Potential of reducing the environmental impact of aviation by using hydrogen Part II: Aero gas turbine design

ABSTRACT

The main objective of the paper is to evaluate the potential of reducing the environmental impact of civil subsonic aviation by using hydrogen fuel. The paper is divided into three parts of which this is Part II. In Part I the background, prospects and challenges of introducing an alternative fuel in aviation were outlined. In this paper, Part II, the aero engine design when using hydrogen is covered. The subjects of optimum cruising altitude and airport implications of introducing liquid hydrogen-fuelled aircraft are raised in Part III.

The study shows that burning hydrogen in an aero gas turbine seems to be feasible from a technical point of view. If the priority is to lower the mission fuel consumption, the results indicate that an engine employing increased combustor outlet temperature, overall pressure ratio and by-pass ratio, seems to be the most attractive choice. The mission NO_x emissions, on the other hand, seem to be reduced by using engines with a weak core and lowered by-pass ratio.

NOMENCLATURE

Abbreviations and chemical formulae

APU	auxiliary power unit
AMA	Aircraft Mission Analysis (modelling tool)
CEA	chemical equilibrium and applications (modelling tool)
CMR	cryoplane, medium-range
CO	carbon monoxide
EQHHPP	Euro-Québec Hydro-Hydrogen Pilot Project
HE	heat exchanger
H_2O	water vapour
HP	high-pressure
HPC	high-pressure compressor
IAE	International Aero-Engines AG
ISA	international standard atmosphere
LP	low-pressure
LPP	lean pre-mixed pre-vaporised (combustor)
LPT	low-pressure turbine
NASA	National Aeronautics and Space Administration
NGV	nozzle guide vane
N_2O	nitrous oxide
NO	nitric oxide
NO_x	nitrogen oxides ($NO + NO_2$)
NO_2	nitrogen dioxide
ODAC	oil depletion analysis centre
Piano	project interactive analysis and optimisation (modelling tool)
UHC	unburned hydrocarbons

Notations

BPR	by-pass ratio	–
C_p	specific heat at constant pressure	J/kg K
COT	combustor outlet temperature	K

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EI	emission index of pollutant emission	g/kg
ESFC	energy specific fuel consumption	J/N s
F	thrust	N
LSF	linear scale factor	–
MTOM	maximum take-off mass	kg
OFPR	outer fan pressure ratio	–
OME	operating mass empty	kg
OPR	overall pressure ratio	–
P	pressure	Pa
SFC	specific fuel consumption	g/kN s
SPT	specific thrust	N s/kg or m/s
T	temperature	K
TET	turbine entry temperature	K
Δ	change	%
ϕ	equivalence ratio	–

Subscripts and Superscripts

i	generic number
ic	initial cruise
N	net
st	stoichiometric
*	nominal measured quantities

1.0 INTRODUCTION

Since commercial aviation started in 1920 it has undergone spectacular growth, and today it has become a fundamental part of business and commerce. Mainly for business purposes, engine and aircraft manufacturers, and airlines compile forecasts for future traffic growth. Over the next 20 years global passenger aviation traffic is expected to grow, averaging around 5·3% annually (Rogers *et al*[1]), implying an almost threefold traffic volume at the end of this period. It is expected that capacity growth, i.e. number of seats offered, will expand at a slower rate, provided that passenger load factors maintain their improving trend over the long term.

In addition to the anticipated air traffic growth, the dwindling fossil oil resources raise concerns. In 2000 the United States Geological Survey estimated that the ultimate recoverable world oil resources would supply an expansion of 2% per annum through 2025, reaching around 125 million barrels per day (Schnieder and McKay[2]). Thereafter a plateau could occur for around two decades before depletion would lead to a permanent decline in production. Other specialists believe that the plateau will be reached earlier. The Oil Depletion Analysis Centre (ODAC) believes that a plateau of conventional oil production will be reached before 2010, with significant price rises thereafter.

Moreover, the greenhouse effect generated by emissions produced by human activity, particularly carbon dioxide, has become an increased concern. The majority of scientists today are in agreement that discharging greenhouse gases (or their precursors) and particulates into the atmosphere has an impact on the global climate (see Part I). It is, therefore, essential to find an alternative to kerosene for civil aviation. In the context of this paper, hydrogen, in the longer term produced from renewable energy sources, is addressed as a suitable future fuel in order to cope with these concerns. Because the introduction of hydrogen would influence the complex interactions among a number of different fields affected by this change and because hydrogen is not considered a practical alternative at present but a number of years ahead, this paper covers a number of selected fields. This broad study results in making it possible to define those fields of which more research and development are necessary in order to move towards liquid hydrogen-fuelled aircraft in civil aviation. Furthermore, a broad approach will provide knowledge on the technical feasibility of this fuel change, taking practical aspects into account.

The objective of the paper is to evaluate the potential of reducing the environmental impact of civil subsonic aviation by changing the source of energy from kerosene to the energy carrier hydrogen. In addition, the practical and technical feasibility of introducing hydrogen as fuel is investigated. The paper is divided into three parts of which this paper constitutes Part II. In Part I the background, prospects and challenges of introducing an alternative fuel into aviation were outlined. In this paper, Part II, the fundamental effects on aero engine pollutant emissions, performance and design of changing to hydrogen fuel are covered (Section 2). In particular, the NO_x issue is raised by studying the engine and combustion chamber design in detail, including the potential to achieve low NO_x emissions, when using hydrogen and kerosene. Moreover, the subject of designing an aero gas turbine using hydrogen for low environmental impact is raised (Section 3). Here, the mission fuel consumption and mission NO_x emissions are calculated for a medium-range liquid hydrogen-fuelled aircraft. The paper ends with a summary and discussion of the conclusions (Section 4).

In the last part, Part III, the optimum cruising altitudes for minimum global warming for a liquid hydrogen-fuelled aircraft and a conventional aircraft are investigated. Furthermore, the feasibility of introducing cryoplanes on a regional level is explored. The infrastructure changes of introducing cryoplanes and possible hydrogen production methods at Stockholm/Arlanda airport are discussed.

2.0 EFFECTS OF USING HYDROGEN ON AERO GAS TURBINE POLLUTANT EMISSIONS, PERFORMANCE AND DESIGN

2.1 Introduction

As for the aero engine, the cycle and combustion chamber design need careful attention to secure a safe and reliable operation, as well as to exploit fully the favourable properties offered by using hydrogen for aero gas turbines. The engine could be designed either by making a minimum number of hardware changes or by employing unconventional cycles, which exploit the cryogenic properties of hydrogen, in order to improve the performance.

This section covers the main effects on aero gas turbines when changing to hydrogen fuel, with emphasis on environmental issues. The objective is to answer the question whether there are any significant gains, in terms of engine performance and pollutant emissions, as well as to assess the technical feasibility of changing to hydrogen fuel. The effects on the performance are outlined in Section 2.2, while the emissions are discussed in Section 2.3. For reasons discussed below, there might be the potential for lowering the NO_x emissions compared with conventional kerosene combustors. Using simulation tools, the potential of achieving low NO_x emissions is illustrated for a conventional as well as for a hydrogen-fuelled engine by calculating the flame temperature for relevant operating conditions (Section 2.3.4). In Section 2.4 design and operational matters of particular interest when using hydrogen are discussed.

This section provides an overview of the performance, emission and design issues covered in much more detail in Haglind and Singh[3] and Svensson and Singh[4]. In addition, some previous NO_x emission results of an APU engine provided with a micromix combustor are included.

2.2 Engine performance

When burning hydrogen in an aero gas turbine there are several issues that need to be regarded. In addition to re-designing the combustion chamber, the minimum change that needs to be adopted with the engine cycle is the implementation of facilities to evaporate the hydrogen

† Oil Depletion Analysis Centre (ODAC) is a charitable organisation in London that is dedicated to researching the date and impact of the peak and decline of world oil production due to resource constraints, and raising awareness of the serious consequences of oil depletion. The assertion is made by Colin Campbell who is a Trustee of ODAC (see http://www.hubbertpeak.com/campbell/).

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(which is stored in the tanks in a liquid state at 24K) prior to its entry into the combustion chamber. The fuel heating can be accomplished either by an external heat source or a heat exchanger (HE) located at a suitable engine location. Placing the HE outside the engine does not affect the engine performance; however, it might cause practical problems when the fuel system is integrated with the other aircraft systems. Available external heat sources are, for instance, cooling systems of hydraulic fluids, pumps, electric equipment and the cabin area (Boggia *et al*[5]).

