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Lecture Notes on

Propulsion Mechanics

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Prologue

The universe is a vast ocean of mysteries, ever-changing phenomena, and forces that shape our reality. Among these forces, one of the most fundamental and captivating is that of propulsion. Throughout the ages, humanity has sought to harness this force to conquer the skies and space, to travel to unexplored worlds, and to challenge the limits of human understanding.

This course handout will take you on a fascinating journey into the heart of *propulsion mechanics*, a discipline that transcends the boundaries of science, engineering and innovation. Whether you are a curious student looking to understand the fundamentals or a seasoned engineer looking to push the boundaries of technology, this book is designed to guide you through the complexities of propulsion with clarity and precision.

Propulsion mechanics is much more than just a collection of equations and theories. It is a science that allows us to transform the raw power of nature into controlled movement, turn dreams of exploration into reality, and rise above the boundaries of our planet. Throughout the chapters, you will discover the basic principles of propulsion, from jet engines to thrusters, including space vehicles, supersonic planes, and rockets. You will also explore the latest technological advances and upcoming challenges for the future of propulsion.

Prepare to be amazed by the power of *propulsion mechanics*, to be inspired by the discoveries that have shaped our history, and to be motivated to play a role in the next era of space exploration and energy.

The journey begins here. Welcome to the world of *propulsion mechanics.*

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Summary

CHAPTER I
PRINCIPLE OF PROPULSION BY REACTION

1-1 Introduction :

The purpose of this chapter is to introduce the principle of jet propulsion and how it was understood and applied historically. Before jet and rocket propulsion became quite a common concept, the principle of jet propulsion had seemed mysterious. There was a lot of confusion over such questions as:

- Could a jet plane or rocket exert thrust by discharging into a vacuum (without an atmosphere to "counter-push" it)?
- Could a rocket be propelled at a speed much higher than the speed at which the jet escapes the ejection nozzle?
- Could a rocket lift a loaded weight far into space?

Experiment has now decisively answered such questions and the theory of jet propulsion (which can be said to have taken centuries to develop) is now so well established that today it appears accepted beyond doubt. Nevertheless even today the theory deserves careful, careful and critical examination because it is the central point for understanding the behavior of real engines and their potential for improvement. The implications of the theory for various kinds of engines are given in some detail in later chapters (although here we begin with an elementary explanation of how jet thrust is quantitatively related to fluid flow).

Within the framework of Newtonian mechanics and classical thermodynamics, this course aims to provide a unified explanation not only of how jet propulsion devices work, but also of why they developed in certain ways and that the keys are to future developments. An adequate understanding of mechanics and thermodynamics is essential for the design and development of all kinds of jet propulsion engines.

The rocket and aviation engine designs provide many examples of how it is possible to develop the best technology from an economic point of view. This, not only provides great motivation to the designer for every possible means to build more powerful, efficient, durable and safe engines; it also motivates the development of new technology and explains why in recent decades there have been enormous improvements in design techniques as well as materials and manufacturing. The computer has opened up wonderful possibilities for translating basic knowledge, and the results of experiments, into better designs. A study of what is already being achieved and consideration of the reasons behind these events can help us see where future development is likely to be successful.

Before beginning a serious examination of any type of propulsion device, it is logical to consider the characteristic features of various kinds of rocket and turbine engines and why they have their own particular applications and limitations. For example: Since chemical rockets are so powerful, why are they not used to power large passenger planes? Why are other types of rockets more suitable than chemical ones for long space trips? Why is the conventional thruster limited, but still the best choice in its characteristic application? Why is the turbojet decisively superior to conventional propellant for ultrafast flight? What about propeller planes and ramjets? It seems in this chapter, that to answer such questions it is necessary to indicate the preliminary importance of fluid dynamics and thermodynamics.

1-2 Moment and reactive force in a fluid :

An easy way to appreciate the idea of jet propulsion is to consider the tube shown in figure (1.1). Part (a) shows a closed tube of cross section A_i; the internal pressure is p_0 and the external (ambient) pressure is p_a. The internal and external pressure forces in the *x direction* are clearly in balance and no net thrust is exhibited by the internal liquid or external atmosphere on the tube. We now assume that the right half of the tube is instantly removed at time t_0. Figure 1.1 (b) shows the thrust $\wp$ that would be required at time $t_0 + dt$ to keep the left half of the tube stationary. At this moment the fluid has not yet left the tube and the internal pressure on the left end is still p_0. By applying a force balance to the tube, the instantaneous thrust is

$$\wp = A_i (p_0 - p_a) \qquad 1.1$$

before a finite velocity has developed at the exit of the tube. After a small finite time increment, the fluid instantly leaves the tube at an exit velocity u_e, as shown in part (c), while the internal pressure at the left end drops to p. The push is now

$$\wp = A_i (p - p_a) \qquad 1.2$$

and drops very quickly towards zero when $p \to p_a$.

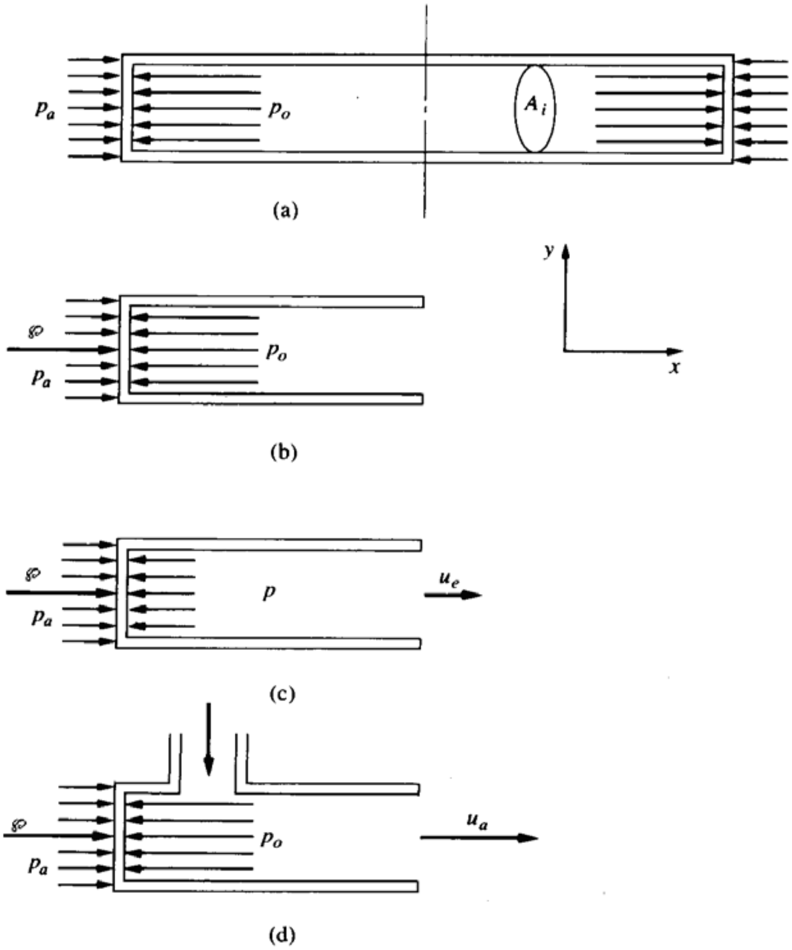


Fig. 1.1 : La propulsion par jet (a) Tube clos; aucune poussée. (b) Poussée instante ℘ résultant du déplacement brusque de la fin droite du tube. (c) Poussée transitoire du jet alors que *p* reste supérieure à p_a. (d) Poussée permanente avec ajout d'air pour maintenir $p_0 > p_a$.

The thrust in the stationary case could be produced, as shown in part (d) of figure (1.1), by supplying the fluid (at the same rate as it leaves the tube) so that the internal pressure remains at p_0 and the push continues to

$$\wp = A_i\,(p_0 - p_a)$$

These estimates ignore the effect of shear forces that the fluid exerts on the wall (tending to reduce thrust), but they illustrate the general idea that if we know the internal and external pressure distributions (and the stresses of shear on the walls) of a device capable of generating thrust, we can calculate the thrust by adding the components of all the forces exerted on all the surfaces.

The difficulty with the general application of this procedure is that the internal geometry of the latter device such as a turbojet is so complex that thrust evaluation would thereby become unacceptably boring. Fortunately, as we will see in what follows, we can estimate the stationary thrust in a much simpler way. Newton's laws can be used to show that, for the steady-flow tube shown in figure (1.1d), the thrust can be estimated by

$$\wp = \dot{m}\,u_e$$

where $\dot{m}$ is the flow rate at which mass exits the chamber and u_e is the mass-averaged exhaust velocity.

The existence of such an equation is quite plausible since we realize that a moment can be created only by the application of a force. This equation, however, is true only if the pressure at the exhaust section is equal to p_a and if external shear forces are unimportant. Under such simplified conditions, it can be seen that, for a given output moment, the thrust is independent of the internal pressure distribution and internal geometry details. It is also independent of the ambient pressure, p_a. Such an equation is directly applicable to a chemical rocket; here the production of the output moment in an ejection nozzle follows the combustion of a fuel stored in solid or liquid form in the vehicle and having no moment, initially, with respect to the rocket nozzle.

If the burning of rocket fuel is "quasi stationary," the thrust can be estimated by $\dot{m} u_e$ as before; if it is highly unsteady, then a more complicated expression is required, resulting from the application of Newton's laws to rapidly accelerating flow.

So far we have discussed the application of thrusts by a stationary "rocket". What if the rocket is moving quickly? If it travels in an almost straight line at almost constant speed, then an observer traveling with the rocket (and measuring the escape velocity u_e relative to the rocket) would invent the same thrust that we have discussed already, viz. $\dot{m} u_e$. If the rocket itself accelerated extremely quickly, then dynamic corrections to this expression would be necessary.

It is worth noting here that, for a given output moment relative to the rocket, the thrust is independent of the flight speed of the vehicle. Thus the vehicle can be propelled at a speed much higher than u_e.

About two thousand years ago Hero of Alexandria demonstrated the idea of reactive thrust with his steam "turbine", imaginarily depicted in figure (1.2). Steam produced by heating (probably at a near steady rate) and at pressures significantly greater than ambient, flowed tangentially through the nozzles and produced tangential thrust in each. A fundamental explanation of how this could be produced had to wait until the ^{17th} century Newton and his formulation of the laws of dynamics.

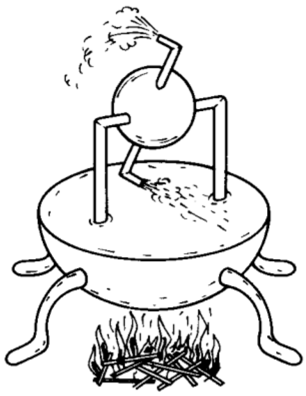


Fig. 1.2 : Eolipile conception of Heron of Alexandria.

It should be noted, however, that Hero and many others; for example, the Chinese with their black powder projectiles dating from the twelfth century manifested the principle of reaction thrust long before Newton explained it. Thus the art of rocket array became established long before the science behind it. Such a situation has not been uncommon in engineering history. It has been said that the development of thermodynamics, for example, owed more to the steam locomotive than the steam locomotive to thermodynamics.

1-3 Rockets :

The solid fuel chemical rocket can be considered the oldest technical development of jet propulsion. In such a rocket the moment at exit is due to the flow of hot gases produced by the rapid combustion of a solid material which contains both the fuel and the oxidant necessary for combustion. The gas pressure inside the combustion chamber may or may not be stable during the combustion period.

The powder appears to have appeared in China around 850 BCE, as an accidental result of alchemists' work on its constituent (potassium nitrate), sulfur and charcoal. A formula for the powder was printed as early as 1044 CE. It was reported that Chinese engineers "first made rockets... Probably around 1150 CE, or soon after that." The first examples of rockets, called by their inventors "running rats," probably each consisted of a bamboo tube filled with powder; on ignition the combustion products escaped at high speed through a small hole in one end of the tube. This is because the tube was fired all over the earth, giving a lot of fun in fireworks displays. It was shortly before someone discovered that, attached to a stick for flight stability, such a rocket could actually fly. The flying models were named "meteoric running rats," the new adjective brilliantly predicting the age of true space flight.

For a period of seven centuries or so, the Chinese, Mongols, Indians, and Arabs worked extensively on the empirical development of solid fuel rockets for military use. In England, William Congreve developed solid fuel rockets with a range of 3000 yards, which were used with questionable accuracy in the War of 1812 between Great Britain and the United States and in other conflicts. By the end of the nineteenth century non-military uses of solid fuel included; whale fishing, signaling and the transfer of lifelines between boats.

1-4 The thrusters :

For propulsion aviation the great advantage of using a propellant rather than a rocket is that most of the fuel does not need to be carried by the vehicle. The airflow rate through the thruster can be two to three times greater as the rate of fuel at which the engine is supplied. Having to carry so much fuel does not mean that the aircraft can travel much greater distances before refueling. A second advantage is so much better propulsion efficiency is possible with a booster than with a rocket. To show the importance of these advantages, we can resort to some simplified and approximate estimates (which will be considerably improved in subsequent chapters.).

As shown in Figure (1.3), the task of the thruster is to accelerate a stream of air passing through it from an approach (arrival) speed u to exit at ejection speed u_e .

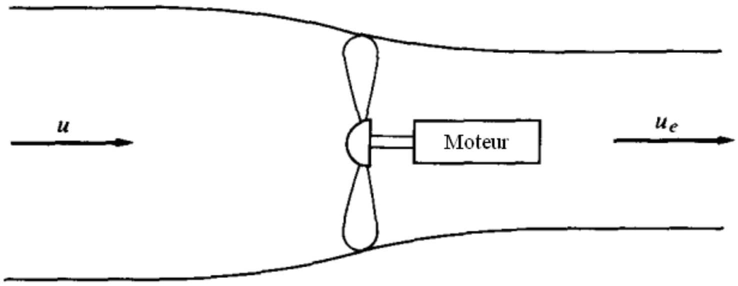


Fig. 1.3 : Accélération d'un courant d'air à travers un propulseur.

The thruster operates in steady state at a moderate speed such that the cross section of the tube decreases as the fluid flow accelerates as it passes through the thruster. Without focusing here on the distribution of pressure forces on the thruster blades, we could show by a careful application of the momentum equation that the thrust developed by the thruster is approximately

$$\wp = \dot{m}_a (u_e - u) \qquad 1.3$$

where $\dot{m}_a$ is the rate of airflow passing through the thruster. The minimum possible energy added to the airstream as it passes through the propeller is the change in kinetic energy. If the engine thermal efficiency is η_e (the ratio of engine power to the rate of consumption of supplied chemical energy), the minimum possible rate of fuel energy consumption will be

$$\dot{E} = \frac{\dot{m}_a}{\eta_e}\left(\frac{u_e^2}{2} - \frac{u_a^2}{2}\right)$$

thus the amount of thrust corresponding to the minimum possible fuel energy consumption rate will be

$$\frac{\wp}{\dot{E}} = \frac{2\eta_e}{u_e + u}$$

A better value η_e might be 2/5. For positive thrust, u_e must be greater than u , so that the maximum possible value of the thrust proportion for the thruster is

$$\left(\frac{\wp}{\dot{E}}\right)_{\substack{\max \\ prop}} \approx \frac{\eta_e}{u} \approx \frac{2}{5u}$$

where u is the flight speed. For the chemical rocket we can write the thrust as

$$\wp \approx \dot{m}_p u_{ef}$$

and the minimum possible energy consumption as

$$\dot{E} \approx \dot{m}_p \frac{u_{ef}^2}{2}$$

in which $\dot{m}_p$ is the fuel consumption rate and u_{ef} is the ejection speed of the rocket nozzle. Most chemical rockets have an appreciable amount of unused thermal energy in the exhaust, so the aforementioned estimate $\dot{E}$ is still quite an approximate value. By combining the equations we can write

$$\left(\frac{\wp}{\dot{E}}\right)_{\substack{\max \\ fusée}} \approx \frac{2}{u_{ef}}$$

For the same (minimum) rate of chemical energy consumption, we can approximately estimate the ratio of propellant and rocket thrusts as

$$\frac{\wp_{prop}}{\wp_{fusée}} \approx \frac{u_{ef}}{5u}$$

For maximum propulsion effect, chemical rockets have high exhaust velocity; for example 5000 m/s is not out of range for an oxygen-hydrogen rocket. By taking this value, we have

$$\frac{\wp_{prop}}{\wp_{fusée}} \approx \frac{1000}{u}$$

where u is the aircraft flight speed. So for a slow aircraft (say, 100 m/s or 360 km/h), the booster would develop perhaps 10 times the thrust (per unit of energy consumption) of the rocket. By using ambient air as most of its "fuel", the propeller has the advantage that it can adapt to minimize wasted kinetic energy in propelling a vehicle at a given speed. With the ejection velocity set high (to minimize wasted thermal energy), the rocket would inevitably waste a lot of kinetic energy in propelling a low-speed (slow) vehicle.

1-5 Thrust and efficiency :

Quite general equations of the thrust and efficiency of jet engines in air can be derived from the laws of momentum and energy without needing to consider the detail of the internal mechanisms of particular engines. Consider, for example, the generalized thrust-producing device shown in Figure 1.4, as observed from a stationary position with respect to the device. While this device resembles a ramjet or turbojet, the results obtained will be applicable to the calculation of thrust produced by any jet engine in air with a simple exhaust

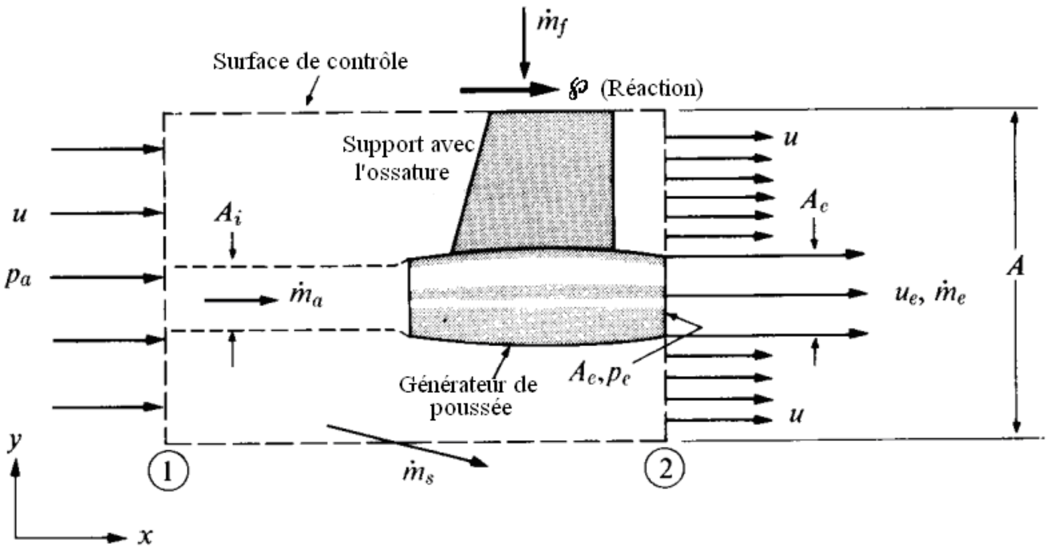


Fig. 1.4 : Schéma général du dispositif de production de poussée.

In a previous section we derived thrust from a stationary jet engine; we now generalize this result to show the influence of flight speed on thrust. An additional effect of flight speed (which is the drag at the outer surface of the engine casing) is of secondary importance in this discussion, so for simplicity we will assume that the flow rate external to the engine is reversible.

In figure (1.4); a control surface is assumed to have passed through the fuel outlet section at ② and ① and extends upstream to . The side surfaces of the control volume are parallel to the upstream (flight) speed $\vec{u}$ and away from the thrust device. We also assume that the thrust and conditions at any point in the control volume do not change with time.

