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AIRCRAFT PROPULSION

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Humans have always dreamed with flying, from Icarus' wax-glued feathered-wings myth, to Da Vinci's flying machines, but, leaving aside lighter-than-air balloons, only in the early 20th century enough specific propulsion power was available to sustain human flight, with the compact petrol engine of 80 kg of Flyer-I, supplying 9 kW to a twin propeller. you may compare this engine power-to-mass ratio, $\dot{W}/m$ =9000/80≈100 W/kg, with a typical man power of (150 W)/(75 kg)=2 W/kg, or with modern airliner engines of (25 MW)/(6400 kg)≈4000 W/kg.

But the foundations of powered flight can be set in 1799 when Sir George Cayley, the father of Aerodynamics, first identified the four basic forces of flying: weight, lift, drag, and thrust, and set forth the concept of the modern aeroplane as a fixed-wing flying machine (no flapping wings like in bird flight), with separate systems for lift (tilted planes), propulsion (engine), and controls, designing the first successful glider to carry a human being aloft. Sustained aerodynamic flight requires almost-permanent propulsion, as opposed to other transportation means (land vehicles, aerostats, surface vessels, submarines, or spacecraft), which can live for some time without propulsion. We say in Spanish: "Le dijo el ala al motor: tu empuja, que yo te subo" (y el motor respondió: "por ahorrar combustible, que si no, no me haces falta").

Taking advantage of moving within a fluid, aircraft propulsion is achieved by air-breathing engines, i.e. engines that take a stream of air and throw it at higher speed backwards. The energy source is the combustion of a fuel (carried onboard) with oxygen in the air, but it might also be solar power or nuclear power. The standard in aircraft propulsion is the jet engine, basically consisting on a gas turbine delivering most of its work through a shaft that drives either a few-large-blade propeller or a many-small-blade ducted fan. Even for the same type of engine (e.g. a gas turbine), different notations are used in specific propulsion fields, like aviation, than on general power plants or in basic thermodynamic studies, further modified by different traditions and language.

Modern aircraft engines routinely stayed in service for 20 or 25 years, often without experiencing a problem significant enough to warrant removal from aircraft. They are extremely reliable; engine go-out rate is less than 2 times per 100 000 hours of operation (engine life). Pilots could go through an entire career without a single engine emergency. More than 50 % of the reduction in energy intensity of airliners (from 6 MJ/pkm in 1950 to 1.5 MJ/pkm in 2000) has been due to improvements in jet engines (some other 25 % to increase in aerodynamic efficiency and the rest to more efficient aircraft use). The tendency has been towards twin-engine aircraft (with engine size matching aircraft size), in spite of the oversizing implied by the requirement of safe take-off with one engine out (maximum engine thrust is dictated by meeting the climb requirements with one engine out, where available thrust reduces by more than 50 % because of the extra drag associated with the failed engine and the need to trim with asymmetric thrust).

Aircraft propulsion is very effective because a small power plant is able to yield a large thrust, i.e. $F/(m_{eng}g)>>1$ (e.g. for RR Trent 900 engine, $F_{max}/(m_{eng}g)=350\cdot 10^3/(6600\cdot 9.8)=5.4$), and the aerodynamic gain is large, i.e. $L/D>>1$ (e.g. for A380, $L/D|_{max}=W/F=17$), providing large lifting force (e.g. the four Trent 900 in a A380 support 560 t of aircraft weight with just 4·6600=25 t of engine weight). Large airliners weight share is as follows: some 35..40 % fuel, 30..35 % equipment, 15 % structure, 10 % passengers and luggage (payload), and 5 % for all the engines. The loaded-fuel share is: 2 % take-off, 10 % ascent, 75 % cruise, 2 % descent, 1 % landing, plus 10 % reserve.

Supersonic propulsion ($1<M<5$) is much harder because $L/D|_{max}\approx 4$ instead of $L/D|_{max}\approx 20$ (sailplanes may reach $L/D=50$), and consequently $W_{eng}/W_{craft}\approx 0.2$ instead of 0.05 as in airliners. Hypersonic propulsion ($M>5$) is still a research topic (the only operative hypersonic aircraft-like vehicle was the Space Shuttle, gliding down without propulsion (from $M\approx 25$ to $M=0$). Hypersonic flow is not delimited by a precise Mach number, but by the appearance of new phenomena like gas dissociation, and curved strong shocks, which depend on geometry too, although the landmark $M=5$ is often assumed. It seems that hypersonic aircraft design would be based on the engines, with integrated payload-volume and lifting-surface, instead of the current aircraft, in which the three components (engines, cabin, and wings) can be clearly distinguished and designed almost separately.

It may be convenient to recall the aircraft thermal environment before propulsion is dealt with in detail.

SOME FREQUENTLY ASKED QUESTIONS

Q. Why jet engines are preferred for aircraft, instead of reciprocating engines like in cars?

A. Because we want to fly at high speed, say >100 m/s (>500 km/h). In fact, to travel at <100 m/s (<350 km/h) aircraft with reciprocating engines are used (or high-speed trains in populated areas).

Q. Why airplanes fly so fast? Or better, why they cannot fly slowly (e.g. for take-off and landing)?

A. Because aircraft weight is supported by aerodynamic forces proportional to v^2. Explanation: for given $\{W,A,\rho\}$, from $W=\frac{1}{2}\rho v^2 c_L A$ we deduce that $c_L v^2$=constant, and as c_L is small and upper-bounded (e.g. $c_L=2\pi\alpha$, with $c_L<1$ to avoid flow detachment), flight speed v must be large; e.g. for a wing load of W/A=5 kPa (→500 kg/m^2), minimum flight speed is around $v=\sqrt{2(W/A)/(\rho c_L)}=\sqrt{2\cdot 5000/(1\cdot 1)}$=100 m/s (airliners take off at about 70 m/s). Associated to the need of fast motion is the need of powerful engines for aircraft. In fact, aircraft might have sustained flight without wings, like rockets, but not without a propulsion plant.

Q. Why we talk about thrust instead of power, in jet engines?

A. Because jet engines are only used for propulsion, and what matters in propulsion is propulsive power, Fv_0, and the advance speed is understood. But when the engine is separately built from the propeller (as in reciprocating engines, turboshafts, and so on), the engine performances are given separately, as shaft power and shaft speed (or a combination, as shaft torque). The jet engine directly pushes forward; with a shaft engine, however, it is the propeller that pushes. Notice that the interest is in propulsive power, Fv, and not just on thrust, but jet engines yield a thrust nearly independent on flight speed (for the same ambient conditions), and hence it is simpler to talk about thrust than about mechanical power produced by the engine: $Fv_0+\frac{1}{2}\dot{m}_a\left(v_e-v_0\right)^2=\eta_t\dot{m}_f h_{LHV}$.

Q. Why take-off thrust must be nearly four times larger than cruise thrust? (e.g. A380: F_{TO}=4×350 kN, F_{cruise}=4×90 kN).

A. Because we not only need thrust (F) to compensate drag (D) and create lift (L) to compensate aircraft weight (W), but to accelerate and compensate aircraft-weight-projection on climbing (the latter without the 15× aerodynamic efficiency); i.e. under horizontal steady flight, $F=D$ and $L=W$, but on climbing, $F=D+W\sin\theta+ma$ and $L=W\cos\theta$, but as $L/D\sim 15$, $F=D+L\tan\theta+L(a/g)=D[1+(L/D)(\tan\theta+a/g]\approx D[1+15\times(0.1+0.1)]=4D$.

Q. Why there are much more blade-discs in a jet-engine compressor than in its turbine? Even more when realising that in turbofans the turbine must provide more power than the compressor takes.

A. Because the axial pressure gradient is favourable in the turbine ($\nabla p<0$ along the flow) and blades can be heavily loaded aerodynamically without boundary-layer detachment, whereas the contrary happens during compression: $\nabla p>0$ and air pressure ratio in a single step (rotating plus steady blade-discs) cannot by greater than p_2/p_1=1.2 (whereas expansion ratios of p_2/p_1=1/5 are achieved in a single turbine step). Compressor blades with air suction have been tried to control boundary layer separation, but without much success.

Q. What is the difference between a turbine blade and a compressor blade?

A. There are many, the most apparent being its camber (Fig. 1): turbine blades have large camber (so that the flow can be deflected a lot to extract much work), whereas compressor blades have little camber because otherwise the boundary layer would separate. Besides their shape, different alloys are used to make them: turbine blades require special nickel alloys (directionally solidified or mono-crystalline) resistant to very high temperatures (above 1500 K), and even special thermal coatings and a set of holes for cold-air cooling may be apparent, whereas compressor blades have no such requirements. Turbine-blade requirements are so high that, in the common case of separate high-pressure turbine and low-pressure turbine, their blades are significantly different in material and cooling (but similar in shape).

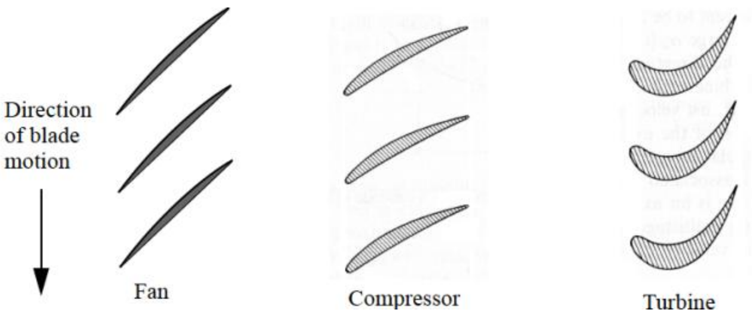


Fig. 1. Blade shapes in a turbofan engine: fan blades, compressor blades, and turbine blades.

Q. Why aircraft fly so high (they must climb a lot to later descend)?

A. Because fuel consumption per km decreases with altitude, as explained below. Their ceiling is limited by flight speed, because higher means faster, and turbofans are limited to M<0.9 and turboprops to M<0.5. The explanation is given by Breguet's equation:

$$R=-\frac{v_0}{g\dfrac{\dot{m}_\mathrm{f}}{F}}\frac{L}{D}\ln\frac{m_\mathrm{TO}-m_\mathrm{final}}{m_\mathrm{TO}}=-\frac{v_0}{\dot{m}_\mathrm{f}}m_\mathrm{mean}\ln\frac{m_\mathrm{TO}-m_\mathrm{final}}{m_\mathrm{TO}}=K\frac{v_0}{\dot{m}_\mathrm{f}}$$

showing that for given initial and final mass $\{m_\mathrm{TO}, m_\mathrm{final}\}$, maximum range R_max requires $v_0/\dot{m}_f=\max$, i.e. large speed and small fuel-consumption. But speed is limited by transonic effects, and minimum fuel-consumption implies minimum thrust, since overall propulsion efficiency is nearly invariant with flight speed, what finally arrives to minimising ambient air density; i.e.:

$$\eta_\mathrm{tp}=\frac{Fv_0}{\dot{m}_\mathrm{f}h_\mathrm{LHV}}\quad\rightarrow\quad\dot{m}_\mathrm{f}=\frac{Fv_0}{\eta_\mathrm{tp}h_\mathrm{LHV}}=\frac{\rho}{\rho_0}\frac{F_0v_0}{\eta_\mathrm{tp}h_\mathrm{LHV}}=K'\frac{\rho}{\rho_0}$$

Q. How safe is flying?

A. Flying in an airliner is the safest mean per distance travelled (the third in terms of travel time, after bus and rail, but behind most other travel modes by number of journeys).

AERO-ENGINE TYPES

An engine is a machine designed to convert energy into useful mechanical motion. Most devices used in the industrial revolution were referred to as engines, and this is where the steam engine gained its name.

Engines may be used for several purposes, from lifting water from a pit (the initial application of steam engines), to propulsion for transporting people and goods, which is our present focus. Once the need identified (engines for propulsion), the engineer should know the state of the art: systems available, principles and functionalities (degrees of freedom, ranges), performances (static and dynamic response), and their limits (it is very important to know the limits, because, once in operation, systems are invariably pushed to their limits).

The present practical engines for aerospace propulsion (manned and unmanned vehicles included) can be classified as:

- Shaft engines, usually driven by reciprocating engines or by turbine engines (e.g. turboprops), but any other motor may work (electric motors powered by solar cells or fluid cells, or even mechanical means like rubber bands or human muscles). The power plants is basically independent of the propeller; the engine applies a torque, Q, to a shaft at a certain rotation speed, ω, producing a shaft power $\dot{W}_{shaft}=Q\omega$. The shaft is mechanically coupled to the air-propeller itself (in the same way as for water-propellers in ships, or friction wheels in land vehicles). The propeller is usually ahead of the engine, and propulsion is achieved by traction (i.e. the shaft transmits torsion and tensile stresses).
- Jet engines, where part or all of thrust is due to high-speed exhaust of gases from inside the engine. They may be air-independent jet-engines (better called rockets, the only engine able to operate in outer space, if solar wind sailing is not considered), or air-breathing engines. Propulsion may be achieved:
    - Exclusively by reaction, as in rockets, ramjets, scramjets, pulsejets and turbojets.
    - By combination of shaft and jet, as in turbofans and propfans. Pure air-breathing jet-engines (turbojets) are obsolete because of its poor performances; nowadays, all jet engines in aviation are combined shaft-and-jet engines (apart of simple shaft engines).

Reciprocating engines

The first aero-engine

The first aircraft engine (used in Wright Flyer-I, 1903-12-17, with four flights on the same day) had to be developed on purpose by the Wright brothers (they could not find a "little gas motor" in the market); they started its development in Dec-1902, and the first prototype broke in the tests. It was a horizontal 4-cylinders in-line 4-stroke cycle piston engine of 80 kg, with water-cooled aluminium-alloy frame, burning carburated gasoline within 0.1 m bore 0.1 m stroke pistons (3.2 L total displacement), with a compression ratio of r=4.4, delivering 9 kW at 1000 rpm to twin two-blade propellers of 2.4 m in diameter (also of Wright's invention), chain-driven and counter-rotating at the rear of the wings, with 3 m between their axes. The engine was to the right of the pilot (the right wing was 10 cm longer to compensate for the extra engine weight over the pilot's weight, who was to the left), who had command of the fuel valve (the only engine control); after opening the valve (intended to operate fully open or fully closed), two assistants pulled the two propellers through, ignition was provided by a low-tension magneto (driven by friction from the flywheel), and the engine got its regime. The fuel tank, of 6 L capacity, was on a front strut and had about 1.5 litres of gasoline (from Standard Oil, with an estimated octane number of 38), feeding the downdraught carburettor by gravity. Also on the front strut near the pilot was a water tank piped down (by gravity) to the engine casing, to provide some evaporative cooling. The entire power plant including the engine, magneto, radiator, tank, water, fuel, tubing, and accessories weighed a little more than 90 kg, being able to lift some 350 kg while advancing at 48 km/h (the aircraft was 280 kg without the pilot).

Flyer-I had no wheels but a sledge frame (Fig. 2) optimised for landing on the soft beach sand chosen for the trials, and consisted of two long wooden skids (extending well in front of the wings to prevent rolling over on landing) separated 1.5 m, and with several strengthening crosspieces. For take-off, a 18 m long by 0.2 m wide wooden rail was built using planks of 4.5×0.1 m^2, with a thin metallic top. Over this rail ran a small truck made of two rollers with guiding flanges (made from modified bicycle hubs) with their axles connected to a transversal plank of 1.8 m that supported the whole aircraft (by the skids) some 0.2 m off the ground. For take-off, the machine was lifted onto the truck, with the skids resting on the truck's crossbeam (another bicycle hub on the forward crosspiece between the skids provided a nose support). A wire from the truck attached to the end of the starting rail-track held the plane under tension while the engine was warmed up, until released by the pilot. The propeller's thrust accelerated the plane along the rail and, added to the 10 m/s head wind, made it airborne after running 12 m along the rail, and hence the plane lifted off the truck, which continued rolling on and ran off the rail.

Flyer-I was the first aircraft with roll control (previous flyers only thought of pitch and yaw controls); the propellers and the engine were also new designs by the Wright brothers. The use of wheels for take-off and landing is due to Alberto Santos Dumont, who on 23 October 1906 flew his Oiseau de proie in Paris, powered by a 37 kW Antoinette engine (eight-cylinder in V, liquid-cooled gasoline engine), the first heavier-than-air flight to be certified in public (by the FAI, Fédération Aéronautique Internationale, founded in 1905). His Demoiselle aircraft was a monoplane with tail (and a wheel-tail), with a pilot's seat below the wing and between the main wheels of a tricycle undercarriage. Meanwhile, the Wright brothers used catapults to improve their rail-based take-offs.

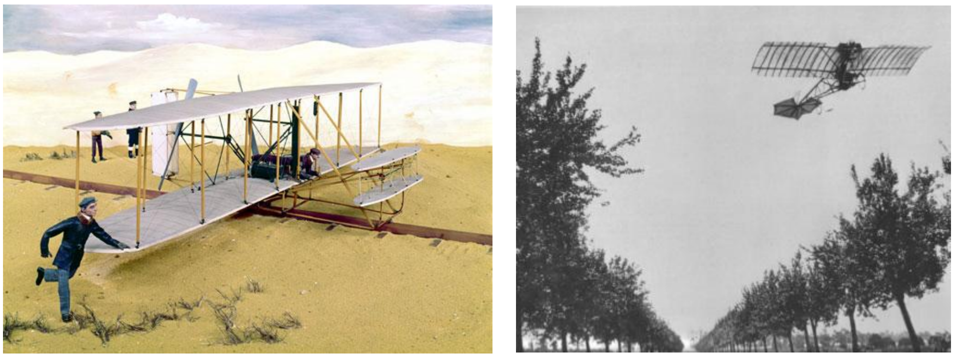


Fig. 2. a) Artist impression of Flyer-I flight. b) Dumont's Demoiselle flight.

