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CONTRIBUTION OF ENGINE IMPROVEMENT ON NEXT FUTURE ROTORCRAFT

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ABSTRACT

Applying foreseen advanced technologies to rotorcraft engines under development, performances might be improved in such a way that benefits should become increasingly important to the whole helicopter configurations, airframe and direct operating costs.In the present paper, different compressor and turbine configurations, heat recovery and material projections, are considered to establish how specific fuel consumption might be decreased.
Through an analytical study of an improved engine performance, it has been possible to reach conclusions about engine and airframe advances, in order to have a better helicopter generation in next future, regarding payload and mission range.
Improvements in material technology allow higher stresses, fewer stages and smaller engines, at a given power level. Specific fuel consumption influences helicopter operating costs and engine costs. Using technology projections for the engine, the process of optimizing the engine/rotorcraft system involves more parameters.
All comparisons are made referring to: baseline rotorcraft/engine system; baseline missions for rotorcraft; technology projections regarding materials, engine cycles,and manufactoring and operating costs.

1. INTRODUCTION

A next generation medium power helicopter engine will have to show significant advances, when compared with today engine high standard of technology. Towards this end, reduced fuel consumption, weight, size and cost,will have an increasing impact on advanced helicopter system configurations and associated operating costs. Availability of improved engine cycles and component technologies, mission requirements, will be primary contributors for engine selection. Design activities need to start well in advance of the actual engine program to define the environment for component development, with the target to have technology and design metodology available at the beginning of engine development. For both the engine and airframe manufacturers the challange of defining the benefits of engine improvements will become increasingly important to increase helicopter marketability. The engine/aircraft optimization process for the year 2000 potential helicopter configuration, to satisfy improved mission requirements and direct operating cost goals, has to take into account: technology advances (aerodynamics, material systems, cycle approaches), engine improvements (fuel consumption, cost,weight, size), to have improved engine options, reduced system weight because of composite materials, and enhanced control systems).

The year 2000 rotorcraft is incorporating advancements,in respect to the actual rotorcraft, in material technology, specifically the use of composites in the airframe. Comparisons were made at the same operating conditions, regarding standard altitude and engine power, for two reference missions. Manufacturing cost, fuel cost and maintenance cost, were considered for evaluating the direct operating costs of the year 2000 rotorcraft.

2. ENGINE IMPROVEMETS IN NEXT FUTURE

Reducing specific fuel consumption (SFC) cannot be the sole justification for new development.In spite of increasing fuel cost, it remains one important aspect of engine economics, in which further reduced weight and volume, higher reliability together with better maintainability appear equally as high. It is this requirement for better system performance and overall economics, resulting in reduced operating cost, that the industry is reacting to, when proposing a next generation of engines.The basis for success still remains the application of advanced components in terms of aerodynamic and mechanical design.
We assume that new technologies will be available through proper effort and funding support and ready for full-scale engine development by the year 2000: i.e. in the areas of materials, aerodynamics/thermodynamics, heat recovery,and mechanical improvements.
Material improvements are expected for both compressor parts (cold) and for the turbine (hot). Regarding the cold section, cost and weight saving in respect to present metallic parts may be respectively:30 and 40% for vanes,blades and rotors, using high-temperature powder metallurgy aluminum alloy; no cost saving and 30% for gearboxes, using polymeric composites; three times the present cost and 30 percent for shafts, using titanium metal matrix composite. Regarding the hot section, cost and weight saving in respect to present metallic parts may be respectively:0 and 15% for turbine blades/vanes, using nickel alumide (Ni_3Al); 0 and 60% for turbine vanes, blades, rotors, combustors, transition liners and recuperators, using ceramics.

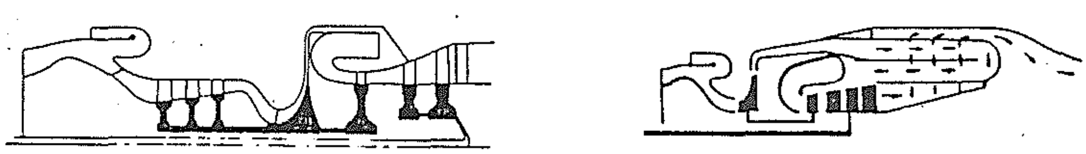
The projections forecast improvements in technology derived from existing work, from advances in materials processing and manufacturing methods, as well as improvements from advances in design techniques and computer aids. Material improvements for hot-end parts allow increased engine performance by permitting enhanced cycle temperature and component stress capabilities, which will increase the need for better cooling of liner materials, for example with a plasma-sprayed ceramic on a porous metal substrate. Material research is primarely directed at life-cycle cost requirement with emphasis on metallic coatings for oxidation and corrosion protection and ceramic coatings for thermal protection and reduced cooling requirements.