1-5.1 The thrust equation :

The reaction to the thrust $\wp$ transmitted by the support (support) of the frame is shown in figure (1.4). In this sense we can define the engine thrust as the vector addition of all the forces on the internal and external surfaces of the engine and the carcass (the framework). The generalized thrust produced by the generator can be taken from Newton's second law of dynamics applied to a steady flow:

$$\sum \vec{F} = \int_{sc} \vec{u}\,\rho(\vec{u} \bullet \vec{n})dA$$

Considering the force and moment components in the *x direction* only, we have

$$\sum F_x = \int_{sc} u_x\,\rho(\vec{u} \bullet \vec{n})dA \qquad 1.4$$

With the assumption of reversible external flow, the pressure and velocity can be assumed constant over the entire control surface, except at the exhaust section A_e of the engine. If the ejection velocity u_e is supersonic, the exhaust pressure p_e may differ from the ambient pressure p_a. The net pressure imposed on the control surface is therefore $+(p_a - p_e)A_e$. The only other force acting on this control volume is the thrust reaction $\wp$. Summing the forces on the control surface that act in the *x direction* , we get

$$\sum F_x = (p_a - p_e) A_e + \wp \tag{1.5}$$

Far upstream at the section ① the air which is entrained into the engine crossing the control surface by the incidence (capture) sector A_i at a flow rate $\dot{m}_a$ given by $\dot{m}_a = \rho\, u A_i$, in which ρ is the ambient density and u is the flight speed. The mass flow crossing the exhaust surface A_e is $\dot{m}_e = \rho_e u_e A_e$. Taking into account the fuel flow $\dot{m}_f$, we have $\dot{m}_e = \dot{m}_a + \dot{m}_f$, Or

$$\dot{m}_f = \rho_e u_e A_e - \rho u A_i \tag{1.6}$$

Now, if we consider the continuity requirement for the control volume overall and assume that the fuel flow comes from outside the control volume, the continuity equation for steady flow is

$$\int_{sc} \rho \vec{u} \bullet \vec{n}\, dA = 0$$

So for the present case we can write it as

$$\rho_e u_e A_e + \rho u (A - A_e) + \dot{m}_s - \dot{m}_f - \rho u A = 0$$

where A is the cross section of the control volume perpendicular to the velocity u and $\dot{m}_s$ is the mass flow of air passing through the side surfaces of the control volume (Fig. 1.4). By rearranging, we arrive at

$$\dot{m}_s = \dot{m}_f + \rho u A_e - \rho_e u_e A_e$$

When we use equation (1.6), the latter equation becomes

$$\dot{m}_s = \dot{m}_f + \rho u (A_e - A_i) \tag{1.7}$$

If the sides of the control volume are far enough from the thrust generator, we can assume that this flow crosses the control surface with a very small velocity in the *y direction* and an essentially stable velocity component in the *x direction* . Thus the momentum carried out by the control volume with this flow is simply $\dot{m}_s u$, and when we take components only in the *x direction* , the right-hand side of equation (1.4) can be written

$$\int_{sc} u_x \rho (\vec{u} \bullet \vec{n}) dA = \dot{m}_e u_e + \dot{m}_s u + \rho u (A - A_e) u - \dot{m}_a u - \rho u (A - A_i) u$$

which is the net momentum flux exiting the control volume in the *x direction* . Using equation (1.7), we can reduce this to

$$\int_{sc} u_x \rho (\vec{u} \bullet \vec{n}) dA = \dot{m}_e u_e - \dot{m}_a u \tag{1.8}$$

If we use equations (1.5) and (1.8), the momentum equation (1.4) becomes

$$\wp = \dot{m}_e u_e - \dot{m}_a u + (p_e - p_a) A_e$$

however, by defining the fuel-air ratio $f = \dot{m}_f / \dot{m}_a$, we have

$$\wp = \dot{m}_a \left[(1+f) u_e - u \right] + (p_e - p_a) A_e \tag{1.9}$$

The term $(p_e - p_a)A_e$ is not zero only if the exhaust jet is supersonic and the nozzle extends its jet towards ambient pressure. This can, however, be a substantial contribution to the thrust if $p_a << p_e$.

Taking into account that in the derivation of equation (1.9) the flow outside the engine was assumed to be reversible. If this is not so, because of significant boundary layer effects such as separation, the actual force transmitted by the supporting frame in Figure (1.4) could be significantly less than that in Equ.(1.9).) will provide. Alternatively, one could consider the effect of aerodynamic friction on the framework and external surface of the engine to be a part of the total drag of the aircraft.

For engines that have two distinct exhaust streams, equation (1.4) must be applied separately to each stream. As we will see, the complementary fuel flows of the turbojet and turboprop engines can be distinguished according to whether they pass through or around the combustion chamber and the engine turbine. Fuel streams can be labeled “hot stream” and “cold stream,” respectively. The thrust equation for such an engine, if we neglect pressure terms, becomes

$$\wp = \left(\dot{m}_{aH} + \dot{m}_f\right)u_{eH} - \dot{m}_{aH}\,u + \dot{m}_{aC}\left(u_{eC} - u\right)$$

where the subscripts *H* and *C* refer to hot and cold flows, respectively. Based on the definition of the fuel-air ratio that is mixed with the fuel $f \equiv \dot{m}_f / \dot{m}_{aH}$, we obtain the thrust equation

$$\wp = \dot{m}_{aH}\left[\left(1+f\right)u_{eH} - u\right] + \dot{m}_{aC}\left(u_{eC} - u\right) \qquad 2.10$$

1-5.2 Performance of turboshaft engines :

In describing aviation engine performance, it is useful to first define several efficiencies and performance parameters. We will now give various definitions and, for simplicity, present representative expressions as they would apply to an engine with a simple fuel flow (i.e., a turbojet or ramjet). The turbofan and turboprop would require slightly more complex expressions, but the qualitative conclusions regarding comparative engine performance would be similar.

Propulsion efficiency . The product of thrust times flight speed $\wp u$ is sometimes called thrust power. A measure of the performance of a propulsion system is the ratio of this thrust power to the rate of production of the kinetic energy of the fuel. This ratio is generally known as the propulsion efficiency η_p, for a single flow of fuel,

$$\eta_p = \frac{\wp u}{\dot{m}_a\left[(1+f)(u_e^2/2) - u^2/2\right]} \qquad 2.11$$

With two reasonable approximations we can considerably simplify this relationship. First, for air-breathing engines in general, f << 1 and can therefore be ignored in equations (1.9) and (1.11) without making a serious error. Second, the pressure term in equation (1.9) is often much smaller than the other two terms, so $\wp \approx \dot{m}_a (u_e - u)$. So

$$\eta_p \approx \frac{(u_e - u)u}{(u_e^2/2) - u^2/2} = \frac{2u/u_e}{1 + u/u_e} \qquad 2.12$$

The thrust equation shows that u_e must exceed u for the right-hand side of equation (1.12) to have a maximum value of unity for $u/u_e = 1$. However, for $u/u_e \to 1$, the thrust per unit mass is practically zero; and of course for finite thrust the engine required would be infinitely large. Thus it is unrealistic to try to maximize the propulsion efficiency of a jet engine, and other parameters are required to evaluate its overall performance.

It is interesting to note that if the fuel of a traction vehicle on the surface of the earth is considered to be the earth itself, then according to this expression the propulsion efficiency of such a vehicle is unity.

Thermal efficiency . Another important performance ratio is the thermal efficiency η_{th} of the motor. For ramjets, single- and dual-flow turbojets it is defined as the ratio of the rate of addition of kinetic energy to the fuel to the rate of total energy consumption $\dot{m}_f Q_R$, where Q_R is the heat of reaction of the fuel. So we can write the thermal efficiency, again for a simple fuel flow, as

$$\eta_{th} = \frac{\dot{m}_a \left[(1+f)(u_e^2/2) - u^2/2 \right]}{\dot{m}_f Q_R}$$

Or

$$\eta_{th} = \frac{\left[(1+f)(u_e^2/2) - u^2/2 \right]}{f Q_R} \qquad 2.13$$

The power produced by a turboprop or shaft turbine engine is largely the mechanical power transmitted by the shaft. In this case the thermal efficiency is defined with respect to other power shaft devices, by

$$\eta_{th} = \frac{\wp_s}{\dot{m}_f Q_R} \qquad 2.14$$

where $\wp_s$ is the power of the tree.

Propellant efficiency . Shaft power is converted into thrust power in an aircraft moving by a propeller. Thruster efficiency η_{pr}, is usually defined as the ratio of thrust power to shaft power, or

$$\eta_{pr} = \frac{\wp_{pr} u}{\mathcal{P}_s} \qquad 2.15$$

where, in this case, $\mathcal{P}_s$ is the part of the thrust due to the propellant. However, since many turboprops derive appreciable thrust from the exhaust of hot gases from their turbines, an equivalent shaft power $\mathcal{P}_{es}$, can be defined such that the product of η_{pr} and $\mathcal{P}_{es}$ is equal to the total thrust power (at a certain arbitrarily chosen flight speed). Thus;

$$\eta_{pr} = \frac{\wp u}{\mathcal{P}_{es}} \qquad 2.16$$

Overall performance . The product of $\eta_p \eta_{th}$, or $\eta_{pr} \eta_{th}$ as applicable, is called the overall η_G yield , and is defined by

$$\eta_G = \eta_p \eta_{th} = \frac{\wp u}{\dot{m}_f Q_R} \qquad 2.17$$

Using equation (1.12), we obtain (for $f \ll 1$)

$$\eta_G = 2\eta_{th} \left(\frac{u / u_e}{1 + u / u_e} \right)$$

Thus the overall efficiency depends only on the speed ratio u / u_e and the thermal efficiency η_{th} , which depends in one way on the speed ratio. The importance of overall efficiency will be demonstrated by an analysis of a simple aircraft range, but first we consider takeoff thrust.

Takeoff Thrust . One of the most important characteristics of a turbine engine installed in an aircraft is its ability to provide static and slow thrust so that the aircraft can take off under its own power (ramjets cannot provide static thrust and therefore are excluded from this discussion). We can derive the static thrust from equation (1.9), ignoring the pressure term and neglecting f with respect to unity:

$$\frac{\wp}{\dot{m}_a} = u_e \text{ (static)} \qquad 2.18$$

Thus the static thrust per unit mass of air is directly proportional to the speed of gas ejection. But another important question arises: For a given fuel flow, how does thrust depend on exhaust velocity? The answer to this question illuminates one of the key differences between single-flow turbojets, turbofans and turboprops and it plays a significant role in the choice of propulsion system for many applications. Using equations (1.13) and (1.18) for the static case, we have (for f << 1)

$$\wp = \frac{2\eta_{th} Q_R \dot{m}_f}{u_e} \quad \text{(static)} \qquad 2.19$$

This equation shows that for a given fuel flow $\dot{m}_f$ and thermal efficiency η_{th}, takeoff thrust is inversely proportional to ejection speed. That is, for a given rate of power consumption, takeoff thrust can be increased by accelerating a greater mass flow rate of air at a smaller exhaust velocity.

The importance of thrust per unit fuel burn rate, other than fuel economy (which is relatively unimportant during takeoff), is that it approximately determines the engine size for a requirement of given thrust. Turbine engines that have been developed for high performance have approximately the same maximum temperature limit and therefore the same fuel-air proportion. Thus equal fuel flow rates imply approximately equal "hot" air flow rates, and it is the hot air flow rate that largely determines the size of the gas generator and engine.

Aircraft range . In many cases the distance or range that an aircraft can travel on a given mass of fuel is an important criterion for the performance excellence of that aviation engine combination. Ignoring the climb (takeoff) and descent (landing) from cruise altitude and assuming that for all aircraft and engine characteristics except total weight are constant during cruise, one can easily obtain an assessment of aircraft range.

In constant-speed flight, engine thrust and vehicle drag are equal, as are vehicle weight and lift. SO

$$\wp = \mathfrak{D} = L\left(\frac{\mathfrak{D}}{L}\right) = \frac{mg}{L/\mathfrak{D}} \qquad 2.20$$

where m is the instantaneous mass of the vehicle, g is the acceleration of gravity, $\mathfrak{D}$ is the drag force, and $L/\mathfrak{D}$ is the lift-to-drag ratio. The thrust power is then

$$\wp u = \frac{mgu}{L/\mathfrak{D}}$$

Using equation (1.17) with this expression, we obtain

$$\dot{m}_f = \frac{mgu}{\eta_G Q_R \left(L/\mathfrak{D}\right)} \qquad 2.21$$

However, since fuel is part of the total aircraft mass m,

$$\dot{m}_f = -\frac{dm}{dt} \quad \text{or} \quad \dot{m}_f = -u\frac{dm}{ds}$$

where s denotes the distance along the flight path. Substituting this expression into equation (1.21), we have

$$\frac{dm}{ds} = -\frac{mg}{\eta_G Q_R (L/\mathfrak{D})} \qquad 2.22$$

If, as an approximation, the denominator $\eta_G Q_R (L/\mathfrak{D})$ is assumed constant, the range s of the vehicle can be obtained by integrating Eq. (2.22) to obtain Breguet's range formula,

$$s = \eta_G \left(\frac{L}{\mathfrak{D}}\right) \ln \frac{m_1}{m_2} \frac{Q_R}{g} \qquad 2.23$$

where the m_1 and m_2 are the initial and final masses of the vehicle, the difference being the mass of the fuel consumed e. Thus the range is directly proportional to the overall efficiency η_G.

The importance of this term prompts us to be interested in its variation with flight speed and engine characteristics. If $f << 1$ and the pressure term in the thrust equation is neglected and we again consider a simple fuel flow for simplicity, it follows that

$$\eta_G = \frac{(u_e - u)u}{fQ_R}$$

and this for u_e, η_G given is maximum for $u = u_e / 2$.

We can show that this can be modified to express the overall efficiency of a dual fuel flow engine. It is only necessary to replace the ejection speed u_e with the ejection speed of the average thrust $\bar{u}_e$, which is defined by

$$\bar{u}_e = \frac{\dot{m}_{aH}(1+f)u_{eH} + \dot{m}_{aC}u_{eC}}{\dot{m}_{aH}(1+f) + \dot{m}_{aC}} \qquad 2.24$$

So to replace the fuel-air ratio f with the term $f / \left[1 + (\dot{m}_{aC} / \dot{m}_{aH})\right]$; this ratio is defined by $f = \dot{m}_f / \dot{m}_{aH}$. Therefore :

$$\eta_G = \left(1 + \frac{\dot{m}_{aC}}{\dot{m}_{aH}}\right) \frac{(\bar{u}_e - u)u}{fQ_R} \qquad 2.25$$

For a given proportion $\dot{m}_{aC} / \dot{m}_{aH}$ and a given exhaust speed u_e, the overall efficiency is approximately maximized for $u = \bar{u}_e / 2$.

We show the reasons why and the ways in which the exhaust speed u_e and the fuel-air ratio f vary with flight speed (or flight Mach number M). Figure (1.5) shows that with certain engine materials and best design configurations, typically η_G increases continuously with flight Mach number.

This does not mean, however, that for maximum range the aircraft must fly at maximum Mach number; The lift-drag ratio $L / \mathfrak{D}$ also depends on the flight Mach number, as shown in figure (1.5). The ratio is typically much lower for supersonic flight than for subsonic flight and the product $\eta_G L / \mathfrak{D}$ peaks at a subsonic flight Mach number. In long-range commercial passenger aviation, 30% to 50% of the aircraft's total direct operating expense may consist of fuel loading costs. Thus there is the great motivation to maximize the range s for a given fuel consumption m_1 - m_2 (i.e., minimizing fuel consumption per unit distance traveled). This is why longer range commercial passenger turbojet aircraft are designed to fly at subsonic Mach numbers. For supersonic flight in which fuel economy is important, Figure (1.5) suggests that the higher the flight Mach number, the better the economy. But maximizing $\eta_G L/\mathfrak{D}$ isn't the whole goal; the higher the flight Mach number, the higher the surface temperature of the aircraft and thus the heavier the structure must be to hold tensions at safe levels. Therefore the proportion m_1/m_2 must also be considered in determining the most economical Mach number of supersonic flight. If we use equation (1.17), we can transform the scope formula from equation (1.23)

$$s=\frac{L}{\mathfrak{D}}\ln\frac{m_1}{m_2}\frac{\wp}{\dot{m}_f}\frac{u}{g}$$

which shows that for a given flight speed, the range of an aircraft is directly proportional to the ratio of thrust and fuel flow. This proportion is so important that its inverse, the thrust-specific fuel consumption, $TSFC$, defined by

$$TSFC=\frac{\dot{m}_f}{\wp}$$

is widely used as an experimental engine quality indicator. For a turbojet (with $p_a \approx p_e$) equation (1.9) shows that

$$TSFC=\frac{\dot{m}_f}{\dot{m}_a\left[(1+f)u_e-u\right]} \qquad 2.26$$

This expression can be used for turbofan and turboprop engines by replacing the symbol u_e with $\bar{u}_e$ equation (2.24).

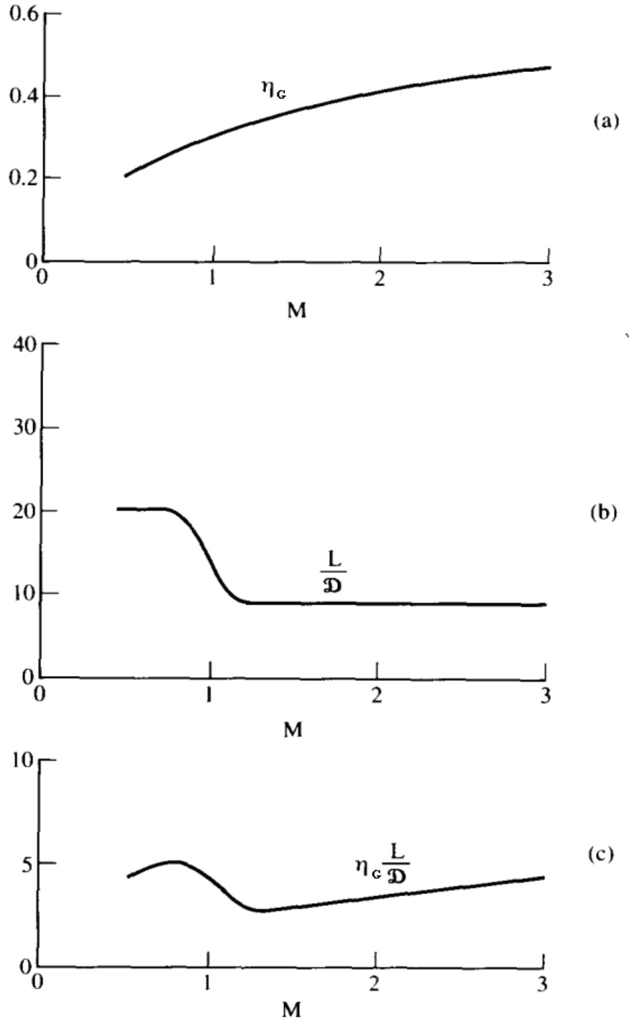


Fig. 1.5 : Variations typiques du rendement global η_G et le rapport portance-traînée $L/\mathfrak{D}$ avec le nombre de Mach de vol.

Equation (2.26) indicates that the *TSFC* of a given engine depends strongly on flight speed. Typical *TSFC* values for modern engines are as follows:

	Kg/N.hr	lb/lb $_f$.hr
Ramjets (M=2)	0.17 – 0.26	1.7 – 2.6
Turbojets (static)	0.075 – 0.11	0.75 – 1.1
Turbofan engines (static)	0.03 – 0.05	0.3 – 0.5

For turboshaft engines which produce mechanical power (shaft rotation), the fuel consumption specific to braking, *BSFC* , is defined by

$$BSFC = \frac{\dot{m}_f}{\mathcal{P}_s}$$

To represent hot gas thrust, we can define the equivalent braking-specific fuel consumption, *EBSFC*, as

$$EBSFC = \frac{\dot{m}_f}{\mathcal{P}_{es}} = \frac{\dot{m}_f}{\mathcal{P}_s + \wp_e u}$$

in which $\mathcal{P}_s$ is the shaft power supplied to the propeller, $\wp_e$ is the thrust produced by the turboshaft exhaust and *u* is an arbitrarily chosen flight speed. Typical values of equivalent fuel consumption specific to braking are

$$EBSFC = 0.45 - 0.60\,\frac{lb/hr}{hp}\ \text{Or}\ 0.27 - 0.36\,\frac{kg}{kWh}$$

In this respect the best turboprops are as efficient as the best piston engines. Additionally, the turboprop is considerably lighter and smaller (in frontal dimensions) than a piston engine of equal power, at least in the very powerful sizes.

CHAPTER III
TURBOSHAFTS

2-1 Introduction :

Modern aircraft engines have the capacity to drive massive columns of air. The total airflow ingested by the engines of a large passenger aircraft during takeoff is of the order of one ton per second. The engine airflow rate is perhaps 50 times the fuel flow, so the term "air-breathing engine" is quite appropriate. It is with the air-breathing jet and propeller engines that this and subsequent chapters are concerned.

2-2 Turbojets, turbofan engines and ramjets :

In the year 1926, Frank Whittle was a Royal Air Force cadet in Cranwell, England, pondering the future of flight as he prepared an article. At that time the maximum airplane speed was around 150 mph and 10,000 feet was the maximum flight altitude. Whittle deduced that to improve efficiency, the future aircraft must fly much faster and higher. At the same time he recognized the limitation of the propellant discussed previously. He saw that the rocket was not the answer and concluded that a new principle of aviation engine was needed namely, an ultra-fast jet produced by a propeller of a pipeline. For some time he worked on the idea that the propeller should be turned by a piston engine. But still thinking about the problem three years later, it came to mind that one could turn the propeller by a turbine if a combustion was placed in the middle, and the exhaust from the turbine could accelerate in a nozzle to form an ultra-fast jet plane. His idea met with a discouraging response from experts. After enormous technical, financial and bureaucratic difficulties, Whittle was able to run the engine for 20 minutes at 16,000 rpm on June 30, 1939, in a demonstration that finally convinced the authorities that his concept was valid and worthy of substantial support.

The introduction of the turbine engine and the principle of the turbojet in aircraft propulsion led to a revolution in the field of transport:

- It made supersonic flight possible.
- It has greatly reduced air travel expenses.
- It contributed to a radical improvement in aviation safety.

Reductions in cost were due partly to increasing flight speed and partly to the ability to build larger aircraft. The turbine engine was capable of developing much greater thrust per unit weight and per unit area than its piston counterpart. Also it turned out to have a much lower maintenance cost. Increasing thrust per unit mass of the engine directly led to improvements in payload weight, while increasing thrust per unit area reduced drag. of the engine framework and made possible enormous thrusts by the engine developed by modern long-range passenger aircraft. (See figs (2.1 and 2.2) for the configuration of typical turbine engines. Turbojet, turbofan, and turboprop versions are shown, each with a gas turbine compressor generator).

2-3 Definition and classification :

Since thrust is proportional to the rate of airflow and large thrust per unit size of the engine must be a design objective, it follows that the jet engine designer will generally try to maximize the rate of airflow per unit engine size. This means maximizing the speed at which air can enter and the fraction of the intake that can be devoted to airflow. Turboshaft engines are generally vastly superior to piston engines in these respects, so you never hear of a piston-type jet engine.

For low flight speeds, thrusters (ingesting 20 to 30 times the engine airflow rate) are well capable of handling the massive airflow rate required for propulsion. Even though piston engines may have superior efficiency at low power levels, gas turbines are favored for running large propellers because they can generally be designed to do so with much less engine mass per unit of output. power. This is also related to their ability to accommodate more airflow for a given engine size.