Looking at the option of employing a HE, there are various possible engine locations which have been studied within the CRYOPLANE project (Corchero and Montañes[6]; Boggia and Jackson[7]; Boggia *et al*[5]; Boggia[8]). When deciding where to place the HE, possible benefits in performance need to be weighed against increased engine complexity and safety issues. Aiming at minimising the number of hardware changes and prioritising safety issues, a HE located at the LPT (low-pressure turbine) exit is employed here. Other possible engine configurations are listed in Section 2.2.3.

Practically, the heat exchanging might be accomplished by employing the struts, which are the mechanical structures in the exhaust which hold the rotors in place and are connecting the outer structure of the bearings. These are already-existing structures that do not add any pressure loss and do not cause any significant changes in engine complexity or engine weight. The idea is to pass the hydrogen through the hollow struts which are heated on the outside by the exhaust gases. There are, however, safety issues associated with this design, due to the potential detrimental consequences of a fuel leak. In order to employ this idea, the design would have to meet the airworthiness requirement. Simplified calculations outlined in Haglind and Singh[3] suggest that the heat transferring area available from the exhaust struts is sufficient to accomplish only a minor portion (about 10%) of the desired temperature rise of the hydrogen fuel. One option would, therefore, be to use an external heat source for the remaining temperature rise prior to the struts.

An alternative is to place a heat exchanger in the exhaust, featuring a simple coil tube placed over the inside face of the jet pipe casing (Corchero and Montañes[6]). Though this solution may incur additional pressure losses in the exhaust, it is believed that it would give a relatively aerodynamically clean jet pipe.

In all calculations presented in this paper, the energy necessary to raise the fuel temperature from the storage temperature is taken into account by decreasing the exhaust gas temperature accordingly. No reasonable figures on possible pressure losses or weight penalties due to the inclusion of a heat exchanger are available. Therefore, no other effects of a heat exchanger than an exhaust gas temperature drop are taken into account in subsequent calculations.

In Section 2.2.1 the engine used in the simulations is described. Effects on performance are illustrated in Section 2.2.2 by carrying out engine simulations. In addition, a number of suggested unconventional engine cycle configurations are mentioned (Section 2.2.3).

2.2.1 Choice of engine for simulations

In order to investigate the consequences on performance when changing to hydrogen, a particular engine, namely the V2527-A5 engine manufactured by IAE[2], is simulated. This is a two-shaft boosted turbofan engine suitable for short- and medium-range aircraft. The jets from the by-pass and core are unmixed. It features a single-stage fan, a four-stage LP (low-pressure) compressor and a ten-stage HP (high-pressure) compressor. The inlet guide vane and the first three stages are variable. The HP inlet guide vane and the first three vane stages are variable (not modelled in the calculations). The reason for choosing this engine is that it is a typical, reasonably modern, medium-thrust turbofan engine, and that there are sufficient public data on it to create a sufficiently detailed model of its performance. Furthermore, this engine is suitable for the main study aircraft within the CRYOPLANE project, namely the A320, on which this work is based.

With small modifications (lower HPC isentropic efficiency and inclusion of losses in the exhaust duct) the model is based on the one proposed within the CRYOPLANE project (Boggia and Jackson[7]; Boggia *et al*[5]; Boggia[8]). Also, as opposed to their model, a thrust coefficient, which is calculated depending on the expansion ratio of the nozzle and hence varies with the operating condition, is applied here. Using this methodology, a thrust coefficient equal to about 0·98 is applied. In total, therefore, a slightly decreased performance in terms of SFC (specific fuel consumption) and SPT (specific thrust) is obtained here compared with that presented in the CRYOPLANE project. The main design parameters at the design point are as follows:

- Design point: take-off, static sea-level, ISA+10 K
- BPR: 4·8
- OPR: 28·5 (1·5 x 2·0 x 9·5)
- Thrust: 117·9 kN

The component efficiencies and pressure ratios assumed are given in Table 1, while the assumed pressure losses taken account for are displayed in Table 2. Neither overboard bleeds nor power extraction is included. The outer fan pressure ratio chosen here is lower than the optimum, i.e. the value that minimises SFC and maximises specific thrust for a given engine core and BPR. A fan pressure ratio lower than the optimum is often applied to reduce design and handling problems, at the expense of a very small penalty in SFC (Pilidis[9]).

To enable higher turbine entry temperatures than the maximum allowable metal temperature, cooling of hot parts is required. This is accomplished by bleeding off a part of the compressed air which then passes through cooling passages inside the blades. Generally the NGV (nozzle guide vane) and HP turbine rotor need to be cooled. Also air needs to be taken off for sealing, in order to stop expanding gases from penetrating the disk system. In order to simulate these effects, the same numbers as applied by Boggia *et al*[5] are applied. It is assumed that 15% of the core air is bled off after the HP compressor. Depending on where expansion work takes place, this air is mixed with the hot gases at different points along the expansion. To simulate NGV cooling, 12·5% of the bleed is injected immediately before the HP turbine, and the remaining 2·5% is injected immediately before the LP turbine to be used for HP rotor cooling and seals.

2.2.2 Engine performance estimations

The effects on the engine performance by changing to hydrogen fuel are estimated for the V2527-A5 engine using TurboMatch (Cranfield University's own code for gas turbine performance estimation) (Svensson and Singh[4]). Three engines with the same BPR, OPR and inlet mass flow – one using kerosene and two using hydrogen – are

Table 1
Assumed component data for the V2527-A5 engine

Component	Pressure ratio	Isentropic efficiency
Outer fan	1·7	0·88
Inner fan	1·5	0·89
Booster	2·0	0·88
High pressure compressor	9·5	0·87
High pressure turbine	–	0·91
Low pressure turbine	–	0·91

Table 2
Losses taken account for to model the V2527-A5 engine

Component	Pressure loss (%)
Intake	0·2
By-pass duct	1·5
Combustor	3·0
Exhaust duct	1·0

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simulated for the design point. When changing to hydrogen, either the combustor outlet temperature or the net thrust could be retained.

According to Svensson and Singh[4], when the COT (combustor outlet temperature) is preserved, the net thrust increases by 3·2% for this particular case, resulting in a corresponding increased specific thrust (SPT). The energy consumption, reflected as energy specific fuel consumption (ESFC) (SFC times the lower heating value), increases slightly, but less than 1%. The net thrust is retained by lowering the COT by 33K. In terms of energy consumption, this is slightly beneficial, since ESFC decreases by 1·4%.

Following Boggia and Jackson[7], the performance improvements could be explained by the two fundamental changes when using hydrogen: reduced mass flow and changed composition of the gases expanding through the turbine(s). The reduced mass flow is a consequence of the considerably higher heating value of hydrogen, and it will have a detrimental effect on the performance. However, the loss in thrust owing to the reduced mass flow, is offset by an increased thrust due to an increased C_p (specific heat at constant pressure) value of the combustion gases. In total, the positive effect of increased C_p value outweighs the effect of reduced mass flow, and hence results in a slightly increased performance.

The effects on engine performance are quite small; but still there are some desired features that could be exploited. If the COT is kept the same, the TET (turbine entry temperature) is also about the same, thus requiring the same cooling technology. An increased specific thrust implies that the inlet mass flow, and thereby the physical sise of the engine, may be reduced slightly to achieve the same net thrust. As for the option of lowering the COT to preserve the net thrust, there are a few interesting consequences worth mentioning. The decrease in TET of more than 30K will require less advanced cooling technology as well as having a favourable effect on turbine blade life. The lowered ESFC will reduce the amount of energy that needs to be provided by the fuel for a certain thrust level. Moreover, designing for a lower maximum cycle temperature will help to suppress the NO_x emissions (see Section 2.3.1).

2.2.3 Unconventional engine cycles

As for the case of evaporating the hydrogen using a HE suited within the engine cycle, the cryogenic properties of hydrogen could be exploited to improve the performance through the usage of unconventional cycles. Some of these cycles are also proposed for kerosene-powered cycles; however, the main advantage when using hydrogen is the involvement of an additional fluid with favourable properties. Considering unconventional designs, there are basically four options, and various combinations of these that could be employed (Boggia and Jackson[7]; Boggia *et al*[5]; Boggia[8]; Baerst and Riple[10]; Payzer and Renninger[11]):

- Pre-heating the hydrogen fuel with exhaust gases
- Cooling the compressor air with hydrogen fuel
- Cooling turbine cooling air with hydrogen fuel
- Hydrogen topping cycle

The design principle, objectives, advantages and drawbacks of these concepts are outlined in Svensson and Singh[4].

