Aircraft Propulsion
Leading the way in Aviation

Prof. H. Wittenberg
AIRCRAFT PROPULSION - LEADING THE WAY IN AVIATION

by

Prof. H. Wittenberg
Faculty of Aerospace Engineering
Delft University of Technology, the Netherlands

SUMMARY

Some fundamental aspects of airbreathing propulsion systems are considered with emphasis on fuel economy in relation to the thermal and the propulsion efficiency. The state of the art of piston-engines in general aviation is surveyed shortly. Next, the application of gasturbines in civil transports at high-subsonic speeds is considered. Trends of future developments with respect to ducted- and unducted (propulsion) concepts are discussed. Furthermore some features of the propulsion of supersonic transports are reviewed, based on the Concorde and future developments. In the final section propulsion concepts for hypervelocity vehicles are reviewed shortly. Throughout the article the leading role of propulsion in aviation is emphasized.

Contents:

Preface
1. Introduction
2. Some fundamentals on propulsion
3. Piston-engines
4. High-subsonic propulsion systems
5. Propulsion at supersonic speeds
6. Propulsion at hypersonic speeds
7. References
PRE FACE

This report on aircraft propulsion is based on an introductory lecture by the author at the symposium "Developments in Aircraft Propulsion", organized by the Association of Aeronautical Students at the College of Advanced Technology in Haarlem, the Netherlands on March 18, 1987. The paper is published in the Proceedings of the Symposium. A revised and extended version of the original lecture is presented in this report. The author is aware of the fact that writing of the report had not been possible without the extensive use of sources from the literature. In this way many authors have contributed indirectly to its contents, which is mentioned here with appreciation.

With this publication the author intends to honour the late Professor Gordon C. Oates, who died suddenly and unexpectedly on November 1, 1986. At the time of his death Professor Oates was a member of the Propulsion and Energetics Panel of AGARD, then under the chairmanship of the author. Professor Gordon C. Oates was a well-known educator and research scientist in aerospace propulsion. His last post was Professor at the Department of Aeronautics and Astronautics, University of Washington, in the United States. By his professional work he has given many valuable contributions to the propulsion field. He was author of two well-known volumes in the AIAA Education Series: "Aerothermodynamics of Gas Turbine and Rocket Propulsion" and "Aerothermodynamics of Aircraft Engine Components", both textbooks of a high standard for students and practicing engineers. Professor Oates will not only be remembered by his colleagues and friends for his competent professionalism, but also for his dedication to human relationships. The author is thankful for having known him personally for some years.
1. Introduction

During the development of aerospace over the years propulsion has been one of the main driving forces. The first successful powered flights of the Wright Brothers in 1903 were achieved because a suitable engine was built by themselves. Size and speed of the aircraft could be continuously increased by developing more powerful and efficient piston engines, until jet propulsion offered the possibility of a quantum jump in these aircraft characteristics into the high-subsonic speed range. The supersonic speed regime could also be conquered only by the forceful thrust of the propulsion system. The same holds for the launching of spacecraft through rocket power.

Today new developments in the propulsion field are on the horizon: unducted and ducted very high by-pass ratio engines for subsonic transports, variable cycle engines for supersonic speeds, ramjets with solid fuels or with supersonic combustion for missiles, and other advanced air-breathing propulsion systems for hypersonic vehicles for the launch of spacecraft or for ultrafast transport over the Earth.

In an introductory survey of limited length, it is quite impossible to cover all propulsion subjects adequately and a selection must be made. The choice of the author has been to concentrate on propulsion systems for fixed-wing aircraft in the field of civil aviation.

For every propulsion system a large range of characteristics is important for its application in mostly every aircraft design:
- the speed range, which can be covered,
- the available thrust and (specific) fuel consumption within the envelope of operating conditions,
- configuration characteristics, such as overall size, weight, frontal area and installation concepts,
- operational characteristics: reliability, maintainability and repairability,
- environmental characteristics: noise and pollution,
- costs of development and production.

A family-tree of airbreathing propulsion systems is given in fig. 1; for each system the design Mach number range is indicated in this figure. Most of these types of powerplants will be discussed in this survey, putting emphasis on the fuel economy as one of the prime characteristics of every powerplant. Before
the propulsion systems for the various speed regimes are considered in detail, some fundamentals on aircraft propulsion are given in the next section.

2. Some fundamentals on propulsion

All propulsion systems are based on the third law of Newton: action = reaction. Mass is accelerated with respect to the vehicle (action) to obtain a propulsive force (thrust) in the opposite direction (reaction). For air-breathing propulsion systems the atmospheric air is used to create the propulsive force. For rocket propulsion, on the other hand, all mass required for propulsion is carried on board of the vehicle and no atmospheric air is required. Thus only this last type of propulsion is suitable for operation in the space environment.

All propulsion systems have two conversion systems: the energy conversion system and the power conversion system (or thrust generator; Fig. 2). Excluding nuclear power, all energy conversion systems make use of the chemical energy of fuels; for air-breathing powerplants the fuel is burnt with atmospheric air.

For piston engines and propeller-gasturbines the energy conversion is separated from the thrust generation. The energy generator delivers mechanical shaft power through the action of reciprocating pistons or rotating turbine blades. The shaft power is transmitted to the propeller, which produces the thrust by the acceleration of atmospheric air only.

For the pure turbojet-engine and ramjet, however, the energy conversion system produces gaspower, which is directly converted into the thrust generating (hot) jet by the exhaust nozzle.

The ducted or unducted turbofan engines combine both schemes: the available gaspower of the energy conversion system is partly used to generate a hot propulsive jet and partly to accelerate atmospheric air through the action of a propeller or fan.