At low flight speeds, thruster propulsion is considerably more efficient than jet propulsion. Conventional propellers, however, become inefficient and noisy at flight speeds higher than 0.5 or 0.6 times the speed of sound. In contrast, turbofan and turbofan engines can operate efficiently and quietly at flight speeds as high as 0.85 times the speed of sound. Turbojets can also operate at supersonic flight speeds. It is ultra-fast flight which is the primary advantage of using jet engines, rather than propellants. In very small size piston engines can be made to operate more efficiently than turboshaft engines so, at the low flight speed of very small aircraft, the piston engines will have an uninterrupted role in driving the propellers. Here we will not discuss piston engines, but will return to a discussion of thrusters. Much work has been done in recent years to develop new designs of propellers that may be capable of operating efficiently (and with tolerable noise) at flight speeds as high as 0.6 or 0.7 times the speed of sound.

These courses are based on the air-breathing engines that operate in the gas turbine cycle. As shown schematically in Figure (2.1), they can take several forms depending on what is added to the basic gas generator in Figure (2.1a). The gas generator's output can be used entirely in a single propulsion nozzle (Figure 2.1b) or, prior to expansion into the main engine nozzle, to activate additional turbine stages needed to drive a propeller (figure 2.1c) which accelerates a large flow of air passing around the engine core. The flow through the outer part of the propeller can be five or six times the flow through the motor core. The engine produces thrust, simultaneously by the hot jet passing through the main nozzle and by accelerating the cold bypass flow outside the core (core) of the engine.

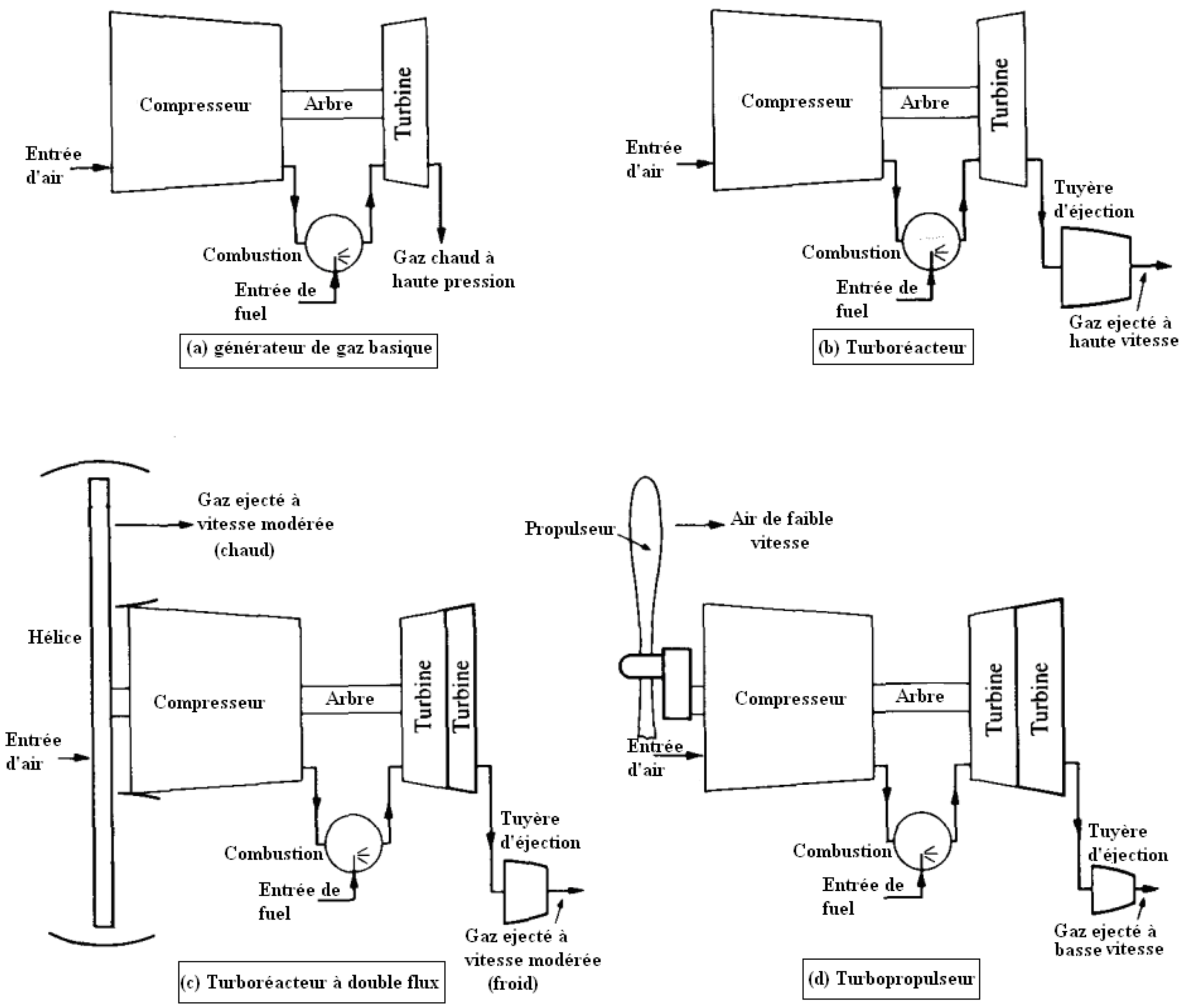


Fig. 2.1 : Turboshaft engines development from basic gas generator.

The turboprop concept in figure (2.1d), whose development preceded that of the turbojet, could be considered an extension of this concept; The rate of airflow through the thruster can be 25 to 30 times that of airflow through the main engine. The gearbox is necessary so that the thruster and main engine can operate at the optimum rotating speed. For subsonic flight, there are substantial efficiency improvements to be gained in the change from turbojet to turbofan engine. The turboprop may still be more efficient, but the use of a propeller without an encircling drive limits the application of the turboprop to relatively low flight speeds.

Figure (2.2) shows to scale drawings a typical axial flow turbojet, a turbofan and a turboprop as well as the concept of the propfan (combination of a turbofan and a propellant). In the turbojet diagram (figure 2.2a) we can see the individual compressor and turbine rotors mounted, in this case, on a single shaft. Figure (2.3) shows, for this engine, the typical pressure distribution by the engine. Later we will see why the compressor has many more rotors than the turbine, although the pressure change through the compressor is only about 20% greater than through the turbine. The combustion chambers between the compressor and the turbine are shown. The exhaust nozzle is short, provided with a simple converging internal cone, i.e. designed for subsonic (or sonic) output flow.

Figure (2.2b) shows a typical sketch of a turbofan engine. This one has three concentric shafts. The exterior transmits power from the single high-pressure turbine rotor to the high-pressure compressor rotor, which has six sets of rotor blades. The intermediate shaft takes power from the single-stage medium-pressure turbine to drive the six-stage low-pressure compressor. Three stages of the turbine drive the propeller through the innermost shaft. In this drawing we see the relatively small convergence of the hot gas nozzle section and even less in the propeller nozzle. Figure (2.2c) shows a two-shaft axial turbine engine. Hidden in a front housing is the planetary gear reduction unit connecting the innermost engine shafts to the propeller (not shown). Figure (2.2d) shows a drawing of a so-called "propfan" engine, showing two counter-rotating thrusters (or "non-driven" propellers, as they are sometimes called) driven by separate turbines. The thrusters differ from conventional ones in which they are multi-bladed and particularly manufactured for aerodynamic efficiency at high flight speeds.

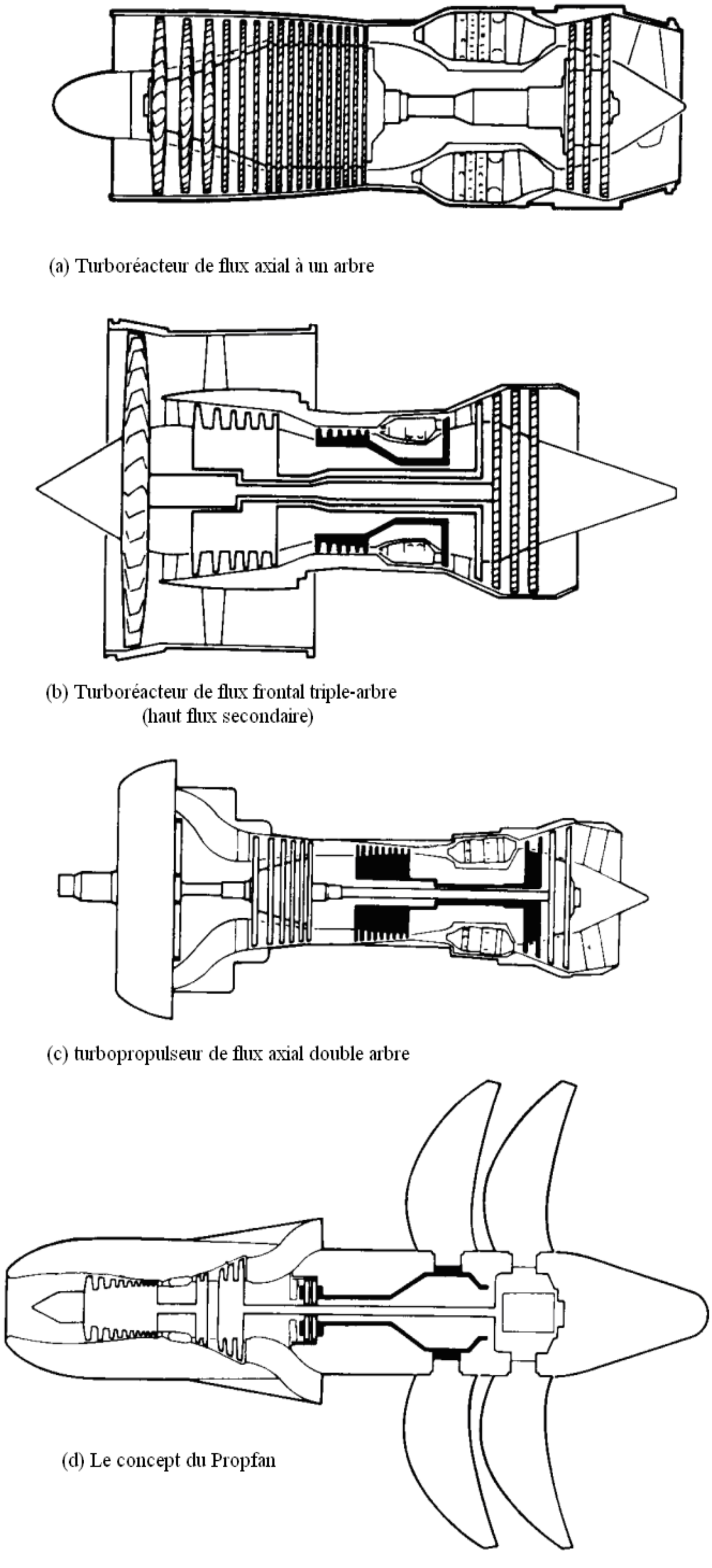


Fig. 2.2 : Turboshaft engines. (Courtesy Rolls-Royce, pic.)

Any jet engine develops thrust by imparting momentum to the fluid passing through it. With Newton's laws in mind, it is easy to say that the thrust exhibited by the jet engine is the reaction to the net force that the engine exerts on the fluid passing through it. However, the internal distribution of the components of this net force is complex. Figure (2.3) shows internal forces calculated on a turbojet whose net thrust is 11.158 lb $_f$. As shown, the total internal forces on the turbine and nozzle are 46.678 lb $_f$ and act in the wrong direction! The thrusts on the turbine exhaust cone, combustion chamber and compressor are positive, however and total 57,836 lb $_f$; so the net force imposed on the metal surfaces of the motor is positive after all.

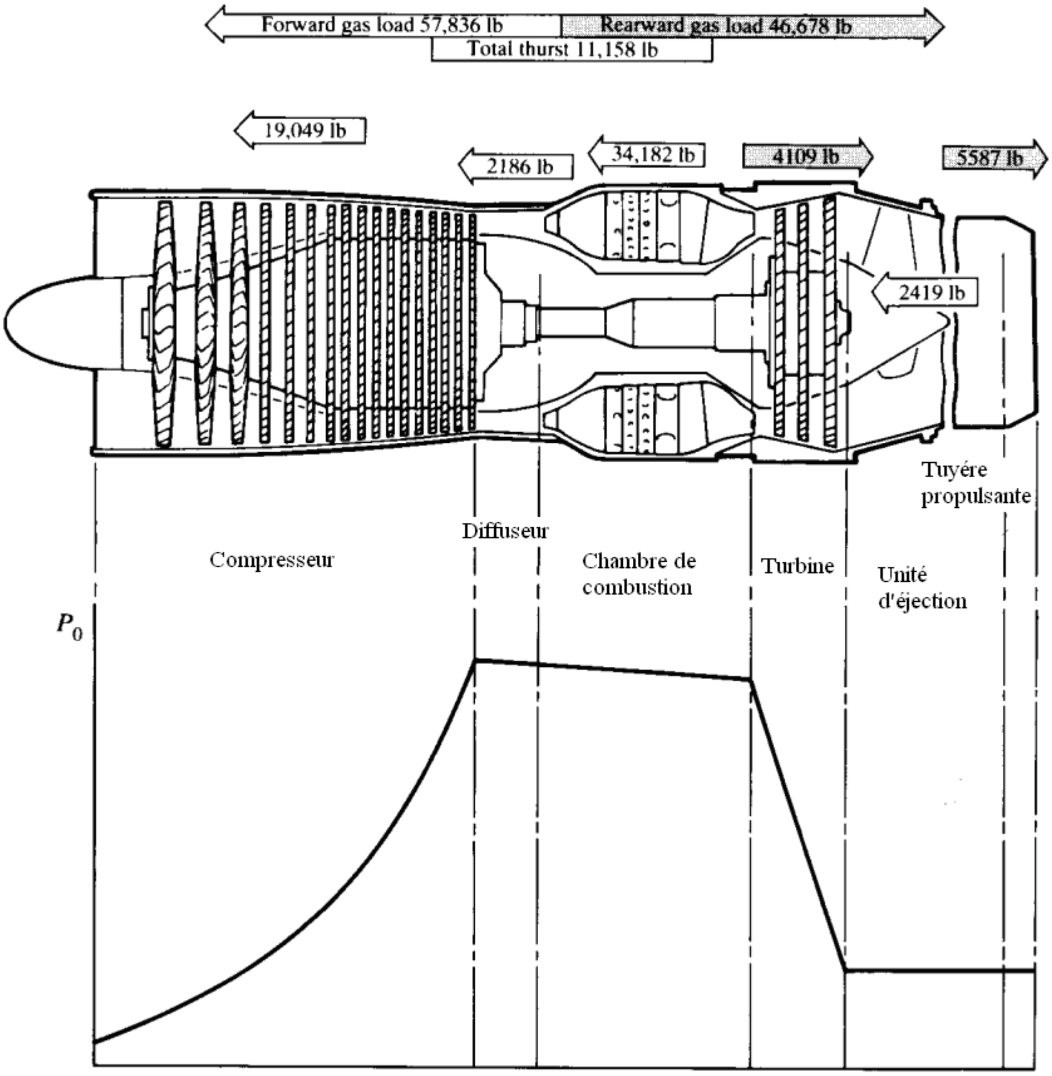


Fig. 2.3 : Distributions interne de la pression totale et la poussée dans un turboréacteur à un arbre. (Adapted from Rolls-Royce, pic.)

Evaluating an integral of the pressure and shear forces acting on each surface element within the engine to determine the net thrust at all flows and speeds would be an unacceptably boring and uncertain procedure. Calculations such as those shown in Figure (2.3), which are used in a wide variety of design evaluations, are best performed using the control volume method.

CHAPTER IV
AVIATION ENGINES

3-1 Introduction :

The general goal of this chapter is to apply the laws of thermodynamics and mechanics to air-breathing jet propulsion engines in such a way that one can:

- Estimate the best possible engine performance as a function of key design parameters - for example, maximum allowable engine temperature, pressure ratio, flight speed and ambient conditions;
- Evaluate the effects of feed idealization ("losses") in engine components - for example, compressor, turbine and nozzle;
- Appreciate the reasons for developing specific engine configurations – for example, the turbofan and afterburner turbojets;
- Establish a method to evaluate the effects of future increases in turbine inlet temperature, component efficiency, etc.;
- See how major engine components – for example, compressors and turbines – can be designed to match each other under design conditions;
- See how, in principle, essential engine and aircraft performance characteristics can be considered together in evaluating flight vehicle performance.

3-2 The turbojet :

In later sections we will see that one of the disadvantages of the ramjet is that its pressure ratio depends on the flight Mach number. It cannot develop takeoff thrust and, in fact, it does not perform well unless the flight speed is considerably above the speed of sound. One way to overcome this disadvantage would be to install a mechanical compressor in the intake pipe, so that even in zero speed flight the air could be drawn into the engine, burned and then expanded through a nozzle. However, this presents a need for the power to drive the compressor. If a turbine is attached to the compressor and driven by the hot gas leaving the chamber and advancing towards the ejection nozzle, the ramjet has become a turbojet. The introduction of turbomachinery, however, entirely changes the performance characteristic of the engine.

The internal configuration of the turbojet is shown schematically in figure (4.1). As it flows through the machine, the air undergoes the following processes:

ⓐ ar u① ream, where the air speed relative to the engine is the flight speed, the air is brought the input, often with an acceleration or deceleration.

① ②

- Air velocity is decreased as air is carried to the compressor inlet by the intake diffuser and the driving system.

(2) - (3) The air is dynamically compressed in a compressor.

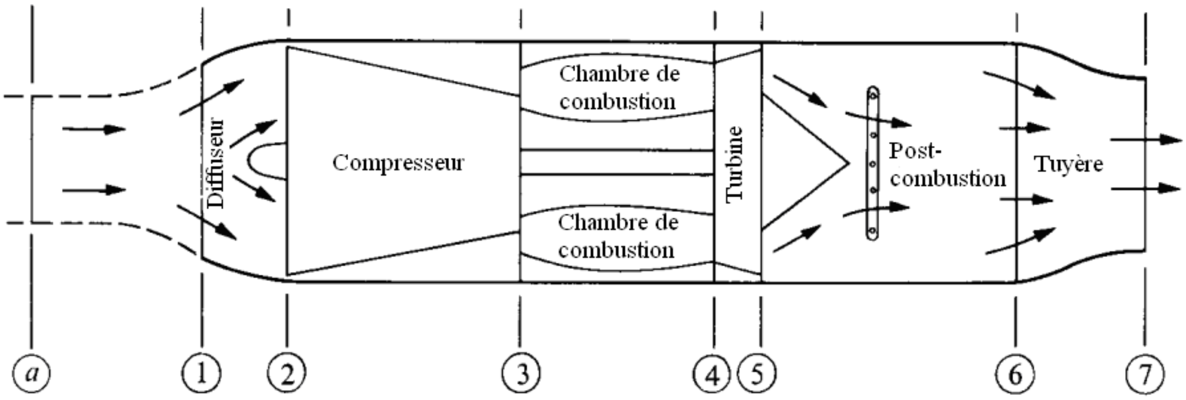


Fig. 4.1 : Scheme of turbojet engine.

(3)he a(4)s “heated” by mixing and burning the fuel in the air.

(4)he a(5)s expanded by a turbine in order to obtain the power necessary to drive the compressor.

(5)ir m(6)or may not be “heated” by the addition and combustion of more fuel in a post-combustion phase.

(6)he a(7)s accelerated and ejected through the exhaust nozzle.

Before proceeding with the thermodynamic analyses, we look at examples of turbojet engines, each with distinctive features.

Figure (3.2) shows the F404 jet engine with afterburner from General Electric. All airflow enters the three low-pressure stages of the compressor, but there is a bypass flow around the high-pressure compressor and the combustion chamber, which provides a cooling flow to the outer region of the compressor. post-combustion downstream of the turbine. The bypass flow is typically about 34% of the flow in the high-pressure compressor; the engine could be called a turbofan with low bypass flow. The high and low pressure compressors of the F404 are each driven by single-stage turbines using concentric and independent shafts.

The section of the post-combustion nozzle is variable to accommodate flow rates of different volumes with and without post-combustion action. Because the temperature of the hot gas entering the turbine must be kept far below the adiabatic combustion temperature (and because the combustion process is almost adiabatic), the abundance of oxygen is left for a second combustion to take place in the post -combustion chamber. Shown in the sectional views are two concentric V-section rings, which are used to stabilize the flame in the afterburner.

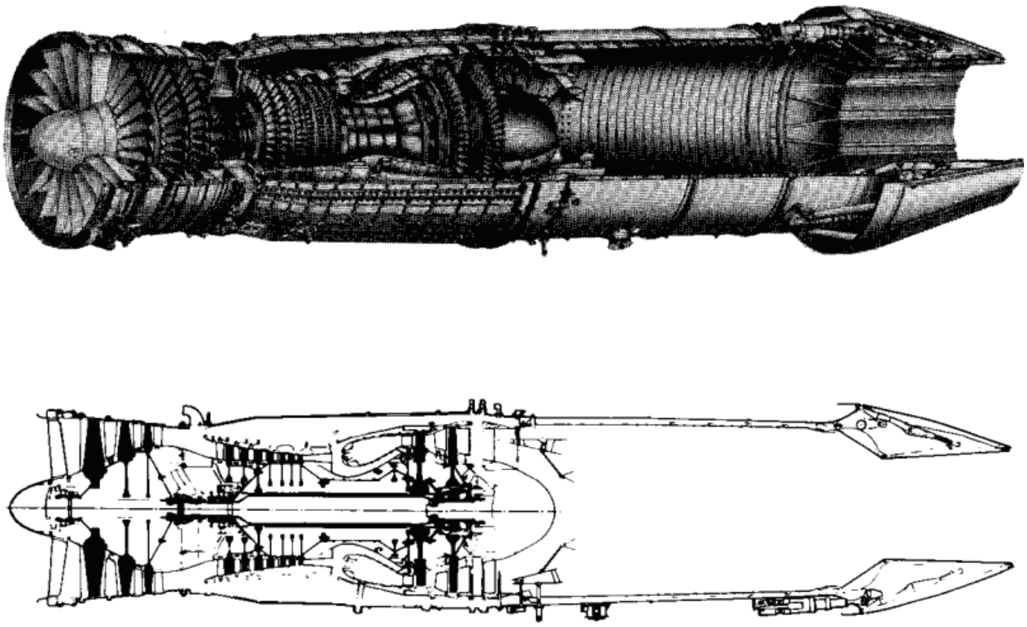


Fig. 4.2 : Turbofan F404 of General Electric
with weak secondary flux and postcombustion.
(Courtesy GE Aircraft Engines.)