According to Boggia *et al*[5] who studied and compared the four different cycles described above, the pre-cooled and high TET cycles appeared to be the most promising, offering a reduction in operational cost in the order of 3%. Furthermore, they point out that the positive aspect of these two cycles is that the innovations are technically feasible and do not involve any additional turbomachinery. The configuration with pre-heating of the fuel is the simplest solution and would require only minimal modifications to the base engine. Payzer and Renninger[11] conclude that there is a definite thermodynamic benefit in switching from hydrocarbon fuel to hydrogen fuel due to the differences in fuel properties. However, having evaluated eight different unconventional cycles, both in terms of uninstalled engine performance and aircraft mission performance, they state that the additional complexity associated with these cycles does not appear to be justified.

2.3 Pollutant emissions

Given that liquid hydrogen is produced from renewable energy sources, the emissions are reduced to water vapour (H_2O) (which increases by a factor of about 2·6) and small quantities of oxides of nitrogen (NO_x) when burning hydrogen. All emissions containing carbon and sulphur are eliminated. In order to achieve satisfactory emission characteristics, the challenge is to minimise the NO_x emissions while preserving or reducing the SFC. Performing research focused on reducing NO_x emissions of gas turbines is highly important, since the effects of these emissions upon the environment and on human health are considerable (see Part I).

This section is introduced by a brief discussion of the NO_x formation processes (2.3.1), followed by a section dealing with the consequences on the NO_x emissions of using hydrogen (2.3.2). In Section 2.3.3 combustion chamber design is discussed, and finally in Section 2.3.4 the potential of achieving low NO_x emissions is illustrated by outlining a summary of simulations for the V2527-A5 engine.

2.3.1 NO_x formation processes in combustion

Most of the nitrogen oxides formed during combustion are in the form of NO; however, subsequently the NO oxidises to NO_2. Usually these are lumped together, and the result is expressed in terms of oxides of nitrogen (NO_x). The formation processes of NO in combustion are complex and generally comprise different mechanisms: thermal NO, prompt NO, nitrous oxide (N_2O) mechanism and fuel NO (Lefebvre[12]). In burning hydrogen, thermal NO formation is the most contributing mechanism to the total NO formation. Thermal NO is produced from the nitrogen and oxygen present in the air in the high-temperature regions of the flame and in the postflame gases. The process is endothermic and it proceeds at a considerable rate only at temperatures above around 1,850K (Lefebvre[12]).

Prompt NO is a designation for NO that under certain conditions is formed very early in the flame region. For lean premixed combustion this formation mechanism can be a significant contributor to the NO emission produced (Correa[13]). Prompt NO may play a significant role for lean systems burning kerosene, whereas it does not contribute to the NO formation in combustion systems using hydrogen (since there are no carbon atoms present in the fuel). As for the nitrous oxide mechanism, Miller and Bowman[14] report that the measured N_2O levels in stack gases of gas-fired combustors are low. Furthermore, when burning hydrogen, for high engine load operating conditions where the main part of the NO_x is formed, the nitrous oxide mechanism is less important (Dahl and Suttrop[15]). This mechanism is therefore not believed to play a significant role for neither the kerosene nor the hydrogen combustion, and it is not considered here.

Fuel NO is formed when nitrogen which is organically-bonded to the fuel reacts with oxygen during the combustion. Since neither kerosene nor hydrogen contains any nitrogen, this mechanism is not relevant here.

2.3.2 Effects on NO_x emissions by using hydrogen

In order to address properly the issue of NO_x production of hydrogen combustion compared with kerosene combustion, several different combustion characteristics of these fuels need to be considered. The engine load in a gas turbine is controlled by varying the TET, which is determined by the overall fuel-air ratio in the combustion chamber. When a conventional kerosene-fuelled combustor is operating at full power, the primary zone operates at roughly stoichiometric fuel-air mixtures, where the flame temperature is highest and the chemical reactions are fastest (Ziemann *et al*[16]). At low power idle conditions, the fuel-air ratio is essentially leaner and the primary zone fuel-air ratio has to be maintained above the

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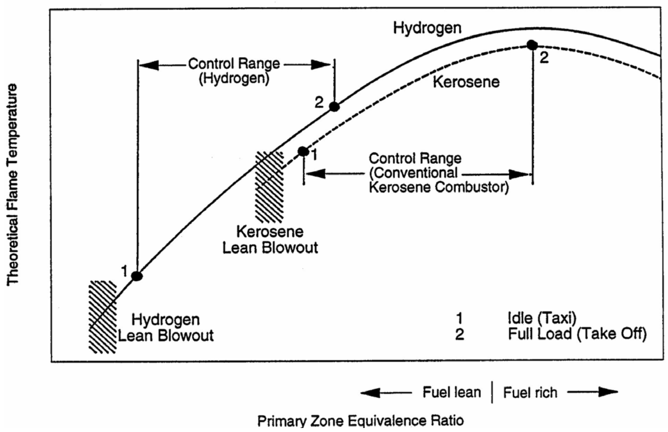

Figure 1. Temperature characteristics of the combustion chamber primary zone (Ziemann *et al*[14]).

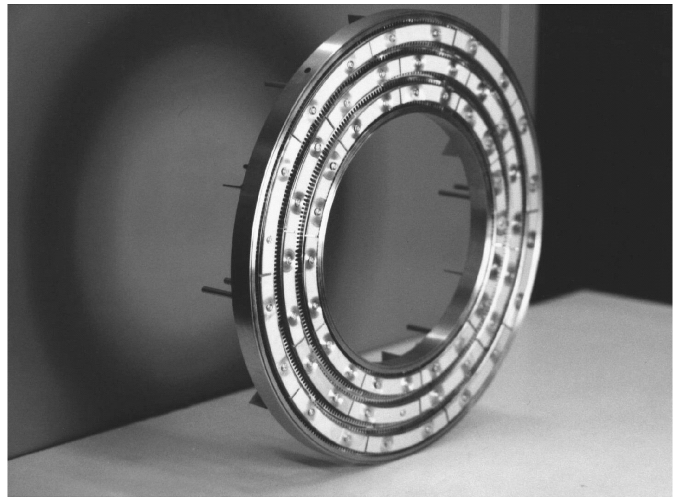

Figure 2. Exit surface of micromix hydrogen burner bulkhead (flame tubes are removed). The concept is based on regular diffusion (Dahl and Suttrop[18]).

flame-out limit with a sufficient margin. Hence, in order to minimise the NO_x emissions, it is desirable to modify the fuel-air ratio in the combustor primary zone in a way that lean combustion is achieved at all engine load conditions without suffering a flame-out (Ziemann *et al*[16]). In Fig. 1 the flame temperature of kerosene and hydrogen combustion versus primary zone equivalence ratio is shown.

The comparison illustrated in Fig. 1 applies only to a conventional kerosene-fuelled combustor featuring a primary, intermediate and dilution zone. If the kerosene-powered combustor would employ any unconventional technology such as, for instance, a staged or an LPP (lean pre-mixed pre-vaporised) combustor, the situation would change. Generally, the combustion zone of these combustors is operated at an equivalence ratio which is closer to the lean blow-out limit than the conventional, and hence it is not applicable to the situation sketched in the figure.

As illustrated in Fig. 1, the stoichiometric flame temperature of hydrogen-air flames is in fact higher than that of kerosene-air flames (about 100K higher), which would have a detrimental effect on the NO_x production. For a given equivalence ratio, the flame temperature is higher when burning hydrogen, which means that more NO_x would be generated. However, more importantly is that the hydrogen flame has a wider flammability range; particularly the lean limit is much lower than that encountered for kerosene flames (Ziemann *et al*[16]). Therefore the entire operating range may be shifted further into the lean region, with reduced NO_x emissions as a consequence.

Due to the rapid reaction kinetics of hydrogen-air flames, the burning velocity increases about eight times when changing to hydrogen (Ziemann *et al*[16]). The higher flame speed will result in a shorter combustor, hence reduced cooling requirements. Owing to this, more air will be available for the combustion, hence offering lower flame temperatures which will reduce the NO_x emissions. Another effect of the increased burning velocity is that the actual dwell time in the hot flame zone decreases, which is beneficial for hindering NO_x formation.