By applying the second law of Newton, we can obtain a simple expression for the thrust force of a propulsion system (Fig. 3):

\[
T = m (w - V) + m_f w
\]  
(1a)
where: $T = \text{thrust}$, $m = \text{air mass accelerated per unit time}$, $m_f = \text{fuel mass consumed per unit time}$, $w = \text{jet or wake velocity}$, $V = \text{flight speed}$. This expression holds for rockets ($m = 0$) as well as for air-breathing propulsion systems. In the latter case, $m_f$ is very small in comparison with $m$ and eq. (1a) reduces approximately to:

$$T = m(w - V) \quad (1b)$$

The overall efficiency of the air-breathing powerplant at a given speed is the ratio between the propulsive power $TV$ and the energy content of the fuel, consumed per unit time. With the heating value $H$ per unit fuel mass and $F$ the fuel consumption per unit time, the overall propulsion efficiency is:

$$\eta_{\text{tot}} = \frac{TV}{HF} \quad (2)$$

For all common hydrocarbon fuels the heating value is about the same; here we will use: $H = 43.1 \times 10^3 \text{kJoule/kg}$ ($H/g = 4395 \text{ km}$, $g = 9.80665 \text{ m/s}^2$).

To analyse the characteristics of powerplants, it is useful to divide the overall efficiency into two components, related to the conversion steps in Fig. 2:

- the thermal or cycle efficiency: $\eta_{\text{th}} = \frac{P}{HF}$ = power output / heat content of fuel.

  The power output is the shaft horsepower or the gaspower of the energy conversion system. The thermal efficiency includes all heat and mechanical losses in the conversion system.

- the propulsive efficiency: $\eta_p = \frac{TV}{P}$.

Here $TV$ is the propulsive power produced by the thrust generator.

Obviously the following relation holds:

$$\eta_{\text{tot}} = \eta_{\text{th}} \cdot \eta_p \quad (3)$$

For these efficiencies physical limits can be derived from fundamental laws. The thermal efficiency depends on the thermodynamic cycle of the engine. For the Otto cycle of piston engines and the Brayton cycle of gas turbines the maximum thermal efficiency is given by the well-known expression for ideal cycles (Fig. 4):
\[ \eta_{Br} = 1 - \frac{1}{\frac{\varepsilon}{\gamma}} \]  \hspace{1cm} (4)

where \( \varepsilon \) = overall pressure ratio of the cycle and \( \gamma = \frac{C_p}{C_v} = 1.4 \) (air cycle).

The upper limit of the propulsion efficiency is given by the Froude efficiency. In this case the power output \( P \) is expressed as the increase of kinetic energy of the propulsive jet:

\[ P_j = \frac{1}{2} m (w^2 - V^2) \]

which leads to:

\[ \eta_{Fr} = \frac{TV}{P_j} = \frac{\frac{m(w - V)V}{\frac{1}{2} m(w^2 - V^2)}}{1 + w/V} = \frac{2}{1 + w/V} \]  \hspace{1cm} (5a)

We introduce here two other parameters:
- the specific thrust: \( \psi = \frac{T}{mg} \) (dimension: second)
- the thrust coefficient: \( C_F = \frac{T}{mV} \) (dimensionless)

With these two expressions, we get by combining with eq. (1b):

\[ \eta_{Fr} = \frac{2}{2 + \frac{\psi g}{V}} = \frac{2}{2 + C_F} \]  \hspace{1cm} (5b)

Only for \( C_F = 0 \) (\( T = 0 \)), we obtain \( \eta_{Fr} = 1 \). For a given thrust the Froude efficiency \( \eta_{Fr} \) increases with increasing mass flow \( m \) and with increasing flight speed \( V \). As we will see later on this conclusion is very important in the analysis of trends in the development of all air-breathing propulsion systems.

The overall propulsion efficiency is directly related to the specific fuel consumption (SFC), which is generally used in engineering practice. The SFC can be expressed as the fuel consumption per unit of power \( (c_p = \frac{F}{P}) \) or per unit of thrust \( (c_T = \frac{F}{T}) \). One finds easily the relationships:

\[ \eta_{tot} = \frac{TV}{HF} = \frac{V}{HC_T} = \frac{\eta_p}{HC_P} \]  \hspace{1cm} (6)
Table 1 illustrates the characteristics given above for a typical large turbofan engine in cruise condition at altitude. We find from eq. (6) that even in the ideal case of no losses ($\eta_{th} = \eta_p = \eta_{tot} = 1$), the SFC based on thrust has a non-zero value: $c_T = 0.20 \text{ kg/daN-h}$ (approximately also $1\text{ lb/1F}_{f\text{-h}}$). For the actual values of $c$ and $\phi$ of the engine, the SFC for the theoretically maximum efficiencies is more than doubled ($c_T = 0.43 \text{ kg/daN-h}$). Further losses in the engine increase the SFC-value in practice to: $c_T = 0.62 \text{ kg/daN-h}$ (1970 technology, e.g. Rolls Royce RB-221-22B, Pratt and Whitney JT-9D series).

The overall propulsive efficiency $\eta_{tot}$ can be easily related to the specific range of the aircraft:

$$\frac{V}{F} = \frac{V}{c_T} = \frac{V}{c_p} = \eta_{tot} \frac{H}{g} \frac{L}{M}$$  \hspace{1cm} (7)

Here we have used the well-known relation for steady horizontal flight ($D = \text{drag, } L = \text{lift, } M = \text{aircraft mass}$):

$$T = D = \frac{D}{L} M g$$  \hspace{1cm} (8)

Assuming constant values of $\eta_{tot}$ and $L/D$ during the flight, we get by integration of eq. (7) a generalized expression for the Bréguet range equation with $F = -\frac{dM}{dt}$:

$$R = \int V \, dt = - \int \frac{M_1}{V} \, dM = \eta_{tot} \frac{H L}{g} \ln \frac{M_1}{M_0} = -\eta_{tot} \frac{H L}{g} \ln \left(1 - \frac{M_F}{M_0}\right)$$  \hspace{1cm} (9)

Here $M_0 = \text{initial aircraft mass, } M_1 = \text{final aircraft mass and } M_F = M_0 - M_1 = \text{fuel mass.}$