Ceramics have the potential for use in combustors and liners, permitting increased temperature capability without cooled turbine blades; the maximum material inlet temperature limit is projected to be 1800 K and more.

Improvements in component aerodynamic performance, including compressors, combustors, turbines and regenerators, are also foreseen, in terms of increased component efficiencies, reduced losses, and higher aerodynamic loading capabilities. Almost 4% increase in polytropic efficiency is expected by the year 2000 for an axial-centrifugal compressor, fig. 1 a), in respect to the present baseline, as result of improved clearance control, reduced vane inlet losses, and reduced impeller shock and secondary losses.

Advanced simple-cycle engines, with compression ratio ranging from 20 to 26:1 and turbine rotor inlet temperature from 1480 to 1800 K, in multistage axial turbines and two stage centrifugal compressors configurations, will be considered.

Heat recovery devices are used to recapture waste energy from the gas turbine exhaust, fig. 1 b), to preheat the combustor inlet air. Material improvements are expected in metallics and non-metallics. Recuperator technology projections for year 2000 consider, as metals, nitride-dispersion strenghtened 300 stainless steel (fin density: 15 fins/cm) for maximum temperature between 1250 and 1365 K and, as ce



SIMPLE CYCLE
a)

RECUPERATED CYCLE
b)

Fig. 1

ramics, plate-fin units (fin density:14 fins/cm) for maximum temperature between 1690 and 1810 K. Reduction of overall recuperator weight and its low-cost manufacturing process will be the potential success of ceramic plate-fin counterflow.

The comparison between conventional and regenerative helicopter engine shows that the installation properties of the two engines are roughly the same. For a typical attack helicopter mission, the composite of power system weight (including the IR suppressor) and fuel/mission weight will be noticeably lower in the case of the regenerative engine. This is mainly a consequence of the latter's better fuel consumption. The variable power turbine, being an indispensable part of a regenerative engine, at the same time leads to an extremely favourable transient behaviour with a moderate influence of severe cyclic loading on hot-part lifetime. The life-cycle costs of a future fleet of attack helicopters equipped with regenerative engines can be lower than with conventional engines. This holds true in spite of higher development and production costs of regenerative engines, if the present trend of growing fuel costs continues.

In order to adapt the simple cycle engine to operation with heat exchanger design,modification are required: new compressor outlet casing for ducting air from the compressor to the heat exchanger; new combustion chamber casing with connections for ducting air from the heat exchanger to the combustion chamber; new power turbine casing required by the introduction of variable vanes for the power turbine; new gas generator turbine bearing housing.

Ceramics are used in the combustor and turbine blades and vanes. Convenient pressure ratios are in the field of 8:1 to 12:1. For the compressor driving turbine, advanced metallics or ceramics may be used; and turbine rotor inlet temperature from 1480 to 1800 K may be adopted. For the ceramic turbine configuration, specific fuel consumption improvements of over 30 percent are found to be possible.However, although high recuperator effectiveness and low pressure drop are highly desirable in terms of engine performance,a penalty is payed in terms of size, weight, and cost.

Regarding mechanical improvements, inputs to the cycle study in terms of turbine hub speed, temperature constraints, and cooling flow requirements, have to be considered. Face contact seals are used to reduce leakage rate. A high speed spiral-groove has been developed. In the area of shaft dynamics, there are programs to develop improved balancing techniques for rotors. In addition, a new computer code ca pable of predicting non-linear rotor dynamics, has been developed. This code allows investigation of transient rotor motion during adverse operating conditions, such as a blade loss with rub. Damper concepts also are being analyzed, primarely to explore rotor systems that will be more tolerant to a large imbalance, as in cases equivalent to the loss of a blade or foreign object damage. Research on bearings will continue to focus upon optimizing design through improvement in materials and lubrication for higher-load capacity and longer life.