The F404 is the successor to General Electric's J79, which was in service for many years and has almost the same thrust (approximately 16,000 lb $_f$, or 71 kN). Compared with the J79, the F404 has half the weight and double the pressure ratio (25:1); this is an indication of the progress that has been made in turbojet development since the 1960s. The gross weight of the F404 is approximately 1,000 kg (2,200 lb $_m$). The inlet diameter (of the inlet section) is 0.79 m (31 in.) and the air flow when operating statically at sea level is 64.4 kg/s (142 lb/sec).

Figure (3.3) shows the two Rolls-Royce Olympus 593 engines used in the Concorde, which cruises at twice the speed of sound, at altitudes up to 60,000 ft (18,300 m) and with a range of 4000 miles (6440km). Each engine produces 38,000 lb $_f$ (169 kN) at takeoff; With the flight Mach number of 2 and an altitude of 53,000 ft (16,100 m), the thrust per engine is approximately 10,000 lb $_f$(44.6 kN). Engine weight is 7465 lb (3386 kg).

While the Concorde is cruising at Mach 2, the engine's specific fuel consumption is approximately 1.19 lb/hr/lb $_f$(33.7 g/kNs). Rough cycle calculations indicate that in this condition the exhaust velocity ratio u/u_e is about 0.55, so that the propulsion efficiency is about 0.70. The estimated thermal efficiency of the engine during cruise is 0.60, so the overall efficiency η_G is approximately 0.42.

The Olympus engine compressor has a pressure ratio of 11.3:1. And the two low-pressure and high-pressure sections of the compressor (driven by concentric and independent shafts) are driven by single-stage turbines. Each compressor section has seven stages. The air intake (not shown in Fig. 3.3) to the low-pressure compressor has a variable geometry to provide very different air intake conditions corresponding to takeoff, subsonic and supersonic cruise.

The maximum turbine inlet temperature for the Olympus engine is approximately 1350 K. The afterburner or "heat" chambers of the Concorde Olympus engines are used for increased thrust during takeoff and in acceleration by almost sonic flight speeds.

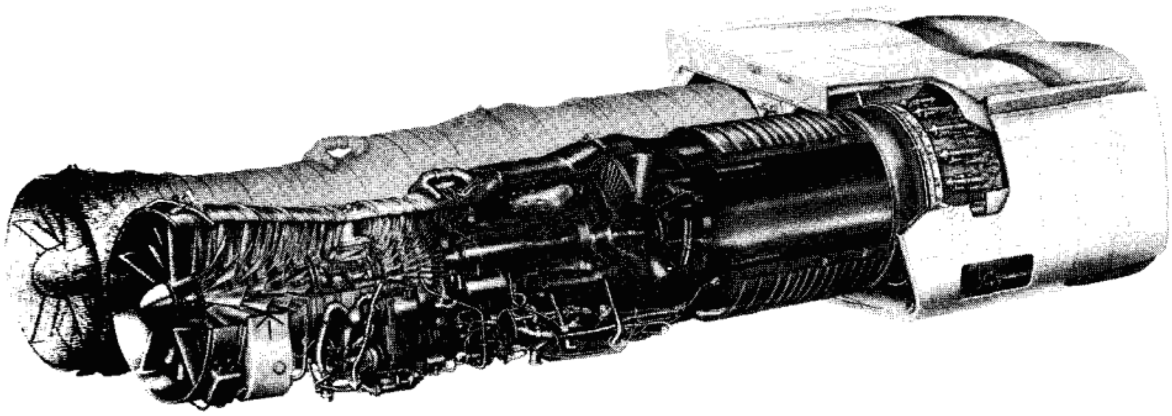


Fig. 4.3 : The Olympus 593 engine from Rolls-Royce used in Concorde. (Courtesy Rolls-Royce, pic.)

The Concorde is the first supersonic commercial passenger aircraft in the world; it is unlikely to be the last. Designers are studying some concepts that could reduce the costs of supersonic flight. Currently the Concorde produces only about 13 miles per gallon of fuel, while the large subsonic passenger plane is capable of delivering about 75 miles per gallon. Part of the reason for the difference is size (the Concorde having only 100 seats compared to 400-500 for the Boeing 747); the other part is due to the necessary compromises in the design of subsonic and supersonic cruise. The afterburner is, for thermodynamic reasons, fuel inefficient (although it can provide a large increment of thrust without requiring a large complement to the engine weight). One concept being explored for the supersonic passenger aircraft involves a variable engine configuration: the engine would act as a turbofan during takeoff and subsonic cruise, but would revert to turbojet operation for flight at the Mach number of 2 or 3. To preserve a higher Mach number, one could bypass the turbomachines to provide ramjet propulsion. In this way the propulsion efficiency could be increased across a Mach number range . The selection of the cruise flight Mach number that will provide maximum room per gallon will undoubtedly be greatly affected by new developments in aircraft structures and hardware. In flight at very high Mach numbers, cooling will be necessary because of high wall temperatures.

We can conveniently show the thermodynamic path of the fluid in a turbojet on an enthalpy-entropy *(hs)* or temperature-entropy *(Ts) diagram* . To understand the complete process, it is useful to first study a simplified model. Let us therefore assume that all components except the combustion chambers are reversible and adiabatic, that the combustion chambers can be replaced by simple frictionless heaters, and that the velocities at the sections at are negligible. We show the *Ts diagram* for such an engine in figure (3.4) for engines without and with afterburner, assuming that the operating fluid is an ideal gas.

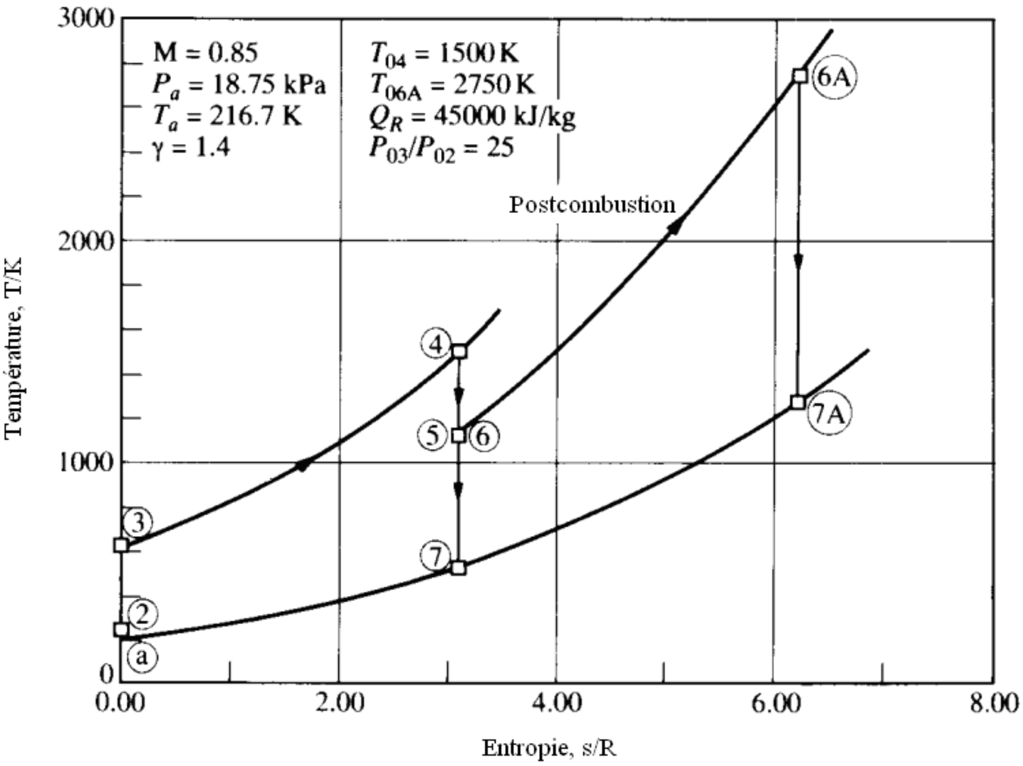


Fig. 3.4 : Theoretical *T-s* Diagram (ideal) of a turbojet.

In the ideal case the pressure increases are from to (a) and always more from (1) to as the (1) air is (2) slowed down relative to the engine. Since the speed at is assumed to (2) zero and the slowdown is isentropic, p_2 is the stagnation pressure of the states , and . Also, T_2 is the (a) stagnation (1) temperature (2) for these states. The power consumed in the compression of
(2) to (3) must be supplied by the turbine in the expansion of (4) Hence (5), if the mass flow rates of the compressor and turbine are equal, $h_3 - h_2 = h_4 - h_5$ and the specific heat is constant, the corresponding temperature differences are also equal.

In the case without post-combustion, the states (5) and (6) are identical and the enthalpy decrease from (5) to (7) is proportional (6A) (7A) to the square of the exhaust speed. In the case of post-combustion, the air is heated between and . From the (6) shape (6A) the constant pressure curve we can see that $(T_{6A} - T_{7A})$, and thus the exhaust velocity, will be greater in the case of afterburner. In fact, as we will see later, the absence of highly stressed material in the post-combustion allows it to T_6 be much higher even than T_4, so that the increase in exhaust speed can be of the order of 50%.

Although very simplified, this model illustrates the functions of the various components and the relationships between them. It clearly shows that the kinetic energy output of the exhaust fluid is, in a sense, the remainder after power has been extracted from the fluid to drive the compressor.

Real cycle:

A real engine differs from this ideal model in several respects. First and most important, no component is actually reversible, although it is usually reasonable to assume them to be adiabatic. Second, combustion chambers are not simple heating devices and the composition of the operating fluid will change during combustion processes. Third, the fluid velocities in the engine are not negligible. If the fluid velocity in the combustion chamber were actually zero (as constant pressure combustion requires), it would be impossible to have a stable flame, since the flame propagates relative to the fluid at fairly high velocities. There is a fourth difference, in that the mass flow rates of the turbine and the compressor cannot be equal, on the one hand fuel is added between the two and on the other hand air can be extracted at Various positions for cooling purposes.

Figure (3.5) shows an entropy-temperature diagram *Ts* of a real engine with reasonable irreversible effects and typical temperatures, for a compressor pressure ratio of 10. Afterburning and non-afterburning processes are also shown. combustion, with an exhaust pressure equal to ambient pressure in both cases.

The process begins with atmospheric air at h_a, p_a. By virtue of the relative (flight) speed between the air and the engine, this air has a stagnation enthalpy h_{0a}, greater than h_a. Further, since no work or heat transfer takes place between and , the stagnation enthalpy is constant in the station . The air is externally slowed down from to . For all practical purposes this external slowing is an isentropic process (unless an external shock arrives); from there the state is on an isentropic with the state and $p_{01} = p_{0a}$. Air is slowed down, with an increase in entropy due to the effects of friction and a decrease in stagnation pressure. Air is compressed, again with an increase in entropy due to irreversibilities in the compression process.

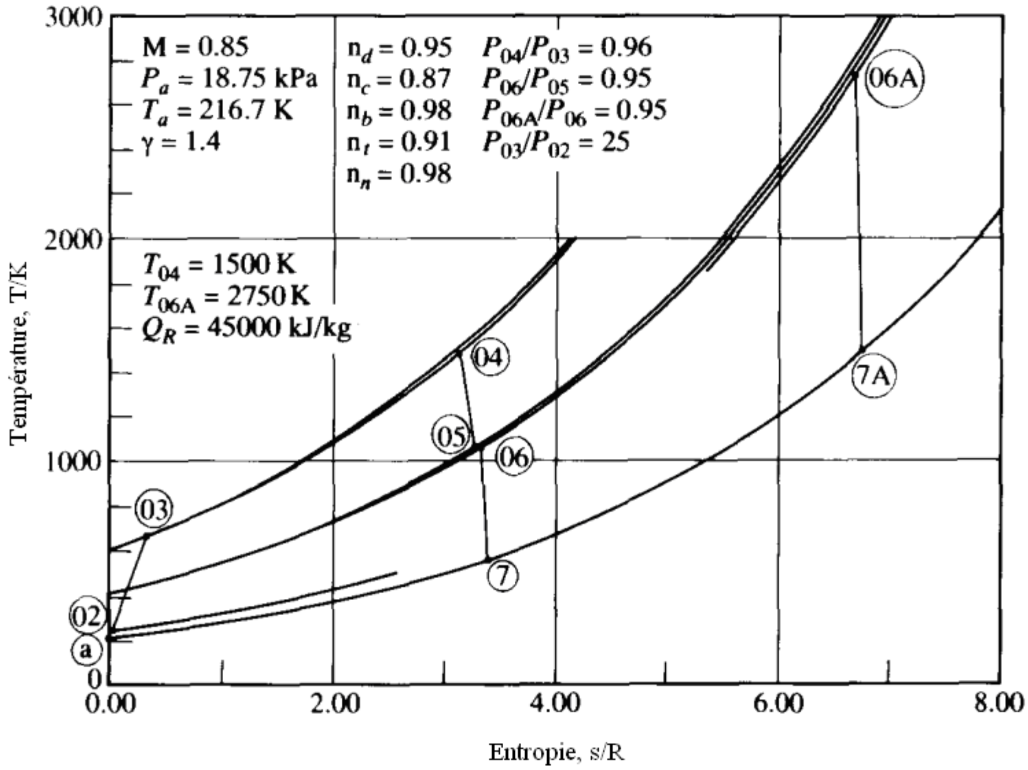


Fig. 3.5 : Real *T-s* diagram of turbojet engine.

The sta 03s defined as the state that would exist if the air could be compressed isentropically to the actual outlet stagnation pressure. The state is the a 03 l stagnation exit state. We will discuss these states later when we define compressor efficiency.

From station to station (see Figure 3.1), a quantity of the fuel is mixed with air, and combustion takes pl 3 In other w 4 s, the composition of the fluid changes between these stations, so there is not a continuous path between them. However, since the characteristics of the fluid do not change remarkably, there is no difficulty in exhibiting the two substances on different parts of the same diagram. The stagnation pressure at must be less than at because of fluid friction and also beca 4 of the drop in stagna 3 n pressure due to additional heating at finite speed. As we'll see later, it's beneficial to make it T_{04} as high as hardware limitations will allow. Thus the states and are fairly well fixed.

From to the fluid ex 04 ds 4 ugh the turbine, providing shaft power (mechanical) equal to the shaft p 4 er 5 plied to the compressor (plus any mechanical losses or additional power). Since no work or heat transfer takes place downstream of 6 the station, the stagnation enthalpy remains 5 stant throughout the rest of the machine. The state depends on the geometry involved, 6 but p_{06} must be less than p_{05}. Exhaust pressure p_7 is us 6 ally equal to atmospheric pressure p_a, but they can be different if the exhaust flow is supersonic. 06A we know the state, the speed u_7 can be calculated from h_{07} known (or h_{05}) independently of the properties in the states. If post-combustion is present, the fluid undergoes a rise in temperature to state , then it extends in the nozzle towards state .

Again we can see that the exhaust kinetic energy is the relatively small difference between the total available enthalpy drop of the state and the work supplied to the compressor. For a given compressor pressure ratio, irreversibilities increase the compressor power requirement while at the same time increasing the necessary turbine pressure drop. Both effects reduce the exhaust kinetic energy, which is why the overall performance is very sensitive to the performance of the turbine and the compressor.

Using the fact that the compression and expansion processes in turbojet engines is quasi-adiabatic, we can make realistic assessments of engine performance by defining the adiabatic component efficiency as follows:

For the intake diffuser (air inlet), an adiabatic efficiency η_d can be defined as the ratio of the ideal to the actual enthalpy change during the diffusion process (for the same pressure ratio p_{02} / p_1), or

$$\eta_d = \frac{h_{02s} - h_a}{h_{02} - h_a}$$

Also, for a compressor, a useful definition of adiabatic efficiency η_c is the ratio of the work required in an *isentropic process* to that required in the *actual process* , for the same ratio of stagnation pressure and inlet condition:

$$\eta_c = \frac{h_{03s} - h_{02}}{h_{03} - h_{02}}$$

For the turbine, the adiabatic efficiency can be defined as

$$\eta_t = \frac{h_{04} - h_{05}}{h_{04} - h_{05s}}$$

which is the ratio of the actual work of the turbine to that which would be obtained during isentropic expansion at the same exhaust stagnation pressure. The adiabatic efficiency of a nozzle can be defined as

$$\eta_n = \frac{h_{06} - h_7}{h_{06} - h_{7s}}$$

In addition to these four adiabatic efficiencies, a fifth kind of efficiency is often employed: burner (combustion chamber) efficiency η_b, which is simply the fraction of the chemical energy of the fuel that is released into the combustion chamber.

For well-designed motors the above efficiencies will generally be in the following ranges:

$0.7 < \eta_d < 0.9$ (strongly dependent on the flight Mach number),

$0.85 < \eta_c < 0.90$, $\quad 0.90 < \eta_t < 0.95$,

$0.95 < \eta_n < 0.98$, $\quad 0.97 < \eta_b < 0.99$.

Note that these definitions used enthalpy stagnation values. It is usually more convenient experimentally to measure the stagnation values, rather than the statics, of pressure and temperature in a fluid flow. Stagnation values are convenient analytically, since they contain the kinetic energy terms.

If we use thrust equation (1.9) for the case in which the aircraft exhaust pressure is atmospheric, the thrust per unit mass of air is

$$\frac{\wp}{\dot{m}_a} = \left[(1+f)u_e - u\right] \qquad 3.1$$

and we give the fuel consumption specific to the thrust

$$TSFC = \frac{\dot{m}_f}{\wp} = \frac{f}{(1+f)u_e - u} \qquad 3.2$$

Given the flight speed u, the ambient conditions p_a, T_a the compressor pressure ratio p_{03}/p_{02}, and the turbine inlet temperature T_{04}, our task is to determine the fuel-air ratio f and the ejection speed u_e. Then we can evaluate the engine thrust and specific fuel consumption from equations (3.1 and 3.2). We proceed as follows:

1. Compressor inlet: With the flight Mach number $M = u/\sqrt{\gamma R T_a}$, in which γ is the ratio of specific heats (assumed constant for the diffuser process), we evaluate the stagnation temperature T_{02} with

$$T_{02} = T_a\left(1 + \frac{\gamma - 1}{2}M^2\right) \qquad 3.3$$

Given the definition of the adiabatic efficiency of the diffuser η_d, we can evaluate the corresponding stagnation pressure with

$$p_{02} = p_a\left[1 + \eta_d\left(\frac{T_{02}}{T_a} - 1\right)\right]^{\gamma_d/(\gamma_d - 1)} \qquad 3.4$$

2. Compressor outlet: Since the compressor pressure ratio $p_{rc} = p_{03}/p_{02}$ is specified, we can determine the outlet stagnation pressure of

$$p_{03} = p_{02}\,p_{rc} \qquad 3.5$$

and the total outlet temperature of

$$T_{03} = T_{02}\left[1 + \frac{1}{\eta_c}\left\{p_{rc}^{(\gamma_c - 1)/\gamma_c} - 1\right\}\right] \quad 3.6$$

in which γ_c is the ratio of specific heats (assumed constant) for the compression process and η_c is the adiabatic efficiency of the compressor.

3. Burner fuel-air ratio: With the burner outlet (combustion chamber) temperature T_{04} given, we can determine the fuel-air ratio from

$$f = \frac{T_{04}/T_{03} - 1}{Q_R / c_p T_{03} - T_{04}/T_{03}} \quad 3.7$$

4. Turbine inlet pressure: Given the pressure ratio p_{04}/p_{03} across the combustion chamber, we find the total turbine outlet pressure to be

$$p_{04} = p_{03}(p_{04}/p_{03}) \quad 3.8$$

5. Turbine output: Since the turbine must provide the power required by the compressor, we write (for the stable adiabatic flow in both turbomachines):

$$\dot{m}_t(h_{04} - h_{05}) = \dot{m}_c(h_{03} - h_{02})$$

Or

$$\dot{m}_t c_{pt}(T_{04} - T_{05}) = \dot{m}_c c_{cp}(T_{03} - T_{02}) \quad 3.9$$

The mass flow rates $\dot{m}_t$ and $\dot{m}_c$, are not quite the same, since fuel is added into the combustion chamber and air is drawn from the compressor for turbine cooling. Also, are the average specific heats c_{pt} and c_{pc}, the same. For modest compressor air extraction cooling rates, we can say, as a first approximation,

$$\dot{m}_t c_{pt} \approx \dot{m}_c c_{pc}$$

Thus

$$T_{05} \approx T_{04} - (T_{03} - T_{02}) \quad 3.10$$

Then, from the definition of the adiabatic efficiency of the turbine, we can evaluate the total pressure p_{05} of

$$p_{05} = p_{04}\left[1 - \frac{1}{\eta_t}(1 - T_{05}/T_{04})\right]^{\gamma_t/(\gamma_t - 1)}$$

in which γ_t is the ratio of specific heats (assumed constant) of the turbine.