The fact that hydrogen is a gaseous fuel while kerosene is liquid, is also an argument in favour for hydrogen. The reason is that for liquid fuels there is always the potential of near-stoichiometric combustion temperatures in local regions adjacent to the fuel drops, and thereby high NO_x formation, even if the average equivalence ratio throughout the combustion zone may be appreciably less than stoichiometric (Lefebvre[12]). However, for increasing flame temperatures, the bulk flame temperature becomes closer to the stoichiometric value, and local conditions around the fuel drop have less influence on the overall combustion process and NO_x emissions. Hence, the difference in NO_x production decreases, and for equivalence ratios close to the stoichiometric value, the difference becomes negligibly small. Moreover, the minimisation of the NO_x emissions of hydrogen combustion is facilitated by the fact that no other secondary combustion products (e.g. CO, UHC, soot) need to be regarded in the combustor design process.

2.3.3 Combustor configurations burning hydrogen

In order to exploit the favourable characteristics of hydrogen combustion, the combustor needs to be re-designed compared with the conventional designs using regular diffusive combustion. When using conventional combustors, with their limited number of fuel injection points, the mixing of hydrogen and air tends to be incomplete. Moreover, the large-scale hydrogen diffusion flames form layers of stoichiometric mixtures of high local temperature and viscosity in which the NO formation rate remains high and acts as a barrier against the further progress of mixing (Ziemann *et al*[16]).

Generally when burning hydrogen the attempts to reduce the NO_x emissions are focused on lowering the flame temperature, eliminating hot spots from the reaction zone and reducing the duration and exposure in the flame region (Ziemann *et al*[16]).

In 1992 during the third phase of the Euro-québec hydro-hydrogen pilot project (EQHHPP), analytical modelling and experimental tests of low-NO_x combustors for aero engines were performed (Ziemann *et al*[16,17]). Various burner concepts for lean hydrogen combustion were evaluated in a generic can-type combustor.

The above cited studies and other preceding studies show that lean premixed combustion is undoubtedly superior to any combustion scheme without premixing in terms of temperature pattern uniformity and NO production. However, premixing implies the major drawback of premature burning and flashback danger, which may cause structural damage and compromise the operational reliability (Ziemann *et al*[17]). The risk of auto-ignition for premixed systems and the problems of large-scale hydrogen diffusion flames, have led to the lean non-premixing concept of micromix combustion, which is based on miniaturised diffusive combustion. Employing this principle, imperfections in fuel-air mixing cannot be avoided; however, the local standard deviations from the nominal equivalence ratio are kept to a minimum (Ziemann *et al*[16]).

The micromix combustor consists of a very large number (typically more than 1,000) diffusion flames uniformly distributed across the burner's main cross section, thereby minimising the geometric size of the combustion zone (Dahl and Suttrop[18,19]; Ziemann *et al*[16]). Theoretically, the lowest NO_x production would be achieved by an infinite number of miniaturised flames. The

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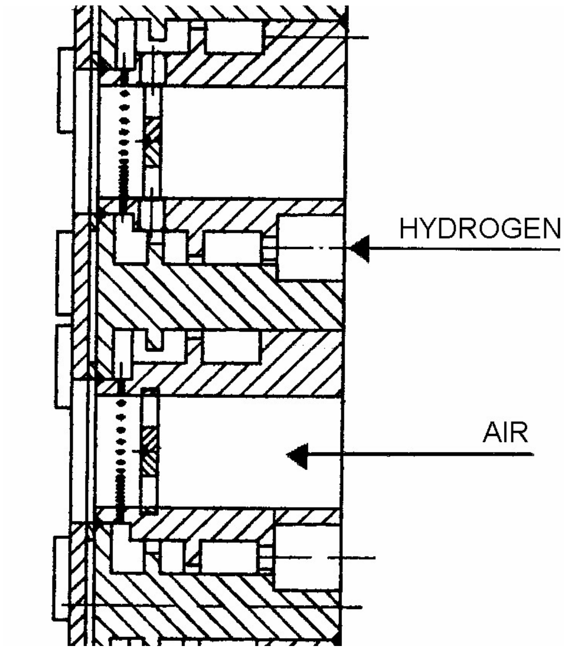

Figure 3. Air and hydrogen admission for diffusive combustion in the micromix hydrogen combustor (Dahl and Suttrop[18]).

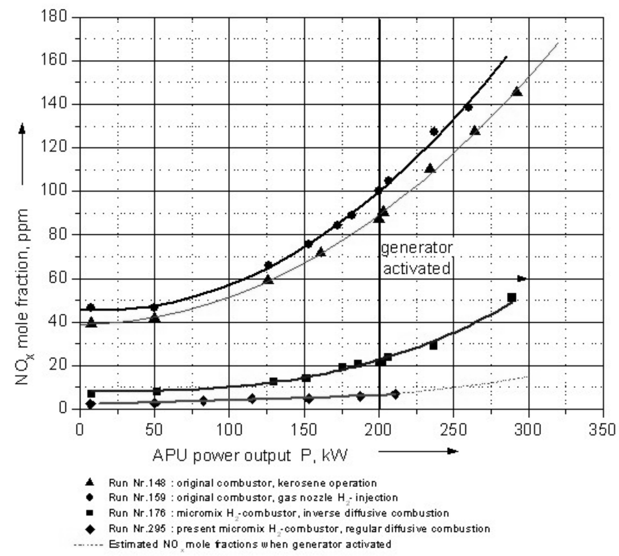

Figure 4. NO_x emissions of the Honeywell APU GTCP 36-300 fitted with different combustion systems (Dahl and Suttrop[18]).

minimum size is, however, restricted by the manufacturing cost and the combustion stability at engine idle conditions; the latter being deteriorated with reduced flame dimensions. In addition to minimising the scale of the combustion zone, the micromix concept aims at optimally utilising the available pressure loss (providing energy for the dissipative turbulent mixing process) in the combustion system to enhance the mixing process of hydrogen and air. Simultaneously minimising the scale of combustion and maximising the intensity of mixing would minimise the number and size of local stoichiometric flame regions, where the gas phase NO formation processes are most likely to occur. Due to the high flame speed and high reactivity of hydrogen compared with other fuels, the degree of miniaturisation of hydrogen-air diffusion flames very well exceeds the possibilities of other fuels (Ziemann *et al*[16]).

Having evaluated a large number of different micromix burner configurations, it was concluded that the burner based on regular diffusive combustion (called the concept of third generation) offered the lowest NO_x emissions, and the best thermal and mechanical stability, see Figs 2 and 3.

The Honeywell APU engine GTCP 36-300 was fitted with various combustion systems and the NO_x mole fractions were measured. The combustion systems were as follows: original combustor using kerosene, original combustor using gas nozzle hydrogen injection, micromix hydrogen combustor using inverse diffusive combustion and present micromix combustor using regular diffusive combustion (shown in Figs 2 and 3), see Fig. 4.

From Fig. 4 it may be concluded that significantly lower levels of NO_x may be achieved with the micromix combustors compared with conventional combustors using kerosene. Moreover, it can be observed that the lowest level of NO_x is attained with the present configuration of a micromix combustor employing regular diffusive combustion. There is, however, no use in simply changing the injection nozzles of the original combustor to hydrogen injection nozzles; this would even increase the NO_x emissions. It is worth mentioning that at the time of the test runs, the APU output was restricted to about 210kW because the electric generator was not available. The curve in Fig. 4 is however extrapolated for higher power loads. A cautious estimation of the NO_x emission level for main engine start or full power operation, corresponding to an APU power output of roughly 300kW, may yield about 20ppm (Dahl and Suttrop[18]). The figure shows that for main engine start conditions, the mole fractions of NO_x emissions are reduced by about 88% compared with the minimum change configuration (only hydrogen injection nozzles), and by roughly 85-86% when compared with the emissions of the kerosene-fuelled engine. As the output power is reduced the difference increases.

The next steps in the development of the micromix combustion system should be to prove long duration structural stability at extended operating times and check long time embrittlement (Dahl and Suttrop[18]). Moreover, high altitude re-light capability and high altitude flame stability at idle condition need to be proven.