Improving the overall efficiency of the engine, we get advantages in helicopter payload mass for the same range, or more range for the same initial aircraft mass. Here we are showing how remarkable this gain can be.
The overall efficiency at a given forward flight speed is the ratio between the product of the directional thrust component T and the speed and the product of the heating value H per unit fuel mass and the fuel consumption F per unit time including the thermal or cycle efficiency (power output/heat content of fuel = P/HF) and the propulsion efficiency TV/P).
The range R equation, assuming constant values of η_{tot} (overall efficiency) and lift/drag, L/D, ratio during the flight, is

$$R = \int V\,dt = -\int_{M_0}^{M_1} \frac{V}{F}\,dM = \int_{M_0}^{M_1} \frac{V}{dM/dt}\,dM = \eta_{tot}\frac{H}{T}\int_{M_0}^{M_1} dM =$$

$$= \eta_{tot}\frac{H}{g}\frac{L}{D}\int_{M_0}^{M} \frac{dM}{M} = -\eta_{tot}\frac{H}{g}\frac{L}{D}\ln\left|1 - \frac{M_F}{M_0}\right|$$

where M_0 , M_1 , $M_F = M_0 - M_1$ are, respectively, initial aircraft mass, final aircraft mass and fuel mass.
The aircraft mass ratio's equation (M_b = basic operational mass, M_p = payload mass) is

$$\frac{M_b}{M_0} + \frac{M_p}{M_0} + \frac{M_F}{M_0} = 1$$

Combaining the two equations, we find for the payload mass ratio of the rotorcraft

$$\frac{M_p}{M_0} = e^{\frac{R}{\eta_{tot}\frac{H}{g}\frac{L}{D}}} - \frac{M_b}{M_0}$$

Now , we consider an example for a small helicopter of the class AGUSTA BELL AB206B1, with fuel at heating value H = $43 \cdot 10^3$ kJ/kg.
Its specifics are: Allison 250-C20 engine; fuel density, 0.78 kg/liter; M_b = 800 kg; range in stationary flight,555

kg; M_0 = 1436 kg; L/D = 3.85; η_{tot} = 0.19. Being H/g = km 4395, we obtain from the last equation M_p = 415 kg, i.e. , the relative mass ratio's

$$\frac{M_P}{M_0} = 0.289 \quad \text{and} \quad \frac{M_F}{M_0} = 0.154 \; ; \; M_F = 221 \text{ kg}$$

Now, considering an improvement of the overall efficiency to η_{tot} = 0.25, due to engine thermal efficiency (because propulsion efficiency is depending on the main rotor efficiency), we obtain from the previous range R equation, for the same values of R and L/D, a fuel mass ratio

$$\frac{M_F}{M_0} = 0.123$$

Without any further change of the rotorcraft design, considering the same values of the basic operational mass and the payload mass M_b = 800 kg and M_p = 415, we obtain from the aircraft mass ratio's equation M_0 = 1385 kg and M_F= kg 0.123·1385 = 170. The fuel mass is reduced by 51 kg or 23%. Due to the reduced rotorcraft mass, the improvement of specific fuel consumption is 22%.

The basis for success of the year 2000 rotorcraft engine remains the application of advanced components in terms of aerodynamic and mechanical design.
Recognizing the small size of future power plants, particular attention has to be given to the selection of an optimum cycle, fig. 2. Aside from the components' technology

