engine/airframe/drive train dynamic interface problems experienced by helicopters and the methodology used to avoid dynamic interface problems. The problem of low torsional frequency of the engine/airframe/drive train dynamic system that results from the design philosophy of using a stationary main rotor mast enclosing a separate 'floating' torque drive shaft is addressed. The use of supercritical shafts and vibration isolators are described and mobility methods to preclude engine/airframe vibration problems are discussed.
RPT#: AD-A056956 HH-78-31 USARTL-TR-78-12 78/05/00 79N10064

UTTL: Engine/airframe dynamic interface investigation using T63/LOH helicopter system
AUTH: A/PARKER, W. H. In: American Helicopter Society, Annual National Forum, 32nd, Washington, D.C., May 10-12, 1976, Proceedings. (A77-26851 11-01) Washington, D.C., American Helicopter Society, 1976,

p. 1074-1 to 1074-7.
ABS: An investigation is presented to develop a common language for use in engine/airframe translational vibration specification and analysis of future helicopter systems. Mobility and modal synthesis techniques are applied to the coupled dynamic analysis of the T63/OH58 helicopter installation. Correlations with shake test and flight vibration data are presented for each analysis. It is shown that the mobility technique, applied with shake test mobilities results in an inadequate correlation while the modal synthesis technique, using analytical modal descriptions of the engine and airframe, provides a reasonable correlation. Recommendations are formulated for the use of the modal synthesis method of analysis as a specification methodology. 76/00/00 77A26894

UTTL: Helicopter vibration suppression using simple pendulum absorbers on the rotor blade
AUTH: A/PIERCE, G. A.; B/HAMOUDA, M. N. H. CORP: Georgia Inst. of Tech., Atlanta. CSS: (School of Aerospace Engineering.)
ABS: Simple pendulums are installed on the blades of a helicopter rotor to suppress the root reactions. A frequency response analysis is conducted of typical rotor blades excited by a harmonic variation of spanwise airload distributions as well as a concentrated load at the tip. The effect of pendulum tuning on the minimization of the hub reactions is considered. A properly designed flapping pendulum attenuates the root out of plane force and moment whereas the optimum designed lead lag pendulum attenuates the root in plane reactions. A properly tuned pendulum can attenuate the vibratory loads by generating appropriate forces at its attachment point with the blade. These forces redistribute the loads on the blade so that only a small portion of the reactions is transmitted to the hub. For optimum pendulum tuning the parameters to be determined are the pendulum uncoupled natural frequency, the pendulum spanwise location and its mass. It is found that the optimum pendulum frequency is in the vicinity of the excitation frequency. A pendulum can be tuned and its optimum mass determined by excitation with a concentrated simple harmonic load at the tip. However, it is necessary to utilize distributed airloads to accurately determine the attenuation of the root reactions.
RPT#: NASA-CR-3619 NAS 1.26:3619 82/08/00 82N33734

UTTL: Standardization of helicopter certification requirements in the western world
AUTH: A/PORTET, H. CORP: Societe Nationale Industrielle Aerospatiale, Marignane (France). CSS: (Airworthiness and Certification Dept.) Presented at 10th European Rotorcraft Forum, The Hague, 28-31 Aug. 1984
ABS: European cooperation in drawing up airworthiness codes is reviewed and the need to extend this to the helicopter industry is stressed. Regulations for the power transmission system endurance test, takeoff, and engine power ratings are discussed.
RPT#: SNIAS-851-210-103 84/00/00 85N31048

UTTL: Engine/airframe/drive train dynamic interface documentation
AUTH: A/RICHARDSON, D. A.; B/ALWANG, J. R. CORP: Boeing Vertol Co., Philadelphia, Pa.
ABS: Engine/airframe/drive train dynamic interface problems of Boeing helicopters are described. The investigation leading to the problem solution, the solution, and its limitations are discussed. Forecasts of potential future problems, recommendation for investigations, and specifications are included.
RPT#: AD-A055766 D210-11328-1 USARTL-TR-78-11 78/04/00 78N31114