We introduce also the so-called unit equation for the mass ratio's of the aircraft ($M_D = \text{basic operational mass, } M_p = \text{payload mass}$):

$$\frac{M_D}{M_0} + \frac{M_p}{M_0} + \frac{M_F}{M_0} = 1$$  \hspace{1cm} (10)
Combining eqs. (9) and (10) we find for the payload mass ratio of the aircraft:

\[
\frac{M_p}{M_o} = e^{-\frac{\eta_{\text{tot}} L}{g D}} \frac{M_b}{M_o} - \frac{M_b}{M_o}
\]  

(11)

Now we will consider an example for a long-range aircraft of the Boeing 747-class. We assume a payload \( W_p = 42,000 \text{ kg} = 42 \text{ (metric) ton} \) and an equivalent still-air cruise range of \( R = 11,000 \text{ km} \) (\( \approx 6000 \text{ n.m.} \)). For the datum-aircraft we take: \( \frac{M_b}{M_o} = 0.47, \frac{L}{D} = 15 \) and \( \eta_{\text{tot}} = 0.32 \), which corresponds at Mach number 0.8 and 10.67 km altitude with \( c_T = 0.62 \text{ kg/daN-h} \) (Table 1). From eq. (11) we derive for the initial aircraft mass (take-off mass): \( M_o = 340 \text{ ton} \) with \( M_f = 138 \text{ ton} \) and the relative mass ratio's:

\[
\frac{M_p}{M_o} = 0.124 \quad \text{and} \quad \frac{M_f}{M_o} = 0.406
\]

Now we will consider an improvement of the overall propulsion efficiency to \( \eta_{\text{tot}} = 0.35 \), corresponding with \( c_T = 0.567 \text{ kg/daN-h}_f \) (\( M = 0.8, h = 10.67 \text{ km} \)). Now we find from eq. (9) for the same values of \( R \) and \( L/D \) a fuel-mass ratio:

\[
\frac{M_f}{M_o} = 0.379.
\]

Without further change of the aircraft design, we may assume for the absolute basic operating mass and payload mass the same values as above. Then we find from eq. (10) for the take-off mass: \( M_o = 325 \text{ ton} \) and \( M_f = 123 \text{ ton} \). Compared to the datum-aircraft the fuel mass is reduced by 15 ton or 10.9 %. Due to the reduced aircraft mass this is higher than the 8.5 % improvement of S.F.C.

We also consider the case that the aircraft is redesigned for the lower S.F.C.-engines. If we assume the same basic operational mass ratio by applying also improved aircraft technology \( \frac{M_b}{M_o} = 0.47 \), we find from eq. (10):

\[
\frac{M_p}{M_o} = 0.151. \quad \text{Thus the payload mass ratio with respect to the datum-aircraft is increased with 21.8 %!}
\]

For the same payload of \( W_p = 42 \text{ ton} \), the initial mass is now: \( M_o = 278 \text{ ton} \) and the fuel mass: \( M_f = 105.4 \text{ ton} \). Compared to the datum-aircraft the fuel mass

*) The actual range will be less because of extra fuel consumption in climb, reserve-fuel requirements, etc. Also uninstalled S.F.C.-values have been used here for simplicity.
reduction is increased to 32.6 ton or 23.6 %. This is about twice the fuel mass reduction for the previous case of a fixed aircraft design. The simple example above shows clearly the large effect of the overall propulsion efficiency (or S.F.C.) on the overall characteristics of the aircraft. This explains very well the efforts by the engine manufacturers to improve the fuel consumption of their engine designs. The state-of-the-art of these efforts for subsonic powerplants will be described in section 4.

Obviously, another important parameter is the heating value \( H \) of the fuel used. The highest value of \( H \) is offered by hydrogen as fuel \( (H = 120.10^3 \text{ kJ/kg}) \) with the large disadvantage of cryogenic liquid state and the low specific mass \( (0.07 \text{ kg/l as compared to } 0.8 \text{ kg/l for hydrocarbon fuels}) \). Despite the high value of \( H \) the use of hydrogen requires very large fuel tanks, not to be accommodated in the conventional aircraft configuration. Several studies have indicated that the common hydrocarbon fuels offer the best compromise between heating value and specific mass for the well-established shape of present aircraft and no other fuel types can be expected for many years to come, except for hypersonic propulsion systems (section 6).

In this respect one should also realize that huge production and logistic facilities are required to fulfil the large fuel demand for world-wide operations in aviation. This may prevent the introduction of radical new fuels for a long time and it is likely that with the exhaustion of the natural oil reserves, synthetic oil from coal, tar sands, etc. will offer the solution for aviation.

3. Piston engines

Until the early fifties the piston engine and propeller dominated the propulsion field as the only propulsion system practically available. The piston engine developed from the 9 kW (12 pk) four cylinder engine of the Wright Brothers in 1903 with a specific engine mass of 10 kg/kW (16.7 lb/hp) to the 18 cylinder Wright Turbo-compound Cyclone with a maximum power output of 2760 kW (3700 hp) and a specific engine mass of 0.56 kg/kW (0.92 lb/hp) in 1950. The second engine type was used in the last generation of large piston transport aircraft in the fifties, such as the Douglas DC-7C.
The piston engine cycle is characterized by a given compression ratio (7-10 based on volume, 13-20 based on pressure), which is independent of the operating conditions (flight speed and engine rpm). Moreover the effect of reduced power at altitude can be offset by mechanical- or turbo-supercharging. Combined with the high propeller efficiency the piston engine offers a good overall efficiency $\eta_{\text{tot}}$ at low subsonic speeds. For the Wright Turbo-compound Cyclone we have the set of data:

$$\eta_{\text{th}} = 0.36, \quad \eta_p = 0.80, \quad \eta_{\text{tot}} = 0.28 ; \quad c_p = 0.24 \text{ kg/kW-h (0.40 lb/hp-h)}$$

As a last unsuccessful effort to compete with the gasturbine, the Napier Nomad engine was developed in the UK in the early fifties (Ref. 1). This compound engine with a maximum take-off power of about 3000 kW (4000 hp) combined the compressor- and turbine-combination of the gasturbine with a piston-diesel cycle for the combustion process (Fig. 5). Under the most favourite cruising condition the specific fuel consumption was: $c_p = 0.20 \text{ kg/kW-h (0.326 lb/hp-h)}$, which corresponds with: $\eta_{\text{th}} = 0.43$. Note that this figure is also achieved by the gasgenerator of current turbofan engines (Table 1).