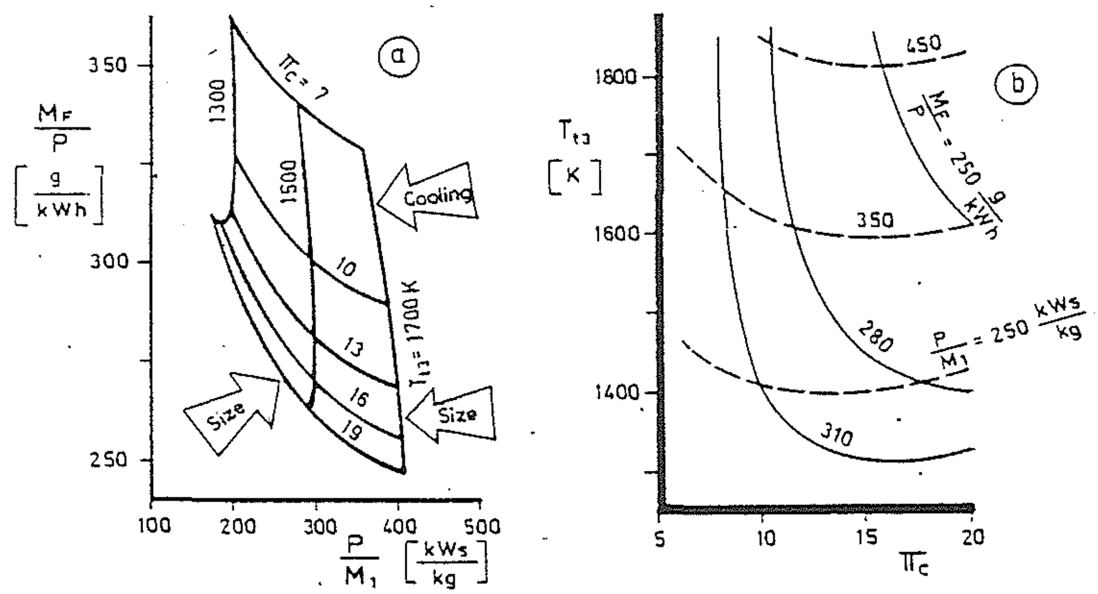


Fig. 2 Thermodynamic cycle trend and limitations

content, limitations have to be accepted resulting from turbine cooling and size effects. Plotted are turbine-stator outlet temperature versus pressure ratio, with specific fuel consumption and specific power used as parameters. This to demonstrate that two important effects need to be taken into account (π_c = compression ratio):

- too high outlet temperature tend to increase cooling air requirements, leading to decreasing returns in cycle efficiency;
- coupled with such too high temperatures T_{t3}, there is an increase in specific power. The resulting reduction in turbomachines size augmented by higher pressure ratios, requires careful trading against the now gaining negative influence of secondary losses on component efficiency.

3. CONTRIBUTION OF ENGINE IMPROVEMENTS ON NEXT FUTURE HELICOPTER

During the development of helicopter over the years propulsion has been one of the main driving forces. Technology advancements will permit in the year 2000 weights reductions in airframe and subsystems of the order of 20%, aerodynamic improvements in main rotor and reduced drag in airframe. Defining optimum future helicopter engines for year 2000, high payload mission, high altitudes and speeds will be possible. In engine cycle study, we have considered: compressor efficiency (set by configuration, pressure ratio and flow); turbine efficiency (set by configuration, loading and flow); gas generator spool speed (based on turbine mechanical limits and optimum compressor speed); output spool speed (based upon turbine mechanical limits and maximum speed).
To show influence of engine variables, a study has been conducted starting from present time reference engine, missions, helicopter year 2000 technology, economic model, year 2000 engine technology projections and system analysis. To select and evaluate engine cycle and configuration, simple cycle and heat-recovery cycle have been considered.
As present time reference engine, it has been chosen the configuration of figure 1 a), with reverse-flow annular combustor and two-stage axial power turbine, 5-stage axial plus centrifugal gas generator compressor, pressure ratio 17:1, two-stage axial gas genzerator turbine, turbine rotor inlet temperature 1477 K, power 2000 HP, and 100% of specific fuel consumption, weight, diameter, length and cost.
The reference engine has had to be chosen to make sense a

compared cost analysis with the operating conditions of the 2000 year engine.
Typical operating missions have been defined for future helicopter, the one 250 km long and the other 500 km, both at best cruise sea level range in 5 steps each separated by 20 minutes hovering, the 1° and the 2° to be carried out, respectively, with a rotorcraft gross weight,overall length and rotor diameter, 4180 kg, 52.7 m, 12.7 m, and 4665 kg, 54.2 m, 13.1 m, at critical sizing conditions (1200 m altitude and 308 K ambient temperature).
Direct operating cost model has taken into account various types of input: potential number of aircraft, hour service per year/aircraft, years service life, takeoff gross weight and payload, fixed costs (loan interest rate, imputed interest rate, depreciation and insurance schedules, tax rate, crew wages, hanger rent) and variable costs (engine/aircraft, fuel price, airframe and engine maintenance, crew expenses).
To chose the proper engine on the basis of direct operating cost, it is referred to the percent change of this per 1 percent change in engine parameters (specific fuel consumption, weight, diameter, length and cost).
For both missions, specific fuel consumption percent change has more influence on direct operating cost.