UTTL: Helicopter engine development - New standards for the '80s
AUTH: A/RUEGG, R. G. American Helicopter Society, Specialists Meeting on Rotary Wing Propulsion Systems, Williamsburg, VA, Nov. 16-18, 1982, Preprint. 10 p.
ABS: The T700 Development and Maturity Program included a number of specially designed engine tests to assure early maturity of the engine prior to production and to demonstrate parts-life integrity well in advance of significant field experience. The present investigation is concerned with the methods used to develop the T700 accelerated mission test cycles and compares test cycle severity with both intended and actual field usage, through use of the T700 Engine History Recorder and the Engine Life Usage Monitor. Flight recorder data provide additional information on the relationship between the test cycles and field mission usage. Finally, the experience gained during the T700 Development Program is discussed as it relates to future turboshaft engine development programs. 82/11/00 83A24837

UTTL: Reliability model for planetary gear
AUTH: A/SAVAGE, M.; B/PARIDON, C. A.; C/COY, J. J. CORP: National Aeronautics and Space Administration. Lewis Research Center, Cleveland, Ohio.; Army Aviation Research and Development Command, Cleveland, Ohio. Proposed for Presentation at Design Engr. Tech. Conf., Washington, D.C., 12-15 Sep. 1982; sponsored by ASME
ABS: A reliability model is presented for planetary gear trains in which the ring gear is fixed, the Sun gear is the input, and the planet arm is the output. The input and output shafts are coaxial and the input and output torques are assumed to be coaxial with these shafts. Thrust and side loading are neglected. This type of gear train is commonly used in main rotor transmissions for helicopters and in other applications which require high reductions in speed. The reliability model is based on the Weibull distribution of the individual reliabilities of the transmission components. The transmission's basic dynamic capacity is defined as the input torque which may be applied for one million input rotations of the Sun gear. Load and life are related by a power law. The load life exponent and basic dynamic capacity are developed as functions of the component capacities.
RPT#: NASA-TM-82859 NAS 1.15:82859 AVRADCOM-TR-82-C-6 82/00/00 82N28643

UTTL: Literature search of publications concerning the prediction of dynamic inlet flow distortion and related topics
AUTH: A/SCHWEIKHHARD, W. G.; B/CHEN, Y. S. CORP: Kansas Univ., Lawrence.
ABS: Publications prior to March 1981 were surveyed to determine inlet flow dynamic distortion prediction methods and to catalog experimental and analytical information concerning inlet flow dynamic distortion prediction methods and to catalog experimental and analytical information concerning inlet flow dynamics at the engine-inlet interface of conventional aircraft (excluding V/STOL). The sixty-five publications found are briefly summarized and tabulated according to topic and are cross-referenced according to content and nature of the investigation (e.g., predictive, experimental, analytical and types of tests). Three appendices include lists of references, authors, organizations and agencies conducting the studies. Also, selected materials summaries, introductions and conclusions - from the reports are included. Few reports were found covering methods for predicting the probable maximum distortion. The three predictive methods found are those of Melick, Jacox and Motycka. The latter two require extensive high response pressure measurements at the compressor face, while the Melick Technique can function with as few as one or two measurements.
RPT#: NASA-CR-3673 NAS 1.26:3673 83/02/00 83N18729

UTTL: Flights of the XH-51A compound helicopter.
AUTH: A/SEGNER, D. R. SOCIETY OF EXPERIMENTAL TEST PILOTS, TECHNICAL REVIEW, VOL. 9, NO. 1, P. 68-79. 68/00/00 68A27821