6. Nozzle inlet conditions: Without afterburner installed, the nozzle inlet conditions are

$T_{06} = T_{05}$ and $p_{06} = p_{05}$

7. Nozzle exit velocity: The kinetic energy of the nozzle exit is, by definition,

$$\frac{u_e^2}{2} = h_{07} - h_7 = \eta_n (h_{07} - h_{7s})$$

where η_n is the adiabatic efficiency of the nozzle. With a stable adiabatic flow in the nozzle $h_{07} = h_{06}$, we have

$$u_e = \sqrt{2\eta_n \frac{\gamma_n}{\gamma_n - 1} RT_{06} \left[1 - (p_a / p_{06})^{(\gamma_n - 1)/\gamma_n}\right]} \qquad 3.11$$

as long as the nozzle is not suffocated.

With the fuel-air ratio f now determined from equation (3.7) and also with the exhaust velocity u_e from equation (3.11) we can determine the engine thrust and the specific fuel consumption from equations (3.1) and (3.2), respectively.

We now consider examples of varying the thrust and fuel consumption of a turbojet with compressor compression ratio, turbine inlet temperature and flight Mach number. For typical calculations we make the assumptions of component efficiency, fluid properties and engine conditions shown in table (3.1). Each flight Mach number is assumed to correspond to different altitudes and thus to a different pair of ambient pressure and temperature p_a and T_a. In these calculations we assume, as before, that the product $\dot{m}c_p$ is the same in the turbine as in the compressor.

Figures 3.6 to 3.9 show the thrust and specific fuel consumption calculated under these conditions for three values of turbine inlet temperature and four values of flight Mach number. All these two parameters, as well as the ratio compression, strongly influence engine performance.

Tab. 3.1: Calculation parameters of a turbojet.

Component	Adiabatic efficiency	Average specific heat ratio
Streamer	$\eta_d = 0.97$	1.40
Compressor	$\eta_c = 0.85$	1.37
Combustion chamber	$\eta_b = 1.00$	1.35
Turbine	$\eta_t = 0.90$	1.33
Nozzle	$\eta_n = 0.98$	1.36
Calorific value of fuel oil, 45,000 kJ/kg		
Flight altitude (N. of cruising Mach)	**Ambient pressure (kPa)**	**Ambient temperature (K)**
Sea level (0)	101.30	288.2
40,000 ft (12,200 m) (0.85)	18.75	216.7
60,000 ft (18,300 m) (2.0)	7,170	216.7
80,000 ft (24,400 m) (3.0)	2,097	216.7

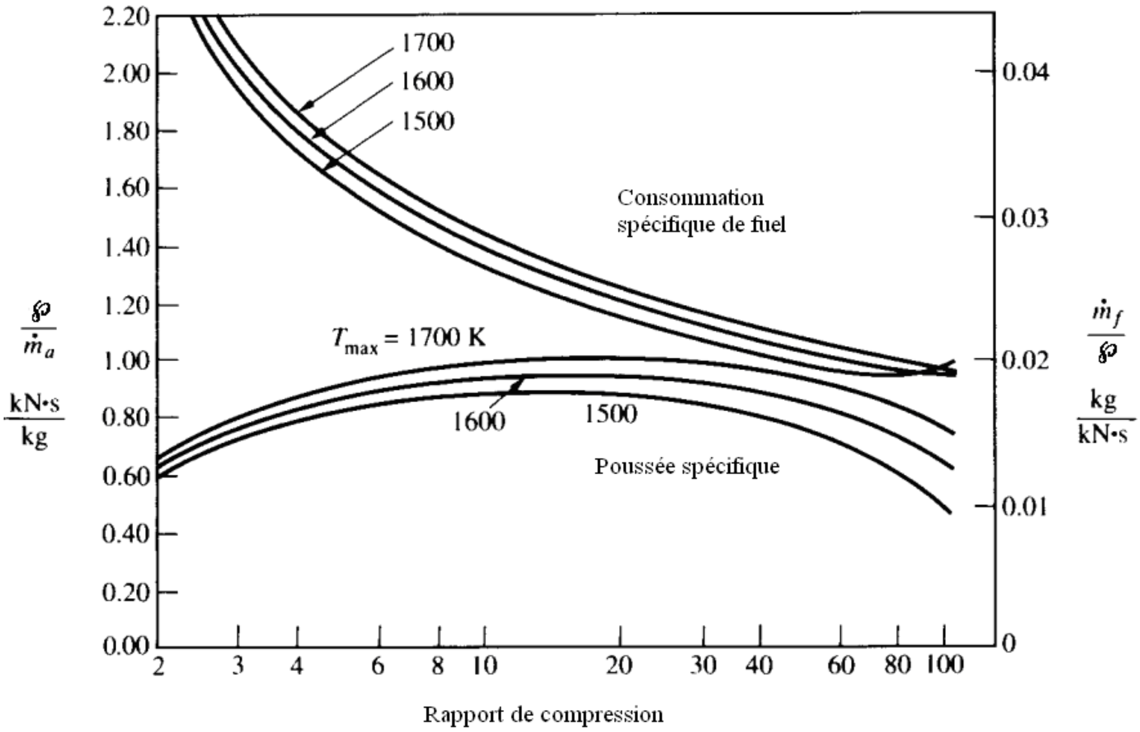


Fig. 3.6 : Static thrust force and specific thrust fuel consumption of a turbojet (M=0).

We see in particular from these graphs that:

1. At a given flight Mach number and turbine inlet temperature, the pressure ratio which maximizes the specific thrust does not present a minimum fuel consumption.

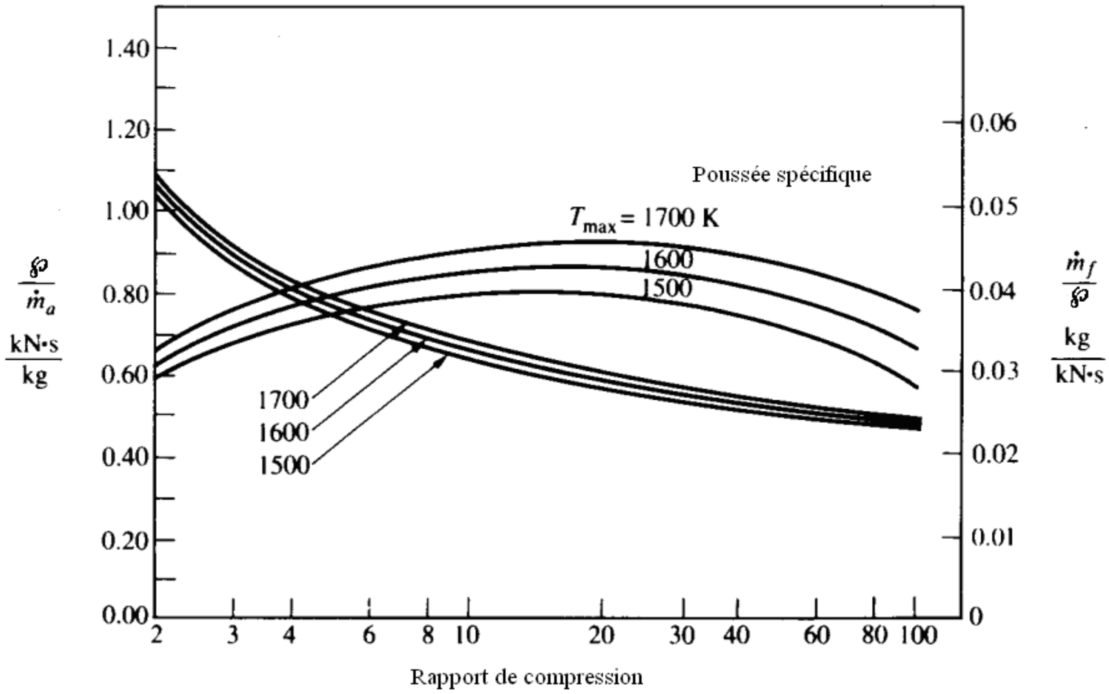


Fig. 3.7 : Static thrust force and specific thrust fuel consumption of a turbojet in cruise (*M*=0.85).

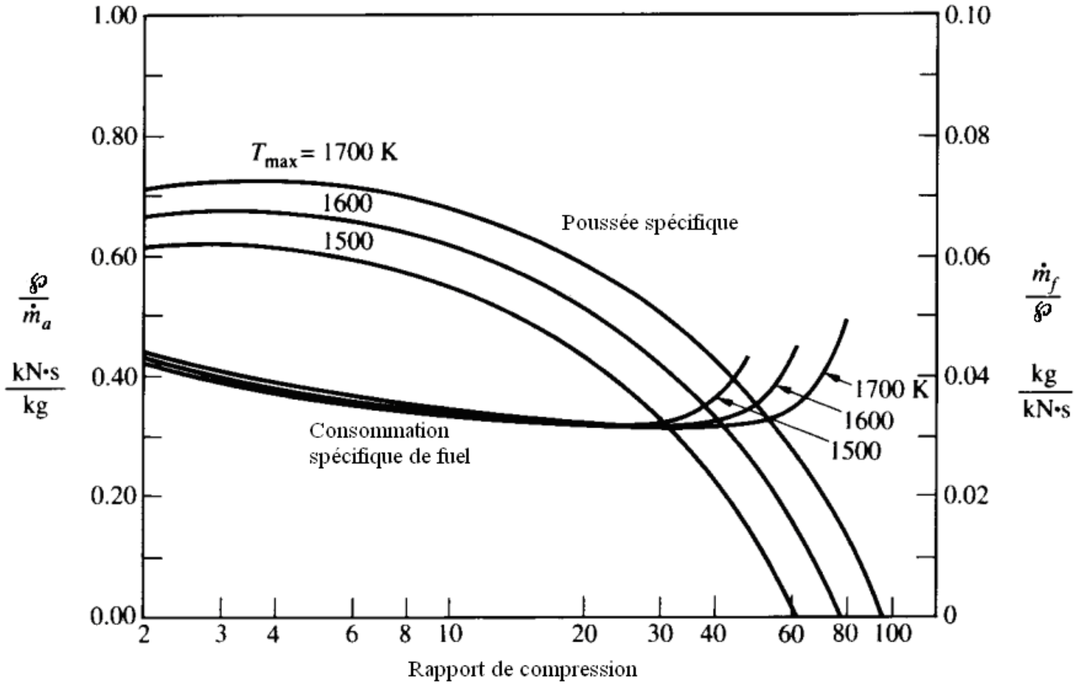


Fig. 3.8 : Static thrust force and specific thrust fuel consumption of a turbojet in cruise (*M*=2).

Since engine mass strongly depends on air flow rate and significantly affects aircraft carrying capacity, specific thrust and specific fuel consumption must be considered in selecting the best compressor pressure ratio. Choosing the pressure ratio for the best flight range will require a compromise that takes into account aircraft and engine characteristics.

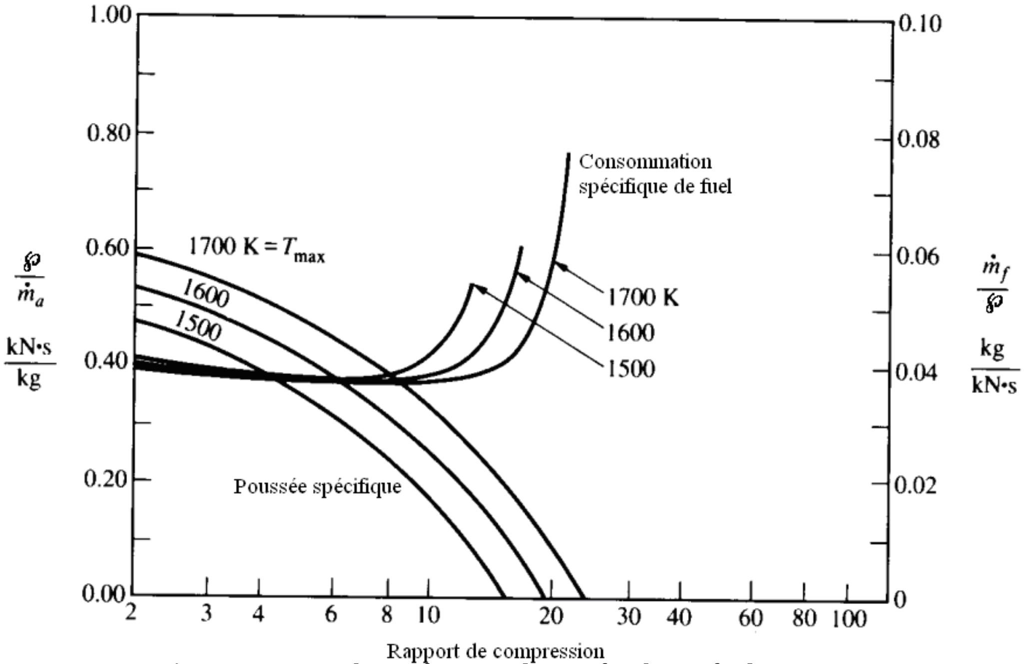


Fig. 3.9 : Static thrust force and specific thrust fuel consumption of a turbojet at cruising (M=3).

2. Raising the turbine inlet temperature can significantly improve specific thrust. The maximum temperature shown, 1700 K, is far below the maximum stoichiometric combustion of hydrocarbon fuels, but requires not only high temperature alloys for the turbine blades, but also quite intensive cooling of dawn too.
3. With a given compressor compression ratio, raising the turbine inlet temperature may or may not increase fuel consumption per unit thrust. For pressure ratios associated with minimum fuel consumption, increasing the turbine inlet temperature can reduce the specific fuel consumption somewhat. (It may, however, require an increase in the flow of cooling air extracted from the compressor, and the present calculations have not taken this into consideration.)
4. The compression ratio required to minimize specific fuel consumption is much lower for supersonic flight than for subsonic. At Mach 3, specific thrust reaches a maximum level with a compression ratio of 1; in this case no compressor or turbine is required and the turbojet becomes a ramjet. The ramjet could, of course, tolerate the maximum high temperature, not having exposed the turbine blades to the stresses of hot gases.

Figures (3.6 to 3.9) do not show the sensitivity of engine performance to component efficiencies. The designer must size the processes of compression, combustion, and expansion as efficiently as possible.

3 -3 The turbofan engine :

As Section (1-5.2) showed, the range of an aircraft is directly proportional to the overall efficiency $\eta_G = \eta_p \eta_{th}$. For f <<1 we give the propulsion efficiency and the thrust approximately by

$$\eta_p = \frac{2u}{u_e + u}$$

and

$$\wp = \dot{m}_a (u_e - u)$$

Using the second of these relations to eliminate u_e, we obtain

$$\eta_p = \frac{1}{1 + \frac{\wp}{2\dot{m}_a u}} \qquad 3.12$$

Thus for a given flight speed the propulsion efficiency can be increased by reducing the specific thrust $\wp / \dot{m}_a$.

One way to do this, for a turbojet, is to reduce the turbine inlet temperature (see, for example, Figure 3.7). However, this idea is not good since η_p it could only be lifted in this way at the expense of an unacceptable increase in engine mass for the same thrust. A better procedure must target changes in engine design.

Figure (3.10) shows that the propulsion efficiency will strongly depend on the type of engine chosen as well as the flight speed. For a turbojet cruise at M = 0.85, the flight speed is around 250 m/s (560 mph), while the exhaust speed can be around 600 m/s, so the propulsive efficiency $\eta_p \approx 2u / (u_e + u)$ is around $2(250)/(600+250) = 0.6$.

At the same flight speed, Figure (3.10) shows that a turbojet with a large bypass flow (secondary flow), — could be a turbojet — has considerably greater propulsion efficiency than a real turbojet. At flight speeds below 400 mph, the turboprop would have the highest propulsion efficiency. Having higher effective "bypass ratio", the turboprop would have the lowest speed averaged by the mass u_e and thus, according to the aforementioned formula, η_p is superior than the turbofan. Figure (3.10) reflects the inability of conventional thrusters to operate effectively at high speeds. For propeller-powered aircraft with the traditional propeller blade design, the flight Mach number is typically limited to less than about 0.5. At higher flight speeds the thruster operation becomes excessively noisy and the thruster η_p efficiency decreases unacceptably because the flow on the outer part of the blade (being supersonic relative to the blade) tends to suffer from waves. shock and flow separation.

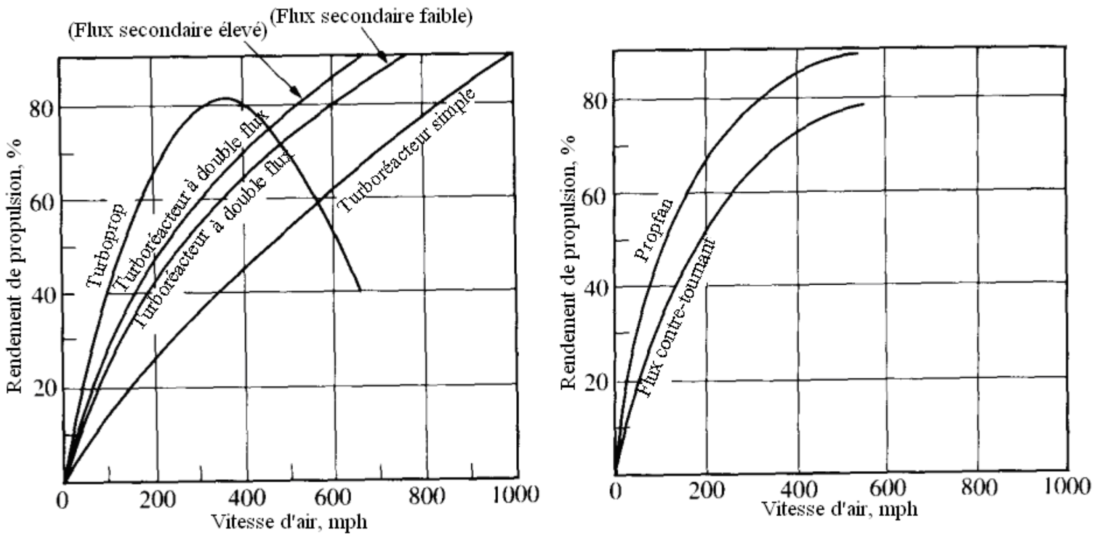


Fig. 3.10 : Typical propulsive efficiencies (Courtesy Rolls-Royce, pic.)

There is a strong motivation for the development of new types of propellers that can operate in high Mach numbers of subsonic flight (supposedly, 550 mph) with higher propulsion efficiency (supposedly, 80%) and a lower acceptable noise. Such ultrafast propellants could significantly reduce fuel consumption costs.

The development of propeller and thrusters has become so important to the development of aircraft propulsion that, after examining some current designs, we extend our thermodynamic analysis to examine these events in some detail. Figure (3.11) shows the General Electric CF6-80C2 turbofan engine, whose takeoff thrust at sea level is approximately 53,000 lb $_f$ (233 kN); the corresponding airflow rate is approximately 1770 lb/sec (800 kg/s). The bypass ratio for this motor is 5, and the propeller tip diameter is 93 in (2.36 m). The total compression pressure ratio is 31:1. The engine weight is 8946 lb (4058 kg) and its full length is 160.9 in (4087 m). The 2-stage high-pressure turbine drives the 14-stage high-pressure compressor. The 5-stage low-pressure turbine drives the 3-stage low-pressure compressor and the propeller by means of a shaft internally concentric with the rotating structure that connects the high-pressure compressor and the turbine.

This high bypass (secondary) flow engine, certified in 1985 and used in the Airbus of Europe and the Boeing 767 aircraft, incorporates the benefits of research conducted (with the collaboration of General Electric) in the NASA Energy Efficient Engine Program. The turbine inlet temperature for this engine is approximately 1600 K (greater than 2400°F).

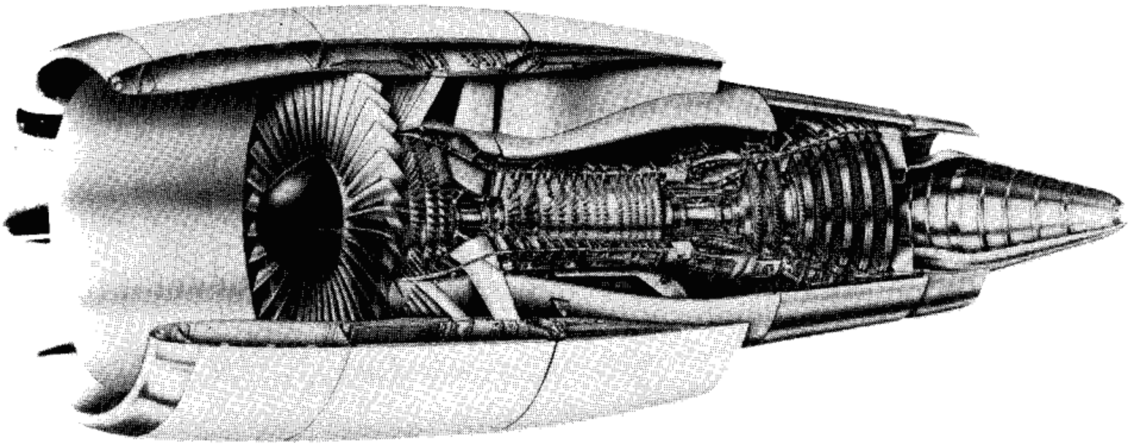


Fig. 3.11 : High Bypass Ration turbjet CF6-80C2
of General Electric. (Courtesy GE Aircraft Engines.)

Figure (3.12) shows the Pratt & Whitney PW4000 engine, which incorporates the results of Pratt and Whitney's collaboration in the Energy Efficient Engine program. Engine thrust is in the range of 222 to 289 kN (50,000 to 65,000 lb $_f$). At the thrust of 252 kN (56,750 lb $_f$), the bypass flux ratio is 4.8 and the overall compression ratio is 30.2. The high-pressure turbine inlet temperature is approximately 1628 K (2450°F) and the total airflow rate is 773 kg/s (1750 lb/sec). The specific fuel consumption at cruise thrust is 0.59 lb/hr/lb $_f$ (17 g/kN·s).