Reciprocating gasoline engines have been used to propel aircraft (by driving a propeller) from the first powered flight in 1903 to the 1950s and beyond, and they are still used on small aircraft, initially in 2-stroke cycles, and later in 4-strokes. These engines first used the same gasoline fuel used in cars, although Standard Oil developed the first aviation gasoline in 1918; lead tetraethyl was added to aviation gasoline since 1921, and the standard avgas was introduced in 1934.

The star engine, having 5, 7, or 9 coplanar cylinders radially distributed, were 4 stroke air-cooled gasoline engines used in the 1920s and 1930s, but their advantage of higher specific power relative to in-line water-cooled engines, was counteracted by higher frontal area drag.

Piston engines, with power from 70 kW to 200 kW, are used in small aircraft; e.g. the UL260i is a 71 kW air-cooled piston engine, with 4-cylinder, 4-stroke, horizontally opposed (H-configuration), with 2.6 L displacement, compression ratio of r=8.2, and a dry mass of m=75 kg; it may run on 95-octane motor gasoline or on avgas, with a fuel consumption of 12 L/h at typical cruise of 2500 rpm (0.25 kg/kWh). Power in piston aero-engines has a practical limit of about 500 kW.

Diesel engines have a key advantage over Otto engines: smaller fuel consumption. They were tried on aircraft in the 1920s and 1930s, both on in-line and in star configurations, but the power-to-weight ratio was poor. The zeppelins LZ 129 Hindenburg and LZ 130 Graf Zeppelin II were propelled by diesel engines. The only successful diesel engine was the Junkers's Jumo 205, a 2-stroke 6-cylinders 12-pistons engine (two opposed piston per cylinder with two crankshafts), developing 450 kW.

Since 2002, new diesel engines, using Jet A-1 fuel are replacing some Otto engines run on avgas (which are more expensive to run, and more contaminant; avgas still uses lead-tetraethyl additive). The first certified modern diesel aero-engine was a 1.7 litres, 101 kW, four-cylinder (based on the 1.7 turbo diesel Mercedes A-class power unit), used for retrofitting Cessna 172s and Piper Cherokees. Ecomotors have a diesel aero-engine of 240 kW (at 3500 rpm) with opposite-pistons opposite cylinders ('opoc' architecture, where each cylinder has two pistons moving in opposite directions), bore of 0.1 m, and specific power of 240/150=1.7 kW/kg. Diesel engines have many advantages: they are more fuel-efficient than gasoline engines and turbines, run on Jet A-1 (and biodiesel), and consequently they drastically reduce CO_2 emissions (up to 40 % in the whole flight envelope of helicopters and small planes); the disadvantages to be solved are: larger weight, and more complex cooling.

The reduction gear is simpler in reciprocating engines (from 3000..6000 rpm engine-shaft, to 1500 rpm propeller-shaft) than in other shaft engines (in a turboshaft the reduction may be from 20 000..30 000 rpm to 1500 rpm, if a single turbine spool is used). Since the 1960s, however, jet engines (first turbojets, and later turboshafts and turbofans) are used in most aircraft with more than four passengers; the first commercial jet aircraft, the de Haviland Comet, introduced in 1952, was powered by two pairs of Ghost-50 turbojet engines, with a mass of 910 kg and a thrust of 23 kN each, spinning at 10 000 rpm.

Jet engines: turbojet, turbofan, propfan, turboprop, and ramjets

A jet engine is a propulsion device producing thrust solely or in part by a high-speed gas-exhaust from inside the engine. Although rocket engines are air-independent jet engines, the term 'jet engine' is usually restricted to air-breathing jet engines, and in particular those with mechanical compression (i.e. to aircraft gas turbines).

Jet engines are better suited to air flight because they are powerful and small. There are almost 100 000 jet engines in service around the world powering every sort of civil aircraft from Jumbo jets to helicopters. The first engine, the Henkel HeS 3B of 1939, had a thrust of F_{max}=4 kN (nowadays they have up to 400 kN).

We said that jet engines may be air-independent (rockets), or air-breathing; we leave rockets for later analysis. Air-breathing engines work on the Brayton cycle, similar to ground gas-turbine engines, with the three basic processes of air compression, heating, and expansion. According to air-compression mode, two kinds of jet engines may be distinguished:

- Jet engines with dynamical compression: ramjets, scramjets, and pulsejets. They are based on the open Brayton cycle like turbojets, but can only work at very high speeds, and are still at the research stage (although some ramjet missiles have been in service since the 1950s). A ramjet decelerates incoming air to get subsonic flow in the combustion chamber (e.g. from M_0=3 to M_{ci}=0.3), whereas a scramjet decelerates the flow but still to supersonic speed (e.g. from M_0=7 to M_{ci}=3). The engine (Fig. 3) is typically a rectangular duct, with a 2D supersonic intake, a diffuser where oblique shocks compress and slows the airflow, fuel injectors, a mixing zone, a combustion zone, and a divergent nozzle. For subsonic combustion. They work with low pressure ratios (the higher the Mach number M the lower the pressure ratio π) and very high maximum temperature. Hydrogen is the only fuel being considered for scramjets because of its high heating power, its fast diffusion and chemical kinetics (notice that the residence time is about 10^{-3} s), and the large heat sink that LH2 provides. For one-time ramjets (missiles), a solid fuel can be used. For subsonic combustion, flame-holders provide the turbulent circulation necessary to stabilize the flame at some 100 m/s (laminar deflagration velocities are of the order of 1 m/s), but when flight Mach number M_0 increases (around M_0=7..8) a transition from subsonic to supersonic combustion takes place, i.e. Mach input to combustion chamber, M_{ci}, suddenly changes from about M_{ci}=0.5 to about M_{ci}=3 because the heat released is not enough to choke the flow. Notice that these engines cannot take-off and fly at low speeds (not enough compression), so that another kind of engine is needed for low speeds and acceleration (an air-breathing engine, or a rocket booster). Ramjets may be the best solution for flight at $2<M<6$ and H=20..30 km altitude, and scramjets for flight at $5<M<15$ (presumably at H=25..50 km altitudes, limited by highest dynamic pressure for structure, and lowest dynamic pressure for combustion, respectively), but for $M>15$ and $H>50$ km, rockets seem to be the only way out. For single stage to orbit (SSTO) flights, air-breathing engines may provide most of the energy requirements, since almost 90 % of the orbital energy is kinetic energy, using small rockets for the final orbiting phase (they ae needed anyway for orbit circularisation and manoeuvring); the last push of the scramjet (at about 50 km altitude), or the initial burning of the orbiting rockets, should have a large vertical-thrust component to quickly cross the 50..100 km altitude band where air drag is still relevant.

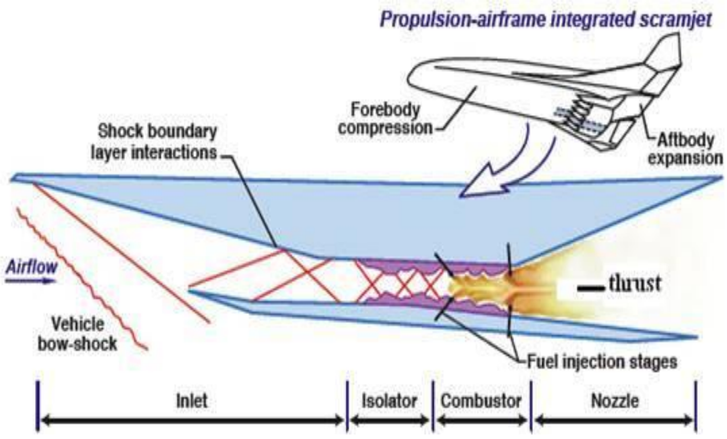


Fig. 3. Basic processes in a scramjet (NASA).

- Jet engines with mechanical compression: the simple turbojet and its derivatives: turbofan, turbofan with post-combustor, propfan (unducted turbofan), turboprop, and turboshaft (not really a jet engine but a gas turbine). Nozzle exit is usually supersonic in turbojets, but subsonic in the others. The main elements of a turbojet (the core of all air-breathing jet engines) are presented in Fig. 4. The flow cross-section area (and the corresponding blade length) decreases downstream along compressor stages (and increases along turbine stages) so as to maintain almost the same gas speed (and to compensate the pressure and temperature effects on gas density); a stage includes the rotating-blade plate and the fixed-vane stator plate (in that order for compressors and in the reverse order for turbines).

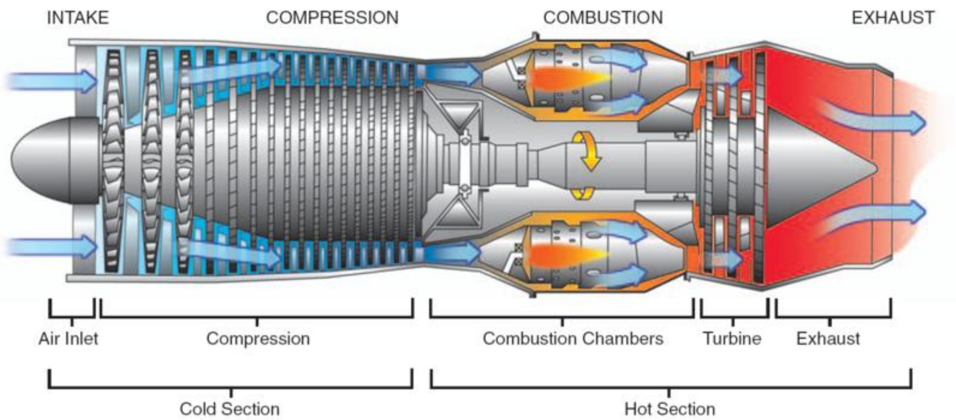


Fig. 4. Basic components of a jet engine (FAA).

The simple turbojet sketched in Fig. 4 is no longer used alone because of its poor efficiency (<20 % at subsonic speeds), but forms a substantial part of all air-breathing jet engines, which, on decreasing order of useful flight speeds are:

- Low-bypass turbofan with after-burner (Fig. 5). It is used when supersonic flight is required (1<M<3; FL<20 km). After Concorde retirement in 2005, they are only used in military aircraft. Mixing of primary and secondary streams take place before the post-combustor, to lower temperatures and to have a single nozzle (a converging-diverging nozzle of variable area). Bypass ratio is $\beta\equiv \dot{m}_{a,fan}/\dot{m}_{a,core}<1$, and fan pressure ratio is FPR=2..6 (two or three fan stages are used). In the long term design, afterburners should be avoided since they are much more pollutant, noisy (180 dB at take-off), and much less fuel efficient (particularly at low speeds), or use it only to cross the transonic barrier. For good performances at both supersonic and subsonic speeds, the bypass ratio should be adjustable (implying a variable geometry inlet).

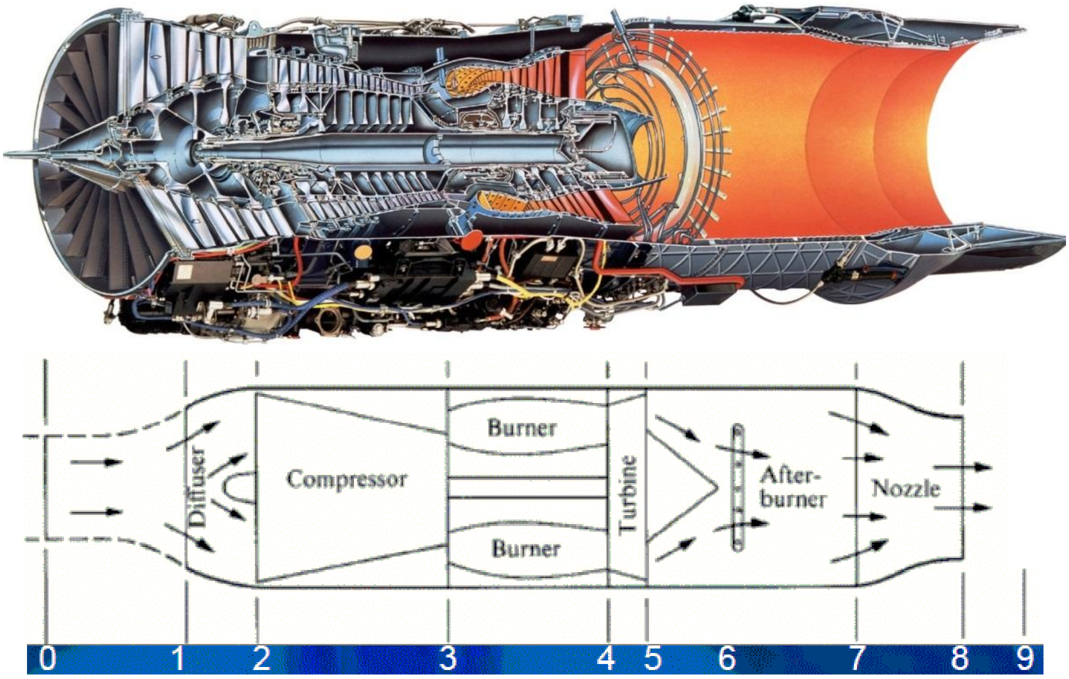


Fig. 5. Cut-away of a turbofan with after-burner (P&W), and standard notation for flow stages in jet engine core stream (bypass stream not shown): 0, unperturbed; 1, at intake; 2, after diffuser; 3, end of compression; 4, turbine entry; 5, end of turbine ; 6, at afterburner (if any); 7, nozzle entry; 8, nozzle exit; 9, free jet.

- High-bypass turbofan (Fig. 6). It is used in all major airliners for high-subsonic flight ($0.6<M<0.9$) and ceiling <13 km. Bypass ratio is $\beta \equiv \dot{m}_{a,fan}/\dot{m}_{a,core}$ =5..10, and fan pressure ratio is FPR=1.4..1.8 (only one fan stage is used). To increase global efficiency, bypass ratio is being increased and fan-pressure-ratio decreased, but the limit may be FPR=1.3; below that, unducted fans (propfans) are better. The basic compressor-turbine spool is supplemented with an additional turbine which drives the fan (and usually a low-pressure compressor), by means of an inner shaft running at lower rotating speed. Since 1960s, spools are supported on foil-air bearings not requiring oil lubrication (the lubricating air layer builds itself at high shaft spin-rate. Core and fan streams exit separately, but both enter through the fan. Overall efficiency is approaching 40 %. In some cases there are three concentric spools, an outer one connecting the high-pressure turbine with the high-pressure compressor (spinning at N_3 rpm), another inner one connecting an intermediate-pressure turbine to the low-pressure compressor (spinning at N_2 rpm), and a central spool connecting the low-pressure turbine to the fan (spinning at N_1 rpm, with $N_1<N_2<N_3$); actual spool-speed on the cockpit is presented as a percentage of nominal rotation rate. For a flight speed of M=0.85, the rotating fan blades may receive the air stream at M=1..1.5 at the tip. All spools work on torsion, i.e. have a driving side at the rear (one or a few turbine stages), and a driven side at the front (compressor stages and fan); the most loaded is the fan spool, N_1, since the fan flow provides >80 % of global thrust, with the core flow contributing the other <20 %. Typical start-up procedure is to feed compressed air from the APU to the N_2 spool, wait some 10..20 s until it accelerates to about 20 % of nominal rotation speed, inject fuel and light up with sparks, wait for N_2 to reach its idle rating (about 60 % of maximum), close the starter air, and wait until N_1 also reaches its idle rating.

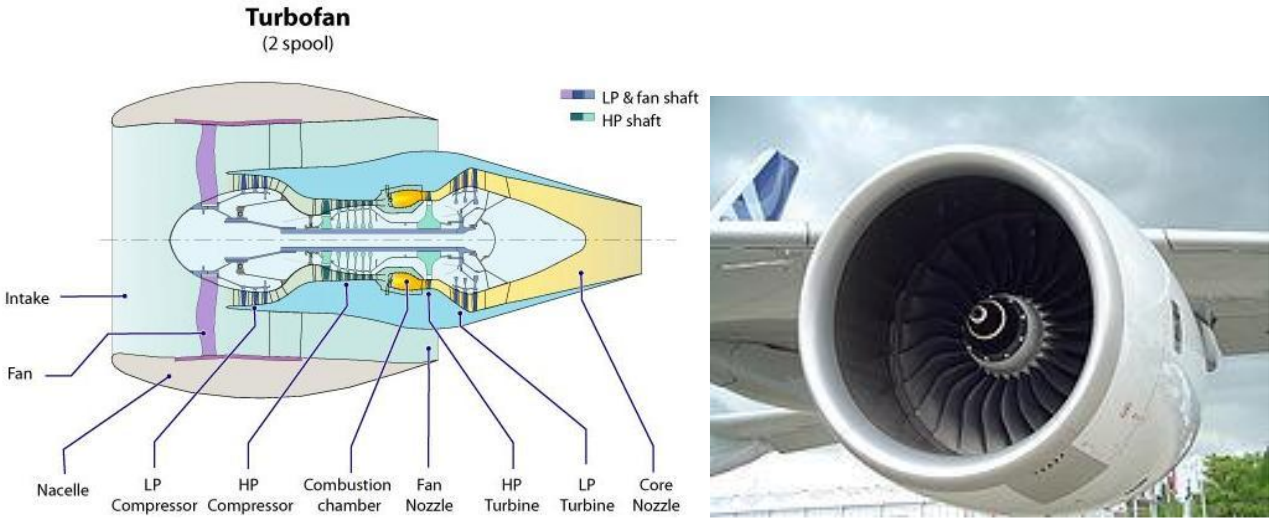


Fig. 6. Sketch of a turbofan (Wiki), and a RR-Trent turbofan installed on A380 (Wiki).