The engine cycle study has been conducted on the basis of technology projections.
As new materials for the engine cold-end are proposed: powdered high temperature aluminum alloy, cast titanium, polymeric composites and metal matrix composites, with cost and weight gains as at page 3.
New materials for engine hot-end are metallics (super single crystal, Ni_3Al) and non-metallics (ceramics).
Compressor performance projections are based on configuration and flow. Best performances are obtained with axial+ centrifugal compressors, as in figure 1 a),in respect to two and one centrifugal compressors for a same compression ratio,with a percent change in polytropic efficiency to 4% for high exit flow. Reasonable stage loading, percent tip clearance and exit flow Mach number, are assumed.
High percent increase in efficiency is obtained for the gas generator turbine, assuming metallic type geometry, unshrouded uncooled blades and two axial stages, as well as for the power turbine with shrouded uncooled blades.

A ceramic plate-fin counterflow recuperator has been chosen.

The simple-cycle configuration selected for the 2000 year

has axial-centrifugal compressor with compression ratio $\pi_c = 24:1$; single-stage axial, ceramic, uncooled, unshrouded, gas generator turbine with turbine rotor inlet temperature 1700 K; multistage axial, uncooled (ceramic as necessary) power turbine. Conventional advanced metallic cycles can reduce specific fuel consumption over 12%, at t. r. i. temperature 1477 K and at a pressure ratio 20 to 22:1.
The use of ceramics for the turbine blades and vanes gratly reduces the need for costly cooling air. The elimination of turbine blade cooling flow allows higher temperature/ pressure cycles to achieve better specific fuel consumption and specific power. Conventional cycles with ceramics improve specific fuel consumption by 20 percent. Ceramic engines have 10 to 15 percent lower weight, and large cost and size advantages.
A comparison of direct operating cost values between the advanced metallic and ceramic engines shows the expected advantage of ceramics, reduction over 8 percent in respect to 5 percent of the best metallic engine. Due to cooling penalties, the metallic engines optimize at relatively modest cycle conditions. However, compressor configuration (2-stage centrifugal and axial centrifugal) has small impact on direct operating cost.

The recuperated configuration selected for the 2000 year , fig. 1 b), has:single stage centrifugal compressor at compression ratio 10:1; single stage axial, ceramic, uncooled and unshrouded blades for 1700 K of turbine rotor inlet temperature; multistage axial, variable geometry ceramic uncooled blades power turbine; ceramic plate-fin recuperator with 0.85 efficiency and 8% of pressure drop.
Year-2000 recuperated engines can improve specific fuel consumption over 30% at turbine rotor inlet temperature high as 1800 K. Increasing recuperator pressure drop increases specific fuel consumption. An 8 to 10 percent specific fuel consumption and a substantial increase in specific power at recuperator effectiveness 0.8 is decreased with ceramic instead advanced metallic engines. Recuperator weight,size and cost, increase with effectiveness. Engines with ceramics have slight weight advantage, have lower costs and are shorter. Both in metallic and ceramic engines, cycle optimizes at pressure ratio 10:1 and longer mission has minimal influence on cycle selection. Recuperated engine has superior part-power performance.

In conclusion, year-2000 engines in recuperated or simple cycle (in brackets) greatly reduce:

- 48.0% (21.1%), fuel burn;
- 2.9% (6.3%), take-off gross weight;
- 1.4% (3.1%), rotor diameter;
- 1.3% (8.8%), wetted area.

In respect to present situation, advancing technology results in smaller, lighter and cheaper engines:
- percent reduction, 40% in weight; 25% in diameter;27% in length; 28% in cost.