2.3.4 Illustration of the NO_X reduction potential

In this section the NO_x reduction potential when using hydrogen and kerosene is illustrated by modelling the V2527-A5 engine, considered to be provided with different combustion systems. A detailed outline of the study is given in Haglind and Singh[3], and here is only given a brief summary of the results. The aim of the study is to quantify and compare the leanest possible attainable levels for the operating conditions take-off and idle (assumed to be 7% power setting) at static sea level, and the combustor primary zone flame temperatures associated with these, for kerosene and hydrogen combustion, respectively. Rather than attempting to design a combustion system, the qualitative differences in flame temperature associated with these different fuels are assessed.

Performance data are obtained using TurboMatch and GasTurb (Kurzke[20]), and the flame temperatures are calculated assuming chemical equilibrium. For the flame temperature computation, a well-established computer program developed at NASA Lewis Research Center is employed (McBride and Gordon[21]; Gordon and McBride[22]). As the dwell time in a gas turbine combustor is slightly too short for chemical equilibrium to be achieved, the flame temperature computed this way tends to be slightly overestimated. Nevertheless, these figures qualitatively indicate differences in flame temperature, and thereby the NO_x levels that may be expected from the different combustion systems. Calculating the flame temperature without assuming chemical equilibrium is beyond the scope of this paper.

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As for the combustor types, a hydrogen-fuelled micromix combustor is compared with two different combustors burning kerosene. For kerosene fuel the combustors are a conventional (featuring a primary, secondary and dilution zone) and a theoretical combustor for which the distribution of air to different zones at different operating conditions can be changed without restrictions (e.g. an extremely effective variable geometry combustor).

The results suggest that at take-off the flame temperature becomes about 1,000K lower for the combustor using hydrogen compared with the conventional kerosene-fuelled combustor. If the kerosene-fuelled engine would be provided with a variable geometry combustor, theoretically the difference might be reduced to about 150K. Comparing these combustors, the hydrogen-fuelled combustor would be operated with a remarkably larger margin to the weak extinction limit at idle than would the kerosene-fuelled combustor provided with a variable geometry combustor, and thus the hydrogen-fuelled combustor is likely to be the more viable of these two (see Section 2.4). The differences in flame temperatures will have a substantial effect on the amount of thermal NO produced by the combustion system. In addition, since the lean kerosene-powered combustor would probably also generate prompt NO, whereas the combustor burning hydrogen would not (see Section 2.3.1), the difference in NO_x formation between these two fuels is likely to be even larger than assessed when looking at the primary zone flame temperatures.

It should be pointed out, however, that since the rate of chemical kinetics and the combustor length of these fuels are different, it cannot be excluded that the difference in flame temperatures would change if the flow field and chemical kinetics were taken into account. Being aware of the simplicity in the approach of this preliminary study, the results of the calculations suggest that there is the potential to design a combustion system using hydrogen that produces less NO_x emissions than a system burning kerosene.

2.4 Design and handling issues

In order to adapt the combustion chamber for hydrogen, some changes are necessary and some others are desired in order to utilise the favourable properties of hydrogen in an optimal fashion. The minimum change that has to be done to adapt a conventional combustion chamber for hydrogen, is to replace the injection system, because when the hydrogen is injected into the combustor it is in the gaseous state, while the kerosene is liquid. In addition, there are several other changes that need to be considered to utilise optimally the changed conditions owing to burning hydrogen. The higher flame speed will result in a shorter combustor, and hence, reduced engine weight and combustor liner cooling requirements.

Generally for combustion gases generated by combustion of kerosene, the total emitted radiation comprises both ‘non-luminous’ and luminous radiation. At high levels of pressure encountered in modern gas turbine combustors, the flame is characterised by a predominance of luminous radiation (Singh[(23)]). When burning hydrogen there are no solid particles (except for those present in the incoming air), which give rise to luminous radiation, present in the flame. Therefore, the thermal energy radiated to the surroundings is essentially lowered when burning hydrogen, thereby beneficially affecting the liner durability and liner cooling requirements.

As discussed, hydrogen-air flames have an essentially wider flammability range, which will allow operation of the combustion zone at an equivalence ratio which has a large margin to the lean blow-out limit. This will facilitate the handling of the engine, as well as reduce the creation of white noise (noise with lots of different frequencies) that might give rise to pulsations and vibrations in the engine, which in turn via resonance can have a detrimental effect on engine components.

3.0 DESIGN OF HYDROGEN-FUELLED AERO GAS TURBINES FOR LOW ENVIRONMENTAL IMPACT

3.1 Introduction

In Section 2 the main effects on aero gas turbines when changing to hydrogen fuel, with emphasis on environmental issues, were covered. The potential of lowering the environmental impact and the technical feasibility of changing fuel was assessed by investigating the effects on pollutant emissions, performance and design. In this section this work is extended by looking at engine design for low environmental impact. In this context, the fuel consumption and pollutant emissions (water vapour and oxides of nitrogen) for complete flight missions are considered.

The water vapour is proportional to the fuel consumption, whereas the NO_x emissions are dependent on the combustor inlet conditions and a number of combustion chamber characteristics. In addition, reducing fuel burn and NO_x emissions might be in opposition to each other, implying that an engine design measure to reduce NO_x might give rise to the fuel burn and vice versa. While the wish to lower the fuel consumption is mainly driven by the objective of delivering products that promise low direct operating cost (and recently also for conforming to future environmental constraints on aviation), it is important to reduce the H_2O and NO_x emissions due to environmental concerns of their impact on the atmosphere and climate. The environmental effects of H_2O and NO_x emissions were discussed in Part I (see also Svensson *et al*[(24)]).

Whether engines should be designed for low mission fuel consumption or for low mission NO_x emissions to minimise the environmental impact is not obvious. Therefore, both options are covered. In practice, of course, a combination of these two aims could be desired. In order to study the potential of lowering either fuel burn and mission H_2O, or mission NO_x emissions, a number of alternative engine cycles are selected, i.e. cycles featuring different combinations of COT, OPR and BPR. The amount of H_2O emissions is known directly from the fuel consumption (since the H_2O emissions are proportional to the fuel burn), while the NO_x emissions are calculated using a semi-empirical correlation. Rather than finding the optimum engine cycle for a specific objective, the aim is to investigate a number of promising cycles, in order to provide some guidance on how to design hydrogen-fuelled aero engines for medium-range aircraft either for minimum mission fuel consumption or mission NO_x emissions.

In order to study the effects on flight mission emissions and fuel consumption of varied engine cycle parameters (COT, OPR and BPR) parametrically, a computer program is developed. In Section 3.2 the methodology which the computer program is based upon is briefly outlined. In Section 3.3 the datum engine cycle is described, and a number of promising alternative engine cycle concepts, either to lower the mission fuel consumption and H_2O emissions or the mission NO_x emissions, are suggested. Evaluations of the alternative engine concepts for flight missions are described in Section 3.4. The study outlined here is also published in a more extensive version in Haglind and Singh[(25)].

3.2 A modelling tool for aircraft mission analyses (AMA)

In order to study the effects on flight mission emissions and fuel consumption of varied engine cycle parameters (COT, OPR and BPR) parametrically, a simulation tool developed by the authors is used. The principles for re-sizing the aircraft and the engines, and the physical principles and equations which the performance estimations for the different flight phases are based upon are described in detail in Svensson[(26)]. Furthermore, the correlation method to estimate the NO_x emissions is outlined more in detail in Svensson[(26)]. In this section only a brief summary of the main features is given.

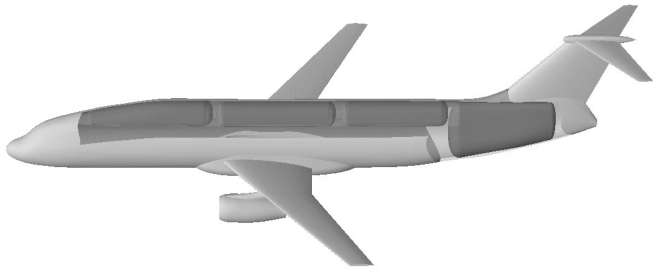

Figure 5. Aircraft layout for CMR1-200 (Oelkers and Prenzel(27)).