- Propfan (or open rotor turbojet) is a kind of un-ducted turbofan where the fan (usually two counter-rotating multi-blade propellers of short length and wide chord) sits near the low-pressure turbine, which drives it directly, to avoid multiple concentric spools (Fig. 7). It is recently being used in short-range medium speed flights ($0.4<M<0.6$). They are more efficient but noisier and heavier than turbofans, but, for BPR $\beta>20$ or so, the extra weight and drag of the ducting in turbofans outweighs their aerodynamic advantage.

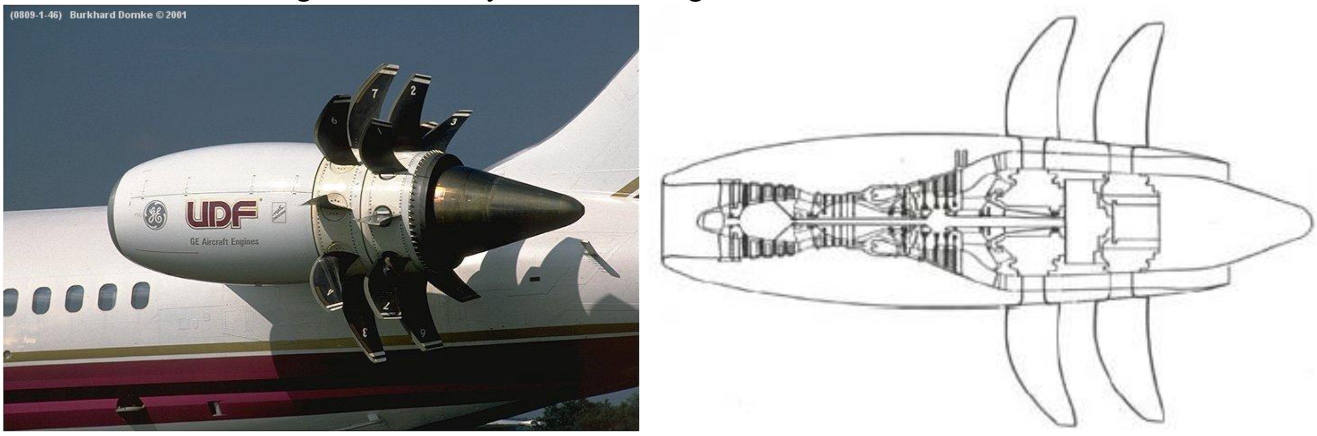
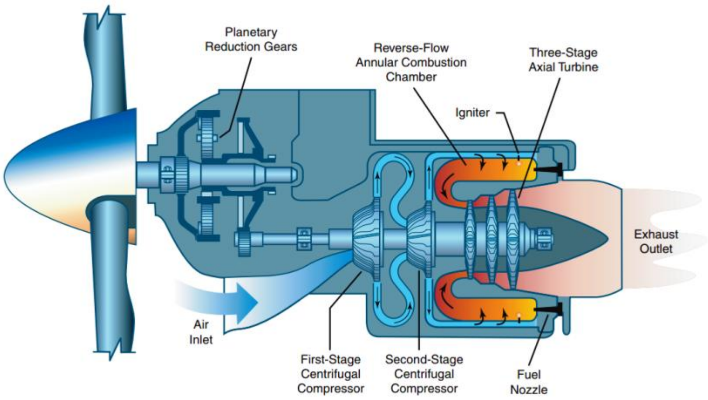


Fig. 7. Propfan (un-ducted fan) propeller (GE), and sketch.

- Turboprop (Fig. 8a) is a gas turbine driving a propeller similar to those used with reciprocating engines, but with a larger gear box to get the greater reduction. Propeller and compressor may be driven by the same turbine (fix shaft configuration, as in Fig. 8a), or by separate turbines (a compressor turbine and a load turbine, like for the turboshaft in Fig. 8b). In case of engine failure (e.g. flame extinction), the propeller works as a windmill (absorbing power from the air instead of forcing it backwards) and it must be put on feather, what is crucial on fix-shaft turboprops because the large power extracted from the wind to drive the compressor cause a great drag (a negative-torque sensor in the shaft automatically puts the propeller on feather). Turboprops are used for slow flight ($0<M<0.4$) in small planes.



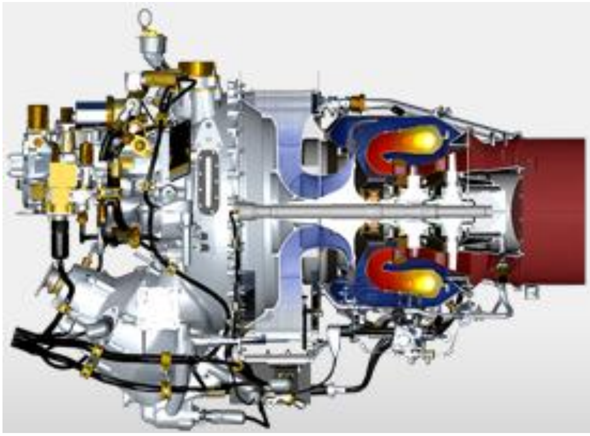


Fig. 8. a) Turboprop (PilotOutlook). b) Turboshaft PW206B used in Eurocopter E135.

- Turboshaft (Fig. 8b) is a kind of turboprop with no useful jet thrust, i.e. it is not really a jet engine but a gas turbine used in helicopters, auxiliary power units, boats and ships, tanks, hovercraft, and stationary equipment.

Helicopters are nowadays powered by turboshafts because these engines are more reliable and lighter than the reciprocating engines initially used, but the rotor is still mechanically driven. The possibility of using compressed air conducted through the blades of the rotor (with or without combustion at the tips), to eliminate the mechanical transmission and the reaction torque, has been attempted but without success. Some helicopters with rotors driven by tip ramjets have been in operation (the only subsonic application of ramjets); tip-rockets were also tried, but fuel-consumption and noise were too high in both cases. As an example, the EC135 from Airbus Helicopters is a twin-engine civil helicopter widely used by police and ambulance services and for executive transport; each of the two PW206B turboshafts has about 450 kW (a torque of 750 Nm at 6000 rpm on the load shaft), 110 kg dry mass (4 kW/kg of specific power), 2 kg/s of air flow rate, 0.04 kg/s of fuel rate (320 g/kWh of specific fuel consumption); it has a single-stage centrifugal compressor with pressure ratio of 8, annular reverse-flow combustion chamber, a single-stage high-pressure compressor turbine (at 55 000 rpm), and a single-stage low pressure power turbine.

ENGINE PARAMETERS AND PERFORMANCES

Engine performance (behaviour) depends on many internal and external parameters, among which the deliverable power (or thrust, in jet engines), and the corresponding fuel consumption, are the main ones, determining engine type and size, and being directly related to safety of operation (especially in aviation). And not only the mechanical performances of the engine are dominated by fuel consumption rate (and related air intake), but emissions, noise, and vehicle integration of engine and fuel tanks, depend on fuel consumption.

Other engine parameters related to size or performances are engine rotation rate, and internal characteristics like operating pressure and temperature. A summary of main magnitudes and symbols used in engine analysis follows, with emphasis in the main aero-engine, the turbofan, using the stage notation of (Fig. 5): 0, far forward (undisturbed); 1, entry to engine; 2, entry to compressors; 3, entry to combustion chamber; 4, entry to turbines; 5, entry to afterburner diffusor in supersonic jet engines; 6, entry to open combustor if any; 7, entry to nozzle; 8, nozzle throat (exit in converging nozzles); 9, nozzle exit in supersonic flow.

- Fuel mass flow rate, $\dot{m}_\mathrm{f}$ (the simpler symbol *c* is often used). Typical values for a jet engine in an airliner are $\dot{m}_\mathrm{f}$=1..5 kg/s at take-off and 0.2..1 kg/s at cruise. For comparison, typical values for a car may be $\dot{m}_\mathrm{f}$=1..3 g/s at 100 km/h; Saturn V (the rocket that put humans on the Moon) consumed 15 t/s at lift-off. The tradition in engine applications is to use the lower heating value of the fuel (h_LHV or LHV; the simpler symbol *L* is often used in spite of being used for length and lift too), with a typical value for petroleum distillates of h_LHV=43 MJ/kg; notice that in other applications the higher heating value (HHV) is used, with h_HHV=47..48 MJ/kg for petroleum distillates, so that care is needed when consulting fuel data (by the way, it is worth recalling that the heating value is ascribed to the fuel because we pay for it, but the heat release is due to a recombination of fuel and oxidiser, the latter being free in the atmosphere, in this case).
- Air mass flow rate, $\dot{m}_\mathrm{a}$ (the simpler symbol *G* is often used; Sp. *gasto*; Fr. *débit*). Typical values for air intake in a large turbofan are $\dot{m}_\mathrm{a}$=200..1200 kg/s at take-off and 40..250 kg/s at cruise. Notice however that most of the air-flow in turbofans is diverted to the fan, and only 10..20 % of entry air goes to the heat engine core (and the symbols $\dot{m}_\mathrm{a}$ or *G* may refer to the former or the latter), i.e. $\dot{m}_\mathrm{a,core}$=20..120 kg/s at take-off. For comparison, typical values for a car may be $\dot{m}_\mathrm{a}$=0.02..0.10 kg/s at 100 km/h. Thermodynamic modelling of jet engines is based to a first approximation in the Bryton cycle, the ideal gas law with cold pure-air flow (i.e. neglecting fuel addition, and taking a molar mass of M=0.029 kg/mol, c_p=1000 J/(kg·K), a thermal capacity ratio $\gamma\equiv c_p/c_v$=1.40), and isentropic efficiencies about 0.85 for compressors and 0.90 for turbines, but this simple model can be enhanced changing to c_p=1150 J/(kg·K) and γ=1.33 for hot flows (combustion chamber, turbine, and exhaust).
- Bypass ratio, $\beta\equiv\dot{m}_\mathrm{a,fan}/\dot{m}_\mathrm{a,core}$ (or BR, or BPR, or sometimes Λ), is the quotient of fan to core air mass-flow-rates in turbofans. Typical values are β=3..10 in subsonic engines, and β=0.1..0.8 in supersonic engines (turbofans with afterburners). If $\dot{m}_\mathrm{a}$ is the mass flow rate of air through the core engine, total air intake is $\dot{m}_\mathrm{a}(1+\beta)$. Engines with large BR are more fuel-efficient and less noisy, and values of β=10..20 are expected for year 2020 designs, for optimum efficiency, but problems arise with fan size (air drag increases, nacelle is closer to ground, weight increases...), and with fan speed (larger fans need lower speeds, requiring a gearbox to match the low-pressure-turbine-spool speed). Bypass ratio in turboprops would correspond to β=30..50.
- Air-to-fuel ratio, *A* (sometimes AFR), or fuel-to-air ratio, f=1/*A* (sometime 'far'), usually refers to the combustion process within the turbine combustor, with $A\equiv\dot{m}_\mathrm{a,cc}/\dot{m}_\mathrm{f}=\left(\dot{m}_\mathrm{a,core}-\dot{m}_\mathrm{a,bled}\right)/\dot{m}_\mathrm{f}$, where cc refers to the combustion chamber; typical air bleeding in the compressor (used for cabin air-conditioning, turbine-cooling, and pneumatic applications) is $\dot{m}_\mathrm{a,bled}/\dot{m}_\mathrm{a,core}$=5..10 %. The stoichiometric value for petroleum-derived-fuels like Jet A-1 is A_0=15 kg/kg (i.e. 15 kg of air per kg of fuel), but more air must be supplied to lower the burned gas temperature before entering the turbine, so that values A=25..50 kg/kg are used (f=0.02..0.04). Current practice to minimise emissions with optimum efficiency is to use double the stoichiometric air in the combustion zone plus another stoichiometric amount to dilute the burnt gases before entering the turbine, i.e. a relative air-fuel-ratio $\lambda\equiv A/A_0$=3.

- Shaft speed (spinning rate), N (usually in revolutions per minute, rpm, or in % of maximum nominal rpm). Piston engines have only one shaft, and operate most of the time in the range of 40..70 % of maximum rotation speed. Jet engines may have 1, 2, or 3 coaxial spools, and operate most efficiently in the 85..100 % range of rated power, with idle running at 50..60 % of maximum rpm. Jet-engine thrust is very sensitive at high spin rates, with larger response time to accelerations (that is why they are operated at high rpm during the crucial final approach to landing phase).
- Thrust, F (symbol T is sometimes used in spite of being used for temperature, and in some texts for torque, too; Sp. *empuje*; Fr. *poussée*). For a turbofan, if the effects of air bleeding, fuel addition, and a possible pressure imbalance term at the exit, are neglected, $F = F_{\text{core}} + F_{\text{fan}} = \dot{m}_{\text{a,core}}\left(v_{\text{e,core}} - v_0\right) + \beta\dot{m}_{\text{a,core}}\left(v_{\text{e,fan}} - v_0\right)$, with typical values for an airliner engine of F=100..350 kN at take-off, and a fourth or so of these maximum values under cruise conditions. In large turbofans, 80..90 % of total thrust is due to the fan. Fan speed (N_{fan}) is the primary indication of thrust on most turbofan engines, with fuel flow rate as a secondary indicator.
- Thrust to engine-weight ratio, $F/W_{\text{eng}} = F/\left(m_{\text{eng}}g\right)$, often named 'thrust/weight ratio' or simply F/W, not to be confused with that of the whole aircraft. Typical values for airliner turbofans are $F_{\text{max}}/W_{\text{eng}}$=3..5 (e.g. engine thrust 4 times its weight), whereas $F_{\text{max-all}}/W_{\text{craft}}$=0.25..0.35 (e.g. total engine thrust equals 30 % of aircraft weight). Jet engines with afterburners may have $F_{\text{max}}/W_{\text{eng}}$=10 or higher (i.e. the aircraft can flight vertically upwards like a rocket).
- Thrust to fuel-rate ratio, $F/\dot{m}_{\text{f}}$ (the name 'specific thrust' should be avoided; 'fuel specific thrust' might be used, but current practice is to name 'specific thrust' in air-breathing engines to $F/\dot{m}_{\text{a}}$, not to $F/\dot{m}_{\text{f}}$, and to name 'specific impulse' to $F/\left(\dot{m}_{\text{f}}g\right)$). Typical values for airliner turbofans are $F/\dot{m}_{\text{f}}$=50..65 kN/(kg/s).
- For shaft engines, shaft power, $\dot{W}_{\text{shaft}}$, is used instead of thrust (which depends on propeller efficiency; $F = \eta_{\text{p}}W_{\text{shaft}}/v_0$). Most usual ratios are:
  - Power to engine-mass ratio, $\dot{W}_{\text{shaft}}/m_{\text{eng}}$, with typical values of 5..15 kW/kg (e.g. for the turboprop used in A400M, the Europro TP400, $\dot{W}_{\text{shaft}}/m_{\text{eng}}$ =8200/1900=5 kW/kg; for the piston engine used in Cessna 172, the Lycoming O-360, $\dot{W}_{\text{shaft}}/m_{\text{eng}}$ =134/117=11.5 kW/kg).
  - Engine-mass to power ratio, $m_{\text{eng}}/\dot{W}_{\text{shaft}}$, is just the inverse of the former (typically 0.1..0.2 kg/kW).
  - Power-specific fuel consumption, $\dot{m}_{\text{f}}/\dot{W}_{\text{shaft}}$ (e.g. 238 g/kWh for TP400, 230 g/kWh for O-360).
- Thrust specific fuel consumption, $\dot{m}_{\text{f}}/F$, is the inverse of thrust to fuel-rate ratio presented above; it is shorthanded to TSFC, or simply to SFC (specific fuel consumption), and symbolised as c_{sp} or c_F (but c_F is used for different thrust coefficients in propellers and nozzles). Typical values for airliner turbofans are $c_{\text{sp}}=\dot{m}_{\text{f}}/F$=0.015..0.025 (kg/s)/kN in cruise conditions, and some 5..10 % more at full load. For a fully loaded modern airliner, this is equivalent to a fuel consumption of 3 kg per 100 km per passenger, i.e. 0.03 kg/pkm (it can be compared to a typical car consumption of 4 kg per 100 km).
- Specific thrust (air-flow specific thrust), $F/\dot{m}_{\text{a}}$. Typical values for airliner turbofans are $F/\dot{m}_{\text{a,global}}$ =0.2..0.3 kN/(kg/s), with $F/\dot{m}_{\text{a,core}}$ =1.5..2.5 kN/(kg/s).

