Hindustan Aeronautics Limited
AERDC

CV Raman Nagar Post, Bangalore

Internship Report

AE14B021: Prem Sagar S
AE14B023: Puneet Kumar
AE14B035: Omkar Walvekar
AE14B051: Abhishek Kelkar

June 21, 2016

Contents

1 INTRODUCTION TO HAL

1.1 History of HAL

HAL was established as **Hindustan Aircraft** in Bangalore in 1940, on 23 Dec 1940. Hindustan Aircraft Company was duly incorporated under the Mysore Companies Act as a private Ltd Company. **Walchand Tulsidas- Khatau Ltd** was the Managing agency. It first directors were: Mr. Walchand Hirachand, Chairman, Mr.Tulsidas Khilachand, Mr.Dharmsey Mularaj Khatau, Mr.A.N. Raghavachar (Mysore State Representative), Mr.Venkatanaranappa (Mysore State Representative). Companys office was opened at a bungalow called "Eventide" on Domlur Road.The initiative was actively encouraged by the Kingdom of Mysore, especially by its Young Maharaja, H.H.Jayachamarajendra Wadiyar and the Diwan, Sir Mirza Ismail. Walchand had fist approached share holders of his own company - The Scindia Steam Navigation Company Ltd for diversifying but was refused. Then he wrote to the Rulers of Baroda, Gwalior and Bhavanagar without success. Only Maharaja of Mysore responded favorably by agreeing to invest 25 lakhs and gave initial 700 acres of land free.

The organisation and equipment for the factory at Bangalore was set up by William D Pawley of the Intercontinental Aircraft Corporation of New York, who had already established Central Aircraft Manufacturing Company (CAMCO) in partnership with Chinese Nationalist government in China. Pawley managed to obtain a large number of machine-tools and equipment from the United States.

The Indian Government bought a one-third stake in the company and by April 1941 by investing 25 lakhs as it believed this to be a strategic imperative. The decision by the government was primarily motivated to boost British military hardware supplies in Asia to counter the increasing threat posed by Imperial Japan during Second World War. The Kingdom of Mysore supplied two directors, Air Marshal John Higgins was resident director. The first aircraft built was a Harlow PC-5[2] On 2 April 1942, the government announced that the company had been nationalised when it had bought out the stakes of Seth Walchand Hirachand and other promoters so that it could act freely. The Mysore Kingdom refused to sell its stake in the company but yielded the management control over to the Indian Government.

In 1943 the Bangalore factory was handed over to the United States Army Air Forces but still using Hindustan Aircraft management. The factory expanded rapidly and became the centre for major overhaul and repair of American aircraft and was known as the 84th Air Depot. The first aircraft to be overhauled was a Consolidated PBY Catalina followed by every type of aircraft operated in India and Burma. When returned to Indian control two years later the factory had become one of the largest overhaul and repair organisations in the East. In the post war reorganisation the company built railway carriages as an interim activity.

After India gained independence in 1947, the management of the company was passed over to the Government of India.

Hindustan Aeronautics Limited (HAL) was formed on 1 October 1964 when Hindustan Aircraft Limited joined the consortium formed in June by the IAF Aircraft Manufacturing Depot, Kanpur (at the time manufacturing HS748 under licence) and the group recently set up to manufacture MiG-21 under licence (with its new factories planned in Koraput, Nasik and Hyderabad).[3] Though HAL was not used actively for developing newer models of fighter jets, except for the HF-24 Marut, the company has played a crucial role in modernisation of theIndian Air Force. In 1957 company started manufacturing Bristol Siddeley Orpheus jet engines under licence at new factory located in Bangalore.

During the 1980s, HAL's operations saw a rapid increase which resulted in the development of new indigenous aircraft such as the HAL Tejas and HAL Dhruv. HAL also developed an advanced version of the Mikoyan-Gurevich MiG-21, known as MiG-21 Bison, which increased its life-span by more than 20 years. HAL has also obtained several multimillion-dollar contracts from leading international aerospace firms such as Airbus, Boeing and Honeywell to manufacture aircraft spare parts and engines.