UTTL: Technology evolution in the Allison Model 250 engine
AUTH: A/STEVENS, E. C. In: European Rotorcraft and Powered Lift Aircraft Forum, 4th, Stresa, Italy, September 13-15, 1978, Proceedings. Volume 2. (A79-18637 06-01) Gallarate, Italy, Costruzioni Aeronautiche Giovanni Agusta S.p.A., 1978, p. 48-1 to 48-20.
ABS: The evolution of compressor and turbine technology in the Allison Model 250 gas turbine engine can be largely attributed to the successful application of finite element analysis techniques. The technology advances in the combustor resulted from the addition of a smaller-diameter prechamber upstream of the main chamber. The combined effects of the component technology improvements on the advanced Model 250 engines are increased power, increased reliability, and improvements in both engine power-to-weight ratio and cruise SFC. 78/00/00 79A18681

UTTL: OH-6A propulsion system vibration investigation
AUTH: A/SULLIVAN, R. J.; B/HEAD, R. E.; C/KORKOSZ, G. J.; D/NEFF, J. R.; E/SOLTIS, S. J. CORP: Hughes Helicopters, Culver City, Calif.
ABS: A study was made of means to improve helicopter engine/airframe vibratory interface compatibility. The study was based on the characteristics of the OH-6A helicopter and the T63 engine in order to utilize existing data to validate the methodology produced in the study. Available engine and airframe data was reviewed and used to prepare a vibration spectrum for a typical mission. Airframe mobility data was acquired during a ground shake test. A finite element model of the OH-6A airframe was prepared and was coupled to a model of the T63 engine, which was based on mobility data supplied by the engine manufacturer. The airframe and engine models were combined using a modal coupling technique, and reasonable correlation with test data was obtained. An impedance/mobility coupling technique is recommended where many flexible modes are in the frequency ranges that could be excited by main-rotor-induced vibratory forces. Based on the

study, recommendations were prepared and circulated to engine manufacturers for future engine vibration specification, vibration parameter selection, and data reduction. The results of the survey and the proposed recommendations are presented.
RPT#: AD-A007225 REPT-369-V-8009(HH-74-114) USAAMRDL-TR-74-85 75/01/00 75N22328

UTTL: Installed engine performance in dust-laden atmosphere
AUTH: A/TABAKOFF, W.; B/HAMED, A. AIAA, AHS, ASEE, Aircraft Design Systems and Operations Meeting, San Diego, CA, Oct. 31-Nov. 2, 1984. 11 p.
ABS: Aircraft engines operating in areas where the atmosphere is polluted with small solid particles are subjected to erosion damage and consequently, their performance deteriorates. This paper presents the results of an investigation of the solid particles dynamics through a helicopter engine with inlet particle separator. The nonseparated particle trajectories were determined through the five stage axial flow compressor. The results are presented to show the particle distribution throughout the compressor and the intensity of the particle blade impacts at the various stages.
RPT#: AIAA PAPER 84-2488 AD-A150110 84/10/00 85A13563

UTTL: Dynamic behavior of transmission systems
AUTH: A/THIBERT, F.; B/MAQUIN, F. Association Aeronautique et Astronautique de France, European Rotorcraft Forum, 8th, Aix-en-Provence, France, Aug. 31-Sept. 3, 1982, Paper. 7 p.
ABS: When designing a helicopter, it is sought to avoid resonance phenomena which might result from the fact that the natural frequency of an assembly would turn out to be located on the excitation frequencies that helicopter blades pass on to the structure. For instance, the natural frequency of the tail rotor drive must be kept away from the torque excitation generated by the main rotor blades, whose frequency is a multiple of the main rotor rotation speed and of the number of rotor blades. Yet other oscillatory phenomena are likely to occur during the first flight tests of the aircraft: torsional oscillations of the drive system at relatively low frequencies (on the order of 3 to 6 hertz) generally excited by a slight disturbance of the resistant aerodynamic torque applied on the tail rotor. Depending on their specific amplitude and damping, these angular oscillations can be deemed excessive and require that the characteristics of the entire linkage be adjusted in flight. 82/08/00 84A19626