- Specific impulse (fuel-weight-flow specific impulse), $I_{sp} \equiv F/(\dot{m}_f g)$. Typical values for airliner turbofans are $I_{sp} \equiv F/(\dot{m}_f g)$=6000..7000 s, but the related parameter TSFC, c_{sp}, is more commonly used, both being related by $I_{sp} \equiv F/(\dot{m}_f g)=1/(c_{sp}g)$.
- Global propulsion efficiency, $\eta \equiv F v_0/(\dot{m}_f h_{LHV})$, is the ratio of thrust power (Fv_0) to raw power (the lower heating value, LHV, of the fuel rate is assumed; for Jet A-1, h_{LHV}=43 MJ/kg). The global or overall efficiency is the product of engine thermal efficiency times propulsion efficiency (see Propulsion efficiency, aside). Typical values for turbofans and turboprops are η=0.30..0.35 (η=0.40 for 2020?).
- Flight speed, v_0, is advancing speed relative to air (not to ground). In air-breathing engines, engine air intake speed is higher than v_0 during take-off and landing (air is aspirated), and lower than v_0 during cruise (some air is diverted away because the engine cannot swallow all of it). Practical flight speed is limited by transonic resistances, and the limit for turbofans is around M=0.85 (v_0=250 m/s=900 km/h). Take-off speed is around 70 m/s (250 km/h). The shape of the engine intake (the cowling) is very important for the aerodynamic efficiency in engine operations, particularly in high speed aircraft, where the cowling design is critical.
- Exit speed, v_e, is the exhaust speed relative to engine (not to ground). In subsonic turbofans, exit speed may be different in the core stream than in the fan stream, but both around 300 m/s, i.e. a little below sound speed (Fig. 9); maximum propulsive efficiency would demand equal exit speeds for primary and secondary streams, but this would imply a multi-stage fan in low-bypass engines, more noisy and heavy (only one fan stage is used in high-bypass airliner engines, but two or three fan stages are used in low-bypass supersonic engines, where both streams are later mixed before the afterburner (both arriving streams must have equal pressure).
- Flight level is a nominal altitude based on pressure measurement, which would coincide with the real altitude of the aircraft if the actual atmosphere coincided with the ISA model, i.e. dry air with precisely p_0=101.325 kPa and T_0=288.15 K at sea level, and a linear temperature descent of Γ=6.5 ºC/km up to 11 km, with T=−56.5 ºC constant from z=11 to z=20 km. This pressure-altitude can be computed from $z/z_{11}=4.03[1-(p/p_0)^{0.19}]$; e.g. to an outside pressure of p=50 kPa corresponds an altitude pressure of $z=11\times4.03[1-(50/101.325)^{0.19}]$=5.6 km. The effect of actual weather causes a deviation of pressure-altitude from true-altitude (as measured by a radar altimeter, or the GPS), but for high altitudes the uncertainty is assumable (even more if all aircraft follow this convention). Instead of using metres, flight-level numbering is traditionally performed in hecto-feet (e.g. FL300 means a 300×100=30 000 ft pressure-altitude, i.e. 9144 m).
- Pressure ratio, π (or PR), is the quotient of total-pressure values (see Stagnation and total conditions, aside):
  - Fan pressure ratio, FPR=π_{fan}. Typical values are π_{fan}=1.5..2 in subsonic turbofans and π_{fan}=3..4 in supersonic-engine fans; notice that in present supersonic-flight engines, the intake is designed to decelerate the flow to subsonic speed in oblique shock waves, remaining subsonic until the exit nozzle; engines with supersonic through flow (SFT) are still at the research state. In most modern turbofans, however, the fan-tip rotation speed is supersonic at high engine power.
  - Global pressure ratio, GPR=π_{23}, or overall pressure ratio, OPR. Typical values are π_{23}=25..40 in turbofans (aiming at GPR=50), and π_{23}=5..15 in the smallest jet engines.

Low-pressure compressor may have π_{LPC}=6..8 and high-pressure compressor π_{HPC}=4..5 (overall pressure ratio is $\pi_{23}=\pi_{fan}\pi_{LPC}\pi_{HPC}$). Pressure ratio in the combustion chamber is about η_{CC}=0.9 (pressure loss may be some 100 kPa at take-off).

- Nozzle pressure ratio, NPR=p_{t8}/p_8, is the ratio between total pressure and static pressure at the nozzle throat (exit in converging nozzles). If the NPR=$p_{t8}/p_8>(\gamma+1/\gamma)^{\gamma/(\gamma-1)}$, the flow becomes choked (M_8=1); these critical value is NPR$_{cr}$=1.89 for cold air (γ=1.40) and 1.85 for hot burned gases (γ=1.33).
- Engine pressure ratio, EPR (or exit pressure ratio; not to be confused with overall pressure ratio); it is $\pi_{19}\approx 1$ for subsonic and adapted nozzles.

- Turbine entry temperature, TET=T_{4t}. It is limited by materials strength, with typical values T_{4t}=1400..1800 K (it started in 1940 at 1000 K). For non-dimensional analysis the quotient between a total temperature and the static outside-air temperature is often used (e.g. $\theta_{4t}=T_{4t}/T_0$=5..6; notice that symbol θ is used for total temperature scaled with outside-air temperature, OAT, but symbol τ is used for total temperature ratio across a unit; e.g. for an isentropic process, $\tau_{12}=\pi_{12}{}^{\gamma-1/\gamma}$).
- Exhaust gas temperature, EGT=T_{9t}. Typical values are T_{9t}=700..800 K for subsonic turbofans. Notice that it refers to the core stream exit; the fan stream is much colder (see Fig. 9). It is usually measured by averaging 6 or 9 thermocouples (type K) placed at the nozzle exhaust.
- Engine temperature ratio (across the engine), ETR=$T_{9t}/T_{0t}=\tau_{eng}$. For subsonic turbofans it is $\tau_{eng}=T_{t,nozzle\text{-}exit}/T_{t,engine\text{-}inlet}\approx 3$. In rockets it is $\tau_{eng}=T_{t,nozzle\text{-}exit}/T_{t,chamber}\approx 1$.

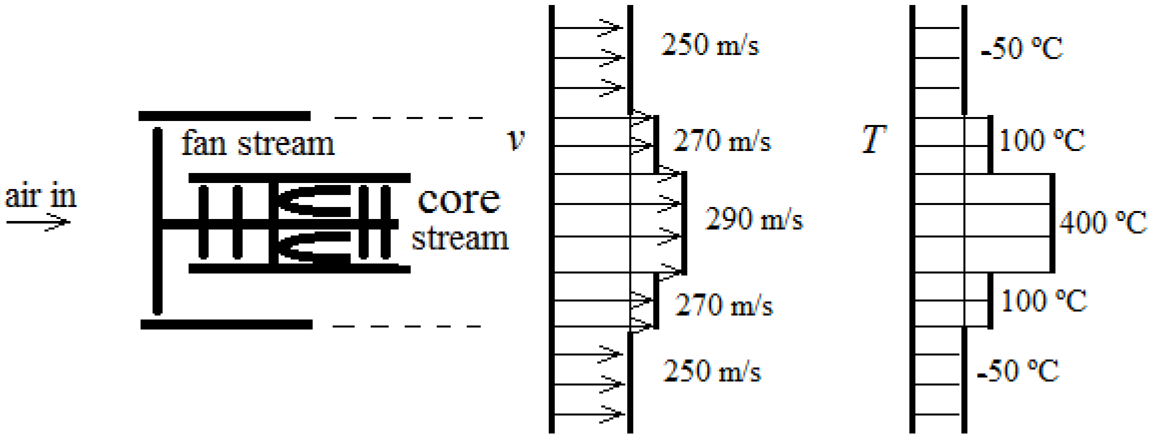


Fig. 9. Sketch of typical velocity and temperature profiles behind a turbofan in cruise flight at M=0.85 (speeds relative to engine frame; sound speed around 300 m/s; core diameter 1 m, fan diameter 3 m).

Jet fuel

On board fuel is not mandatory in aeronautics (neither on land or water vehicles); e.g. aircraft can fly powered by solar cells.

Today, most aircraft use jet engines based on a kerosene fuel (jet fuel), mainly Jet A fuel (used in all civil and some military aircraft), with Jet B fuel being used by other military aircraft. The same jet fuel is to be used by diesel engines envisaged for small airplanes and helicopters, although most small airplanes presently run on avgas (aviation gasoline).

Modern jet engines may run on a mixture of fossil kerosene and up to 50 % biofuel or similar synthetic fuel. Early in 2008 Virgin Airways conducted a demonstration flight of its Boeing 747 jets using biofuel only. On April 2008, Boeing flew a commercial two-seats Dimona glider for 20 minutes at 100 km/h at Ocaña (Spain) powered by a PEM fuel cell (a Li-ion battery was used at take-off to power the electric motor that drew the propeller).

Jet A-1 typical properties are: density at 15 ºC ρ=810 kg/m^3 (in the range 780..840), lower heating value *LHV*=43 MJ/kg (in the range 42.8..43.6). In 2000, USA consumed 2.8 t/s of Jet A, being 42 % of the world 6.7 t/s jet-fuel consumption (for comparison, the Space Shuttle burned 10 t/s at lift-off). Since jet fuels always contain some water, microbial contamination is always a threat, which must be controlled by adding biocides, and inspecting fuel-tank sumps. Although Jet A-1 is guaranty to freeze below −47 ºC (normal T_f=−50 ºC), jet engines are normally equipped with fuel heaters (usually automatic); outside air temperature may sometimes fall below −60 ºC.

Kerosene is a crude-oil distillate similar to petrodiesel but with a wider-fraction distillation (see Petroleum fuels). Jet fuel is kerosene-based, with special additives (<1 %). Rocket propellant RP-1 (also named Refined Petroleum) is a refined jet fuel, free of sulfur and with shorter and branched carbon-chains more resistant to thermal breakdown; it is used in rocketry usually with liquid oxygen as the oxidiser (RP1/LOX bipropellant). The tendency to change to biofuels or GTL fuels is also applicable here. Contrary to its etymology, present-day kerosene and derivatives are less waxy than diesel (i.e. less lubricant). Diesel and kerosene should not be taken as fully interchangeable fuels at present, because kerosene has no cetane-number specification and thus it may have large ignition delays (producing lots of unburnt emissions and engine rough-running by high-pressure peaks); besides, kerosene has less lubricity, and diesel-fuel less cold-start ability.

Jet fuel history

The first jet engines were developed just prior to and during the early part of World War II. Hans von Ohain in Germany developed the first successful aviation turbine engine that flew in the Heinkel He 178 on 27 August 1939. Gasoline was the fuel used because of its ease of evaporation and known performance properties in piston engine aircraft. Across the English Channel Sir Frank Whittle also developed an aviation turbine engine which first flew in a Gloster E28/39 aircraft on 14 May 1941, burning kerosene since gasoline was in short supply because of the on-going war. It was claimed that the jet engine could operate on any fuel, but soon fuel-standards had to be established for proper operation, starting with low-freezing-point military fuel JP-1 in 1944. These fuels (JP-1, JP-2, JP-3, JP-4) were wide-cut fuels (mixtures of naphtha and kerosene) which greatly increase availability; JP-8 is currently the primary jet fuel for NATO (replaced JP-4 in 1979). Early jet-fuel specifications differed significantly in volatility, freezing point, density, flash point, sulphur and aromatic content.

Nearly all jet fuel is presently obtained by crude-oil distillation. As crude oil will become scarce, sooner or later, substitutes have to be found (and they must be liquids of high heating value, since no other fuel type seems suitable). The trends to replace Jet A-1 are:

- Synthetizing it from coal, to stretch fossil fuel lifetime availability (e.g. a 50/50 mix of natural and synthetic Jet A-1 is already supplied in South Africa). The greenhouse-gas problem is not faced up.
- Synthetizing it from non-fossil fuels. On February 2008, a commercial airliner flew powered by biodiesel. Contribution of CO_2 emissions to global warming is balanced by CO_2 capture in the photosynthesis of the original biomass.
- Synthetizing it from non-fuel stuff (e.g. from CO_2 and H_2O, with solar energy, like in natural photosynthesis). The greenhouse-gas problem may be decreased.

Combustion chamber

Easy burning of fuel requires an air-to-fuel ratio near stoichiometry. Best burning occurs when fuel and air are premixed before burning, but this is not the case when a jet of fuel (in aviation, fuel is always in liquid state) is injected in an air stream, even if the liquid jet is sprayed in a swirling injector. Flammability limits of kerosene vapours at room conditions in terms of relative air/fuel ratio to stoichiometry are λ_{LFL}=0.25 and λ_{UFL}=2; very rich and very lean mixtures either do not burn, or do it with unstable or polluting flames (if rich). But, stoichiometric flames are very hot (some 2300 K in standard air, some 2700 K in hot air after compression), and engine materials cannot withstand such high temperatures. To solve the problem of getting an adiabatic combustion temperature (turbine entry temperature, TET) of about 1700 K, fuel is injected and burned with nearly stoichiometric air in a primary zone, and burnt gases are diluted with surrounding excess air, with an overall air/fuel ratio about 3 times the stoichiometry (λ=3). But a homogeneous mixture with λ=3 is too lean to burn (cannot sustain combustion). To solve the problem of too-lean mixtures, the classical two-zone combustion chamber sketched in Fig. 10 has been used since the first gas turbines in the 1940s.

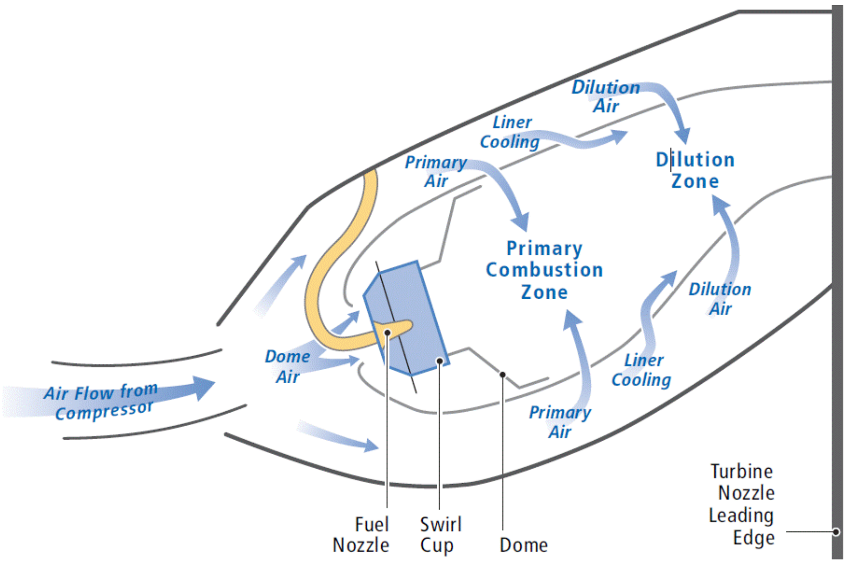


Fig. 10. Air flow in a combustion chamber of a jet engine (Wiki).

The combustion chamber (or burner, or combustor) receives a pressurised air-flow at about 150 m/s from the compressor (adiabatically compressed to some 700 K); this air flow is slowed down in the combustor diffuser, where one part surrounds the fuel injector at about 25 m/s, and mixes with liquid fuel issuing at about 30 m/s as a fine spray within the burner so as to get a rich mixture of about λ=0.8 at take-off and climb (at cruise increased to λ=0.9; at idle it may be a lean mixture, with λ=1.9), and is ignited, attaining a maximum of some 2700 K. To complete the combustion of the rich mixture, and to lower the TET to 1700 K (if the turbine blades are cooled), or below, the secondary air flow is added to the burning gases through the perforated combustor wall, as sketched in Fig. 10. Jet fuel is fed from aircraft tanks by fuel pumps (usually of two-stage centrifugal type) at about 6 MPa (maximum gas pressure there is about 4 MPa at sea level (1 MPa at cruise). In the highly turbulent primary combustion zone, flame speed is similar to main flow speed (around 50 m/s) and the flame is very stable (a flame-out event is a rare event that may occur when flying under heavy rain, and the only way out is to try re-ignition with the spark plugs).

The earliest burners were multiple cans (8 or 10) in an annular configuration, operating independently. A further development, the can-annular burner, retained the multiple flame tubes, but allowed the outer air to flow free around all the flame tubes. Modern combustion chambers are totally annular, reducing emissions (fewer walls); they have a double annular fuel-injection ring, with one row being permanently used as a pilot flame and for low-power needs (idle, and descent conditions), whereas the other ring of injectors is only lit under high power needs (take-off and climbing conditions).

From fuel to thrust: the energy conversion chain in jet propulsion

Raw energy is in the difference in chemical bonds (from fuel and oxidiser, to the combustion products); the fuel is carried aboard, the oxidiser (oxygen) is taken from environmental air, and the products (mainly CO_2 and H_2O) are discarded to the environment. This chemical energy is converted to thermal energy in the combustion chamber with an efficiency of around 99 % (because of the small amount of NO_x and unburnt HC in the exhaust gases). This thermal energy is converted to mechanical energy (excess shaft work and increment in flow kinetic energy) with an efficiency of some 40 % (hot exhaust carries out the rest). Finally, a part of this mechanical energy is used for propulsion (Fv_0) with an efficiency of some 80 % (the rest goes as excess exhaust speed). Energy efficiencies have been analysed under Propulsion efficiency, aside.

The simplest turbojet model

The simplest turbojet model consists of a steady one-dimensional ideal subsonic flow of pure air as an ideal gas, incoming from outside to engine, and going through an isentropic compressor, a heat-addition isobaric chamber, an isentropic turbine, and an isentropic propulsion nozzle that throws the air back to the environment. In spite of the many approximations implied (negligible fuel-mass addition, negligible compressed-air bleeding, same properties for cold air and for hot burned gases, no friction loses...), this analysis offers a multi-parametric analytical model of the core stream in real turbofans. This model also applies to turbojets with supersonic exhaust if the nozzle is adapted (i.e. when the pressure at the end of the nozzle is equal to the surrounding pressure), but we do not insist on it.

A crucial mechanical tie in this turbojet model is that the turbine must supply exactly the power required by the compressor, i.e. there is only one spool, connecting only both items, and engine thrust is due to the high-speed nozzle flow (by the way, in spite of the traditional interpretation and computation of jet-engine thrust as an input-output momentum balance, the pressure forces that push the engine forward are mainly applied at the rear of the compressor blades).

Relative to the engine, air is coming at speed v_0 from an otherwise unperturbed atmosphere at p_0, and T_0. Depending basically on entrance area, but also on engine regime, a certain mass-flow-rate of air, $\dot{m}_a$, is admitted through. Fortunately, v_0 is the only speed that enters in the model (made dimensionless with the local speed of sound, $c_0=\sqrt{\gamma R T_0}$, into the Mach number, $M_0=v_0/c_0$), because the energy balance of the fluid flow at steady state, for a device with one input and one output, can be stated as:

$$q+w=\Delta h_t=\Delta h+\Delta(½v^2+gz) \xrightarrow{\text{PGM}} q+w=c_p\Delta T_t=c_p\Delta T+\Delta(½v^2) \qquad (1)$$

where q and w are the heat-power and mechanical-power inputs through the walls per unit of mass-flow-rate, and Δh_t is the total-enthalpy jump from entrance to exit, per unit of mass-flow-rate. Hence, working with total temperatures, T_t, leaves speeds out of energy balances. Notice that, as total temperatures are used all through the engine, subscript 't' is skipped in some texts, and subscript 's' is used for static values (e.g. T_{s0} and T_0 are used for outside-air temperature and its total vale, respectively, instead of our T_0 and T_{t0}).