By 2012, HAL was reportedly been bogged down in the details of production and has been slipping on its schedules. On 1 April 2015, HAL reconstituted its Board with Mr. TS Raju as CMD, Mr. S Subrahmanyan as Director (Operations), Mr. VM Chamola as Director (HR), Dr. AK Mishra as Director (Finance) and Director (Engineering, Research and Development) to be selected by PESB. There are two Govt. nominees in the Board and six independent Directors.

1.2 About AERDC

AERDC is a Research and Development wing of Design Complex, HAL playing a vital role in the design and development of Gas Turbine Engines in India. It has successfully designed, developed, produced and type certified aero engines (PTAE and GTSU) which are in operation with the defence services. Patents have also been issued by Patent Office, Govt. of India for these two engines.

The Centre started functioning in the year 1960 as a hub of engine research and development with the aim of achieving self reliance in design and development of Gas Turbine Engines and Test Beds in a climate of growing professional competence.

The Hindustan Jet Engine or HJE-2500 project was the first jet engine design and development project in the country taken up by the Centre. A prototype engine was successfully developed and proven on the test bed in 1965 and was proposed for the HJT -16 aircraft. The centre also designed and type certified a piston engine and other engine accessories including a pneumatic starter and hydraulic pump in its early days. The centre also has the unique distinction of being the only design house which has developed test beds for both Western as well as Russian origin engines.

Leveraging on this experience end to end services right from preliminary design to the final product manufacturing and testing are offered.AERDC has fledged its wings and is continuously improvising to meet customer requirements. With the advent of new technologies the division has equipped itself with high end design, analysis and manufacturing tools to meet the challenges of the future.

Developed and Certified Projects

- HJE-2500 engine.
- PTAE-7 engine for Lakshya aircraft

- GTSU-110 starter engine for LCA (Tejas)
- Shakti engine co-development with Turbomeca, France
- Test Bed Projects:
  - TM 333-2B2 / Shakti Engine Test Bed
  - Adour Engine Test Bed
  - Pegasus Engine Test Bed
  - Garrett Engine Test Bed
  - R29B Engine Test Bed
  - LM 2500 Engine Test Bed
- Accessories
  - Pneumatic starter (HFES)
  - Hydraulic pump (HHP)

Current Projects

- GTSU 127 starter for LCA Mk 2 main engine
- Auxiliary power unit for FGFA
- Turbocharger
- Smoke generator
- Air producer for Jaguar aircraft
- Compact auxiliary power unit
- Indigenisation
- Air turbine wheel (ATW)
- Air turbine starter (ATS)
- Test beds
- Shakti engine test bed at Bangalore
- GE F 404 test bed at Sulur
- AI 31 FP Engine Mobile Test bed
- AI 551 engine test bed at Hakimpet
- Future Programmes
- Small turbofan engines for UAV
- APU for MTA
- Starter engine for fighter aircraft engine
- TurboProp engine for trainer aircraft
- Gas turbine engine for industrial application-4 MW class

2 INTRODUCTION TO GAS TURBINES

2.1 Theory of operation

In an ideal gas turbine, gases undergo three thermodynamic processes: an isentropic compression, an isobaric (constant pressure) combustion and an isentropic expansion. Together, these make up the Brayton cycle.

In a real gas turbine, mechanical energy is changed irreversibly (due to internal friction and turbulence) into pressure and thermal energy when the gas is compressed (in either a centrifugal or axial compressor). Heat is added in the combustion chamber and the specific volume of the gas increases, accompanied by a slight loss in pressure. During expansion through the stator and rotor passages in the turbine, irreversible energy transformation once again occurs.

If the engine has a power turbine added to drive an industrial generator or a helicopter rotor, the exit pressure will be as close to the entry pressure as possible with only enough energy left to overcome the pressure losses in the exhaust ducting and expel the exhaust. For a turboprop engine there will be a particular balance between propeller power and jet thrust which gives the most economical operation. In a jet engine only enough pressure and energy is extracted from the flow to drive the compressor and other components. The remaining high pressure gases are accelerated to provide a jet to propel an aircraft.