The weight and complexity of the piston engine, combined with the reduction of the propeller efficiency by compressibility effects at high-subsonic speeds, has brought the development of the large piston engines to a stop and its role for large civil aircraft was rapidly taken over by the gas turbine in the late fifties and early sixties. Today piston engines are only applied in the general aviation field, especially for small aircraft with engines in the power range of 100-300 kW (135-400 hp). In numbers piston-engined aircraft dominate still the world's aircraft fleet. Today in the ICAO-Contracting States (excl. USSR and the Republic China) e.g. 300.000 civil general aviation aircraft with piston-engines are registered against 9000 civil transport aircraft with gasturbine-engines!

Most currently used piston engines are still based on designs of several decades ago. In the wake of the so-called first energy crisis in the early seventies, other engine types for general aviation aircraft were studied, e.g. Diesel engines and rotary-type or Wankel engines (see Ref. 2 and 3). Application of these engines does not only offer advantages in fuel economy, but avoid also the use of aviation fuels of a high-octane grade, which were expected to become scarce in the future. Several development programs of such
engines have been initiated, but no prospect for a large scale application has
emerged until now. On the other hand attempts are made to develop aero-en-
gines based on the technology and hardware of automobile engines, e.g. the
Porsche PFM 3200 (160 kW, 215 kp take-off power, Refs. 4 and 5). Possibly with
the exception of modified car engines, it seems that the designer of general
aviation aircraft has to stick to the piston engines of conventional design or
to look for small gasturbines as alternative. In the latter field the engine
manufacturers are more active in the development of small types (Ref. 7d).
Hence, it seems likely that the gasturbine will gradually push aside the pis-
ton engine in those general aviation aircraft for which the higher capital
costs of the gasturbine in comparison with the piston engine can be accepted.

4. High-subsonic propulsion systems

During World War II the jet engine was developed by the pioneering work
of von Ohain et al in Germany and Whittle in the United Kingdom. A rapid
development of the jet engine and its derivatives took place in the following
decennia and is still going on.

In the very first civil jet aircraft in the fifties (De Havilland Comet,
Sud-Aviation Caravelle, first types of Boeing 707 and Douglas DC-8) straight
(or pure) jet engines were installed. The specific thrust of these engines is
high, leading to a poor propulsive efficiency, even at high-subsonic speeds.
Due to the thirsty engines the range was insufficient for long-range opera-
tions. Moreover, the engine take-off noise level was very high because of the
large velocity of the hot jet. The gasturbine with propeller (turboprop) of-
fered a better fuel efficiency, but the conventional propellers prevented the
application at high-subsonic speeds. This last type of propulsion system has
found its own place for short-range small civil aircraft with moderate cruise
speeds (450-500 km/h), e.g. the Fokker F-27, Fokker 50, Aérospatiale/AirItalia
ATR-42, De Havilland Canada DHC-8 and several other types.

A break-through of the jet engine occurred with the by-pass jet engines
(low by-pass ratio), followed by the turbofan engines (high by-pass ratio).
The core of these engines is the gasgenerator (combination of high-pressure
compressor, combustion chamber and turbine). The gas power produced is divided
between the cold flow of the low-pressure compressor or fan and the hot pro-
pulsive jet. For most engines the fan/low-pressure compressor and high-pres-
sure compressor are driven by separate turbines (two- and three-spool con-
figurations, Fig. 6).
A main characteristic of these engines is the by-pass ratio (BPR): $A = \frac{m_c}{m_h}$
($m_c =$ cold air flow mass, $m_h =$ hot air flow mass). With the corresponding ex-
haust speeds $w_c$ and $w_h$ the thrust for separated propulsive jets is:

$$T = m_c(w_c - V) + m_h(w_h - V)$$  (12)

For any given gasgenerator and flight condition, an optimum split of the
available gas power over the cold and hot flow can be found for which the
thrust and propulsive efficiency are maximum. If the losses in the turbine/fan
combination are neglected this optimum condition is: $w_c = w_h$ and the ratio be-
tween the cold and hot thrust is equal to the BPR. In this case the thrust is
given by the same expression as for the pure jet:

$$T = (m_c + m_h)(w - V) = m(w - V)$$.  (13)

Obviously, this equation holds also for by-pass engines with a fully mixed ex-
haust of the cold and hot flow.

If the losses of the fan/turbine combinations are taken into account, the op-
timum condition for separated jets is roughly given by: $w_c = 0.80 - 0.85 w_h$.
For a given gasgenerator the (optimum) exhaust velocities $w_c$ and $w_h$ decrease
with increasing BPR (increasing mass flow) and a larger part of the available
gaspower has to be diverted to the cold flow. The lower velocity $w_c$ leads also
to a lower fan pressure ratio in the optimum condition.

The first generation of turbo-engines with by-pass flow came into service
during the sixties and only low to moderate BPR were applied (Rolls Royce
Conway, Pratt and Whitney JT-3D- and -8D series, BPR = 0.5-1.7). In the early
seventies the second generation with larger BPR's of 4-6 were introduced
(turbofan-engines: Rolls Royce RB 211-22B, Pratt and Whitney JT-9D series,
General Electric CF6-6 and -50 series, GE/SNECMA CFM-56-2). Contrary to the
first generation, these types of engines have a single stage fan (fan pressure
ratio $\epsilon_f = 1.6 - 1.7$).