With this ideal model, total variables are the same far upstream (stage 0), at the entry cross-section (stage 1), and through the diffuser until entrance to the compressor (stage 2). Using θ for the total temperature at a stage divided by T_0, and τ for the total-temperature ratio in a device (output/input), one has for the entry:

$$\left.\begin{array}{l} \theta_0\equiv\dfrac{T_{t0}}{T_0}=1+\dfrac{\gamma-1}{2}M_0^2 \quad \text{and} \quad T_{t0}=T_{t1}=T_{t2} \quad\to\quad \tau_{02}\equiv\dfrac{T_{t2}}{T_{t0}}=1 \\ \dfrac{p_{t0}}{p_0}=\left(\dfrac{T_{t0}}{T_0}\right)^{\frac{\gamma}{\gamma-1}} \quad \text{and} \quad p_{t0}=p_{t1}=p_{t2} \quad\to\quad \pi_{02}\equiv\dfrac{p_{t2}}{p_{t0}}=1 \end{array}\right\} \qquad (2)$$

For the compressor, with the isentropic model:

$$\pi_{23}\equiv\frac{p_{t3}}{p_{t2}},\quad \tau_{23}\equiv\frac{T_{t3}}{T_{t2}}=\left(\frac{p_{t3}}{p_{t2}}\right)^{\frac{\gamma-1}{\gamma}}=\pi_{23}^{\frac{\gamma-1}{\gamma}},\quad w_{23}=c_p\left(T_{t3}-T_{t2}\right)\ \to\ \frac{w_{23}}{c_pT_0}=\theta_0\left(\tau_{23}-1\right) \qquad (3)$$

For the combustor, with the isobaric assumption:

$$\theta_4\equiv\frac{T_{t4}}{T_0},\quad \tau_{34}\equiv\frac{T_{t4}}{T_{t3}},\quad \frac{p_{t4}}{p_{t3}}=1,\quad q_{34}=c_p\left(T_{t4}-T_{t3}\right)\ \to\ \frac{q_{34}}{c_pT_0}=\theta_4-\theta_0\tau_{23} \qquad (4)$$

For the turbine, with the isentropic model:

$$\pi_{45}\equiv\frac{p_{t5}}{p_{t4}},\quad \tau_{45}\equiv\frac{T_{t5}}{T_{t4}}=\pi_{45}^{\frac{\gamma-1}{\gamma}},\quad w_{45}=c_p\left(T_{t5}-T_{t4}\right)\ \to\ \frac{w_{45}}{c_pT_0}=\theta_4\left(\tau_{45}-1\right) \qquad (5)$$

Recall now the spool energy balance (at steady state):

$$w_{23}+w_{45}=0 \quad \rightarrow \quad \theta_0\left(\tau_{23}-1\right)+\theta_4\left(\tau_{45}-1\right)=0 \tag{6}$$

(mind you, absolute values are often used for w_{45}, and then the spool energy balance is written w_{23}=w_{45}).

For the nozzle (stages 7 to 8 in a converging nozzle, but nothing special happens at the jet-pipe from stage 5 to 7), the isentropic model is:

$$p_8=p_0,\quad \frac{p_{t8}}{p_{t5}}=1,\quad \frac{T_{t8}}{T_{t5}}=1,\quad \theta_8\equiv\frac{T_{t8}}{T_8}=1+\frac{\gamma-1}{2}M_8^2=\left(\frac{p_{t8}}{p_8}\right)^{\frac{\gamma-1}{\gamma}} \tag{7}$$

We want now to get some explicit expressions for the exit conditions (subscript 'e' for exit is used now instead of '8') in terms of the three temperature quotients: θ_0 (related to M_0 by (2)), τ_{23} (related to pressure ratio π_{23} by (3)), and θ_4 (relative turbine entry temperature, TET), before and after imposing the spool energy balance (6). Back chain substitution in the last of (7) yields (notice that p_{t4}=p_{t3}):

$$\begin{gathered}1+\frac{\gamma-1}{2}M_8^2=\left(\frac{p_{t8}}{p_8}\right)^{\frac{\gamma-1}{\gamma}}=\left(\frac{p_{t5}}{p_0}\right)^{\frac{\gamma-1}{\gamma}}=\left(\frac{p_{t5}}{p_{t4}}\frac{p_{t4}}{p_{t3}}\frac{p_{t3}}{p_{t2}}\frac{p_{t2}}{p_0}\right)^{\frac{\gamma-1}{\gamma}}=\frac{T_{t5}}{T_{t4}}\frac{T_{t3}}{T_{t2}}\frac{T_{t2}}{T_0}=\tau_{45}\tau_{23}\theta_0\\ \rightarrow\quad M_e=\sqrt{\frac{2}{\gamma-1}\left(\theta_0\tau_{23}\tau_{45}-1\right)}\overset{w_{23}+w_{45}=0}{=}\sqrt{\frac{2}{\gamma-1}\left[\theta_0\tau_{23}\left(1-\frac{\theta_0}{\theta_4}\left(\tau_{23}-1\right)\right)-1\right]}\end{gathered} \tag{8}$$

Similarly:

$$\begin{gathered}\frac{T_8}{T_0}=\frac{T_{t8}}{T_0}\frac{T_8}{T_{t8}}=\frac{T_{t5}}{T_0}\left(\frac{p_8}{p_{t8}}\right)^{\frac{\gamma-1}{\gamma}}=\frac{T_{t4}}{T_0}\frac{T_{t5}}{T_{t4}}\frac{1}{\tau_{45}\tau_{23}\theta_0}=\frac{\theta_4}{\tau_{23}\theta_0}\quad\rightarrow\quad\frac{T_e}{T_0}=\frac{\theta_4}{\tau_{23}\theta_0}\\ \frac{v_e}{v_0}=\frac{M_e}{M_0}\sqrt{\frac{T_e}{T_0}}\quad\rightarrow\quad\frac{v_e}{v_0}=\frac{1}{M_0}\sqrt{\frac{2}{\gamma-1}\left(\theta_0\tau_{23}\tau_{45}-1\right)\frac{\theta_4}{\tau_{23}\theta_0}}\overset{w_{23}+w_{45}=0}{=}\frac{1}{M_0}\sqrt{\frac{2}{\gamma-1}\left[\theta_4-\theta_0\left(\tau_{23}-1\right)-\frac{\theta_4}{\tau_{23}\theta_0}\right]}\end{gathered} \tag{9}$$

where the latter is obtained by using the Mach number definition ($v=M\sqrt{\gamma RT}$).

The engine thrust, $F=\dot{m}_a\left(v_e-v_0\right)$, made dimensionless with the air flow rate and the speed of sound in unperturbed air, $c_0=\sqrt{\gamma RT_0}$, becomes:

$$\begin{aligned}\frac{F}{\dot{m}_a c_0}&=\frac{v_e-v_0}{c_0}=M_0\left(\frac{v_e}{v_0}-1\right)=\sqrt{\frac{2}{\gamma-1}\left(\theta_0\tau_{23}\tau_{45}-1\right)\frac{\theta_4}{\tau_{23}\theta_0}}-M_0\\ &\overset{w_{23}+w_{45}=0}{=}\sqrt{\frac{2}{\gamma-1}\left[\theta_4-\theta_0\left(\tau_{23}-1\right)-\frac{\theta_4}{\tau_{23}\theta_0}\right]}-M_0\end{aligned} \tag{10}$$

Finally we can find the parametric dependence of the specific fuel consumption (TSFC), using (4) and modelling the heat addition by the energy balance $\dot{m}_{\mathrm{a}} q_{34} = \dot{m}_{\mathrm{f}} h_{\mathrm{LHV}}$:

$$c_{\mathrm{sp}} = \frac{\dot{m}_{\mathrm{f}}}{F} = \frac{\dot{m}_{\mathrm{a}}}{F} \frac{q_{34}}{h_{\mathrm{LHV}}} = \frac{c_p T_0}{c_0 h_{\mathrm{LHV}}} \frac{\theta_4 - \theta_0 \tau_{23}}{\sqrt{\frac{2}{\gamma-1}\left[\theta_4 - \theta_0(\tau_{23}-1) - \frac{\theta_4}{\tau_{23}\theta_0}\right]} - M_0} \tag{11}$$

When testing a given turbojet in bench (at a steady state on a given environment), specific thrust, $F/\dot{m}_{\mathrm{a}}$, depends just on θ_4 and τ_{23} (i.e. on turbine entry temperature, TET, and overall pressure ratio, OPR), although there is only one engine control, the fuel flow rate, which sets θ_4, τ_{23}, $\dot{m}_{\mathrm{a}}$, and F, the latter balanced by the reaction force at the engine supports. On cruise, however, thrust (from all engines in the aircraft) is balanced with aircraft drag (including engine drag), $F_{\mathrm{engines}}=D_{\mathrm{aircraft}}$, and this relationship determines the flight speed (M_0 from (10)).

As a feedback response to the direct control variable 'fuel lever', several related variables (for redundancy) are measured and presented at the cockpit: the spool speed (RPM), the exhaust-gas temperature (EGT), and the actual fuel rate, $\dot{m}_{\mathrm{f}}$. The main goal of engine regulation is to provide a thrust proportional to fuel-lever position. Air flow-rate is proportional to spool speed, and exit speed is proportional to EGT (T_{t7}), which is proportional to TET (T_{t4}, too difficult to measure).

Thermal and mechanical limitations in jet engines

Basic design limitations are imposed by thermal effects on engine materials, mechanical loads, internal aerodynamics, etc.

<u>Thermal resistance</u>

Continuous work at >1300 K is very difficult in general (most materials lose their strength), and jet engines presently work with turbine entry temperatures (TET) of T_{4t}=1300..1800 K (and T_{8t}=1600..2100 K in after-burners). Without cooling, the best metal alloys used in turbine blades can only operate continuously below 1400 K (ceramic blades might overcome this limit). Internal cooling by air convection within the blade has low efficient, so that external cooling with compressed air bled before combustion (about 1..2 % of core flow), is performed by injecting it through laser-drilled holes in all the blade surface (more concentrated near the leading edge), achieving a film cooling of the blades. Water cooling would be more efficient, particularly if let to escape through porous in the blade (evaporative cooling), but it complicates the equipment too much (a water source would be needed).

Maximum temperature within the combustion chamber is around 2700 K some 10 cm downstream of the injectors, at the primary-zone-holes section, where the relative air/fuel ratio is a little below stoichiometry. Adding liquid water at the entrance of the combustion chamber or to the burnt gases before entering the turbine, lowers thermal stresses or provides more power, but, for a given amount of water, it is better to add it at the entrance to the low-pressure compressor; in any case, it can only be an advantage during take-off and initial climb-out, since water weight for cruise would spoil overall performances.

In jet engines with afterburner, a typical thermal problem is the appearance of a non-uniform azimuthal temperature distribution, with permanent hot spots.

Ceramic materials are being tried as coatings and as substrates on hot steady surfaces (e.g. SiC for supersonic nozzle petals (of variable area), able to work at 1500 K). Carbon composites may withstand much higher temperatures, but not in the high-oxidising atmospheres of jet engines (unless properly coated).

<u>Mechanical resistance</u>

Rapid shaft spinning creates very large centrifugal forces in the blades, what may yield creeping (thermal and stresses), and fatigue (particularly due to the high-frequency excitation caused by the rotor blades crossing the periodic wakes of previous stator vanes).

The aerodynamics of cascade vanes imposes some limitations in isentropic efficiencies (the different shape of fan blades, compressor blades, and turbine blades, can be seen in Fig. 1):

- Compressor isentropic efficiencies must be high for the jet engine to work at all. Currently they are η=0.7..0.8 for centrifugal compressors, and η=0.8..0.9 for axial compressors (the less stages they have, the higher η). Centrifugal compressors are used in the smallest jet engines, and in the last compression stage of turboshafts.
- Turbine isentropic efficiencies are always higher, η=0.86..0.92, because of the favourable pressure gradient (thinner, well-attached boundary layers) and the related small number of stages (2 or 3, against 4..10 in compressors).

Internal fluid motion is always quasi-steady because the residence time for a typical 4 m long jet engine at a typical 150 m/s average speed, is 0.025 s, much smaller than the possible internal or external changes (≈1 s).

Overall pressure ratio (OPR) has been increasing over the years from OPR=10 to 40 for the primary stream. Fan pressure ratio (FPR) varies inversely with bypass ratio (BPR). At present:

- FPR=1.5..1.8 and bypass BPR=5..9 in large subsonic engines, with a trend to FPR<1.4 and BPR>10 for future engines. However, the aerodynamic behaviour of such large-BPR low-FPR fans depends so much on advancing speed that a variable area fan-nozzle will be required between take-off and cruise.
- FPR>2 and BPR<1 in supersonic engines.

Jet engine auxiliaries: start-up, electrical, pneumatic and hydraulic services

<u>Start-up</u>

Heat engines cannot start-up alone (e.g. by just a spark); they must first be run by another means to compress the air and to overcome friction, either using an electric motor, a fluid motor (pneumatic, hydraulic, or pyrotechnic systems, previously charged), or just by hand in small vintage engines (on cars, motorboats, and aircraft).

Large jet engines in an airliner are started by a smaller jet engine (the auxiliary power unit, APU, Fig. 11), which is previously started by an electric motor; i.e. the electric batteries in the aircraft make spin a relatively large electric motor (some 100..150 kW) that engages by gears to the APU (as when starting a car), which, when spark-ignited and fuel fed, may yield up to 1 MW in terms of compressed air and electricity (to provide on-board electricity, up to 100 kW at 115 V at 400 Hz three-phase); the compressed air stream (around 1 kg/s at 250 kPa) can be used to drive the air-conditioning system on board, and is diverted to the main engine for start-up, what is achieved by means of a dedicated small lateral turbine (spinning up to 60 000 rpm) that engages the main engine shaft through a gear reduction, until the main engine ignites with sparks and fuel injection, and gets disengaged (at about 15..20 % of nominal spin rate), stabilising at 'ground idle' state.

The igniter plugs are only used on ground (engine-start) or exceptionally in air (restart after engine stops); once the start is completed, igniters are turned off, sometimes automatically, although they may be left to operate permanently as added safety during take-off, heavy rain or severe turbulence. On rear-mounted engines, sudden changes in pitch angle due to severe turbulence or a quick manoeuvre may cause flame extinction (flameout), compressor stall, and other engine malfunctions.

Some airports reduce the use of APUs due to noise and pollution, and electric power and compressed air are provided from ground.

The APU is primarily designed for ground use, but some models may be started aloft too; e.g. this is the case for Extended-range Twin-engine Operations (ETOPS) aircraft, where they are a critical safety device, as the APU must supply backup electricity and compressed air in place of the dead engine or failed main engine generator. (as a last in-flight resource, an impact-air turbine might be used). The APU compartment (as well as the main engines) incorporates fire protection measures, including fire detection and extinguishing systems.


Fig. 11. a) The APU exhaust at the tail end of an A380. b) B787 Dreamliner APU.

The B787 Dreamliner is the most-electric airliner at present. It has two identical Li-ion rechargeable battery packs: the main pack at the front, and the auxiliary pack at the rear, near the APU. Each pack has 8 cells, and can provide 4.8 kW at 32 V with a mass of 29 kg. The main pack powers the airplane to life before the APU has been started, and is used to support ground operations such as refueling and powering the braking system when the airplane is towed. The APU battery supplies power to start the APU, which in turn can start the airplane engines, and is a redundancy for the main pack. Once the main engines are started, the electrical energy to run the systems comes from generators, the battery packs remaining a safety backup. The APU in B787 only delivers electricity to the aircraft, and the main engines are started with electric motors (the absence of a pneumatic system simplifies the design, but demand heavier generators).

Expendable APUs (small turbojet engines) are used in some military applications (missiles, decoys) and UAVs.

Electrical, pneumatic and hydraulic services
On ground, with main engines stopped or idle, all energy services are provided by the APU (or a large battery pack in B787, before the APU is switched on). The main power consumer on ground is the air conditioning (aircraft environmental control system, ECS), demanding about 1 kW/p (e.g. 350 kW for a 400 pax airliner, plus another 100 kW for non-ECS secondary power, i.e. non-propulsive).

Rolling and taxiing
On ground, the nose wheel and APU work as an electric vehicle, avoiding main-engine use, and thus minimising airport emissions (or the need of ground tugs).

Braking
At the high speeds of landing (some 100 m/s), aerodynamic dissipation of kinetic energy is much more effective than solid friction in wheel brakes, and usual braking systems are by air-spoilers on the wings, and by thrust-reverse in the engines (achieved by reversing the direction of the fan airflow using pivoting doors, Fig. 12).