The smaller the engine, the higher the rotation rate of the shaft(s) must be to attain the required blade tip speed. Blade-tip speed determines the maximum pressure ratios that can be obtained by the turbine and the compressor. This, in turn, limits the maximum power and efficiency that can be obtained by the engine. In order for tip speed to remain constant, if the diameter of a rotor is reduced by half, the rotational speed must double. For example, large jet engines operate around 10,000 rpm, while micro turbines spin as fast as 500,000 rpm.

Mechanically, gas turbines can be considerably less complex than internal combustion piston engines. Simple turbines might have one main moving part, the compressor/shaft/turbine rotor assembly (see image above), with other moving parts in the fuel system. However, the precision manufacture required for components and the temperature resistant alloys necessary for high efficiency often make the construction of a simple gas turbine more complicated than a piston engine.

More advanced gas turbines (such as those found in modern jet engines) may have 2 or 3 shafts (spools), hundreds of compressor and turbine blades, movable stator blades, and extensive external tubing for fuel, oil and air systems.

Thrust bearings and journal bearings are a critical part of design. Traditionally, they have been hydrodynamic oil bearings, or oil-cooled ball bearings. These bearings are being surpassed by foil bearings, which have been successfully used in micro turbines and auxiliary power units.

2.2 Brayton Cycle

The Brayton cycle is a thermodynamic cycle that describes the workings of a constant pressure heat engine. The original Brayton engines used piston-compressor and expander systems, but more modern gas turbine engines and airbreathing jet engines also follow the Brayton cycle. Although the cycle is usually run as an open system (and indeed must be run as such if internal combustion is used), it is conventionally assumed for the purposes of thermodynamic analysis that the exhaust gases are reused in the intake, enabling analysis as a closed system.

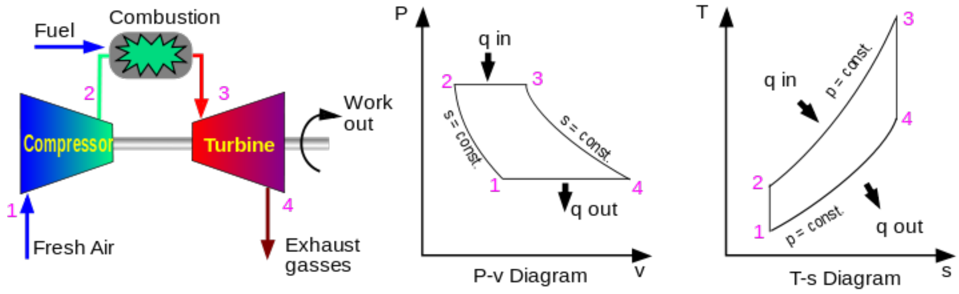


Figure 1: **A schematic of Brayton cycle.**

2.3 Types of gas turbines

- **Jet Engines:**
  Airbreathing jet engines are gas turbines optimized to produce thrust from the exhaust gases, or from ducted fans connected to the gas turbines. Jet engines that produce thrust from the direct impulse of exhaust gases are often called turbojets, whereas those that generate thrust with the addition of a ducted fan are often called turbofans or (rarely) fan-jets.

  Gas turbines are also used in many liquid propellant rockets, the gas turbines are used to power a turbopump to permit the use of lightweight, low pressure tanks, which reduce the empty weight of the rocket.