The improvement of the engine characteristics by increasing the BPR is shown
in Fig. 7 and 8. With the larger mass flow, the specific thrust decreases
which results into a higher propulsive efficiency and lower S.F.C. The improvement of S.F.C. was the first goal of the engine designers, but at the same time it became evident that the high BPR-engines could offer a considerable reduction in aircraft noise, which was very important in view of the growing public pressure to improve the airport noise environment. The noise reduction is shown schematically in Fig. 8. Measures to reduce the mixing noise of the high velocity, free propulsive jet of turbojet engines have shown to be rather ineffective, despite many research efforts. But for the turbofan engines the velocity of the hot jet and its generated noise is reduced considerably with increasing BPR. However, the low-pressure compressor or fan introduces an extra noise source, which at high BPR dominates the jet noise. Because the compressor or fan as noise source are housed within the engine nacelle, effective noise reduction measures with noise absorbing wall materials can be taken. Moreover, in the case of $\lambda = 5$, a single row of fan blades can be applied without inlet guide vanes. With the swirl-eliminating vanes at some distance behind the fan, practically all wake-interactions between rotating and static blades can be eliminated as noise source in the cold flow of the engine. These design features can be clearly seen in Fig. 6.

To discuss further improvements of the fuel efficiency relative to the datum-engine in Table 1 and Fig. 7, we have to take a closer look to the parameters, which affect the thermal and propulsive efficiency of the propulsion system. For a given cruise condition (Mach number and altitude) the thermal efficiency depends on the overall cycle pressure ratio $\varepsilon$, the turbine-inlet temperature TET and the component efficiency level $\eta_{o}$ (compressor and turbine efficiencies); see Fig. 9. At the TET-levels of interest, the thermal efficiency rises slightly with increasing overall cycle pressure ratio. The reduction relative to the Brayton-cycle efficiency $\eta_{Br}$ decreases with increasing TET and increasing component efficiency level $\eta_{o}$. Table 2 illustrates the effects of these parameters on $\eta_{th}$ by extrapolation of the relevant parameters of the datum-engine in Table 1 to target-data for advanced civil turbofans. Note the important effect of the component efficiency level, which explains the efforts of the engine manufactures to reduce the losses by improvement of the design of compressors and turbines. The increase of the turbine-entry temperature will
result from advancements in turbine blade materials, in fabrication- and coating methods and in cooling techniques (e.g. reduction of the amount of cooling air for the turbines).

For the same BPR an increase of TET will result in a higher jet velocity, which effects the propulsive efficiency unfavourable. Thus a higher TET will require also a higher BPR to maintain the same propulsive efficiency. Assuming this adjustment of BPR, Table 2 shows the decrease of SFC corresponding with the improvement of the thermal efficiency only. The combined increase of c_T, TET and n_o gives a SFC of c_T = 0.50 kg/daN-h, which is 20% lower than the level of the datum-engine. Note also that the favourable effect of an increase of TET on n_th and S.F.C. is reduced with increasing efficiency level n_o. This is a consequence of the fact that for the ideal cycle (without losses) the thermal efficiency (Brayton efficiency) is independent of the turbine entry temperature.

From Table 2 we may conclude that worthwhile reductions of S.F.C. can be obtained by improvements of the thermodynamic cycle by advancements in engine technology. These advancements are applied in the third generation of turbofan engines for the eighties. These engines have about the same BPR’s as the second generation engines, but much reduced S.F.C. values in cruise at altitude (c_T = 0.55 - 0.57 kg/daN-h). Some of these engines are derivatives of earlier types (Rolls Royce RB-211-524/535 series, GE/SNECMA CFM 56-80) and others are new designs (Pratt and Whitney PW-2037 and PW-4000, International Aircraft Engines IAE V-2500). These engines will power the new transport types Airbus 320/330/340, McDonnell-Douglas MD-11 and improved versions of aircraft types already in use today.

At a given technology level of the thermodynamic cycle parameters another important possibility for the improvement of S.F.C. is the increase of the cold mass flow accelerated by the propulsion system (higher BPR with lower specific thrust and higher propulsive efficiency). Gas turbine powerplants with a BPR beyond the current value of $\lambda \approx 5$ are called Ultra-High By-pass Ratio (UHBR) engines. Fig. 10 shows schematically various UHBR-types proposed by the engine manufacturers for future civil aircraft. Two different main classes can be distinguished:

- free propellers/unducted rotors,
- ducted fans/ducted rotors.

The high-speed free propellers are pioneered by Hamilton-Standard in the so-called propfan configuration. In comparison with propellers for low speeds,
the diameter of the propfan is reduced to limit the tip speeds and swept-back thin blades are used to reduce compressibility losses. To absorb the large shaft power required in high-speed flight (e.g. 13,500 kW, 18,000 hp in take-off for a 150 passenger aircraft), the propfan has many blades (8-12) and a gearbox is required to reduce the high rpm of the gas turbine to the propeller-rpm. Contrarotating propfans (blade numbers 6+6 to 12+12) offer the advantage of a high power absorption capacity and a reduction of the swirl- or rotation losses in comparison with the single rotating propfan.

To avoid the complex and heavy gearbox for rpm reduction, General Electric has pioneered the Un-Ducted Fan (UDF-) concept. The compact contrarotating propellers are aft-mounted on the engine and directly connected to contra-rotating free power turbines, operating at a relatively low rpm. The BPR is in the order of 50-60.

Most engine manufacturers are also working on ducted-fan concepts with high BPR's, calling them by names as Contra Fan-, Super Fan- or Advance Duct Prop-engines (BPR up to 20).

The several propulsion concepts mentioned above are in different stages of development, ranging from design-studies to actual prototype flight testing. Some of the fundamental characteristics are shown in Fig. 11. For a typical flight condition and gasgenerator figures, the effect of specific thrust on by-pass ratio, fan pressure ratio, propulsive efficiency and SFC is shown. This figure shows clearly the potential for S.F.C.-reduction for future powerplants by increasing of the By-Pass Ratio.

For UHBR-configurations the installation effects of the propulsion system on the thrust becomes very important (nacelle or cowl-drag, interference drag for free propellers, etc.). In a fair comparison the S.F.C. has to be based on the thrust delivered by the installed powerplant. Note the large difference between the uninstallled S.F.C. and the values for the installed case in Fig. 11.