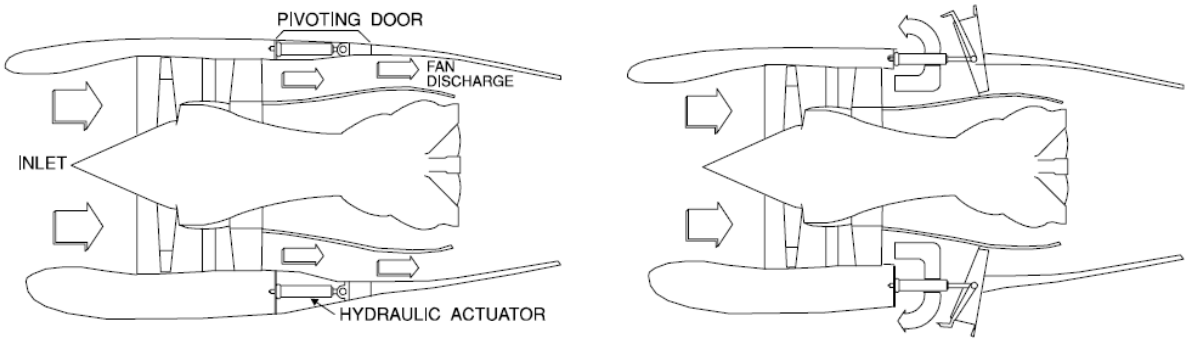


Fig. 12. Engine thrust-reverser.

Engine maintenance

The manufacturer's recommended number of running hours before an aircraft engine requires a general revision (overhaul, Fig. 12) comprising disassembling in a workbench is known as TBO (time between overhauls). At present, jet engines have TBO=10 000 h (industrial gas turbines may have TBO=100 000 h because they are not so much stressed).

Fig. 13. Overhauling on a RR-Trent-900 jet engine (Wiki).

MATCHING ENGINES AND AIRCRAFT

Two kinds of matching may be considered:

- Propulsive matching, i.e. what kind and size of engines are needed to provide the propulsion power (or thrust) to compensate the aircraft drag at cruise, and the extra power for acceleration and climbing. Further engine power must be accounted for non-propulsive duties (e.g. avionics, air conditioning...).
- Location matching and ancillary interfacing, i.e., once engine type and size are identified, define the number of engines to share the total duty, the type of structural links to the aircraft frame, and other interfaces (fluid lines, electrical connections...). Current turbofan location is in nacelles under the aircraft wings, with wing structure below main deck in the airframe, and fuel tanks integrated in the wings. In the past, there were three-engine aircraft, with one of the engines mounted in the aft fuselage, but nowadays most aircraft have just 2 or 4 main engines, always under the wings, what is better because it decreases aerodynamic interference and cabin noise, and enhances mass and force distributions.

Contrary to a family car, aircraft engines are relatively small in comparison with its associated fuel tanks. Among all engines in an airliner, their share of maximum take-off mass is around 5 % (up to 10 % for supersonic aircraft). However, fuel in an airliner may take up to 40% of take-off mass.

Aerodynamic drag and required propulsion power

Let us focus on aircraft at cruise (where it sits most of the times). The force balance in the horizontal and vertical direction are $F=D$ and $L=W$, respectively. Under these circumstances, engine propulsion is needed to overcome aircraft drag, hence, the less drag the less propulsion needed, i.e. less fuel to be loaded (besides smaller engines) and more efficient transportation of payload (people or goods) is achieved. If engine thrust is to overcome aircraft drag, some knowledge of how the latter depends on flying conditions must be acknowledged.

Drag is the component of the total force along the free-stream velocity, ultimately due to air viscosity in subsonic flight, since there is no drag in an inviscid flow (D'Alembert's paradox). However, it is customary to consider two types of drag:

- Viscous drag (or parasite drag), due not only to the unavoidable shear at the body surface (predominant in streamlined bodies), but also to the pressure-force unbalance on BL-detached flows (predominant in blunt bodies and detached flows). The shear can be reduced by avoiding or retarding the laminar-to-turbulent transition in the attached boundary layer (BL). The pressure-force unbalance can be avoided or minimised by making the body streamlined, and by retarding BL-detachment, if any.
- Non-viscous drag (or potential drag), which may have two origins:
    - On 3D lifting bodies, induced drag, due to flow deflection. Also named vortex drag because lift can be explained by vortex lines deflecting downwards the otherwise axial flow. An aerofoil (a wing of infinite span and constant section would produce no induced drag. Induced drag can be diminished by using a high-span tapered wing (an elliptic-shaped wing produces the best vortex distribution; a rectangular wing produces much more severe wingtip vortices than a tapered wing).
    - On transonic and supersonic flight, perturbations travel to the far field as shock waves almost undamped. Wave drag may be almost 80 % of total drag at transonic speed (drag divergence). Wave drag can be reduced by using a swept wing, and by producing wave interference from different sources (wing and body).

The above approach to aircraft drag is based on the far fluid field; locally, on a body surface patch, drag is the contribution of both the tangential force (shear) and normal force (pressure) components along the advancing speed direction.

Aircraft drag D at given air speed v_0 is traditionally analysed in terms of the drag coefficient, $c_D \equiv D/(½\rho v_0^2 A)$, where ρ is air density and A wing area (horizontal projection). At that speed, drag has an almost constant parasitic component (as explained above), c_{D0}, plus a lift-induced component that happens to depend quadratically on lift, $c_{Di}=kc_L^2$, where $c_L \equiv L/(½\rho v_0^2 A)$ is the lift coefficient. An explicit dependence, valid for elliptic wings, is:

$$c_D = c_{D0} + \frac{c_L^2}{\pi \Lambda} \tag{12}$$

where Λ is the wing slenderness (total span divided by mean chord). On a commercial airplane, typical values for cruise are c_{D0}=0.02 and c_{Di}=0.01 (i.e. 60 % parasitic and 40 % induced drag). For comparison, $c_L=c_{L0}+2\pi\alpha$, and a typical value is c_L=0.5 ($c_{L\max}$=2).

When dimensional values are considered, we see that parasite drag, $D_0=c_{D0}(½\rho v_0^2 A)$, grows proportional to v_0^2, and that induced drag decreases with v_0^2 because $L=W$=constant implies that $c_L=L/(½\rho v_0^2 A)$ decreases quadratically with speed, and $D_i=c_{Di}(½\rho v_0^2 A)=(c_L^2/(\pi\Lambda))(½\rho v_0^2 A)$. This drag dependence, and the corresponding propulsion power required, $\dot{W}=Dv_0$, are sketched in Fig. 14.

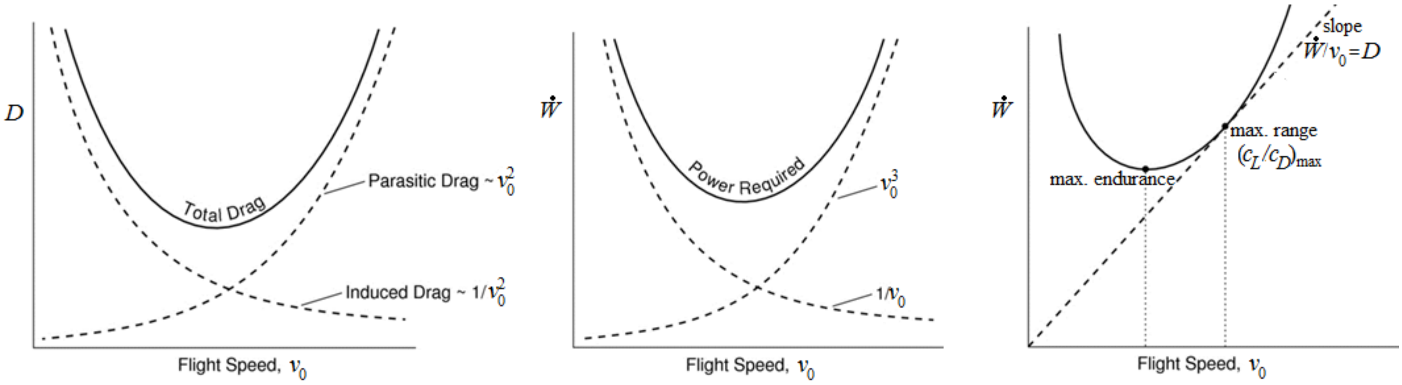


Fig. 14. a) The two terms in aircraft drag, D, (induced and parasitic). b) Propulsion power required ($\dot{W}=Fv_0=Dv_0$). c) Conditions for maximum endurance (minimum required power), and for maximum range (minimum drag for given lift, i.e. maximum aerodynamic efficiency, $(L/D)_{max}$).

Minimum drag occurs at the condition of maximum lift-to-drag ratio; $d(c_L/c_D)/dc_D=0 \rightarrow c_L=(\pi\Lambda c_D)^{1/2}$, and $c_{D,\text{minD}}=2c_{D0}$. Hence, the flight-speed values for minimum drag and for minimum power are, respectively:

$$v_{0,\min D}=\sqrt{\frac{W}{\frac{1}{2}\rho A c_{L,\min D}}}=\left[4\left(\frac{W}{A}\right)^2\frac{1}{\rho^2 c_{D0}\pi\Lambda}\right]^{\frac{1}{4}},\quad v_{0,\min\dot{W}}=\left[\frac{4}{3}\left(\frac{W}{A}\right)^2\frac{1}{\rho^2 c_{D0}\pi\Lambda}\right]^{\frac{1}{4}}\tag{13}$$

where W/A is the wing load (aircraft weight divided by projected area of wings), with typical values of 7 kPa in large airliners, 3 kPa in fighters, and 0.5 kPa in the smallest aircraft. Minimum power occurs when $c_L=(3\pi\Lambda c_D)^{1/2}$, and $c_{D,\text{minD}}=4c_{D0}$, and the flight speed is $3^{-1/4}=76$ % of that for maximum range.

We will see now that maximum endurance (time aloft) occurs when the minimum power is used, whereas maximum range (distance travelled) is obtained when the aircraft is flown at maximum aerodynamically efficient: $(c_L/c_D)_{max}=(L/D)_{max}$. By the way, advance speed is proportional to aerodynamic efficiency and to engine specific power in the way:

$$\left.\begin{array}{l}\dfrac{W}{F}=\dfrac{L}{D}\\ \eta_{tp}\equiv\dfrac{Fv_0}{\dot{m}_f h_{LHV}}\end{array}\right\}v_0=\eta_{tp}\frac{\dot{m}_f h_{LHV}}{W}\frac{L}{D}=\eta_{tp}\frac{\dot{m}_f h_{LHV}}{m_{eng}g}\frac{L}{D}\frac{W_{eng}}{W}\tag{14}$$

For instance, to get a cruise speed of v_0=250 m/s for an airliner with aerodynamic efficiency L/D=17, a power plant is needed such as with global propulsive efficiency of η_{tp}=0.4, engine specific power of $\dot{m}_f h_{LHV}/m_{eng}$ =8 kW/kg, and total engine mass share of 5 % (e.g. for A380 with MTOM of 560 t, four Trent-900 engines of 6.4 t each, burning $\dot{m}_f$ =1 kg/s each).

Recall, however, that engines must provide propulsion power not only to overcome aircraft drag, but to accelerate and climb (besides other non-propulsive duties like powering avionics, air conditioning system, and so on). Typical maximum thrust to aircraft weight ratio are F/W_{craft}=0.4 for executive jets (and Concorde), F/W_{craft}=0.3 for twin-engine airliners, and F/W_{craft}=0.2 for wide-bodies airliners. A jump in propulsion power was required by wide-body aircraft like B747 (in service since 1970), so that the main manufactures (Pratt & Whitney, General Electric, and Rolls Royce) had to build more powerful engines, and the modern high-bypass high-maximum-temperature turbofan was developed (JT-9D, CF-56, and RB-211, respectively, chronologically made available in that order).

Range and endurance: Breguet's equation

Breguet's equation relates the amount of fuel, m_f [kg], and aircraft range, R [m] (we do not use L for range to avoid confusion with lift). Notice that endurance (time on air) is in this case $\Delta t_{flight}=R/v_0$. Breguet's equation is an integrated form of the variation of aircraft mass, m, with distance travelled, $dm/dx=(dm/dt)/v_0=-\dot{m}_f/v_0$, where v_0 is the advancing speed (assumed constant) and $\dot{m}_f$ the fuel mass-flow-rate. Dividing by total mass $m=W/g$, substituting $W=L=LF/D$, rearranging, and, in terms of the global propulsion efficiency, $\eta_{tp}\equiv Fv_0/(\dot{m}_f h_{LHV})$, we get:

$$\frac{dm}{m}=\frac{-\dot{m}_f}{m}\frac{dx}{v_0}=\frac{-\dot{m}_f}{W/g}\frac{dx}{v_0}=\frac{-\dot{m}_f}{F/g}\frac{D}{L}\frac{dx}{v_0}=\frac{-1}{\eta_{tp}h_{LHV}/g}\frac{D}{L}dx \tag{15}$$

which, integrated from the start of the cruise phase (usually approximated as the take-off position) to the end of the cruise phase (usually approximated as the destination location), yields Breguet's equation:

$$\ln\frac{m_{fin}}{m_{ini}}=\frac{-g}{\eta_{tp}h_{LHV}}\frac{D}{L}R \quad\Rightarrow\quad \frac{m_f}{m_{TO}}=1-\exp\left(\frac{\frac{-gR}{\eta_{tp}h_{LHV}}}{\frac{L}{D}}\right) \quad\Rightarrow\quad R=\frac{\eta_{tp}h_{LHV}}{g}\frac{L}{D}\ln\frac{m_{TO}}{m_{TO}-m_f} \tag{16}$$

often written in terms of the thrust-specific fuel consumption, TSFC=$c_{sp}=\dot{m}_f/F=v_0/(\eta_{tp}h_{LHV})$, or its inverse, the specific impulse, $I_{sp}=F/(\dot{m}_f g)=1/(g\cdot c_{sp})$; namely:

$$\begin{aligned} R&=\frac{V}{g\cdot c_{sp}}\frac{L}{D}\ln\frac{m_{TO}}{m_{TO}-m_f}, \quad \frac{m_f}{m_{TO}}=1-\exp\left(\frac{-g\cdot R c_{sp}/v_0}{L/D}\right)\\ R&=v_0 I_{sp}\frac{L}{D}\ln\frac{m_{TO}}{m_{TO}-m_f}, \quad \frac{m_f}{m_{TO}}=1-\exp\left(\frac{-R/(v_0 I_{sp})}{L/D}\right)\end{aligned} \tag{17}$$

Several conclusions can be drawn from Breguet's equation:

- Range is proportional to propulsion efficiency (η_{tp}, or I_{sp}, or $1/c_{sp}$).
- Range is proportional to aerodynamic efficiency (L/D).
- Range increases logarithmically with the load of fuel, m_f.
- Range increases with fuel specific energy, although in practice this is a constant for all kerosene fuels, h_{LHV}=43 MJ/kg.
- Best estimate of mean weight for cruise is the harmonic mean: $W_{mean}=\sqrt{W_{ini}W_{fin}}$.
- Maximum range requires maximum L/D, or $(c_L/c_D)_{max}$, which, with the aircraft polar, $c_D=c_{Do}+kc_L^2$, means $c_{D0}=kc_L^2$ (deduced by equating $dc_D/dc_L=c_D/c_L$). Hence, cruise speed for maximum range is:

$$v_{0,R_{\max}} = \sqrt{\frac{L}{\frac{1}{2}\rho A c_L}} = \sqrt{\frac{W}{\frac{1}{2}\rho A\sqrt{\frac{c_{D_0}}{k}}}} \overset{\text{ellip}}{=} \sqrt{\frac{W}{\frac{1}{2}\rho A\sqrt{\pi\Lambda c_{D_0}}}} \tag{18}$$

which depends on aircraft data (W, A, Λ, and c_{D0}) and on air density ρ. The latter explains why aircraft should fly the higher the better (lowest ρ), limited to flight level ceiling (FL_{max}<5 km in propeller propulsion, limited by propeller efficiency, FL_{max}<13 km in turbofan propulsion to avoid drag divergence at M>0.9; in other aerospace cases limited by structural resistance of the pressurised shell).

- Breguet's equation also allows computing the amount of fuel necessary to cover a desired range (but recall again that this is only the fuel needed for propulsion, and only during cruise; fuel is also needed to take-off, to avionics and lighting, to air conditioning...).
- Breguet's equation in aeronautics, (17), is similar to Tsiolkovsky's equation in astronautics. $\Delta v=-I_{sp}\ln(m_{fin}/m_{ini})$.

For short-range flight where m_f<<m_{TO}, range is proportional to mass of fuel, $R=(\eta_{tp}h_{LHV}/g)(L/D)(m_f/m_{TO})\sim(0.2\cdot(43\cdot10^6)/9.8)\cdot0.2\cdot(15)\cdot(m_f/m_{TO})=(1300\text{ km})\cdot(m_f/m_{TO})$. However, airliner flight does not allows such a simplification, since fuel weight is the largest share; take-off weight is 35..40 % fuel, 30..35 % equipment, 15 % structure, 10 % passengers and luggage (payload), and 5 % for all the engines; fuel-usage share is: 2 % take-off, 10 % ascent, 75 % cruise, 2 % descent, 1 % landing, plus 10 % reserve.

Exercise 1. Determine the cruise range for a Boeing 747-100 flying at 9150 m altitude and Mach 0.8, with the following data. Maximum take-off weight W_{TO}=3286 kN, maximum fuel weight of W_f=1442 kN, fuel consumption on take-off and ascent $W_{f,TA\&AS}$=74 kN, fuel consumption on descent and landing $W_{f,DS\&LA}$=49 kN, fuel reserve $W_{f,RES}$=130 kN. Reference wing area A=511 m^2, wing span b=59.8 m, zero-lift drag coefficient of c_{D0}=0.02, and lift-induced coefficient $k=1/(\varepsilon\pi\Lambda)=0.065$ (with ε=0.7 and wing slenderness $\Lambda=b^2/A=59.8^2/511=7.0$). The airliner is powered by four Pratt and Whitney JT9D-7A turbofans with a cruise TSFC of 0.0694 kg/(h·N).