Table 3
Fundamental performance data for the datum aircraft

Quantity	Value
Design range (km)	6,614
Static sea-level thrust (kN)	134·0
MTOM (tonne)	87·6
OME (tonne)	61·2
Number of seats	185
Payload (tonne)	16·8
Fuel capacity (tonne)	9·6

The aircraft selected to be studied is a low-wing configuration based on the Airbus A321, which is a medium-range aircraft. In order to minimise the tank isolation requirements and the evaporation losses, the aim is to store the hydrogen in tanks that have a high volume to area ratio. For the aircraft under consideration, an aircraft that stores its fuel on top of the fuselage and in a tank outside the pressure cabin in the tail cone of the fuselage turned out to be a viable solution that offers the lowest penalty in energy consumption (Oelkers and Prenzel[27]). The aircraft layout is shown in Fig. 5 and some fundamental performance data are given in Table 3. Further details about the aircraft can be found in Oelkers and Prenzel[27], and Westenberger[28,29]. The performance data in terms of range, thrust and weights used here are slightly different compared with the data given in those sources. All calculations are conducted for a fully-laden aircraft, i.e. the cabin factor is assumed to be 100%. As a result of this the take-off mass is equal to the maximum take-off mass, MTOM.

When changing an engine cycle parameter, the engine and consequently also the aircraft need to be re-sized to provide still the thrust required at the most demanding flight phases. By providing a power setting (i.e. the ratio of COT at that flight phase to that of take-off) for each flight phase, the amount of engine scaling that is required is estimated. The program needs to be provided with engine data (i.e. thrust, fuel flow and emissions versus altitude, Mach number and power setting), which are obtained using the TurboMatch scheme. Various aircraft characteristics (e.g. the aircraft drag polar and weights) are taken from the Piano software (commercial aircraft modelling tool) (Simos[30]) and technical reports from the CRYOPLANE project. Effects on weights and geometries of engines and the aircraft are estimated using simple correlation methods (see Svensson [26]).

The water vapour emission is proportional to the fuel consumption, whereas the NO_x emissions are dependent on the combustor inlet conditions and a number of combustion chamber characteristics. The only available NO_x emissions data (at least to the authors' knowledge) obtained from a hydrogen-fuelled engine are those of the Honeywell APU GTCP 36-300, as outlined in Section 2.3.3. However, compared to main engines this engine is operated under relatively low temperature and pressure conditions. In order to estimate the emissions of NO_x at a specific operating condition, a semi-empirical correlation valid for the micromix combustor concept of third generation (see Section 2.3.3) only is used here. The correlation is derived and evaluated in Svensson[26], following the methodology suggested by Dahl and Suttrop[15], which was proposed within the EC-sponsored project CRYOPLANE. This correlation method involves the measured emissions data of the APU engine being extrapolated for higher combustor inlet pressures and temperatures by appropriate modelling based on similarity considerations. The correlation reads:

$$EI_{NOx} = EI_{NOx}^{*} \times \left(\frac{P}{P^{*}}\right)^{n} \times \left(\frac{\phi}{\phi^{*}}\right)^{m} \times \left(\frac{T_{st}^{*}}{T_{st}}\right) \times \frac{e^{\frac{-37700\,K}{T_{st}^{*}}}}{e^{\frac{-37700\,K}{T_{st}}}} \qquad \ldots (1)$$

where $EINO_x$ is the emission index of NO_x, P and T_{st} are the prevailing conditions of pressure and temperature in the reaction zone, and ϕ is the equivalence ratio related to the core engine airflow and hydrogen fuel. The symbol '*' denotes nominal measured quantities known from experimental work. The measured quantities could be selected from an arbitrary hydrogen operating condition of the Honeywell APU GTCP 36-300 test engine (Fig. 4). As for the exponents, n equals 1·0 and m equals 1·1 for cruise conditions (and lower altitudes) and 0·9 for take-off conditions. The background to the selection of these figures is given in Svensson[26].

For simplicity, the flight mission only comprises the take-off, climb and cruise phases; the descent and landing phases are excluded. Since these two neglected phases only constitute a few percent (about 2%) of the total fuel consumption, and in terms of NO_x, even less, for the medium-range flight mission of about 6,600km considered here, this simplification is not expected to affect significantly the results. Any possible variations in emissions and fuel consumption among the different engine concepts for the neglected flight phases would make a small difference, indeed, on the amounts for the complete flight missions.

3.3 Engine cycle concepts

In this section the datum engine cycle is described, and a number of promising alternative engine cycle concepts either to lower the mission fuel consumption and H_2O emissions or the mission NO_x emissions are suggested. Figures for the cycle parameters for the alternative cycles are chosen so as to cover the available options to attain either of these objectives. More details about the prerequisites for selecting the engine cycle concepts are outlined in Svensson[26].

3.3.1 Datum engine cycle

The datum engine is an up-scaled version of the V2527-A5 engine, i.e. the engine described in Section 2.2.1. This engine is a two-shaft boosted turbofan engine suitable for short- and medium-range aircraft. The losses and efficiencies of various components, cooling flows and the location of the heat exchanger to evaporate the hydrogen prior to its entry into the combustion chamber are the same as assumed in Section 2. The differences are that the datum engine considered here is designed for ISA conditions rather than ISA+10K and that the inlet mass flow is larger to provide the thrust required by the aircraft. Furthermore, in contrast to the study outlined in Section 2, the optimum outer fan pressure ratio is used here. Choosing a value on fan pressure ratio lower than the optimum for practical reasons seemed unnecessary for the preliminary design studies conducted here.

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3.3.2 Alternative engine cycle concepts

In order to study the influence on engine and flight mission performance of different values of cycle design parameters, a number of alternative cycles are selected. These engine cycles are not sufficient to define an optimum parameter combination for a specific objective. Finding the optimum parameter combination for a specific objective requires either that performance software including an optimisation feature is used or that an extensive parameter study, indeed, is performed. The objective here is rather to investigate a number of relevant parameter combinations selected on the basis of general gas turbine performance knowledge, in order to provide some guidance on how to design hydrogen-fuelled aero engines for medium-range aircraft, to minimise either mission fuel consumption or mission NO_x emissions.

Similarly as for the datum engine cycle, the optimum value on fan pressure ratio is used, provided that it is not larger than 1·9, which is the highest attainable pressure ratio of a single fan stage (Walsh and Fletcher[31]). If the optimisation procedure would show that the optimum fan pressure ratio is larger than 1·9, the pressure ratio is set to 1·9. In this study only the pressure ratio of the high-pressure compressor (HPC) is changed when the OPR is changed. In order to make a fair comparison of the different engine cycles, the isentropic efficiency of the HPC is changed such that the polytropic efficiency is the same as for the datum cycle. All the component efficiencies are the same for all engine cycles. As the design point, take-off conditions, i.e. static sea-level, ISA, is chosen.

3.3.2.1 Engine A. Datum core, high BPR

In this cycle the BPR is increased without changing the core engine. This cycle is defined mainly to investigate if it would bring any gains in mission fuel consumption by only increasing the BPR.

When the by-pass ratio is increased the engine frontal area increases, and thus also do the weight and nacelle drag. Moreover, the number of turbine stages needs to be increased (Walsh and Fletcher[31]).

As the by-pass ratio is increased, SFC and the specific thrust are decreased. The latter implies that the engine needs to be up-scaled, i.e. the engine weight and nacelle drag are increased, to provide the thrust required at the most demanding flight condition. A heavier engine with a larger frontal area would have a detrimental effect on mission fuel consumption. If weight and drag penalties are to be minimised, high by-pass ratio engines would therefore require shorter inlet and nozzles than conventional engines (Zimbrick and Colehour[32]). There is an upper limit on BPR beyond which the mission fuel consumption would increase. Le Dilosquer[33] suggests that for kerosene-fuelled engines at about the 1995 level of technology, the optimum BPR with respect to mission fuel burn lies between six and nine depending on the installation.

A detailed investigation of optimum BPR for minimum mission fuel consumption for liquid hydrogen-fuelled aircraft is beyond the scope of this paper. Bearing in mind the discussion above and that the equation to predict the engine weight is not reliable for high values of BPR (Svensson[26]), a BPR equal to 6·5 is chosen for this engine cycle. For the datum core and with a BPR of 6·5, the optimum outer fan pressure ratio is equal to 1·6.