The choice of one of new UHBR-propulsion concepts for future civil aircraft will not be dictated by fuel economy only, but many other aspects have to be taken into account by the aircraft designer. We mention here some of them:
- costs and time of development to the operational status,
- fuel costs, as part of the total operating costs of the aircraft, depending on the fuel price level, which future trend is hard to predict, as we know from the last 15 years,
- effect on aircraft configuration (on-wing installation possible or only fuselage aft-mounted),
- noise in the passenger cabin environment and noise effects on structural acoustic fatigue,
- containment requirements in case of blade failures of propellers or rotors,
- acceptance by the airline companies, taking into account opinions of the travelling public, expectations about safety, reliability, maintainability, etc.

The continuing process of interaction between engine manufacturers, airplane designers and airlines will lead to the best choice for the propulsion system in the years ahead. The choices to be made during this process might be not the same for all transport types, depending on size, speed and range.

Recently the Boeing Company has announced the choice of the General Electric GE 36 Unducted Fan as powerplant for its new short-medium range aircraft project 7-J-7 with 150 passengers. For this propulsion system a cruise S.F.C. of about 0.45 kg/daN-h has been quoted (cruise at M = 0.8 at 11 km, 36,000 ft). A competing engine for this aircraft, the Super Fan with a BRP of 17.5, based on the IAE V 2500 core, was announced by the International Aero Engines consortium (IAE) at the end of 1986, also intended for use in the Airbus 340. Although the development of the Super Fan has now been postponed, its concept can be considered as a typical example of a future UHBR-ducted fan engine (Fig. 12). Considering the future, we always have to keep in mind that new propulsion systems take a long time for development, in the order of 10 years or more (Fig. 13). Hence, new engine concepts, which are not beyond the study phase yet, can only be expected to enter airline service in the second half of the nineties or later.

5. Propulsion at supersonic speeds

Jet propulsion has made supersonic flight a reality and operation in this speed region is now common for military aircraft up to a Mach number of 2 and
in some cases even higher (up to $M = 3$). In the sixties high expectations existed for supersonic civil aircraft, culminating into the successful development of the British-French Concorde, which has now been used in every-day operations for more than 10 years. The American SST as counterpart was abandoned in the early days of conception, mainly due to environmental resistance and economic prospects. Since then the rising fuel prices have hampered any further development and today world-wide civil transport at supersonic speeds seems very remote.

Fundamentally, at supersonic speeds the lift/drag ratio $L/D$ is much lower than at subsonic speeds due to the shockwave drag. One may expect $L/D$-values of 9-10 in the future (Concorde $L/D = 7.4$ at $M = 2$), compared to 15-16 for subsonic types. Any break-through of this aerodynamic barrier seems unlikely. On the other hand the propulsion system offers possibilities to compensate, at least partially, the unfavourable aerodynamic characteristics.

To illustrate some typical features of supersonic propulsion, we will consider firstly the pure jet engine, illustrated by some figures of the Olympus 593 MK610 engine of the Concorde, designed for $M = 2$ in cruising flight (Ref. 15):

- The supersonic airflow is decelerated in the intake to a low subsonic Mach number at the compressor entry. This deceleration gives a considerable rise in the overall cycle pressure ratio. For isentropic pressure recovery the corresponding pressure ratio at $M = 2$ is $\xi_R = 7.82$. The actual deceleration process includes losses due to shockwaves, and in case of the carefully designed Concorde-inlet 94% of the isentropic pressure ratio is recovered. Combined with the engine compressor ratio $\xi_C = 12.07$ this gives a high cycle pressure ratio in cruise: $\xi = 88.5$. The corresponding Brayton-efficiency is $\eta_{Br} = 0.72$. The actual value of the thermal efficiency is $\eta_{th} = 0.58$ (TET = 1350 K). This is considerably above the value of the subsonic turbofan engines of the same state of the art (Table 1).

- The jet velocity of a pure jet engine is highly dependent on the turbine-entry temperature and roughly independent of the flight speed. According to eq. (5a) the propulsive efficiency $\eta_p$ will rise with increasing flight Mach number. For the Concorde with $M = 2$ at 16 km altitude, the corresponding flight speed is: $V = 597$ m/s ($= 2150$ km/h, temperature: ISA + 5 K). With the specific thrust $T_s = 41$ sec or a jet velocity of $w = 1000$ m/s, we find a Froude efficiency: $\eta_{Fr} = 0.75$. 
- Taking into account a 4% loss in the propelling nozzle system, the overall propulsion efficiency of the Concorde powerplant is: \( \eta_{tot} = 0.96 \times 0.58 \times 0.75 = 0.42 \). This is about 1/3 higher than the overall efficiency of the turbofan engine at high-subsonic speed in Table 1. The corresponding SFC for the Concorde Olympus engine is: \( C_T = 1.19 \text{ kg/daN-h} \) at \( M = 2.0 \) cruise. Note that due to the effect of speed the overall propulsion efficiency \( \eta_{tot} \) is a better yardstick for the fuel economy than SFC-values.

- For the so-called range parameter \( \eta_{tot} \frac{L}{D} \) in eq. (9), we obtain for the Concorde a value of 3.1 in supersonic flight. This can be compared with the value of subsonic turbofan transports, for which \( \eta_{tot} \frac{L}{D} = 4.5 - 5.5 \) (section 2).

- At supersonic speeds the propulsion system performance is critically dependent on the design of the engine intake and the exhaust nozzle system. This is well illustrated by the figures in Table 3. To avoid the thrust losses at supersonic speeds, it is necessary to expand the gases in the exhaust nozzle to ambient pressure. This requires the use of a convergent-divergent nozzle instead of the conventional convergent nozzle for subsonic aircraft.

- The acceleration through the transonic speed regime requires a large thrust (low \( L/D \)), which for the Concorde is obtained by use of an afterburner.

- For the design of the propulsion system of supersonic airliners the take-off engine noise and the fuel economy at high subsonic speeds are of critical importance. The last point follows from the requirement to avoid the sonic boom problem in overland flights.