UTTL: The nonlinear instability in flap-lag of rotor blades in forward flight
AUTH: A/TONG, P. CORP: Massachusetts Inst. of Tech., Cambridge. CSS: (Aeroelastic and Structures Research Lab.)
ABS: The nonlinear flap-lag coupled oscillation of torsionally rigid rotor blades in forward flight is examined using a set of consistently derived equations by the asymptotic expansion procedure of multiple time scales. The regions of stability and limit cycle oscillation are presented. The roles of parametric excitation, nonlinear oscillation, and forced excitation played in the response of the blade are determined.
RPT#: NASA-CR-114524 ASRL-TR-166-2 71/10/00 73N17017

UTTL: Exhaust gas reingestion measurements
AUTH: A/TURCZENIUK, B. In: Specialists' Meeting on Helicopter Propulsion Systems, Williamsburg, Va., November 6-8, 1979, Technical Papers. (A81-17501 05-01) Washington, D.C., American Helicopter Society, 1980. 12 p.
ABS: Results of flight tests to measure the magnitude and effects of engine exhaust gas reingestion and inlet pressure distortion are presented. The tandem rotor CH-47 and single rotor YUH-61A helicopters were flown with instrumented engine inlets using fast-response probes. A wide range of flight maneuvers were evaluated, including operating in and out of ground effect, flares, hovering turns, lateral and rearward flight. The dynamic inlet air temperature and pressure patterns, time histories, and variation of basic engine parameters during exhaust reingestion are provided. The effect of instrumentation time constant on the maximum inlet temperature rise is discussed. Also shown is the effect of an inlet screen on pressure distortion. Conclusions are drawn as to the use of this data to establish turbine engine compressor stall margin requirements needed to avoid experiencing problems with the engine/helicopter integration. 80/00/00 81A17521

UTTL: Review of engine/airframe/drive train dynamic interface development problems
AUTH: A/TWOMEY, W. J.; B/HAM, E. H. CORP: United Technologies Corp., Stratford, Conn. CSS: (Sikorsky Aircraft Div.)
ABS: The coupled interaction between two or more helicopter subsystems has often been the source of vibration problems - problems often costly and time-consuming to correct because they have not surfaced until the design and development of the individual subsystems

are far advanced. This report gives a review of Sikorsky experience with such problems over the past twenty years of developing gas turbine powered helicopters. It represents part of an overall Government effort to accumulate data which will eventually lead to solutions of generic problems of this type. The problems presented include forced vibration problems (wherein the excitations come from either aerodynamic loads on the main rotor, or mechanical imbalances in the engine/drive train), self-excited vibrations, and a transient response problem.

RPT#: AD-A057932 SER-510003 USARTL-TR-78-13 78/06/00
79N12092

UTTL: Review of engine/airframe/drive train dynamic interface development problems

AUTH: A/TWOMEY, W. J. In: Specialists' Meeting on Helicopter Propulsion Systems, Williamsburg, Va., November 6-8, 1979, Technical Papers. (A81-17501 05-01) Washington, D.C., American Helicopter Society, 1980. 11 p.

ABS: The coupled interaction between two or more helicopter subsystems has often been the source of vibration problems. This paper gives a review of Sikorsky experience with such problems over the past twenty years of developing gas turbine powered helicopters. This review is part of an overall Government effort to accumulate data which will eventually lead to solutions of generic problems of this type. The problems presented include forced vibration problems, self-excited vibrations, and a transient response problem. Eighteen problems are sumarized and grouped according to type. Five of the problems are described in some detail. Recent trends in problems are discussed. Recommendations are made for future analytical/testing efforts to achieve an improved understanding of interfacing dynamic problems and potential solutions. 80/00/00 81A17516

UTTL: A performance monitoring system for helicopters

AUTH: A/WALKER, G. F. Association Aeronautique et Astronautique de France, European Rotorcraft Forum, 8th, Aix-en-Provence, France, Aug. 31-Sept. 3, 1982, Paper. 17 p.