3.3.2.2 Engine B. Low power core, datum BPR

Aiming for reduced NO_x emissions the COT is reduced by 100K. A reduced COT implies that there is less pressure available downstream of the turbines to produce thrust, and hence the optimum values of OPR and BPR decrease with reduced COT. In addition, a lowered OPR helps to lessen the NO_x emissions. A parametric study shows that for OPRs lower than 20 there seems to be a trend towards a greater penalty in SFC for a fixed gain in specific thrust (Svensson[26]). If SFC versus the NO_x emission index is plotted the trend is less emphasised; nevertheless, for OPRs lower than 20 the SFC penalty seems to rise for a given reduction in the NO_x emission index. Therefore, an OPR equal to 20 is chosen for this cycle, which corresponds to an optimum outer fan pressure ratio of 1·71.

3.3.2.3 Engine C. Low power core, low BPR

As argued for engine B, the optimum BPR decreases with reduced COT. Hence, it seems reasonable to include an option where the low power core defined for engine B is combined with a lower BPR. For this purpose a BPR of 3·8 is chosen. This figure is about 20% lower than the datum BPR, and is considered to be a reasonable value for an engine powering a medium-range aircraft. Due to the reduced BPR the optimum outer fan pressure ratio becomes larger than 1·9, and therefore the value of 1·9 is used.

3.3.2.4 Engine D. High power core, datum BPR

Instead of employing a weaker core as in engines B and C, a stronger core is employed in engines D and E. The reason for this is to study the possibility of lowering the mission fuel consumption, and thus also the mission water vapour emissions, using advanced engine technology. Being aware of the practical limitations of increasing the maximum cycle temperature of gas turbines, the COT is increased by 100K compared with the datum value to 1,655K. Based on the same arguments as raised for the engine with reduced COT saying that the optimum values on OPR and BPR change with COT, the OPR is also increased.

A parametric study shows that the SFC is improved and that the specific thrust is deteriorated with OPR (Svensson[26]). The variation is not large, but for OPRs larger than 35 there seems to be a trend towards a larger penalty in specific thrust for a given SFC gain. In addition, it may be argued that the penalty in the NO_x emission index for a given SFC improvement tends to increase for OPRs larger than 35. Based on these figures an OPR equal to 35 is chosen for this engine cycle. Likewise, for engine C the outer fan pressure ratio becomes larger than 1·9, and thus the value of 1·9 is used.

3.3.2.5 Engine E. High power core, high BPR

Since the optimum OPR and BPR increase with COT, it seems reasonable to include an option where both OPR and BPR are increased with COT. Here the same increased BPR as chosen for engine A, namely 6·5, is chosen. The parameters for this engine are hence: COT is equal to 1,655K, OPR is equal to 35 and BPR is equal to 6·5, and the optimum outer fan pressure ratio becomes equal to 1·74.

3.3.2.6 Performance of selected engine cycle concepts

A summary of cycle parameters for the selected engine concepts is displayed in Table 4, and the engine performances are shown in Table 5. All data refer to the design point, i.e. static sea-level, ISA, and all engines have the same inlet mass flow (and hence also inlet diameter), which is equal to 395kg/s.

Increasing the by-pass ratio and keeping the core engine unchanged for engine A has a substantial effect on the performance. The SFC is improved by 11%, while the thrust (or specific thrust) is deteriorated by 13·2%. This is the option that offers the largest SFC reduction; however, this is accomplished at the expense of the largest fall in specific thrust. As the NO_x emission index ($EINO_x$) only is dependent on the temperature and pressure at the combustor inlet as well as the fuel-air ratio, i.e. quantities that are solely dependent on the core engine, the NO_x emissions are unchanged.

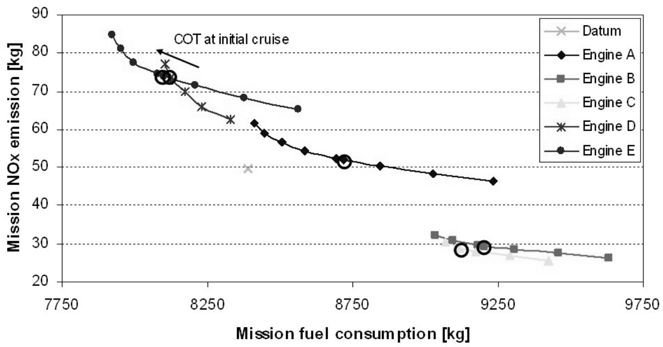

Figure 6. Flight mission NO_x emission versus mission fuel consumption for varied COT at initial cruise (sizes the engine) for the design range mission.

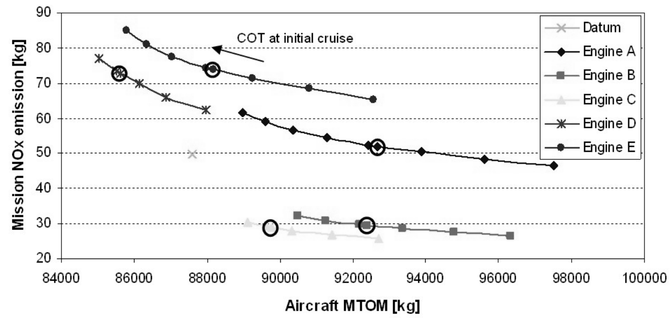

Figure 7. Flight mission NO_x emission versus aircraft MTOM for varied COT at initial cruise (sizes the engine) for the design range mission.

By employing a core engine with lower COT and OPR (engines B and C), a reduction in $EINO_x$ by almost 47% can be achieved. If the BPR is unchanged the SFC is more or less unchanged also, but the thrust is reduced by about 6%. On the other hand, if the BPR also is lowered the thrust increases, at the expense of a 10% SFC deterioration.

Increasing COT and OPR is efficient with respect to SFC provided that the BPR also is increased; otherwise SFC actually increases. See engines D and E. In terms of specific thrust, the most beneficial option is not to increase the BPR. Inevitably a higher COT and a higher OPR imply an essential increase in $EINO_x$. For the engines considered here, the NO_x emission index is increased by more than 50%.

3.4 Flight mission performance

Using the computer program described in Section 3.2, AMA, the mission emissions and fuel consumption are calculated for the datum engine/aircraft and the alternative engines/aircraft for a range equal to the design range of the datum engine/aircraft (6,614km).

Table 4
Engine characteristics of selected engine cycle concepts at the design point, static sea-level, ISA

	COT (K)	OPR (-)	BPR (-)	Opt OFPR (-)
Datum	1,555	28·5	4·8	1·85
A. Datum core, high BPR	1,555	28·5	6·5	1·60
B. Low power core, datum BPR	1,455	20	4·8	1·71
C. Low power core, low BPR	1,455	20	3·8	1·90
D. High power core, datum BPR	1,655	35	4·8	1·90
E. High power core, high BPR	1,655	35	6·5	1·74

Table 5
Performance of selected engine cycle concepts at the design point, static sea-level, ISA

	FN (kN)	FN (%)	SFC (g/kN s)	SFC (%)	$EINO_x$ (g/kgKE)	$EINO_x$ (%)
Datum	134·0	0·0	3·6046	0·0	9·8607	0·0
A. Datum core, high BPR	116·4	–13·2	3·2099	–11·0	9·8607	0·0
B. Low power core, datum BPR	125·6	–6·3	3·6385	+0·9	5·2569	–46·7
C. Low power core, low BPR	139·3	+3·9	3·9635	+10·0	5·2569	–46·7
D. High power core, datum BPR	143·2	+6·9	3·6972	+2·6	15·0560	+52·7
E. High power core, high BPR	124·9	–6·8	3·2764	-9·1	15·0560	+52·7

3.4.1 *Conditions for flight mission simulations*

During the execution of the program the user is asked to provide the power setting in terms of COT for take-off, climb and cruise. While the values of the COT ratio, i.e. COTi to COTmax, at take-off and climb are set to be the same for all engine configurations, the COT ratio at cruise is able to be varied. A regular take-off for a civil airliner is usually performed at full power condition or slightly reduced to increase the engine durability. In the calculations presented here COT is set to the maximum value at take-off. The climb phase is performed at a constant COT which is 96% of the maximum allowable value. The datum engine in this study is designed for a ratio of COT at cruise to COT at max climb equal to 0·877. The cruise phase is conducted at the Mach number of 0·8 at a constant altitude equal to 11,000m.