For a future generation of supersonic airplanes other cycles than the pure jet engine may be applied. The by-pass engine offers some advantages, especially for reduction of the take-off noise and for improvement of the subsonic fuel-economy. The by-pass engine has the disadvantage of increased engine weight and larger installation drag (especially at supersonic speeds), which prevents the application of the high values of BPR of subsonic aircraft. (Modern military jet engines are designed for BPR up to 1-1.5, Ref. 7e). Also several advanced concepts have been studied for supersonic airliners, known as variable cycle engines, which could improve the adaptation of the engine to the conflicting requirements of low- and high speed flight. Various schemes have been suggested (Fig. 14) with:
- varying BPR over the speed range by special flow ducting to obtain a relatively high BPR for subsonic operation and a low value for supersonic flight.
- additional fuel combustion in the by-pass flow (duct burning).

Some of these schemes make use of exhaust nozzles with a so-called "inverted velocity profile" with a thin high-velocity jet coaxially surrounding a low-velocity jet, which reduces the take-off noise level in comparison with conventional nozzle configurations (Ref. 18).

Studies have also shown that the lift-drag ratio of future SST could be increased considerably beyond the state-of-the-art of the Concorde by applying advancements in the aerodynamic design.

Table 4 compares some data of an advanced SST with the data for the Concorde, showing an improvement of the range parameter $\eta_{\text{tot}}$ L/D of nearly 50%. In this example the advanced SST is designed for a Mach number of 2.7, which is usually considered in US-studies. An SST of this type could cover the distance Los Angeles-Tokio (10,000 km) in about 4 hrs as compared to 12 hrs for subsonic aircraft. The increasing economic importance of the Pacific Region could make the development of an advanced SST a viable undertaking in the coming decennia. However, until now no development program has been started and we cannot expect a new SST-airliner before the 21th century.

6. Propulsion at hypersonic speeds

Some in the industry believe even that the SST will be leap-frogged by the transatmospheric (high altitude) transport with hypersonic speeds (up to Mach 8 or even higher). These vehicles for transport between points on Earth could be derived from advanced launch vehicles for spacecraft, which use air-breathing propulsion systems (X-30 National Aero-Space Plane and Orient Express in the U.S., HOTOL and Saenger/Horus concepts in Western Europe).

The hypersonic speed region is defined as the region above flight Mach numbers of 4 to 5, where aerothermal problems dominate the vehicle and engine design, because of the high stagnation temperatures (900 to 1300 K in stratospheric flight).

Considering this flight regime, firstly it should be pointed out that the turbojet and turbofan engines are limited to a flight Mach number of about 3.5. This follows from a simple analysis of the temperatures in the engine
cycle\(^*)\). In the intake the air flow has to be slowed down to a low subsonic Mach number at the compressor entry face. In stratospheric flight (\(T = 217\) K) at \(M = 4\) the stagnation or total temperature at the compressor entry face is about 910 K. With a compressor pressure ratio of \(\varepsilon_c = 8\) and a polytropic compressor efficiency of 0.85, the temperature of the compressed air at the entry of the combustion chamber becomes 1850 K. This is in the range of the maximum allowable turbine inlet temperatures, so no fuel can be burnt without exceeding this turbine temperature. This implies that no thrust will be obtained at \(M = 4\). We note that afterburning will not change this limit, because it is dictated by the turbine temperature limit in front of the afterburner.

For the ramjet with subsonic combustion (Fig. 15a) the turbine temperature does not limit the allowable temperature anymore and a stoichiometric mixture of hydrocarbon fuel can be used. Moreover the temperature rise at the entry of the combustion chamber is caused by the stagnation in the intake only (no compressor). Assuming a combustion temperature of 2350 K, we find easily that the total temperature by stagnation of the airflow in the intake equals this combustion temperature at \(M = 7\). Thus the subsonic combustion ramjet (with hydrocarbon fuel) can be operated up to the Mach range 6-7. However, as the compression for the combustion is through stagnation of the intake flow only, a lower limit for efficient operation exists also: \(M > 2.5-3\). To obtain thrust at lower Mach numbers, the ramjet has to be combined with a turbojet- or turbofan engine.

\(^*)\) The following data are based on the relationships:

- stagnation or total temperature of air in intake:
  \[ T_t = T(1 + \frac{Y-1}{2} M^2) \]; \(T =\) atmospheric temperature, \(M =\) flight Mach number,
  \(Y = \frac{C_p}{C_v} = 1.4\)
- total temperature at compressor outlet:
  \[ \frac{Y-1}{\eta_c Y} \]
  \[ T_{t_c} = T_t \varepsilon_c \eta_c \]; \(\varepsilon_c =\) compressor pressure ratio, \(\eta_c =\) polytropic efficiency.
For hypervelocity vehicles (M > 6-7) the ramjet with supersonic combustion is applicable, using liquid hydrogen as fuel (supersonic combustion ramjet or scramjet, Fig. 15b, Ref. 19). The hydrogen fuel is also a good coolant to limit the temperature of the airframe and/or engine structure, which become very high at hypersonic speeds. Because the engine flow is not decelerated to a subsonic Mach number, the working pressures are also considerably lower than for the conventional ramjet at these high speeds, which reduces the thermal and structural loads of the engine.

To summarize the high-speed propulsion scene, Fig. 16 illustrates the operational Mach numbers and cruise altitudes for the turbo-, ramjet- and scramjet-engines. Research and development of engines for the hypervelocity-regime as discussed here, will offer again new challenges in the propulsion field, which may lead to a new era in aerospace!

7. References


   e. Lewis, G.M.: The next European engine for combat aircraft.


| $M = 0.8; \ V = 850 \ km/h$ |
|------------------|------------------|------------------|------------------|------------------|------------------|
| $h = 10,670 \ m \ (35,000 \ ft)$ |
| $c$ | $\eta_{th}$ | $\psi$ | $C_F$ | $\eta_p$ | $\eta_{tot}$ | $c_T$ (kg/daN-h) |
| **ideal case, no losses** | $\approx$ | 1.0 | 0 | 0 | 1.0 | 1.0 | 0.20 |
| current turbofan engine | Brayton and Froude eff. | 30 | 0.62 | 15 | 0.62 | 0.76 | 0.47 | 0.42 |
| actual efficiencies | 30 | 0.43 | 15 | 0.62 | 0.74 | 0.32 | 0.62 |

Table 1: Example of efficiencies of ideal and current turbofan engine cycles (cruise condition).