ABS: The present investigation is concerned with the development of a performance monitoring system for helicopters. The technique adopted for the development of the performance model represents an extension of the use of energy concepts for helicopter flight path determination. The considered approach is based on the relationship between the power supplied to the rotor system and power required to overcome losses and to change the flight profile. The detection of helicopter performance degradation in icing encounters is identified as a requirement in order to enhance safety of flight. Attention is given to a method for detecting such degradation, using the principles of energy conservation and rotor dynamics. 82/08/00
84A19614

UTTL: A performance monitoring system for helicopters

AUTH: A/WALKER, G. F. IN: NAECON 1983; Proceedings of the National Aerospace and Electronics Conference, Dayton, OH, May 17-19, 1983. Volume 1 (A84-16526 05-01). New York, Institute of Electrical and Electronics Engineers, 1983, p. 260-265. Research supported by the Ministry of Defence /Procurement Executive/.

ABS: A system capable of providing helicopter pilots with a continuously updated display of performance capabilities both under existing atmospheric conditions, and for conditions at a remote site has been developed. In order to demonstrate this capability, the equipment has been adapted to monitor aircraft performance in such a way that degradation through flight in adverse enviroments, particularly icing, is detected and displayed. This paper describes the system and also discusses the type of performance models and monitoring methods upon which the system is based. Results of both bench and flight evaluations are presented, and future possibilities in this field of instrumentation are discussed. 83/00/00
84A16553

UTTL: Development of a helicopter rotor/propulsion system dynamics analysis

AUTH: A/WARMBRODT, W.; B/HULL, R. CORP: National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif.; Systems Control Technology, Inc., Palo Alto, Calif. AIAA, SAE, and ASME, Joint Propulsion Conference, 18th, Cleveland, OH, June 21-23, 1982, AIAA 16 p.

ABS: A time-domain analysis of coupled engine/drive train/rotor dynamics of a twin-engine, single main rotor helicopter model has been performed. The analysis incorporates an existing helicopter model with nonlinear simulations of a helicopter turboshaft engine and its fuel controller. System dynamic behavior is studied using the resulting simulation which included representations for the two engines and their fuel controllers, drive system, main rotor, tail rotor, and aircraft rigid body motions. Time histories of engine and rotor RPM response to pilot control inputs are studied for a baseline rotor and propulsion system model. Sensitivity of rotor RPM droop to fuel controller gain changes and collective input feed-forward gain changes are studied. Torque-load-sharing between the two engines is investigated by making changes in the fuel controller feedback paths. A linear engine model is derived from the nonlinear engine simulation and used in the coupled system analysis. This four-state linear engine model is then reduced to a three-state model. The effect of this simplification on coupled system behavior is shown.

RPT#: AIAA PAPER 82-1078 82/06/00 82A34997

UTTL: A state of the art assessment of turboprop transmission technology and projected needs for the next generation

AUTH: A/WILLIS, R. J., JR. CORP: General Electric Co., Lynn, Mass. CSS: (Gear Systems.) In AGARD Gears and Power Transmission Systems for Helicopters and Turboprops 11 p (SEE N85-23765 14-05)

ABS: The major share of power transmission research and development during the past twenty years has been expended on the improvement of helicopter main rotor drives. Furtunately, most of the advanced technology features resulting from these efforts are directly applicable to turboprop transmission. The technology base is made up of a number of interacting disciplines whose application is tempered by economics as well as the engineering state of the art at any given time. Modern computers are playing an ever increasing role in the design process and promise to be the means of removing gear design from its empirical background. The utilization of state of the art technology as the framework for turboprop transmission design is discussed. 85/01/00 85N23770

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14. Abstract

This Lecture Series will address numerous engine airframe integration aspects relating to rotor wing aircraft. Various areas of engine airframe integration, including the impact of advanced engine design variables to specific propulsion subsystems, i.e. engine air inlets, ~~will be~~ discussed. Other propulsion system integration aspects such as engine/rotor/drive system torsional stability and transient response ~~will be~~ addressed. Additional areas considering the specification and qualification requirements of propulsion items will also be discussed.

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