Sol.: At 9150 m altitude, ISA-values of air pressure, temperature and density are: p=30 kPa, T=229 K, ρ=0.459 kg/m^3; sound speed is $c=(\gamma RT)^{1/2}=(1.4\cdot287\cdot229)^{1/2}=303$ m/s, so that flight speed is $v_0=cM=303\cdot0.8=243$ m/s. We use Breguet's equation (17) accounting for the data c_{sp}=0.0694 kg/(h·N)=19.3·10^{-6} (kg/s)/N, or $I_{sp}=1/(g\cdot c_{sp})$=5300 s:

$$R=\frac{v_0}{g\cdot c_{sp}}\frac{L}{D}\ln\frac{m_{TO}}{m_{TO}-m_f}=\frac{v_0}{g\cdot c_{sp}}\frac{c_L}{c_D}\ln\frac{W_{ini}}{W_{fin}}=\frac{243}{9.8\left(19.3\cdot10^{-6}\right)}\frac{0.38}{0.029}\ln\frac{3212}{2023}=7800\text{ km}$$

where initial weight at cruise altitude is $W_{ini}=W_{TO}-W_{f,TA\&AS}$=3286−74=3212 kN, final weight at cruise altitude is $W_{fin}=W_{empty}+W_{f,rem}=(W_{TO}-W_f)+W_{f,DS\&LA}+W_{f,RES}$=(3286−1442)+49+130=2023 kN, $c_L=2W_{mean}/(\rho v_0^2A)$=0.38 (with $W_{mean}=(3212\cdot2023)^{1/2}$=2549 kN), and $c_D=c_{Do}+Kc_L^2$= 0.02+0.065·0.38^2=0.029. Global propulsion efficiency is $\eta\equiv Fv_0/\left(\dot{m}_f h_{LHV}\right)=v_0/(c_{sp}\cdot h_{LHV})$= 243/(19.3·43)=0.29.

Take-off power

From the different flying phases (take-off, climb, cruise, descent, and landing), take-off is the most demanding in terms of propulsion power, although cruise is the most important for overall propulsion energy and efficiency. Lift-off has the same meaning as take-off, but the latter is preferred for normal airplanes, and lift-off for rockets.

Engine excess power is defined as the capability to increase the kinetic and potential energy of the aircraft, i.e., the total power $\dot{W}_{\text{total}}$ an engine system can produce minus the power used to sustain its steady motion, $\dot{W}_{\text{steady}}$. Excess power (per unit mass of aircraft, m_{AC}) is:

$$\frac{\dot{W}_{\text{excess}}}{m_{\text{AC}}}=\frac{\dot{W}_{\text{total}}-\dot{W}_{\text{steady}}}{m_{\text{AC}}}=\frac{Fv_0-Dv_0}{m_{\text{AC}}}=\frac{\mathrm{d}}{\mathrm{d}t}\frac{E_p+E_k}{m_{\text{AC}}}=\frac{\mathrm{d}}{\mathrm{d}t}\left(gz+\frac{v_0^2}{2}\right) \tag{19}$$

After the brakes are off at the runway head, engine thrust is mainly used to accelerate the aircraft, up to a take-off speed for airliners of $v_{0,\text{TO}}$=70 m/s (250 km/h), so low for lift, that additional lifting devices must be used (flaps, which allow flying at low speed by extending the wing polar, increasing c_L, what lowers the stall speed, but increasing the drag, c_D, too. After take-off, thrust is mainly used to climb, with a minimum climb angle of 4º (climb gradient of 7 %; tan4º=0.07) at least for the first 1000 m in altitude (the typical pitch angle may be 20º, corresponding to some 15º angle of attack over 4º ascent path).

Effect of altitude, flight speed, and revolutions, on engine performances

Atmospheric conditions change all the time and from place to place, but a simple representative model is used in aeronautics since the 1920s: the International Standard Atmosphere (ISA), which assumes dry air of uniform composition, ideal gas model, and approximates mid-latitude average air conditions with sea-level values of T_0=288.15 K and p_0=101.325 kPa, and a temperature lapse rate of $\Gamma\equiv-\mathrm{d}T/\mathrm{d}z$=6.5 ºC/km up to 11 km and then Γ=0 up to 20 km; i.e.:

$$\begin{aligned}&\text{If } 0\le\frac{z}{\text{km}}\le 11 \quad \frac{T_{\text{ISA}}(z)}{T_0}=1-\frac{\Gamma}{T_0}z,\quad \frac{p_{\text{ISA}}(z)}{p_0}=\left(1-\frac{\Gamma}{T_0}z\right)^{\frac{g}{\Gamma R}},\quad \frac{\rho_{\text{ISA}}(z)}{\rho_0}=\left(1-\frac{\Gamma}{T_0}z\right)^{\frac{g}{\Gamma R}-1}\\&\text{If } 11<\frac{z}{\text{km}}\le 20 \quad \frac{T_{\text{ISA}}(z)}{T_{11}}=1,\quad \frac{p_{\text{ISA}}(z)}{p_{11}}=\exp\left(-\frac{g(z-z_{11})}{RT_{11}}\right),\quad \frac{\rho_{\text{ISA}}(z)}{\rho_{11}}=\exp\left(-\frac{g(z-z_{11})}{RT_{11}}\right)\end{aligned} \tag{20}$$

where $\rho_0=p_0/(RT_0)$=1.225 kg/m^3, g=9.80665 m/s^2, R=287.06 J/(kg·K), and T_{11}=217 K, p_{11}=27 kPa, and ρ_{11}=0.36 kg/m^3, are the corresponding values at z_{11}=11 km.

Flight speed is commonly evaluated in terms of Mach number, $M\equiv v/c$, where c is the local speed of sound. With the ideal gas model for air ($\gamma\equiv c_p/c_v$=1.40), $c=\sqrt{\gamma RT}$, and at the stagnation points, where relative kinetic energy adds to internal energy, the total values become:

$$\frac{T_t}{T}=1+\frac{v_0^2}{2c_p}=1+\frac{\gamma-1}{2}M^2,\quad \frac{p_t}{p}=\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{\gamma}{\gamma-1}},\quad \frac{\rho_t}{\rho}=\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{1}{\gamma-1}} \tag{21}$$

For instance, when flying at M=1 (only used as short-time transition to supersonic flight), the incoming air has a total temperature 20 % above that of the unperturbed air (outside air temperature, OAT) at that altitude (T_t=1.2·T, p_t=1.9·p, ρ_t=1.6·ρ); e.g. at z=11 km, with ISA model (T_t=217 K, p_t=23 kPa, ρ_t=0.36 kg/m^3), T_t=260 K, p_t=51 kPa, ρ_t=0.60 kg/m^3.

The effects of altitude and Mach number depend on the type of engine. The two extreme cases are the reciprocating engine with propeller, and the turbojet. As a brief summary, during take-off and initial climb, the performance of reciprocating engines is superior to the turbojet engine, but the latter is superior at cruise.

Reciprocating engine with propeller

Shaft power, $\dot{W}_{\text{shaft}}$, is almost independent of flight speed, v_0 (it decreases slightly with v_0), being proportional to engine regime (ω [rad/s], or n [Hz] though usually stated in [rpm]). Propulsion efficiency grows with advancing speed (see Figs. 1 & 2a in Propellers), but can be assumed almost constant at cruise speeds, particularly for variable-pitch propellers (Fig. 2b in Propellers), so that under these circumstances, propulsive power, Fv_0, is also independent of flight speed, and consequently, thrust ($F=\dot{W}_{\text{shaft}}/v_0$) decreases with increasing speed (Fig. 15a).

Shaft power is proportional to density of aspirated air, although a better empirical fit is $\dot{W}_{\text{shaft}}/\dot{W}_{\text{shaft},0}=1.13\rho/\rho_0$, being also almost proportional to engine speed (rpm), governed by fuel throttle in fixed-pitch blades, or by blade pitch in variable-pitch blades. Consequently, thrust increases with engine regime, and decreases with increasing altitude. But if the piston engine is supercharged, then the shaft power remains constant up until the 'critical altitude' after which it declines linearly with density; the 'critical altitude' is that altitude above which the supercharger can no longer supply sea-level-density air.

The unit-power fuel consumption (or break specific fuel consumption, BSFC, ot just SFC), c_{sp}=$\dot{m}_f/\dot{W}_{\text{shaft}}$, is practically independent of altitude and flight speed, depending mostly on engine regime, with aa typical value of c_{sp}=230 g/kWh=0.23 (kg/h)/kW. A typical engine for small general-aviation aircraft like Cessna-172 may be the Lycoming IO-360-L2A, a 120 kW air-cooled four cylinders (horizontal H layout) four-stroke avgas engine of 120 kg, running at 2700 rpm, with 0.13 m bore, 0.11 m stroke, with compression ratio 8.5, and 5.8 L total displacement. Cessna-172 is a 4-persons high-wing aircraft, with a maximum take-off mass MTOM of 1100 kg, has one Lycoming IO-360 engine consuming (230 g/kWh)·(120 kW)=28 kg/h of avgas, driving a 2-bladed fix-pitch metal propeller of 1.9 m in diameter, and cruises at 230 km/h with 1200 km range and a ceiling of 4.1 km.

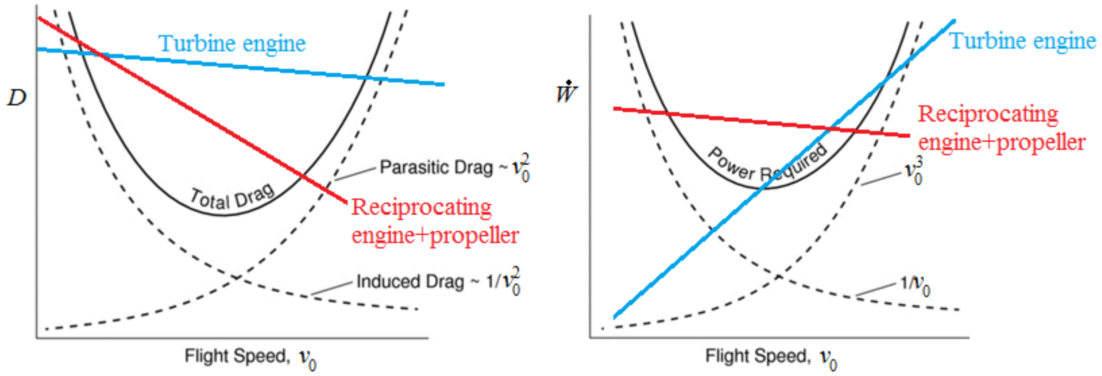


Fig. 15. Available thrust and power of turbojets (blue) and reciprocating engines (red), and their dependence on flight speed. The required thrust and power from aircraft aerodynamics (Fig. 14) is overlaid (available thrust and power must exceed required thrust and power, to accelerate and climb).

Turbojet engine

Jet engines are very sensitive to shaft rotation speed; e.g. a decrease of 10 % in spool spin-rate from nominal value, reduces thrust to 2/3 of nominal value under all conditions (at bench, take-off, cruise, and descent).

For subsonic flight, thrust ($F=\dot{m}_a\left(v_e-v_0\right)=\rho A_0 v_0\left(v_e-v_0\right)$) is nearly independent on flight speed (decreases a little with speed, Fig. 15), because the increase in v_0 is compensated by the decrease of $v_e - v_0$. Consequently, propulsive power, Fv_0, is almost linearly proportional to flight speed (Fig. 15).

Thrust is almost proportional to incoming air density ($F=\dot{m}_a\left(v_e-v_0\right)=\rho A_0 v_0\left(v_e-v_0\right)$), so that $F/F_0=\rho/\rho_0$, i.e. thrust at 12 km is about a fourth of the sea-level thrust: $F_{12}=F_{TO}/4$. The specific fuel consumption, TSFC, $c_{sp}=\dot{m}_f/F$, is practically independent of altitude because both terms (numerator and denominator) are proportional to density, but TSFC grows with flight speed because more power is required and consequently more fuel rate.

For supersonic flight, thrust increases with flight speed roughly as $F/F_1=1+1.18\cdot(M-1)$, whereas TSFC is practically independent of speed and of altitude.

Fuel tank inertization

For years, nitrogen has been used to render inert the headspace in combat aircraft fuel tanks. In 1996, the crash of TWA flight 800 brought the issue of explosive fuel vapours to the forefront for commercial aviation as well. Traditional inertization systems are based on N_2 pressure cylinders.

A system named On-Board Inert Gas Generation System/On-Board Oxygen Generation System (OBIGGS/OBOGS) was studied by Boeing in 2000. The study established the requirements for nitrogen purge (for fuel tank inertization and cargo compartment fire suppression) and oxygen (for passengers and crew in emergencies).

In 2005 Honeywell got FAA certification for its Nitrogen Generation System (NGS) for new B747s. A membrane with micro-fibres separates oxygen from conditioned bleed air from the engines, and the nitrogen-enriched air is injected into the aircraft's fuel tank to displace oxygen.

Fuel tanks are isolated from the cabin by a fume-proof and fuel-proof enclosure that is ventilated and drained to the exterior of the aircraft. Proper fire protection system on modern aircraft (not only at fuel tanks and engines, but in avionic racks and habitable spaces), include not only fire avoiding measures, but a fire detection system and a fire extinguishing system.

ENVIRONMENTAL IMPACT

Aero-engine operations contribute to degrade the environment by consuming non-renewable fuels, by pollutant exhaust emissions, and by the noise they radiate (leaving aside the impact on source materials to build the engines).

On the other hand, aero engines are exposed to damaging environmental conditions, basically by the ingestion of solids: dust from storms and volcanic eruptions, ice particles (and ice chunks from the deicing system), water ingestion from clouds and from wheel's entrainment in wet runways, debris in the runway, birds flying on the airport, etc. Impact of large solid objects may break off fan blades with catastrophic results (e.g. Concorde accident). Massive ingestion of water or ice particles may cool and put out the flame, a serious incident but which can be recovered by reignition. Ingestion of dust may cause erosion of delicate parts, and even engine stop by solidification in the turbine blades of sand or ash particles melted in the combustion chamber. After the large air-travel disruption in Europe in 2010, any airspace where ash density exceeds 4 mg/m^3 is considered prohibited airspace.

Aviation fuel consumption, emissions

Most engines used for propulsion at land, sea, and air, are thermal engines based on the combustion of a fossil fuel with ambient air, contributing to air pollution. Most aircraft engines work on the open Brayton cycle burning kerosene with excess air, typically 40 kg of air per 1 kg of jet fuel in the burnt stream (i.e. the core stream in a turbofan; additional air flows only through the fan). On the other hand, small aircraft utilized in general aviation use avgas (aviation gasoline) with tetraethyl lead (TEL) to increase the octane number, which adds heavy-metal particles to aviation emissions.

Using dodecene as surrogate of jet fuel, the stoichiometry is $C_{12}H_{24}+18O_2=12CO_2+12H_2O$, and substituting molar-mass values: (168 g of fuel)+(576 g of oxygen)=(528 g of carbon dioxide)+(216 g of water), to which the non-reacting nitrogen in the air should be added, in molar basis 79/21=3.76 times the entry O_2 (or (79·0.028)/(21·0.032)=3.29 in mass basis). Assuming a typical relative air-to-fuel ratio three times that of stoichiometry (λ=3), the molar and mass relations would be: $C_{12}H_{24}+3\cdot(18O_2+3.76\cdot 18N_2)= 12CO_2+12H_2O+36O_2+203N_2$, and substituting molar-mass values: (168 g of fuel)+(1728 g of oxygen) +(5685 g of nitrogen)=(528 g of carbon dioxide)+(216 g of water)+(1152 g of oxygen)+(5684 g of nitrogen); as a check, there are 7581 g in the 258 mol entering the combustor, and 7580 g in the 263 mol exiting, i.e. mass is conserved (within the uncertainty in the calculation), and the amount of gases increases. The molar composition of this gas exhaust is 12/258=4.6 % CO_2, 4.6 % H_2O, 13.7 % O_2 and 77 % N_2, and the mass fractions 528/7580=7.0 % CO_2, 2.8 % H_2O, 15.2 % O_2, and 75.0 % N_2.

The above figures are obtained with the complete combustion model. The actual exhaust of the burnt stream in a turbofan has, in addition to the main gas mixture stated (3.1 kg of CO_2 by 1 kg of fuel, and 1.3 kg/kg of water), some 15 g/kg (10..20 g/kg,) of nitrogen oxides (collectively named NO_x), some 1 g/kg (0.5..2.5 g/kg) of carbon monoxide (CO), some 0.5 g/kg (0.1..0.9 g/kg) of volatile organic compounds (VOC; also known as unburnt hydrocarbons, UHC), and some 0.03 g/kg of solid carbonaceous nano-particles (soot, also known as PM, particulate matter), with sulfur compounds being negligible nowadays. The formation of the different components is due to different causes:

- CO_2 production is due to the fuel used, and is fixed in terms of fuel used: 3.1 kg of CO_2 per 1 kg of jet fuel. It can only be eliminated using a carbon-free fuel like hydrogen. It can only be reduced by increasing global propulsion efficiency. CO_2 concentration is usually measured with non-dispersive infrared (NDIR) spectrometry. CO_2 production contributes to global climate change by increasing the greenhouse effect.
- H_2O production is due to the fuel used, some 1.3 kg of H_2O per 1 kg of jet fuel. This water enters the hydrological cycle with minimum local or global effects, with a residence time of 10..15 days. However, water emission at altitude may have a greenhouse effect by the formation of cirrus clouds from contrails.
- NO_x production is due to oxidation of nitrogen from the air at the high-temperatures found in the primary combustion zone within the engine (in the fuel-lean regions of the diffusion flame at high temperature), remaining in metastable state downstream because of the low decomposition kinetics. NO_x emissions are maxima at maximum load, i.e. at take-off and climbing, with some 40 g of NO_x per 1 kg of fuel, but the total flight average is about 15 g/kg in modern turbofans (no NO_x emissions during descent and idle). NO_x concentration is usually measured by chemiluminescence. Maximum concentration takes place near the end of the combustion chamber because of the time-delay in the formation of NO_2 (NO takes much less time to form), in spite of the maximum temperatures found some 20 % combustor-length downstream of the injectors (it takes some 10 ms for the gas flow to cover a combustion-chamber length of some 0.5 m at 50 m/s). These emissions contribute to acid rain, and smog formation, with typical residence time associated to the water cycle (10..15 days in the atmosphere); they also contribute to ozone layer depletion in the stratosphere. For the same engine rating, NO_x emissions decrease with flight altitudes. On ground applications, NO_x emissions are avoided by using regenerative adsorption filters (changing from an oxygen-rich flow to an oxygen-lean flow, from time to time), or by adding reducers (urea, ammonia, or some fuel), but no clear solution is foreseen for aviation.
- CO and VOC (and soot) emissions depend a lot on operating conditions, being highest at partial load (idle, descent, and taxiing), because of the lower burning temperature of lean mixtures, with values of 15 g/kg of CO and 2 g/kg of VOC at idle, although the total average per flight may be 5 g of CO and 0.5 g of VOC per 1 kg of fuel. For the same engine rating, these emissions grow with flight altitudes, and decrease with ambient temperature (hotter burning). These emissions contribute to global greenhouse effect, and to local chemical pollution. CO concentration is usually measured by NDIR, and VOC (UHC) by Fourier transform infrared (FTIR) spectrometry.