From the previously developed computations,it is emerging a quite high reduction of specific fuel consumption. New materials can permit very high service temperatures even lowering engine component dynamic stresses.
The year 2000 gas turbine combustor, faced with rapidly changing environment, has to adjust to higher cycle pressure ratios and turbine rotor inlet temperature (TRIT) levels,and increase of heat release rate. Higher air temperature leaving the compressor has made cooling of the liner more difficult. In addition, the combustor designer faces the problem of exhaust emissions that are coupled with inherent demand to operate with wider range of new types of fuel.The necessary solutions are particularly demanding with the small combustor typical of rotorcraft powerplants.
New combustor design and development, as in fig. 3, are enabling these challenges to be met succesfully.Great effort

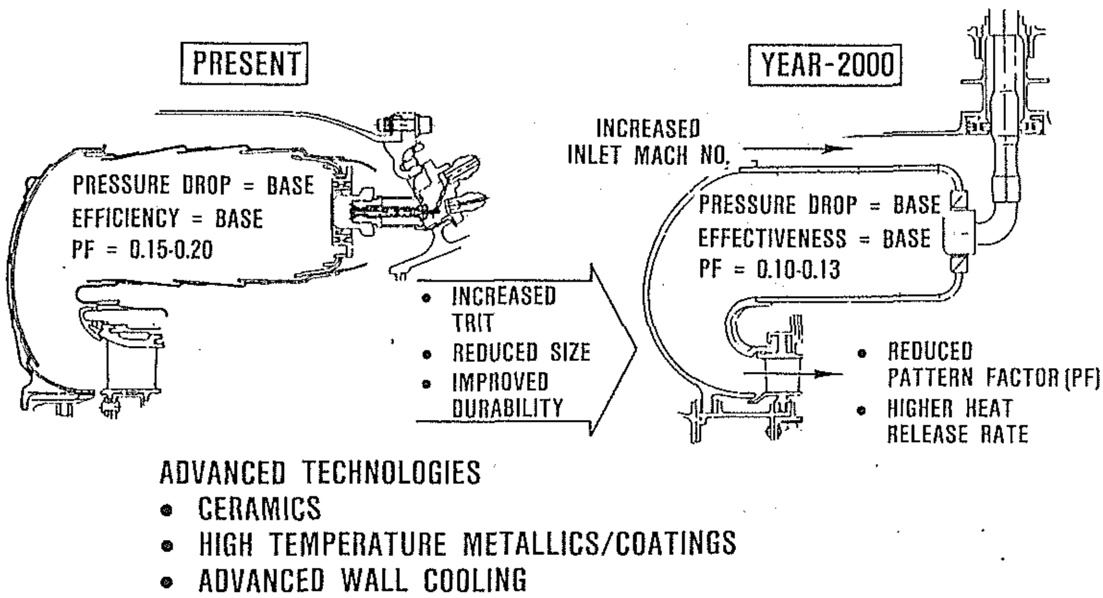


Fig. 3 Combustor for increasing heat release rate

is being devoted to the development of accurate analytical models of combustion systems. Overall design approaches are being developed and used. Interactive design methods have been started. Fuel-spray modeling has yielded performance benefits and has been successfully coupled with experimental techniques to verify its accuracy. Liner-wall analysis techniques are particularly useful in the development of long-life combustors because of the high-temperature cooling air and the higher combustor exit temperature od advanced engines. Characteristic time-modelling procedures have yielded benefits in reducing exhaust emissions.

4. CONCLUSIONS

Rotorcraft engines of year 2000 will exibit significantly improved performance if current research and development programs are pursued to their full potential.
Single cycle turboshaft engines are projected to have cycle pressure ratio in the range of 22-26:1 and turbine inlet temperatures of about 1800 K.
Compressors used in future engines will achieve this cycle pressure ratio by using fewer stages. Configurations consisting of a combination of axial and centrifugal stages or dual centrifugal stages are viable approaches. The centrifugal compressors will have leanback impellers to improve efficiency and range. The turbines will be aerodynamically highly loaded with active clearance control to minimize tip clearance and secondary losses. Advanced materials such as single crystal blades will be used to extend life and reduce cooling requirements. Environmental controls will become more severe requiring that more emphasis be placed on developing combustors having low emissions. And, finally, because of the escalation of fuel prices, the regenerative turboshaft engine will receive more attention for rotorcraft applications with long duration mission requirements.
Using technology projections for the engine, direct operating cost savings for future engines can be quantified from 7 to 12%, depending on the engine cycle,mission,and fuel price.

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