The end-to-end diameter of the PW4000 propeller is 2.38 m (93.4 in.) and its pressure ratio approximately 1.72. The 38 propeller rotor blades are made of titanium; The propeller stator is made of a carbon fiber composite. The rotor and stator blades in the low-pressure compressor are made of titanium, as are the rotor blades of the high-pressure compressor (the corresponding stator blades are made of nickel steel).

At the inlet of the high-pressure turbine, the vanes are air-cooled and are constructed of nickel steel. The first stage turbine blades each contain single crystals to improve their resistance to high temperatures. The tip diameter of the last stage of the turbine rotor is 1.36 m (53.4 in.).

Figure (3.13) shows the Rolls-Royce RB211 turbofan engine (in its 535E4 version). In the Boeing 757 this engine-aircraft combination achieved record fuel economy. The overall thrust of the engine is approximately 41,000 lb $_f$ (92 kN) and the bypass (secondary) ratio is 4. (This is not the largest of the RB211 family of engines; the 523L version develops 65,000 to 70,000 lb $_f$ of thrust.) The overall compressor compression ratio is 28.5:1. The tip diameter of the propeller is approximately 74 in. (1.88 m) and motor length is 117.9 in. (3 m). The mass of the engine is 7264 lb (3300 kg) and the breathing air flow is 1150 lb/sec (523 kg/s).

The engine has three concentric shafts; the outer shaft connects the single-stage high-pressure turbine to the six-stage high-pressure compressor. The intermediate shaft connects the single-stage intermediate turbine to the six-stage intermediate compressor. The three-stage low-pressure turbine drives the single-stage propeller, which requires approximately 36,000 hp

(27,000 kW). With directionally solidified nickel alloy blades, the maximum turbine inlet temperature is approximately 1530 K.

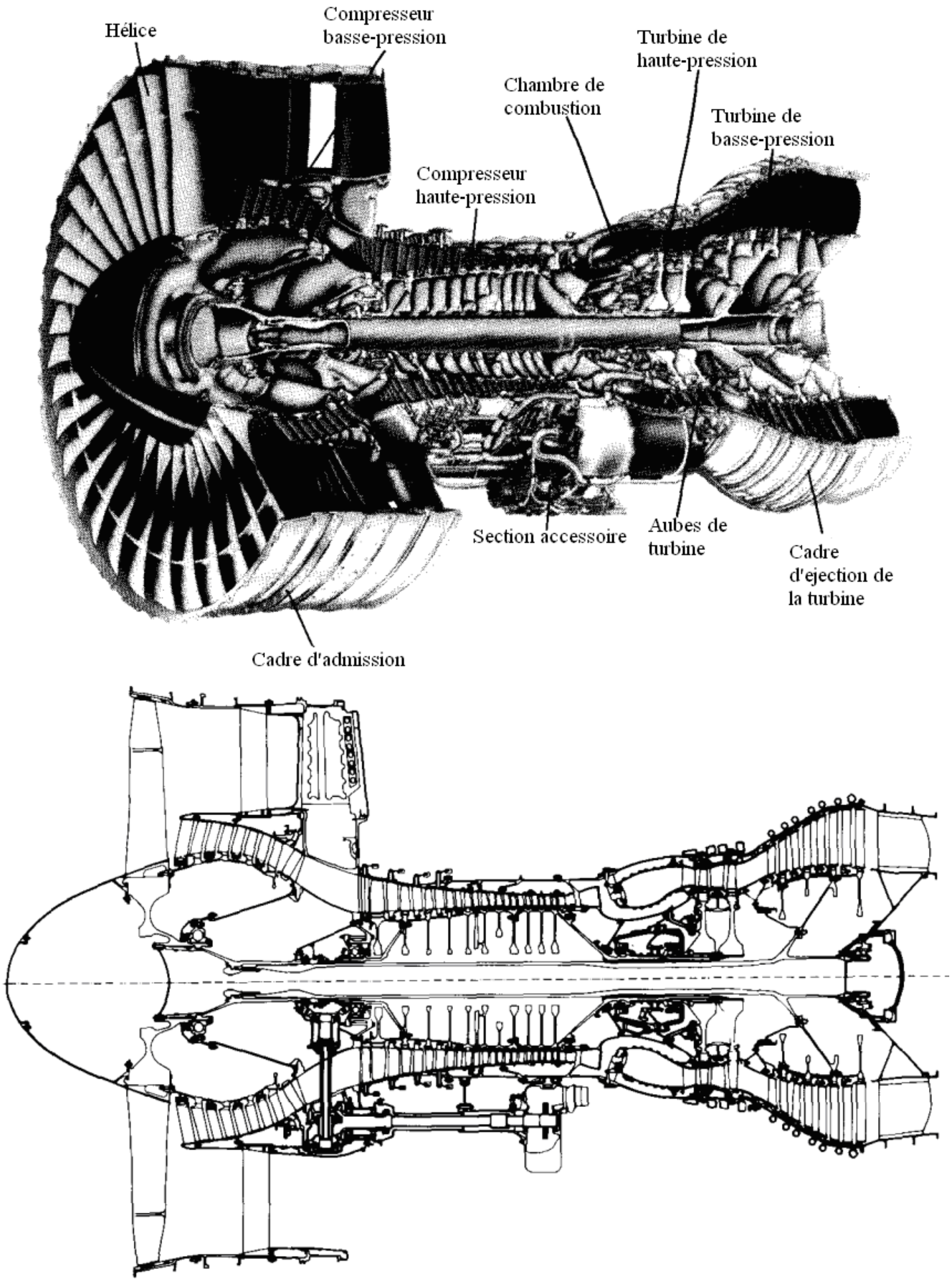


Fig. 3.12 : Turboréacteur à double flux PW4000 de Pratt & Whitney. (Courtesy Pratt & Whitney, a division of United Technologies Corp.)

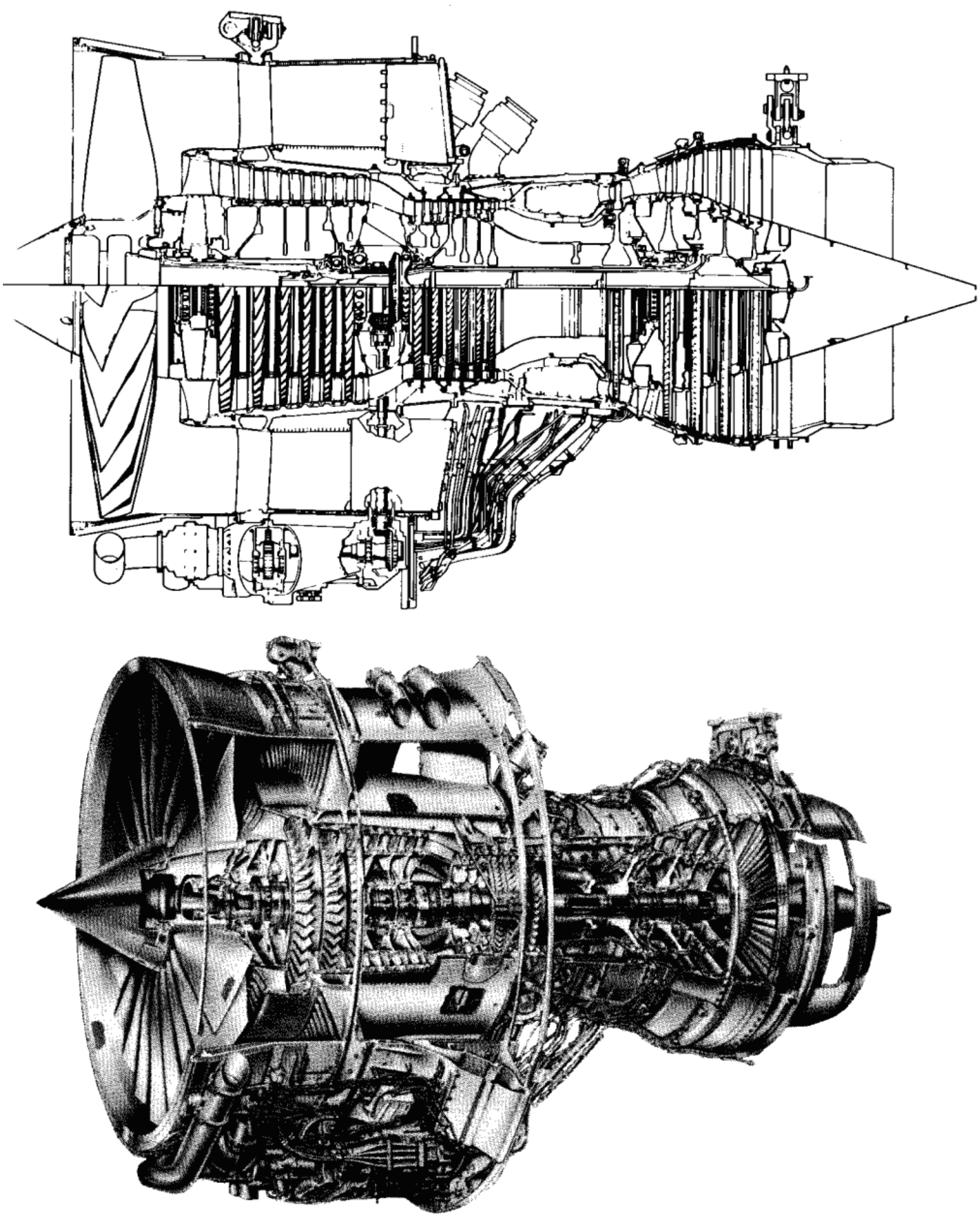


Fig. 3.13 : Turboréacteur à double flux RB211-535E4 de Rolls-Royce.
(Courtesy Rolls-Royce, pic.)

The propeller tip speed is approximately 450 m/s (1500 ft/sec), so the propeller blades are heavily loaded; a 15 lb fin is subjected to a centrifugal force of approximately 60 tonnes. The propeller must be designed to survive bird strikes (a test requiring 30 minutes of continuous engine operation after the propeller is struck with eight birds at 1.5 lb for a period of one second). The RB211 propeller blades are manufactured from titanium alloy surface panels which include a volume filled with a titanium honeycomb. The honeycomb is precision formed from 0.075 mm (0.003 in.) thick titanium sheet and is bonded to the surface panels. The structure has a high strength-to-weight ratio, good impact resistance and a long fatigue life under vibration stresses .

An interesting feature of this engine is that the propeller exhaust flow and the basic engine exhaust flow are mixed before expansion in a common nozzle. Figure (3.14) shows the mixing section of the Rolls-Royce RB211-524L engine. We can show the mixing of the primary flow with the bypass flow, for thermodynamic reasons, to provide a small but important efficiency advantage. Also, it can reduce the noise emission of the jet. The jet noise intensity is shown to be proportional to the eighth power of the jet's speed relative to the ambient air; such small reductions in speed can mean significant reductions in noise. On the other hand, much of the noise of a jet engine, particularly those sound emissions with frequencies that are uncomfortable for people, can come from the compressor.

Figure (3.15) shows an entirely different bypass flow engine, the Rolls-Royce Pegasus used to operate the Harrier V/STOL (vertical or short takeoff and landing) aircraft. This engine develops a vertical takeoff thrust of 98 kN (22,000 lb_f), which is 5.5 times its own weight. The horizontal thrust under static conditions at sea level is approximately 77 kN (17,000 lb_f). Figure (3.15b) shows the exhaust nozzles, which can be rotated to provide any combination of horizontal or vertical thrust.

The Pegasus engine has three low-pressure compressor stages and a secondary flow (for flow around the eight-stage high-pressure compressor) of 1.4. The high- and low-pressure compressors are on separate concentric shafts, each driven by two-stage turbines. The low-pressure compressor flow rate is 198 kg/s (435 lb/sec) and the overall pressure ratio is 14:1. The low-pressure compressor casing diameter is 1,219 m (48 in.). The first stage of the high-pressure turbine has single crystal blades.

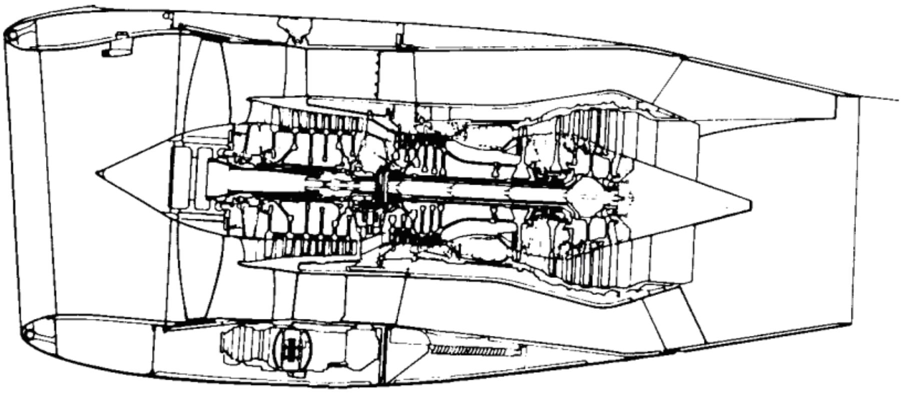
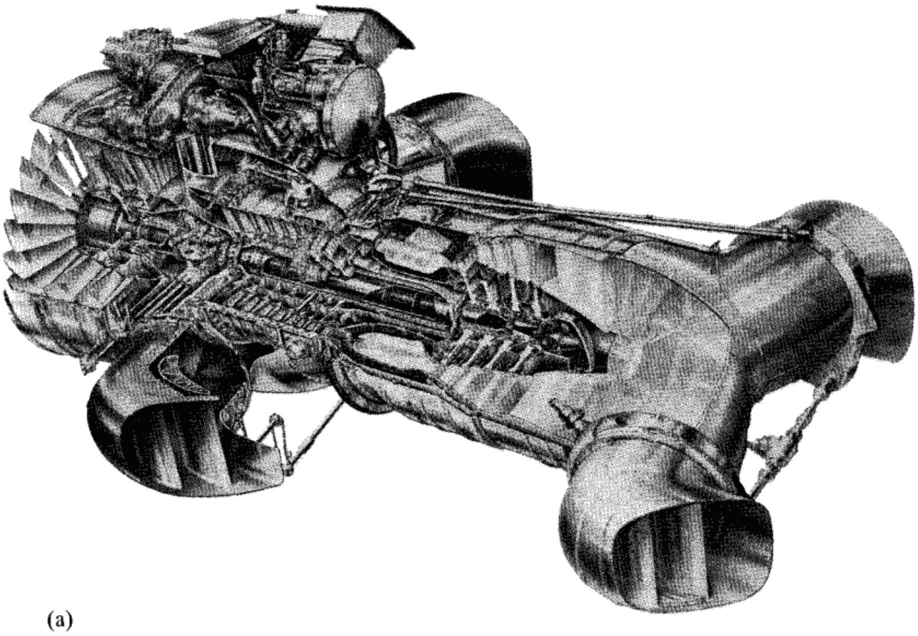
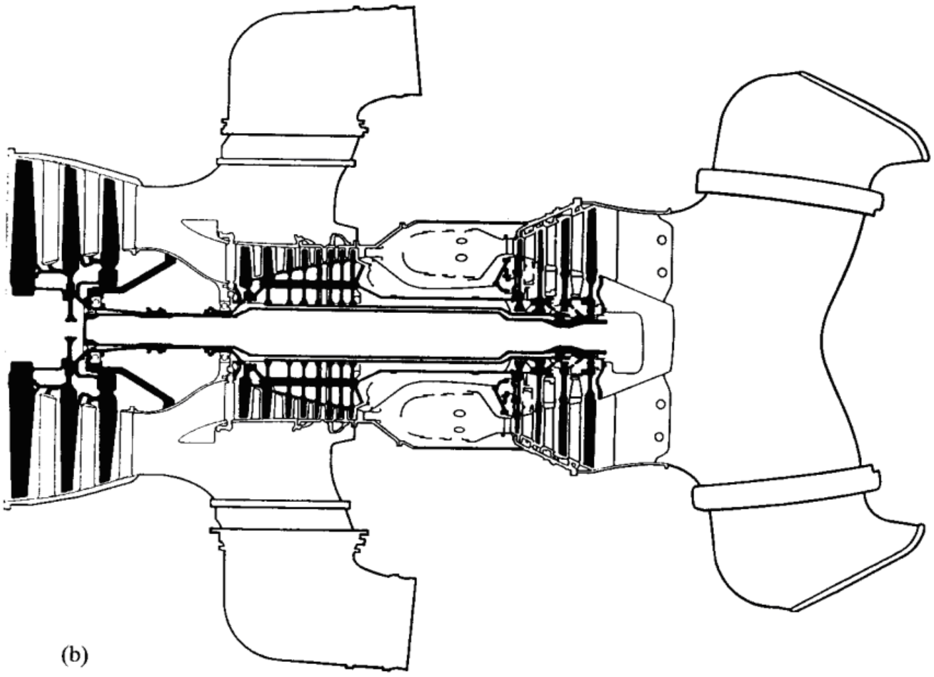


Fig. 3.14 : Vue en section de RB211-535E4 de Rolls-Royce.



(a)



(b)

Fig. 3.15 : Moteur Pegasus V/STOL de Rolls-Royce.
(Courtesy Rolls-Royce, plc.)

Having briefly reviewed some of the main features of examples of modern turbofan engines, we use fundamental thermodynamic analysis to examine the rationale for the development of turbofan engines and to preliminarily assess their turbojet benefits.

To analyze the performance of the engine propeller, we modify the performance analysis of the turbojet engine by taking into account the bypass ratio β (the ratio of the secondary (bypass) airflow rate to the primary airflow rate engine). In the following we assume that the primary engine flow and the secondary flow expand separately at ambient pressure. If the two streams are mixed before expansion, the analysis must be modified somewhat.

First we recognize that the equations for specific thrust and fuel consumption must be modified such that

$$\frac{\wp}{\dot{m}_a} = (1+f)u_e + \beta u_{ef} - (1+\beta)u$$

in which u_e is the primary exhaust speed and u_{ef} is the exhaust speed of the secondary flow. The thrust-specific fuel consumption equation can be written

$$TSFC = \frac{\dot{m}_f}{\wp} = \frac{f}{(1+f)u_e + \beta u_{ef} - (1+\beta)u} \qquad 3.13$$

The internal analysis of the turbofan engine is the same as that of the turbojet engine for steps 1 to 4 already shown. The rest of the modification procedure is as follows:

5. Inlet conditions of the propeller: Since the propeller and the main engine compressor receive air from the same diffuser, the inlet stagnation conditions of the propeller are T_{02} and p_{02}, previously determined from the Mach number of vol M, the ambient temperature and pressure T_a and p_a the efficiency of the diffuser η_d.

6. Propeller outlet conditions: We assume that the propeller has a design pressure ratio $p_{rf} = p_{08} / p_{02}$ and adiabatic efficiency η_f, in the case where the total propeller outlet pressure is

$$p_{08} = p_{02} p_{rf} \qquad 3.14$$

and the total outlet temperature is

$$T_{08} = T_{02}\left[1 + \frac{1}{\eta_f}(p_{rf}^{(\gamma_f - 1)/\gamma_f} - 1)\right] \qquad 3.15$$

in which γ_f is the (assumed constant) ratio of specific heats for the propeller flow.

7. The propeller nozzle exit velocity: By a derivation similar to that of equation (3.11) we can show that the propeller nozzle exit velocity u_{ef} is given by

$$u_{ef}=\sqrt{2\eta_{fn}\frac{\gamma_f}{\gamma_f-1}RT_{08}\left[1-(p_a/p_{08})^{(\gamma_f-1)/\gamma_f}\right]} \qquad 3.16$$

where η_{fn} is the adiabatic efficiency for the propeller nozzle.

8. Turbine output conditions: Here, taking into account the work done by the engine on the propeller, we write the power balance of the turbine in the form

$$\dot{m}_t c_{pt}(T_{04}-T_{05})=\dot{m}_a c_{pc}(T_{03}-T_{02})+\beta\dot{m}_a c_{pc}(T_{08}-T_{0a})$$

In which the second term on the right is the power absorbed by the propeller flow. As before, we make the approximation $\dot{m}_t c_{pt}\approx\dot{m}_a c_{pc}$, so that

$$T_{05}=T_{04}-(T_{03}-T_{02})-\beta(T_{08}-T_{0a}) \qquad 3.17$$

where T_{08} is the propeller outlet stagnation temperature. As before

$$p_{05}=p_{04}\left[1-\frac{1}{\eta_t}(1-\frac{T_{05}}{T_{04}})\right]^{\frac{\gamma_t}{\gamma_t-1}}$$

9. Nozzle admission conditions: As before, without post-combustion, $T_{06}=T_{05}$ and $p_{06}=p_{05}$.

10. Nozzle exit conditions: Again we see that the primary ejection speed of the engine can be written

$$u_e=\sqrt{2\eta_n\frac{\gamma_n}{\gamma_n-1}RT_{06}\left[1-(p_7/p_{06})^{(\gamma_n-1)/\gamma_n}\right]}$$

With $p_7=p_a$ unless the outlet nozzle is choked.

To illustrate the possible performance improvement with turbofan engines, we will use the same calculation assumptions as shown in Table (3.1) and the bypass flow assumptions through the propeller shown in Table (3.2) to obtain the results shown in figures (3.16 to 3.21).

We can see the effect of the bypass ratio β on takeoff thrust by comparing Figures (3.6) and (3.16). Other conditions remaining the same, the takeoff thrust per unit of primary engine flow $\dot{m}_a$ was almost doubled, as the bypass ratio was changed from 0 to 5. At the same time the thrust-specific fuel consumption decreased considerably.