- Soot, i.e. solid carbonaceous particulate matter (PM) is due to the incomplete combustion of hydrocarbons, and is produced by pyrolysis in the fuel-rich regions of the diffusion flame, and by quenching near the walls of the combustion chamber. Soot may clog to solid surfaces, but most of it is entrained in the gas exhaust (about 0.03 g of solids per 1 kg of fuel) as nanometric particles in the 50..5000 nm range size (see Atmospheric particulate matter), and is measured by gravimetric analysis on filters. Maximum soot mass-fraction occurs around mid-length in the combustion chamber (may reach 0.1 g of soot per kg of fuel), but decreases downstream. The smaller particles, PM2.5 (less than 2.5 μm in size) are particularly deadly as they can penetrate deeper into the lungs (it was found that air with 10 $\mu g/m^3$ of PM2.5 cause a 36 % increase in lung cancer). Smoke Number is calculated from the loss in reflectance of a filter paper measured before and after the passage of a known volume of a smoke-bearing sample. At take-off conditions the smoke number may be 4 % after 1 minute.

The emissions index (EI) is the mass of pollutant (CO, HC or NO_x), in grams, divided by the mass of fuel used in kilograms. Emission detectors are dealt with aside. Fortunately, emissions reduction has gone in parallel to economic benefit (lower fuel consumption), and, at least until now, with noise reduction (the major advance in jet engines has been the increasing bypass ratio and decreasing exit jet speed, but more-efficient unducted fans are noisier)..

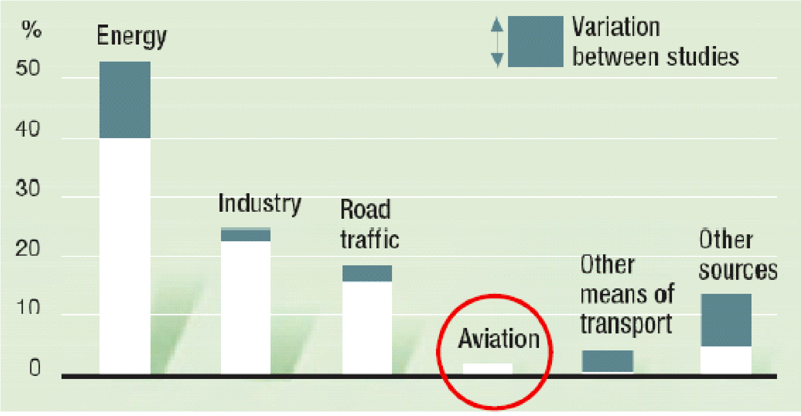
From 1970 to 2010 fuel consumption and emissions per passenger and km in a typical commercial aircraft has been reduced by one third, half and half due to engine efficiency and aerodynamic efficiency (just the winglet devices at the wing-tip save 3 % of fuel), with some 10 % due to more efficient air traffic control. The 10..15 % saving in frame weight due to the substitution of aluminium by carbon composites, will add to fuel savings. Emissions reduction has not been uniform though: CO_2 and H_2O have decreased by 30 %, unburnt hydrocarbons by 70 %, but NO_x by only 5 %.

Commercial aircraft comprise almost 70 % of all aircraft emissions and is the fastest growing segment (the other being military, and general aviation). In 2010 there were 20 000 commercial aircraft, with 95 000 commercial flights per day, some 7 500 aircrafts in the air at a time, transporting $5\cdot10^{12}$ pkm per year (growing almost 5 % annually, doubling every 15 years), and burning $150\cdot10^9$ kg/yr of kerosene (half of that in long-haul routes of >3000 km; only a 20 % in short-haul flights of <1500 km), what produced the following emissions:

- CO_2: $500\cdot10^9$ kg/yr. The contribution of civil aircraft-in-flight to global CO_2 emissions has been estimated at around 2 % (Fig. 16) of the total $30\cdot10^{12}$ kg/yr (the IPCC foresees a rise to 3 % for year 2050). Aviation in Europe represented 14 % of CO_2 emissions of transportation sector in 2011. ACARE has foreseen the halving of specific CO_2 emissions of new aircraft to be built in 2020 (including air-traffic management measures), compared to aircraft built in 2000; this 50 % reduction is mainly attributed to mass saving in the aircraft frame (some 20..25 %) by composites substituting aluminium allows, plus some 15..20 % attributed to engine enhancements, with the 5..15 % remaining due to better air traffic management. But total aviation fuel consumption is expected to increase >50 % from 2000 to 2020.
    - Domestic, short distance (<500 km) emit 257 g/pkm CO_2 (or 259 g/pkm CO_2 equivalent).

- Domestic, mid distance (500..2000 km) emit 177 g/pkm CO_2 (or 178 g/pkm CO_2-equiv.).
- Long distance (>2000 km) emit 113 g/km CO_2 (or 114 g/pkm CO_2-equiv.).
- In average, aviation contributes just above 0.1 kg_{CO2}/pkm.

- H_2O: $180\cdot10^9$ kg/yr. Above 9000 m, contrails may produce cirrus clouds (global warming).
- NO_x: $2500\cdot10^6$ kg/yr.
- CO: $250\cdot10^6$ kg/yr.
- SO_2: $130\cdot10^6$ kg/yr.
- PM: $13\cdot10^6$ kg/yr.



Contribution to man-made carbon dioxide emissions according to different key studies

Fig. 16. Man-made CO_2 emission share (aviation contributes with some 2 %).

The social cost in monetary units of these emissions has been roughly estimated as:

- For CO_2, 0.05 €/kg.
- For NO_x, 20 €/kg.
- For CO, 4 €/kg.
- For HC, 8 €/kg.
- For PM, 200 €/kg.

Although the world average fuel consumption for aircraft is about 0.2 L/pkm, the newest planes like the A380 go down to 0.03 L/pkm, less than the typical 4 L every 100 km of a modern car (with just the driver). By the way, a reason why air travel is so cheap (low cost fares), is that, under international law, aviation fuel for international flights is exempt from taxation, what is a handicap to mitigating emissions, reducing the incentive for airlines to invest in more efficient aircraft (Europe tried in the 2010 to impose a CO_2 tax, but had to desist due to intercontinental refusal). The 'polluter pays' principle should be enforced in all kinds of transport (and any other business).

Aircraft emissions, depending on whether they occur near the ground or aloft, are primarily considered local air quality pollutants, or greenhouse gases, respectively. The largest share of aviation emissions are found to occur during approach, take-off, and climb (the Landing & Take-Off cycle, LTO, below 900 m height). Major aviation pollution at airports are NO_x, PM2.5 (particulates of size <2.5 μm), and VOC. At high altitude, water in engine exhaust may have a greenhouse effect.

Leaded gasoline, used in high-compression internal-combustion engines since 1920s, has been phased out from road engines since 1980s because of deadly heavy-metal contamination (and its incompatibility with catalytic converters), but it is still used in aviation (EPA estimates that approximately $60\cdot10^9$ litres of leaded avgas were consumed between 1970 and 2007, emitting approximately $34\cdot10^6$ kg of lead). The most used avgas is 100LL blue, with 0.3..0.6 g/L of TEL. By May 2012, the US Federal Aviation Administration had put a plan in conjunction with industry to replace leaded avgas with an unleaded alternative within 11 years.

The International Civil Aviation Organization (ICAO) sets international standards for smoke and certain gaseous pollutants for newly-produced large jet engines; it also restricts the venting of raw fuels.

Aircraft noise

Noise is unwanted sound heard, from quiet but annoying to loud and harmful. All sounds are due to density fluctuations in a continuum, which creates acoustic waves propagating in all directions; sound intensity from a point source decreases with the square of the distance (i.e. at double distance 6 dB less), and is proportional to air density (no sound propagates under vacuum). Hearing is the sensation detected by the human ear caused by sound, basically restricted in intensity to $>20\cdot10^{-6}$ Pa (rms), and in frequency from 20 Hz to 20 kHz. In terms of radiated power, whispering generates about 10^{-9} W of acoustic power, shouting generates about 10^{-3} W, and a large turbofan at take-off some 10^3 W.

Noise levels (acoustic intensity, in general) can be measured from the amplitude (or related value like the root-mean-square) of any of the fluctuating variables: fluid-particle displacement, speed, acceleration, pressure, temperature, etc., the most used being sound pressure level (SPL), although sound power level (SWL or PWL), and sound intensity level (SIL), may be used too. All these 'levels' are made dimensionless with a reference pressure value, p_{ref} (ANSI S1.1-1994 established p_{ref}=20 μPa in air, but for underwater acoustics, p_{ref}=1 μPa is used). The measurement of acoustic amplitude (sound pressure) is based on the use of a microphone, which gives an electrical signal proportional to the root mean square of pressure fluctuation at the point in space where it is located. Sound pressure is the root-mean-square of the pressure fluctuations at a point, p_{rms} (should not be called intensity, neither level). Peak-to-peak pressure amplitude is $2\sqrt{2}\approx 3$ times the rms-value.

Sound pressure level (SPL, or just sound level), L_p, is a logarithmic measure of a sound pressure ratio, defined following the pioneering work of G. Bells by:

$$L_p \equiv 10\,\log_{10}\frac{p_{\mathrm{rms}}^2}{p_{\mathrm{ref}}^2} = 20\,\log_{10}\frac{p_{\mathrm{rms}}}{p_{\mathrm{ref}}} = 8.7\,\ln\frac{p_{\mathrm{rms}}}{p_{\mathrm{ref}}} \quad \text{with} \quad \begin{cases} p_{\mathrm{ref}} = 20\cdot10^{-6}\ \text{Pa for air-acoustics} \\ p_{\mathrm{ref}} = \ \ 1\cdot10^{-6}\ \text{Pa for water-acoustics} \end{cases} \tag{22}$$

with non-dimensional units of decibel [dB]. The logarithmic function is based on biological response (Weber–Fechner law of human response to physical stimuli; see also Stevens' power law). The 10 factor is to shift the original Bell's scale to more convenient figures for sound levels (normal sounds are 10..100 dB). The reference pressure p_{ref} was so chosen because the human ear only detects sounds with >20 μPa (audible threshold) at a reference frequency of 1 kHz (we feel pain if p_{rms}>2 Pa). Since the human ear does not have a flat spectral response, an effective sound pressure level is defined by:

$$L_{p,\mathrm{eff}} = \int_{20\ Hz}^{20\ kHz} L_{p,f}\alpha_f \mathrm{d}f \tag{23}$$

where a_f is a standard human-ear frequency-sensitivity relative to 1 kHz standard ($a_{f\,1\,\mathrm{kHz}}$=1). The human ear has a large sensitivity range (7 orders of magnitude in sound pressure, p_{rms}, and 13 orders of magnitude in sound power, $I \equiv \dot{W}/A = p_{\mathrm{rms}}^2/(\rho c)$, where I is sound power intensity, i.e. power per unit frontal area, ρ the density of the propagating medium, and c its speed of sound, around 340 m/s in air and 1500 m/s in water). Some typical noise levels are presented in Table 1.

Table 1. Some typical sound levels in air at 1 kHz (with the receiver at 1 m from the source).

Source	Sound Pressure Level L_p [dB]	Sound Pressure p_{rms} [Pa]	Sound Power Intensity I [W/m^2]
Imperceptible	<0	$<20\cdot10^{-6}$	$<10^{-12}$
Modern air conditioning / computer	30	$0.6\cdot10^{-3}$	10^{-9}
Normal quiet conversation (at 1 m)	40	$2\cdot10^{-3}$	$10\cdot10^{-9}$
Maximum allowed sleep disturbance	50	$6\cdot10^{-3}$	$0.1\cdot10^{-6}$
Maximum allowed daytime disturbance	70	0.06	$10\cdot10^{-6}$
Disco (rock concert)	100	2	0.01
Accelerating motorcycle	110	6	0.1
Jet engine at take-off (receiver at 100 m)	120	20	1
Pain / pneumatic hammer at 1 m	130	63	10
Jet engine at 3 m / Siren at 10 m	140	200	100
Damage (irreversible)	150	630	1000
Rocket launch equipment acoustic tests	165	3600	$31\cdot10^{3}$
Shuttle launch*	190	63 000	$10\cdot10^{6}$

*Largest man-made continuous sound (130 dB at 3 km, 100 dB at 40 km; Orlando is 84 km West KSC).

In terms of frequency, the human ear has a sensitivity range of 3 orders of magnitude (from 20 Hz to 20 kHz), usually divided in octaves; an octave is the frequency interval between one sinusoidal waves and another with half or double its frequency (i.e. there are 10 octaves in the 20 Hz to 20 kHz range).

Aircraft noise causes discomfort to flying passengers and crew, and especially to people living close to airports, which are exposed to it regularly and without any advantage.

Several kinds of noise sources can be considered in a flying aircraft:

- Shear noise, from the velocity gradients in a steady flow around objects (fixed or rotating). This noise originates at the frame surface and wake (mainly around the wings), and is called aerodynamic noise, taking place even when gliding. Aerodynamic noise is proportional to aircraft speed, and is greater on blunt objects (e.g. landing gear) and slots (e.g. clearance gaps in high-lift device).
- Jet noise, from core and fan streams having a large velocity gradient with the environmental air, generating turbulent eddies. Jet noise is proportional to relative jet speed to the eight power. It seems that the small-scale turbulence is the dominant source of noise in subsonic jets (but large-scale eddies are the dominant source in supersonic jets). A serrated trailing edge on the fan nozzle is used in B787 to reduce jet noise.

- Rotodynamic noise, which may be internal to a ducted jet engine (from the wake of rotor blades being periodically cut by the stator blades, also known as blade-vortex interaction, BVI), or external to the engine (from the wake of the rotor blades being periodically cut by other parts of the aircraft, as in propeller's aircraft and helicopters). Rotodynamic noise is proportional to rotor spin-rate.
- Combustion noise, from the sudden fluid expansion caused by energy release.
- Mechanical noise, due to vibrations in machinery (piston engine, pumps, gears...).
- Shock noise in supersonic flight. All supersonic airplanes produce two sonic booms, created by both the nose and tail shock waves, but they cannot be resolved by the human ear because they happen so close to each other that you hear them as one single sound. But the space shuttle was so large, 37 m, that both sonic booms were distinguishable when the shuttle was approaching the landing site (they were received about one-half second apart).

Engine noise can be categorized as either tonal noise (which occurs at discrete frequencies, basically harmonics of the shaft spinning rates, and vortex shedding at the wing trailing edge), and broadband noise (which is composed of random and uncorrelated pressure fluctuations over a broad range of frequencies, basically due to turbulent fluctuations).

Relative importance of noise sources depend on aircraft type. In piston-engine planes, major noise sources are mechanical (inside the engine) and aerodynamic (at the propeller). In helicopters with piston engine the main noise source is the engine, but in turbine engines the main noise contributor are the rotors. In turbofans it depends on the flight phase:

- At taxiing, rolling on the runway, the major fraction of acoustic power emitted by a modern turbofan engine is rotodynamic (generated by interaction of the blade wakes and the stator blades).
- At take-off and ascent, the share of jet-generated noise in modern turbofans is comparable to rotodynamic noise, but dominates in the case of low bypass turbofans.
- At cruise, the major noise source is rotodynamic, with a large component of mechanical noise from air conditioning inside the cabin.
- At descent and landing, the main source is aerodynamic noise at the airframe. With landing gear and flaps deployed, airframe noise increases some 10 dB.

Jet noise is generated outside the engine, by the turbulent mixing of the exhaust jet with the air around, within an axial distance of up to ten diameters, with a noise power proportional to the eighth power of relative speed. Subsonic exhaust in turbofans has significantly reduced jet noise from early turbojets. Furthermore, there is a tendency to incorporate acoustic barriers at the inner surface of the fan duct (before the rotor and after the stator).

Noise levels standards are set by ICAO and monitored in airports at three basic locations on ground: 2 km backwards of runway beginning, 6.5 km forwards of runway beginning, and 450 m laterally to point where aircraft are 300 m above ground. Because of noise, night flying restrictions apply in most large airports. The ultimate goal in aviation noise reduction is to go down to 60 dB (typical urban environment) outside the airport perimeter.

Significant noise is defined as Day Night Average Sound Level (DNL) of 65 decibels (dB). The number of people exposed to significant noise levels was reduced by approximately 90 % between 1975 and 2000, although the frequency of disturbances has increased with the number of operations.

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