3.4.2 *Results*

The amount of engine scaling influences the aircraft size, mission fuel consumption, mission emissions and the flight envelope (rate of climb). In Fig. 6 the mission NO_x is plotted versus the mission fuel consumption, and in Fig. 7 the mission NO_x is plotted versus the aircraft MTOM, for a range of COT values at initial cruise (COT_{ic}) for the datum and the selected engine cycles. Going from right to left in the figures, the maximum power setting at cruise increases, which means that the necessary engine size decreases. Smaller engines mean that the aircraft could be smaller and lighter for a given payload and range, which also implies that it consumes less fuel (see Fig. 6). The mission NO_x emissions, on the other hand, increase, since a higher power setting during cruise where the main share of the NO_x emissions are formed enhances the NO_x production. In addition, the take-off distance gets longer and the rate of climb gets lower when the engine size is decreased (even if the aircraft is lighter), and hence the climb phase becomes longer and more time-consuming. All engine concepts are evaluated for the same power settings, and the COT at initial cruise is increased until the limit where take-off becomes the operating point that sizes the engines.

In Table 6 the effects on engine size, MTOM, mission fuel consumption and mission NO_x emissions of the different engine concepts are summarised. These data are obtained for ratios of COT at initial cruise to COT at max climb equal to the value for the datum engine (the encircled points in Figs 6 and 7), which is 0·877. The linear scale factor (LSF) is defined as the ratio of the actual diameter to the diameter of the datum engine. Using the datum COT ratio for engine E, the fuel consumption is reduced by 3·5%, and the MTOM is more or less unchanged at the expense of increased mission NO_x emissions by about 49%. To achieve this, the engine diameter is increased by 6·9%.

If the increased COT and OPR are combined with the datum BPR (engine D), the MTOM is reduced for a given COT_{ic}, but the potential

Table 6
Flight mission performance for the various engine concepts for the design range of 6,614km

	COT_{ic} (K)	LSF (-)	MTOM (kg)	MTOM (%)	Mis Fuel (kg)	Mis Fuel (%)	Mis NO_x (kg)	Mis NO_x (%)
Datum	1,336·0	1·000	87,596	0·0	8,389·9	0·0	49·63	0·0
A. Datum core, high BPR	1,336·0	1·157	92,687	+5·8	8,718·8	+3·9	51·89	+4·6
B. Low power core, datum BPR	1,250·1	1·072	92,400	+5·5	9,203·5	+9·7	29·39	–40·8
C. Low power core, low BPR	1,250·1	0·973	89,767	+2·5	9,120·0	+8·7	28·98	–41·6
D. High power core, datum BPR	1,421·9	0·947	85,613	–2·3	8,130·2	–3·1	72·76	+46·6
E. High power core, high BPR	1,421·9	1·069	88180	+0·7	8,100·2	–3·5	73·77	+48·6

of lowering the fuel consumption is deteriorated compared with engine E. Assuming the COT_{ic} is the same as for the datum case, the engine diameter is decreased by 5·3%, MTOM is decreased by 2·3%, the fuel burn is decreased by 3·1%, and the NO_x emissions are increased by 47%. By allowing for a higher power setting during cruise, and thereby decreasing the engine size, it would be possible to raise the fuel burn and MTOM benefits further. However, the NO_x emissions would also increase. The extent to which the engine size can be decreased is limited by the thrust requirement at take-off.

As can be seen in Table 6 the mission fuel burn, mission emissions and MTOM are all detrimentally affected if the BPR is increased and the core is kept unchanged (engine A). It would be possible to keep the fuel consumption at about the same level as the datum engine/aircraft combination if the COT_{ic} is increased maximally and thereby the engine size is decreased to a minimum. However, it may be accomplished at the price of a MTOM increase of 1·6% and NO_x emission increase of 24%.

Employing engine C, the mission NO_x emissions can be reduced by 42% at the expense of increased MTOM by 2·5% and increased fuel consumption by 8·7%. Owing to an increased specific thrust, the engine diameter is actually decreased by 2·7% compared with the datum engine. If the BPR is not lowered the engine size needs to be increased (engine B). The reduction in mission NO_x emissions is about the same, but the enlargements in MTOM and fuel consumption become larger: compared with the datum engine the MTOM increases by 5·5% and the fuel burn increases by 9·7%. Looking at engine C and employing a larger engine and thereby allowing for a lower COT during cruise (and the other phases), the reduction in the mission NO_x may be increased further, at the expense of an increased MTOM and increased fuel burn. The extent to which the engine size may be increased is limited by how large an increase in aircraft weight and fuel consumption is considered to be acceptable, and ultimately the maximum engine size will also be limited by practical geometric constraints.

4.0 CONCLUSIONS AND DISCUSSION

In this second part of the paper (constituted of three parts), the subject of engine design when burning hydrogen is raised.

- From a technical point of view, it seems to be feasible to use hydrogen for aexro gas turbines. The main changes comprise re-design of the combustion chamber and fuel control system, as well as the implementation of facilities to evaporate the hydrogen prior to its entry into the combustion chamber. The fuel heating can be accomplished either by an external heat source or a heat exchanger located at a suitable engine location. Small performance gains, which depend on the fuel temperature and cycle configuration, in the order of a few percent may be obtained by changing to hydrogen fuel. By employing unconventional cycles, it would be possible to increase the performance gains. However, it appears to be questionable if these benefits justify the increased complexity imposed by unconventional cycles.
- In terms of pollutant emissions, hydrogen use offers the possibility of a significantly reduced number of emission species, resulting in only H_2O and NO_x emissions. All emissions containing carbon and sulphur are eliminated. Preliminary calculations suggest that there is the potential to design a combustion system using hydrogen that produces less NO_x emissions than a system burning kerosene. This assertion is based mainly on the simplified analysis comparing primary zone flame temperatures calculated under the assumption of chemical equilibrium for kerosene and hydrogen fuel (Section 2.3.4). In addition, the qualitative discussion of the effects on the NO_x emissions when burning hydrogen (Section 2.3.2) points in that direction. However, from these analyses it cannot be said for sure that there actually is a potential to design hydrogen-fuelled combustors that give lower NO_x emissions. Further studies on this topic, taking into account all relevant NO_x formation principles, the chemical kinetics and the flow field, are recommended as future work.
- Due to the risk of auto-ignition for premixed systems and the problems of large-scale hydrogen diffusion flames, the lean non-premixing concept of micromix combustion, which is based on miniaturised diffusive combustion, is suggested as a promising hydrogen combustor configuration.
- Due to the wider flammability range of the hydrogen-air flames (compared with kerosene-air flames), which allows operation of the combustion zone at an equivalence ratio which has a large margin to the lean blow-out limit, the engine handling is facilitated and the creation of white noise is reduced. Furthermore, when burning hydrogen the thermal energy radiated to the surroundings is lower than that of kerosene, thereby beneficially affecting the liner durability and liner cooling requirements.
- In order to lower the fuel consumption and thus also the mission H_2O emissions for a medium-range liquid hydrogen-fuelled aircraft, the results indicate that an engine employing increased COT, OPR and BPR, seems to be the most attractive choice. A reduction in mission fuel consumption of 3·5% and an aircraft maximum take-off mass more or less unchanged at the expense of increased mission NO_x emissions of about 49% is demonstrated. For minimum mission NO_x emissions, on the other hand, the results suggest that an engine with a low power core engine, i.e. reduced COT and OPR, along with a lowered BPR, should be used. According to calculations presented in the paper, the mission NO_x emissions can be reduced by 42% at the expense of increased aircraft maximum take-off mass by 2·5% and mission fuel consumption by 8·7%.
- ‘Technologies involving a high power core accompanied with increased BPR are changes that are subject to improvements. Increasing the temperatures and pressures in the gas turbine cycle requires advancements in high temperature materials and cooling technology. An increased by-pass ratio requires an increased number of turbine stages, a shorter inlet and shorter nozzles than conventional engines. It is therefore likely that engines based on

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such technologies raise the development, production and maintenance costs. Moreover, the durability of the core engine is likely to decrease. In contrast, engines designed for lowering mission NO_x emissions employing a core engine with lowered COT and OPR along with reduced BPR, are based on reliable technology that may eliminate the development cost, reduce the production and maintenance costs and increase the durability.

In the last part, Part III, the optimum cruising altitudes for minimum global warming for a liquid hydrogen-fuelled aircraft and a conventional aircraft are investigated. Furthermore, the feasibility of introducing cryoplanes on a regional level is explored. The infrastructure changes of introducing cryoplanes and possible hydrogen production methods at Stockholm/Arlanda airport are discussed.

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† 'Fredrik Svensson' has changed his surname to 'Haglind'.