Comparing Figures (3.7) and (3.17) shows the same sorts of benefits for the cruise thrust of high subsonic flight (M =0.85), although the relative benefits are considerably less. For supersonic flows the dual flow idea would have little potential benefit, and the problem of shock losses associated with a large frame would be formidable, if not prohibitive.

Figure (3.18) shows the efficiency of the turbojet for cruise conditions at M =0.85. On the contrary, Figure (3.19) shows the corresponding efficiency for the turbofan engine $\beta = 5$. The overall efficiency is significantly greater than for the turbojet.

Tab. 3.2: Secondary flow characteristics of a turbofan engine.

Component	Yield	Specific heat ratio
Streamer	$\eta_d = 0.97$	$\gamma_d = 1.4$
Helix	$\eta_f = 0.85$	$\gamma_f = 1.4$
Propeller nozzle	$\eta_{fn} = 0.97$	$\gamma_n = 1.4$
Propeller Compression Ratio $p_{rf} = 1.50$		

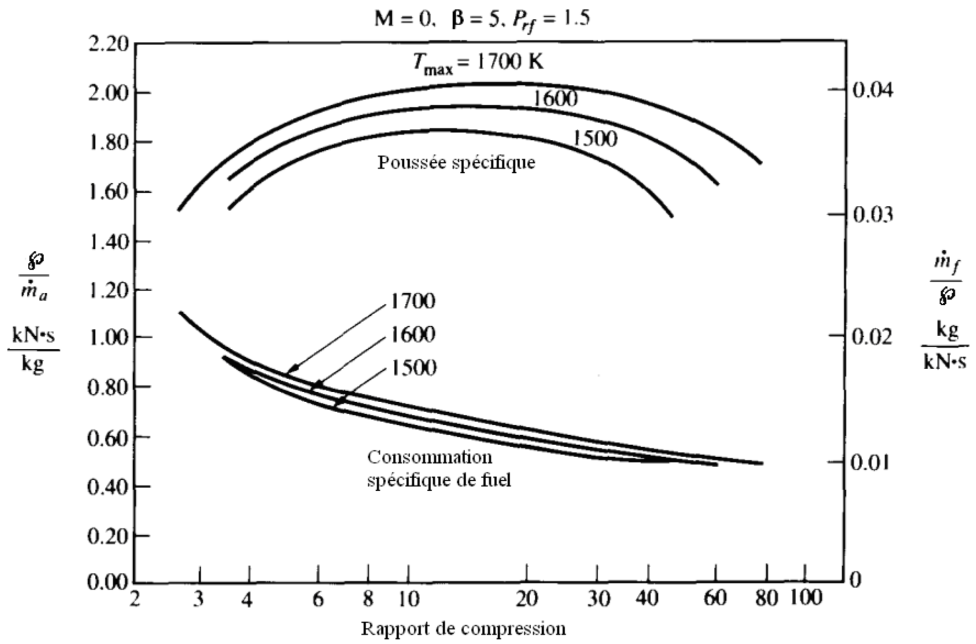


Fig. 3.16 : Poussée et consommation de fuel statiques d'un turboréacteur à double flux.

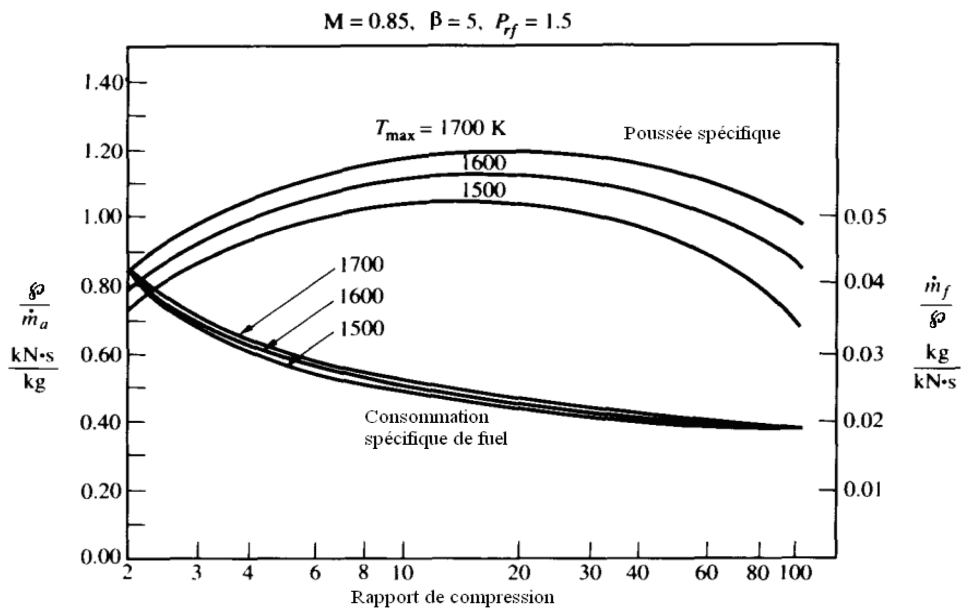


Fig. 3.17 : Poussée et consommation de fuel d'un turboréacteur

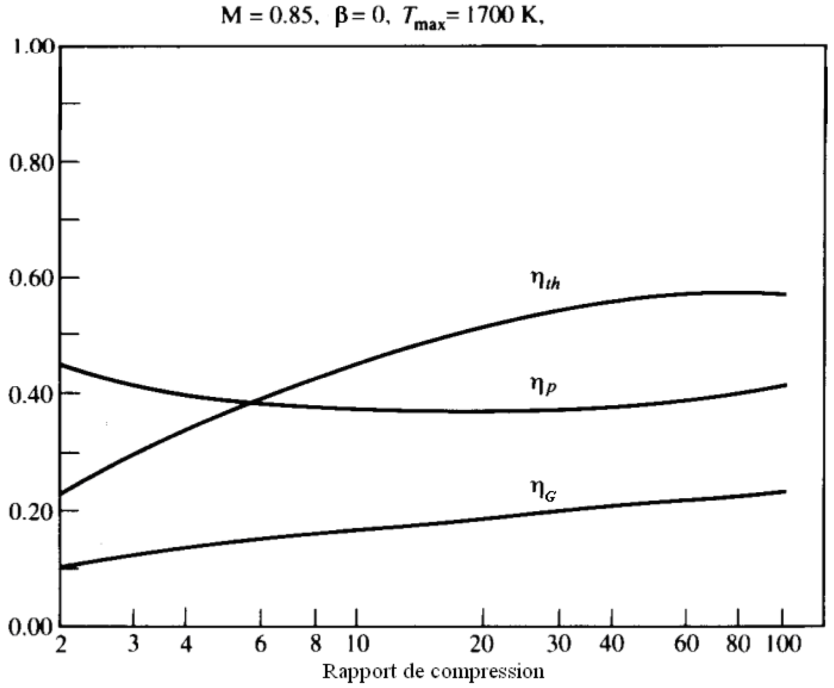


Fig. 4.18 : Rendements thermique et propulsif d'un turboréacteur.

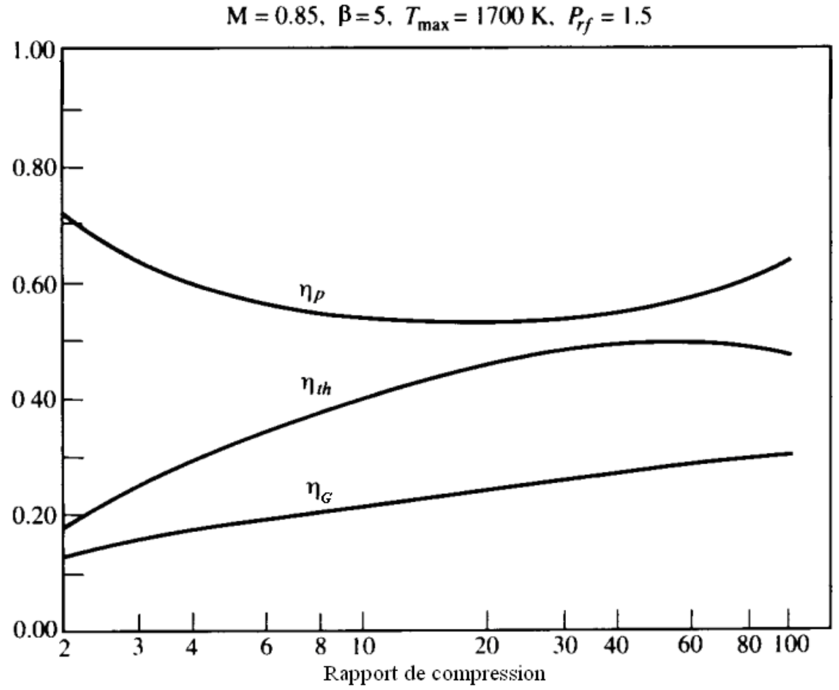


Fig. 4.19 : Rendements thermique et propulsif d'un turboréacteur à double flux.

Figures (3.20) and (3.21) show the large values of the overall efficiency potentially accessible with supersonic flight. The ratio of compressor pressures that maximizes η_G decreases rapidly as the design flight Mach number increases above, say, 2. For an aircraft that is required to cruise, subsonic and supersonic speeds, the choice of compression ratio of the engine involves a significant compromise.

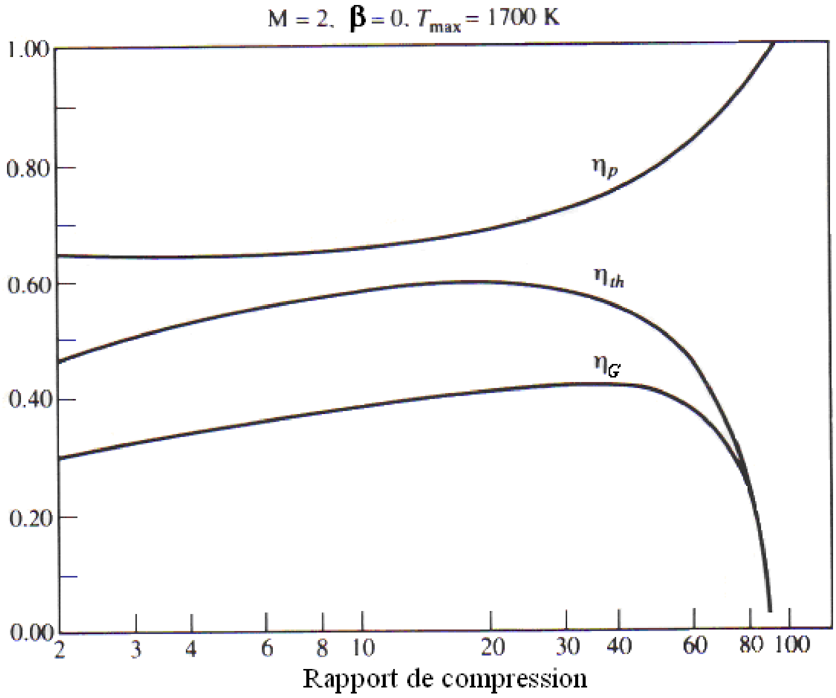


Fig. 4.20 : Rendements thermique et propulsif d’un turboréacteur.

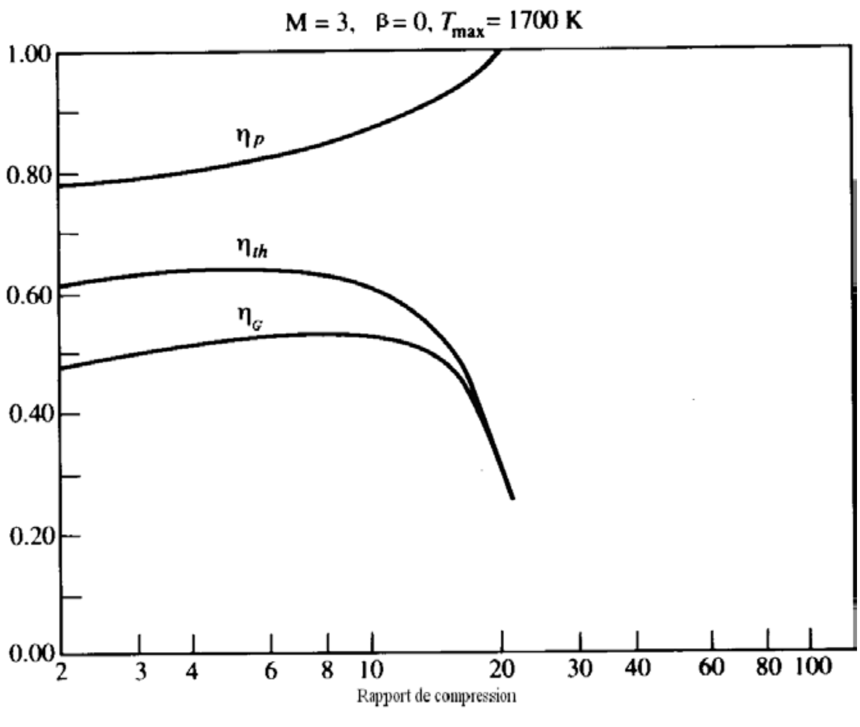


Fig. 4.21 : Rendements thermique et propulsif d’un turboréacteur.

For subsonic flight there is a strong advantage in propulsive efficiency in the use of a secondary flow rather than a turbojet. The higher the turbine inlet temperature (and thus the higher the corresponding turbojet exhaust speed), the greater the advantage. The questions of optimal bypass ratio and optimal propeller compression ratio deserve serious exploration and may lead to further refinements of turbofan designs . In the absence of suitable power transmission to the propeller, the problem of non-corresponding speed between the propeller and its drive turbine tends to limit the bypass ratio of turbofan engines. Other considerations affecting the choice of bypass ratio include the structural weight and aerodynamic drag of the engine frame. Designing a gearbox to transmit 30,000 kW would address serious questions regarding gearbox weight and reliability. However, the adapted propeller is a future possibility.

3-4 The Ramjet :

The simplest of all air-breathing engines is the ramjet. As shown schematically in Figure (3.22), it consists of a diffuser, a combustion chamber and an exhaust nozzle. The air enters the diffuser, where it is compressed before it is mixed with the fuel and burned in the combustion chamber. The hot gases are then expelled through the nozzle under the increased pressure of the diffuser as the incoming air is slowed from flight speed to a relatively low speed in the combustion chamber. Therefore, although ramjets can operate at subsonic flight speeds, the increasing pressure rise accompanying higher flight speeds makes the ramjet most suitable for supersonic flight.

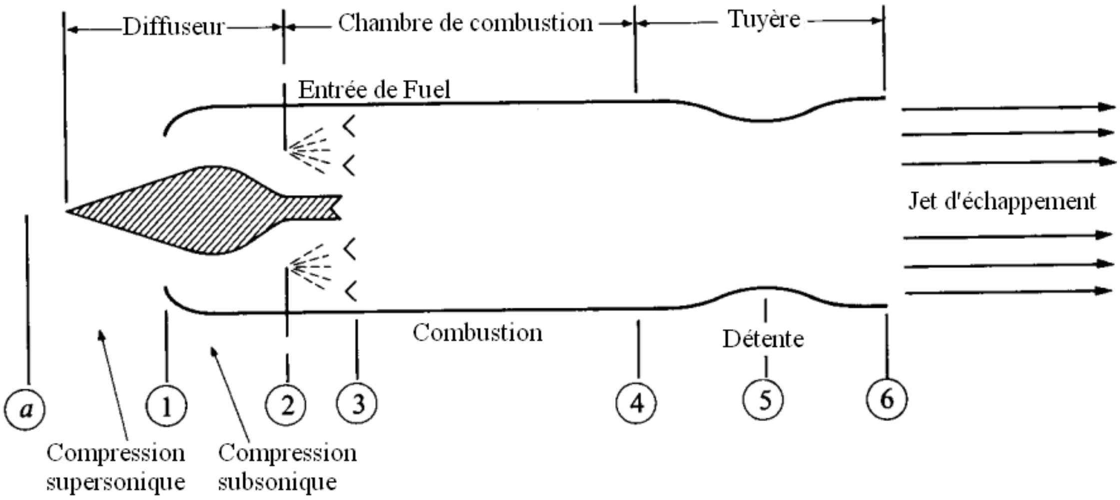


Fig. 4.22 : Diagramme schématique d’un statoréacteur.

Figure (3.22) is a typical supersonic ramjet engine which employs partially supersonic diffusion by a shock system. Since the combustion chamber requires an intake at Mach number of approximately 0.2 to 0.3, the pressure increase at supersonic flight speeds can be substantial. For example, for isentropic deceleration from M =3 to M =0.3, the ratio of static pressures between ambient and combustion chamber pressures would be approximately 34! Only a fraction of the isentropic pressure ratio is actually realized since, particularly at high Mach numbers, the stagnation pressure losses associated with shocks are substantial. After compression the air flows through fuel injectors, which spray a stream of fine fuel droplets so that the air and fuel mix as quickly as possible. The mixture then flows through the combustion chamber, which usually contains a "flame holder" to stabilize the flame, as shown in figure (3.22). Combustion raises the temperature of the mixture to perhaps 3000 K before the combustion products expand at high velocity into the nozzle. The reaction to the creation of fuel momentum is a thrust on the engine according to equation (1.9). This thrust is applied by pressure and shear forces distributed to the internal surfaces of the engine.

The materials now used for the walls of combustion chambers and nozzles cannot tolerate temperatures much higher than 1200 K, but they can be kept much colder than the main flow of fluid by an injection pattern (scheme). of fuel which leaves a protective layer of relatively cool air next to the walls. On the contrary, the turbine engine cannot operate at temperatures as high as 3000 K. The turbine blades are subject to high centrifugal tensions and cannot be cooled so easily. The lower maximum temperature limit of the turbojet greatly affects the relative performance and throws of the two engines, as we will see.

This relatively high limit of maximum ramjet temperature allows operation in flight at high Mach numbers. As the Mach number is increased, however, the combustion chamber inlet temperature also increases, and under any limitation of the Mach number it will approach the temperature limit set by wall materials and cooling methods. For example, in a flight at Mach number 8 in an environment at 225 K, the stagnation temperature is approximately 2500 K. At temperatures above 2500 K, the dissociation of components from the combustion products can be significant. At high temperatures the main effect of fuel injection is dissociation rather than actual temperature increase. If the dissociated combustion products recombine as they extend through the nozzle, the energy of combustion will always be partially transformed into the kinetic energy of the fuel. If not, the occurrence of dissociation could severely penalize performance.

A disadvantage of the ramjet is that the pressure ratio is strictly limited by the flight speed and the performance of the diffuser. The most serious consequence of this is the fact that the ramjet cannot develop static thrust and therefore cannot accelerate a vehicle from rest. In addition, the diffuser, whose behavior is important for the engine as a whole, is difficult to be designed for high efficiency. This is due to the harmful behavior of the boundary layer in which pressure gradients arise, particularly in the presence of shocks, which are practically inevitable during the supersonic regime. Supersonic diffusers designed for best performance at a given Mach number usually have poor performance at other Mach numbers unless their geometry is variable. The development of a large supersonic diffuser of reasonable performance and operating range requires extensive experimental work and substantial test equipment. Figure (3.23) is a view of a ramjet showing the geometry of the supersonic and subsonic diffusers, the fuel injector, the flame holder, the combustion chamber, and the nozzle.

The Ideal Ramjet:

To understand ramjet performance, it is useful to perform a thermodynamic analysis of a simplified model. Let us assume that the compression and expansion processes in the engine are reversible and adiabatic and the combustion process takes place at the constant pressure. These assumptions are not, of course, real. In the real diffuser, there are always irreversibilities due to shocks, mixing and wall friction. Also, we can note that unless combustion occurs at very low fluid velocity, the static and total pressures will drop, as a result of heat addition. The ideal ramjet engine is a very useful concept, however, since its performance is the highest that the laws of thermodynamics will allow and is the limit that real engines will approach if their irreversibilities can be reduced. Using the station numbers in Figure (3.22), Figure (3.24) shows, on a temperature-entropy *T-s* diagram, the processes through which air passes through an ideal ramjet engine. The compression process takes air from its condition at the station i(02)tropically to its (*a*) stagnant state at the station. (2)

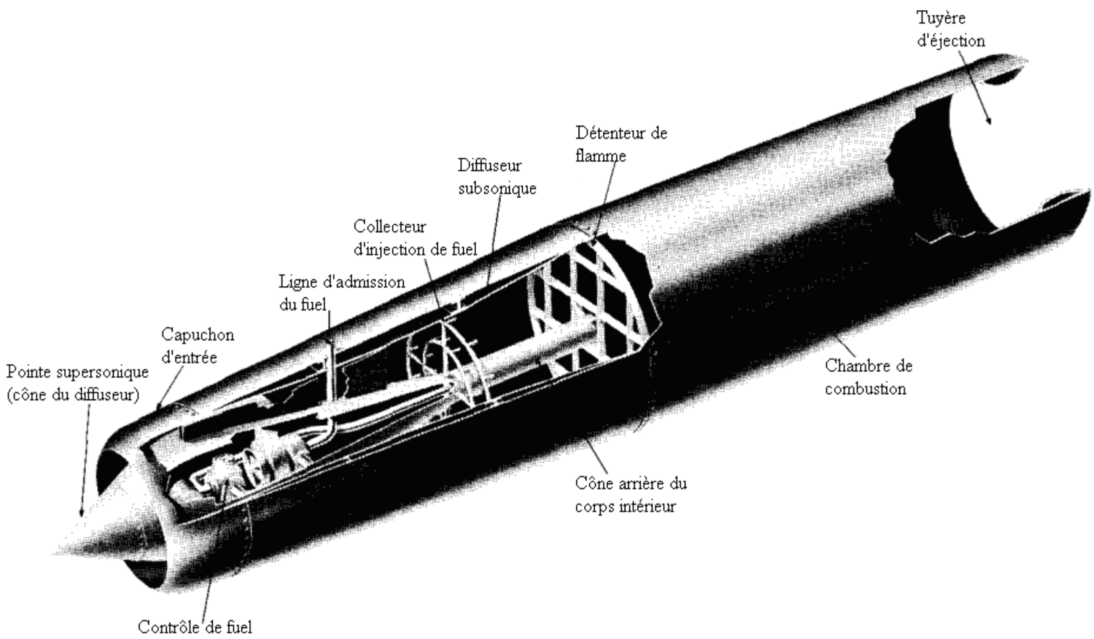


Fig. 4.23 : Vue écorchée d'un statoréacteur.
(Courtesy Marquardt Aircraft Co.) .