<table>
<thead>
<tr>
<th>overall compr. ratio $c$</th>
<th>engine compr pressure ratio*</th>
<th>TET (K)</th>
<th>$\eta_o$</th>
<th>$\eta_{th}$</th>
<th>$c_T$ (kg/daN-h)</th>
<th>$c_T$ (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>datum-engine</strong></td>
<td>30</td>
<td>20</td>
<td>1400</td>
<td>0.88</td>
<td>0.43</td>
<td>0.62</td>
</tr>
<tr>
<td><strong>improved cycle parameters</strong></td>
<td>40</td>
<td>26</td>
<td>1400</td>
<td>0.88</td>
<td>0.46</td>
<td>0.58</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>1800</td>
<td>0.88</td>
<td>0.50</td>
<td>0.53</td>
</tr>
<tr>
<td></td>
<td>1400</td>
<td>0.92</td>
<td>0.53</td>
<td>0.50</td>
<td>80.5</td>
<td></td>
</tr>
<tr>
<td></td>
<td>1800</td>
<td>0.92</td>
<td>0.545</td>
<td>0.49</td>
<td>79.0</td>
<td></td>
</tr>
</tbody>
</table>

*) The overall compression ratio includes the intake pressure recovery.

Table 2: Effect of cycle parameters on fuel economy in cruise ($M = 0.80, h = 11 \ km$, derived from Ref. 8).
<table>
<thead>
<tr>
<th>intake pressure recovery (%)</th>
<th>engine component eff. level (%)</th>
<th>nozzle efficiency (%)</th>
<th>thrust (%)</th>
<th>SFC (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>100</td>
<td>100</td>
<td>100</td>
<td>100</td>
</tr>
<tr>
<td>92</td>
<td>100</td>
<td>100</td>
<td>87</td>
<td>105</td>
</tr>
<tr>
<td>92</td>
<td>90</td>
<td>100</td>
<td>71</td>
<td>116</td>
</tr>
<tr>
<td>92</td>
<td>90</td>
<td>98.5</td>
<td>68</td>
<td>121</td>
</tr>
</tbody>
</table>

Table 3: Effect of intake and exhaust nozzle efficiencies on thrust and SFC (M = 2.2 in stratosphere, fixed engine compressor pressure ratio and turbine inlet temperature; Ref. 14).

<table>
<thead>
<tr>
<th></th>
<th>Concorde (M=2)</th>
<th>Future SST (M=2.7)</th>
</tr>
</thead>
<tbody>
<tr>
<td>By-Pass Ratio</td>
<td>0</td>
<td>1.0</td>
</tr>
<tr>
<td>Turbine Entry Temperature</td>
<td>1350 K</td>
<td>1650 K</td>
</tr>
<tr>
<td>Specific Fuel Consumption</td>
<td>1.19 kg/daN-h</td>
<td>1.36 kg/daN-h</td>
</tr>
<tr>
<td>Overall propulsion</td>
<td></td>
<td></td>
</tr>
<tr>
<td>efficiency $\eta_{tot}$</td>
<td>0.42</td>
<td>0.49</td>
</tr>
<tr>
<td>Cruise L/D Ratio</td>
<td>7.4</td>
<td>9.5</td>
</tr>
<tr>
<td>Cruise parameter $\eta_{totD}$</td>
<td>3.1</td>
<td>4.66</td>
</tr>
</tbody>
</table>

Table 4: Potential improvement of the SST.
Fig. 1: Airbreathing propulsion systems.
Fig. 2: Conversion of fuel energy into thrust.

\[ T = m(w-V) \]

**a: propeller**

Fig. 3: Thrust of propulsion systems.

\[ T = m(w-V) + m_f w \equiv m(w-V) \]
ideal thermal efficiency: \( \eta_{th} = 1 - \frac{Q_2}{Q_1} = 1 - \frac{1}{\frac{p_2}{p_1}} \frac{1}{\epsilon y} \)

\[ \epsilon_v = \frac{v_1}{v_2} \]

\( \epsilon = \frac{p_2}{p_1} = (\epsilon_v)^Y \)

--- spec. volume \( v \)

1 \( \rightarrow \) 2 compression
2 \( \rightarrow \) 3 combustion
3 \( \rightarrow \) 4 expansion
4 \( \rightarrow \) 1 exhaust

(a) Otto-cycle
(heat flows at constant volume)

(b) Brayton-cycle
(heat flows at constant pressure)

Fig. 4: Ideal cycles for piston engine and gasturbine.

Fig. 5: Napier Nomad compound diesel engine (schematic, Ref. 6).
Fig. 6: Example of turbofan engine (Rolls Royce RB 211, three spool engine).
Fig. 7: Effect of B.P.R. on specific fuel consumption.
(1970 Technology)

Fig. 8: Effect of B.P.R. on noise reduction.
Fig. 9: Thermal efficiency of gasgenerator (derived from Ref. 10a).

Fig. 10: Ultra High By-pass Ratio Engines (Ref. 12c).
Fig. 11: Effect of specific thrust on engine characteristics (Ref. 12b).
Fig. 12: Super Fan concept.

Fig. 13: Development time for new technology aero-engines (Ref. 18).
a: Variable stream control engine (VSCE).

TWIN TURBOJET MODE

duct burner on
inverter valve

variable nozzle

TURBO MODE

duct burner off

variable nozzle

b: Rear valve engine (RVE).

Fig. 14: Variable cycle engines (Ref. 17).
Fig. 15: Schematic diagrams of two types of ramjet.

Fig. 16: Propulsion spectrum (Ref. 20)