The combustion process is represented by an addition of heat and mass at constant pressure from to (02) to (04) maximum temperature T_{04}. The outlet nozzle expands the combustion products isentropically at ambient pressure. The ideal engine thrust can be obtained from equation (1.9):

$$\wp = \dot{m}_a \left[(1+f)u_e - u \right] \qquad 4.18$$

With the isentropic processes of compression and expansion, and the slow addition of heat and mass at constant pressure and low speed, it follows that the stagnation pressure must be constant throughout the engine. Thus $p_{0a} = p_{06}$.

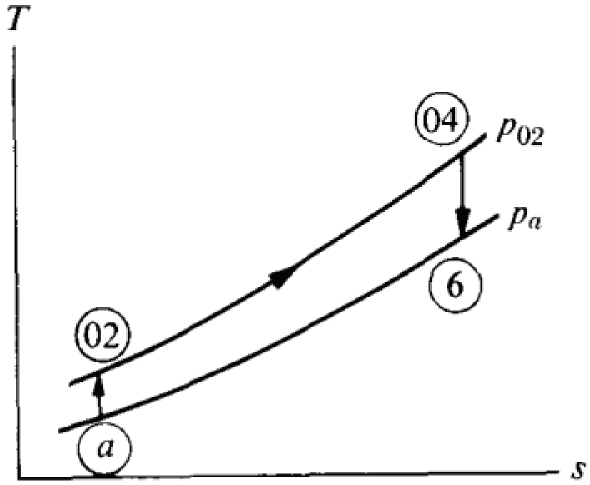


Fig. 3.24 : Cycle thermodynamique du fluide dans un statoréacteur idéal.

If we ignore variations in fluid properties (R,γ) across the engine for this ideal case, then

$$\frac{p_{0a}}{p_a}=\left(1+\frac{\gamma-}{2}M^2\right)^{\frac{\gamma}{\gamma-1}} \text{and} \frac{p_{06}}{p_e}=\left(1+\frac{\gamma-1}{2}M_e^2\right)^{\frac{\gamma}{\gamma-1}}$$

in which M is the flight Mach number and M_e is the Mach number in the ejection plane. So, with the condition $p_e = p_a$, it is clear that

$$\frac{p_{0a}}{p_a}=\frac{p_{06}}{p_a} \text{ and } M_e = M_a$$

So we can determine the escape velocity of

$$u_e=\frac{a_e}{a_a}u$$

where a is the speed of sound. From $a=\sqrt{\gamma RT}$, then $a_e / a_a = \sqrt{T_e / T_a}$. However, for the case $M_e = M_a$, $T_e / T_a = T_{06} / T_{0a}$ and, since $T_{04} = T_{06}$, then

$$u_e = \sqrt{T_{04} / T_{0a}}\,u \qquad 4.19$$

The energy equation applied to the idealized combustion process, if we neglect the enthalpy of the incoming fuel, is

$$(1+f)\,h_{04} = h_{02} + fQ_R \qquad 4.20$$

where f is the proportion of fuel-air and Q_R is the calorific value of the fuel. If the specific heat is assumed to be constant, then equation (3.20) can be solved for f in the form

$$f = \frac{(T_{04}/T_{0a}) - 1}{(Q_R/c_p T_{0a}) - T_{04}/T_{0a}} \qquad 4.21$$

Equations (3.18) and (3.19) can be combined to give the thrust per unit mass of airflow,

$$\frac{\wp}{\dot{m}_a} = M\sqrt{\gamma R T_a}\left[(1+f)\sqrt{T_{04}/T_a}\left(1+\frac{\gamma-1}{2}M^2\right)^{-1/2} - 1\right] \qquad 4.22$$

where f given by equation (3.21). We give the fuel consumption specific to the thrust

$$TSFC = \frac{\dot{m}_f}{\wp} = \frac{f}{\wp/\dot{m}_a} \qquad 4.23$$

Figure (3.25) shows, for the ideal ramjet, the dependence of the specific thrust and the fuel-air ratio on the flight Mach number and the maximum temperature. The calculations are entirely approximate, since:

1. Variations in the ratio of specific heats with temperature have been neglected;
2. No friction or shock losses have been taken into account;
3. No allowance was made for the dissociation of combustion products.

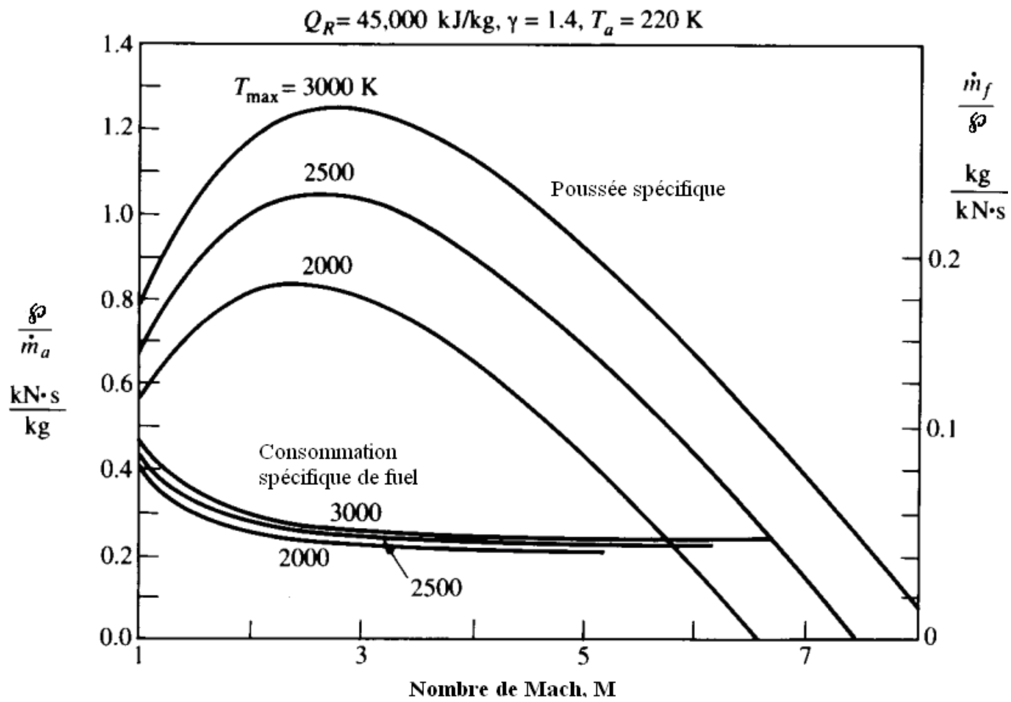


Fig. 3.25 : Poussée et consommation de fuel d'un statoréacteur idéal.

Despite these approximations, however, Figure (3.25) shows qualitatively the same behavior as that of real ramjets, which require supersonic flight speed for acceptable specific thrust and reasonably low specific fuel consumption. Figure (3.26), which also applies to the ideal ramjet, shows that, although the highest specific thrust is associated with a flight Mach number of about 2.6, Mach numbers well above 3 could provide a better reach. Figure (3.26) shows the overall efficiency increasing sharply, while the specific thrust drops sharply, with increasing *M*. This is an example of a common result where the conditions for minimum fuel consumption (or maximum range) differ quite from those where engine size per unit thrust is minimum. As the trends in Figure (3.25) indicate, the thrust-specific fuel consumption remains finite as the specific thrust approaches zero.

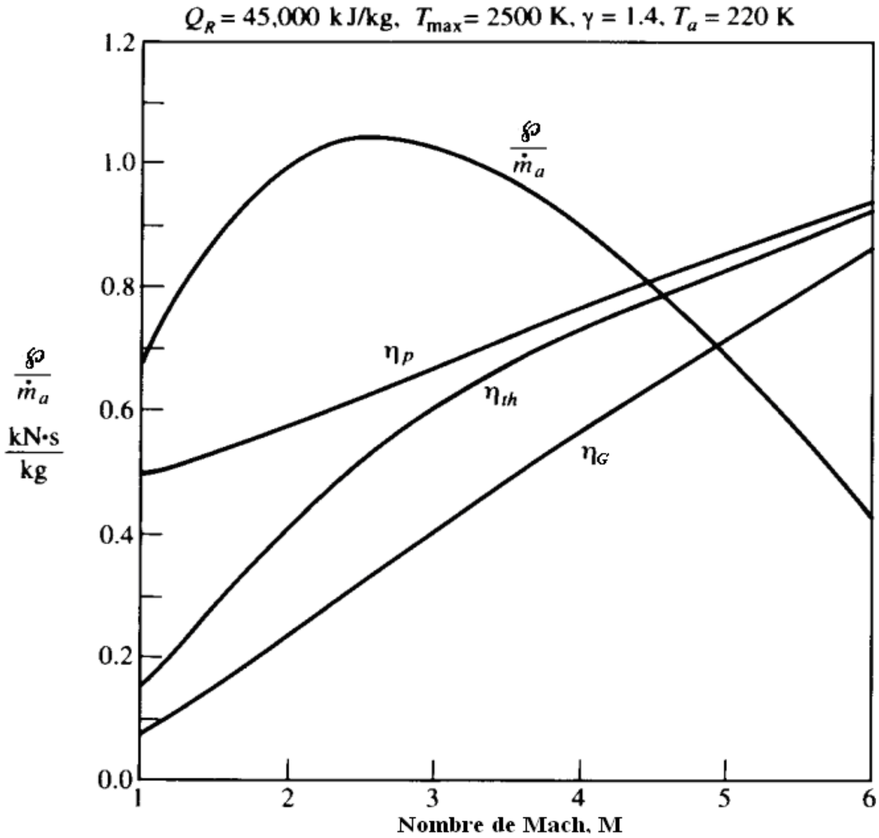


Fig. 3.26 : Poussée et rendements d'un statoréacteur

The effect of aerodynamic losses:

The fuel in an actual ramjet, of course, experiences total pressure losses as it flows through the engine. Figure (3.27) shows, on a *T-s* diagram, the effect of these irreversibilities on the compression, combustion and expansion processes.

The compression process, at a n 02nger isentropic, although the isentropic process is shown for comparison. The stagnation pressure at the end of the process is lower than it would be if the compression were isentropic. The performance of diffusers can be characterized by a stagnation pressure ratio r_d defined by

$$r_d = \frac{p_{02}}{p_{0a}} \qquad 4.24$$

Similarly, the total pressure ratios can be defined for combustion chambers r_c and nozzles r_n as follows:

$$r_c = \frac{p_{04}}{p_{02}}, \qquad 4.25$$

$$r_n = \frac{p_{06}}{p_{04}},$$ 4.26

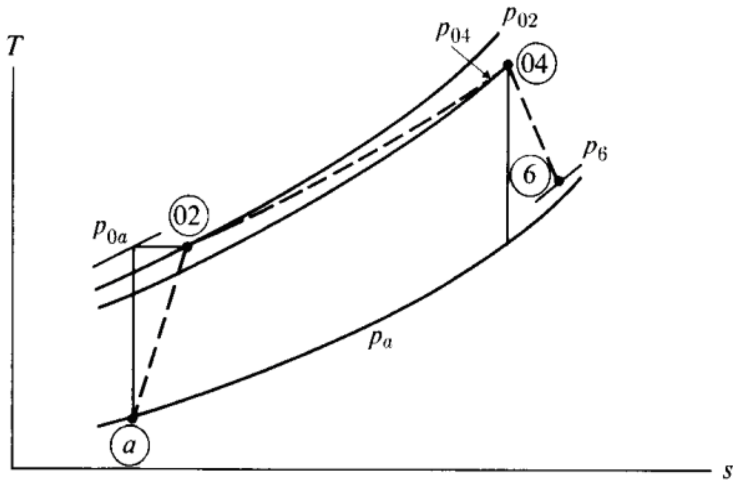


Fig. 3.27 : Diagramme *T*-*s* représentant les pertes aérodynamiques durant les processus d'un statoréacteur.

The overall stagnation pressure ratio is therefore

$$\frac{p_{06}}{p_{0a}} = r_d r_c r_n$$

Further, the actual exhaust pressure p_e or p_6 cannot equal the ambient pressure p_a. However, using the equation

$$\frac{p_0}{p} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}}$$ 4.27

for γ constant we can write the exhaust Mach number as

$$M_e^2 = \frac{2}{\gamma - 1}\left[\left(1 + \frac{\gamma - 1}{2} M^2\right)\left(\frac{p_{06}}{p_{0a}} \frac{p_a}{p_e}\right)^{\frac{(\gamma - 1)}{\gamma}} - 1\right]$$

So, in terms of component total pressure ratios,

$$M_e^2 = \frac{2}{\gamma - 1}\left[\left(1 + \frac{\gamma - 1}{2} M^2\right)\left(r_d r_c r_n \frac{p_a}{p_e}\right)^{\frac{(\gamma - 1)}{\gamma}} - 1\right]$$ 4.28

If engine heat transfer is assumed to be negligible (per unit mass of fluid), then the exhaust velocity u_e is given by $u_e = M_e \sqrt{\gamma R T_e}$ or, in terms of the total exhaust temperature,

$$u_e = M_e \sqrt{\gamma R T_{04} \Big/ \left(1 + \frac{\gamma - 1}{2} M_e^2\right)}$$ 4.29

Since irreversibilities have no effect on stagnation temperatures anywhere in the engine, a modified form of equation (3.21) gives the fuel-air ratio needed to produce the T_{04} desired:

$$f = \frac{(T_{04}/T_{0a})-1}{(\eta_b Q_R / c_p T_{0a}) - T_{04}/T_{0a}},$$

where η_b is the combustion efficiency and $\eta_b Q_R$ is the actual amount of heat output per unit mass of fuel. The thrust per unit mass of the air flow then becomes

$$\frac{\wp}{\dot{m}_a} = \left[(1+f)u_e - u\right] + \frac{1}{\dot{m}_a}(p_e - p_a)A_e$$

or, if we use equations (3.28 and 3.29),

$$\frac{\wp}{\dot{m}_a} = (1+f)\sqrt{\frac{2\gamma R T_{04}(m-1)}{(\gamma-1)m}} - M\sqrt{\gamma R T_a} + \frac{p_e A_e}{\dot{m}_a}\left(1 - \frac{p_a}{p_e}\right), \quad 3.30$$

in which

$$m = \left(1 + \frac{\gamma-1}{2}M^2\right)\left(r_d r_c r_n \frac{p_a}{p_e}\right)^{\frac{\gamma-1}{\gamma}}.$$

Again, the fuel consumption specific to the thrust is given by:

$$TSFC = \frac{f}{\wp/\dot{m}_a}.$$

We can evaluate the effects of aerodynamic losses by assuming the loss coefficients $r_d = 0.7$, $r_b = 0.95$ and $r_n = 0.98$. Their effects are expressed by a change in *m* and figures (3.28 and 3.29) provide the results. The comparison between those and the figures (3.25 and 3.26) shows a reduction of less than 10% in the maximum specific thrust and also in the specific fuel consumption.

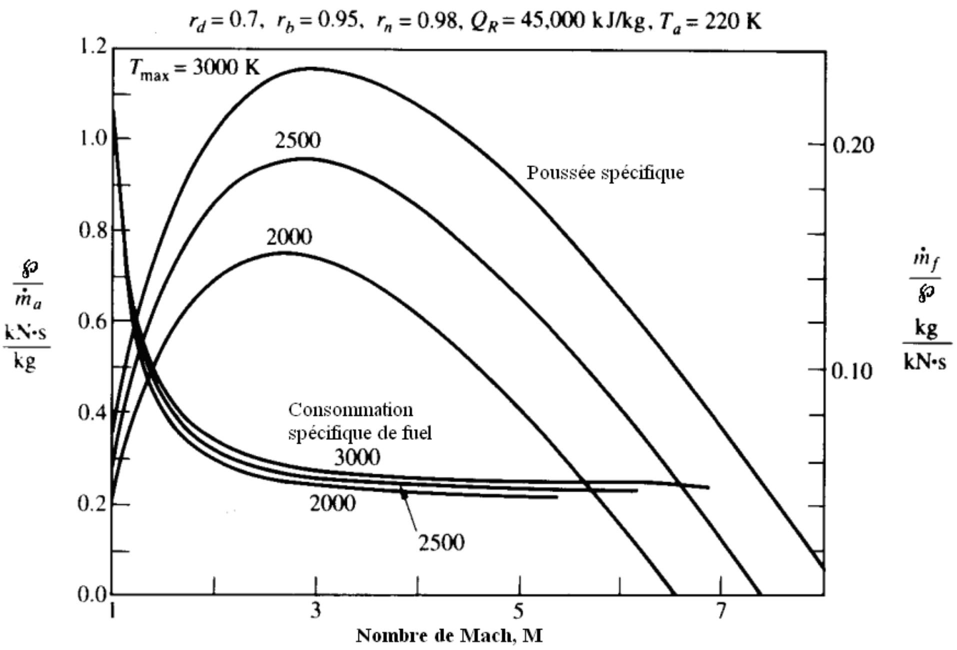


Fig. 3.28 : Poussée et consommation de fuel d'un

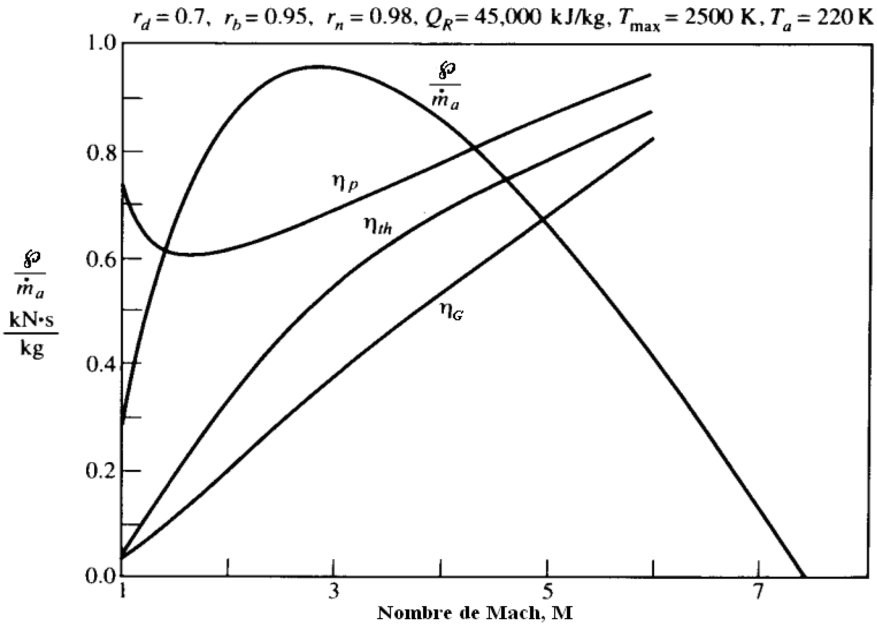


Fig. 3.29 : Poussée et rendements d'un

It is not realistic to assume that these loss coefficients, particularly r_d, are independent of the flight Mach number. Diffuser losses can be expected to depend strongly on the geometry of the diffuser; For efficient flight over a wide Mach number range, the supersonic air intake should have variable (and possibly complex) geometry to avoid excessive shock losses. Part of the reason why variability is necessary is that to keep shock losses low, a combination of oblique shocks would be necessary and the oblique shock angle depends on the upstream Mach number.

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Aerospace - Introduction to Propulsion
www.youtube.com/watch?v=Hlj2eVt1Vbk&list=PLbMVogVj5nJQt5nsksLn4qcsBrDL_JKkd

Aircraft Engine Types and Propulsion Systems | How Do They Work?
www.youtube.com/watch?v=amvrL0FU1ng

Jet Aircraft Propulsion
www.youtube.com/playlist?list=PLbMVogVj5nJS7srFCd_hJwdgHRd27YMgF

Abstract:

Propulsion mechanics is a crucial domain in aerospace engineering and vehicle technology. This course material delves into the fundamental principles of reaction propulsion, with a focus on the performance of aviation engines. Through detailed analysis, it examines the characteristics and applications of ramjets, turbojets, and rocket engines. The introduction provides essential context for understanding these concepts, while each section explores in-depth the mechanisms and performance of each engine type. This document aims to offer a comprehensive understanding of propulsion mechanics and its importance in the aerospace field.

Keywords: Propulsion mechanics, Jet propulsion, Ramjets, Turbojets, Rocket engines.

Résumé:

La mécanique de propulsion est un domaine crucial de l'ingénierie aérospatiale et de la technologie des véhicules. Cette brochure explore en profondeur les principes fondamentaux de la propulsion par réaction, en mettant l'accent sur les performances des moteurs d'aviation. À travers une analyse détaillée, elle examine les caractéristiques et les applications des statoréacteurs, des turboréacteurs et des moteurs de fusées. L'introduction fournit un contexte essentiel pour comprendre ces concepts, tandis que chaque section explore en détail les mécanismes et les performances de chaque type de moteur. Ce document vise à offrir une compréhension approfondie de la mécanique de propulsion et de son importance dans le domaine de l'aérospatiale.

Mots-clés: Mécanique de propulsion, moteurs à réaction, Statoréacteurs, Turboréacteurs, Moteurs des fusées.