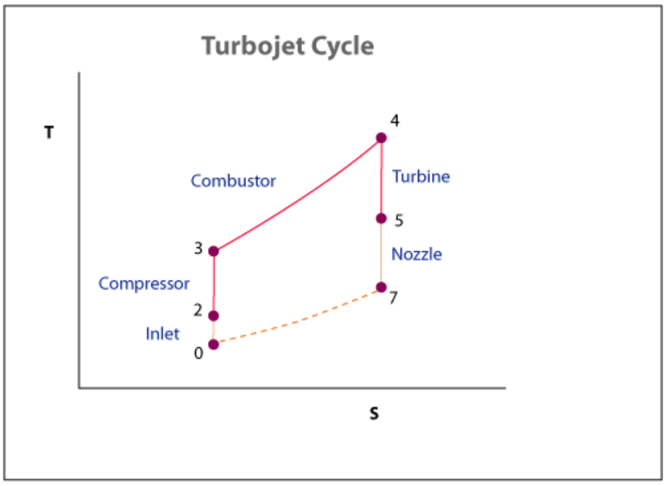


Figure 2: **T v/s S graph for a turbojet cycle.**

- **Turboprop Engines:**
  A turboprop engine is a turbine engine which drives an aircraft propeller using a reduction gear. Turboprop engines are used on small aircraft such as the general-aviation Cessna 208 Caravan and Embraer EMB 312 Tucano military trainer, medium-sized commuter aircraft such as the Bombardier Dash 8 and large aircraft such as the Airbus A400M transport and the 60 year-old Tupolev Tu-95 strategic bomber.

- **Turboshaft Engines:**
  Turboshaft engines are often used to drive compression trains (for example in gas pumping stations or natural gas liquefaction plants) and are used to power almost all modern helicopters. The primary shaft bears the compressor and the high speed turbine (often referred to as the Gas Generator), while a second shaft bears the low-speed turbine.

  In effect the separation of the gas generator, by a fluid coupling (the hot energy-rich combustion gases), from the power turbine is analogous to an automotive transmission's fluid coupling. This arrangement is used to increase power-output flexibility with associated highly-reliable control mechanisms.

- **Turbofan Engines:**
  The turbofan or fanjet is a type of **airbreathing jet engine** that is widely used in aircraft propulsion. The word "turbofan" is a portmanteau of "turbine" and "fan": the turbo portion refers to a gas turbine engine which takes mechanical energy from combustion, and the fan, a **ducted fan** that uses the mechanical energy from the gas turbine to accelerate air rearwards. Thus, whereas all the air taken in by a turbojet passes through the turbine (through the combustion chamber), in a turbofan some of that air **bypasses the turbine**.

  A turbofan thus can be thought of as a **turbojet being used to drive a ducted fan**, with both of those contributing to the thrust. The ratio of the mass-flow of air bypassing the engine core compared to the mass-flow of air passing through the core is referred to as the **bypass ratio**. The engine produces thrust through a combination of these two portions working in concert; engines that use more jet thrust relative to fan thrust are known as **low bypass turbofans**, conversely those that have considerably more fan thrust than jet thrust are known as **high bypass**. Most commercial aviation jet engines in use today are of the high-bypass type, and most modern military fighter engines are low-bypass.

3 GAS TURBINE ENGINE COMPONENTS

3.1 Engine station numbering and nomenclature

In industry the ability to unambiguously transfer performance data world-wide provides **substantial cost savings** due to efficiency gains and avoidance of misinterpretation. Virtually every gas turbine company has some form of **alliance** or **joint venture** with other companies, due to the huge cost of developing new engines. Furthermore customers, such as airframe manufacturers, often have large departments of engineers dealing with gas turbine performance.

3.1.1 ARP 755A station numbering

This section describes the basis for station numbering. Station numbers are appended to symbols, such as for total pressure, to identify exactly at what point in the engine that value of pressure occurs.

The following are the **details** of the **ARP 755A station numbering**:

3.1.2 Fundamental station numbers

The fundamental station numbers for the core stream of an engine are as below:

AMB	Ambient Conditions
0	Ram conditions in free stream
1	Engine intake front flange, or leading edge
2	First compressor/fan front face
3	Last compressor exit face
4	Combustor exit plane
5	Last turbine exit face
6	Front face of mixer, afterburner etc.
7	Propelling nozzle inlet
8	Propelling nozzle throat
9	Propelling nozzle or exhaust diffuser exit plane

3.1.3 Intermediate station numbers

Stations between the fundamental ones are numbered using a second digit suffixed to the upstream fundamental station number. In general this is not formally defined, hence companies have their own practices. **For example** T4 is the combustor exit/turbine nozzle guide vane leading edge temperature, and T41 is usually employed for the first stator outlet temperature.

Where more than ten intermediate stations are required a third digit is used. Continuing the above example the first nozzle guide vane throat, which occurs between station 4 and 41, is usually numbered 405.

3.1.4 Turbofan nomenclature

Here the fundamental station numbers are prefixed with a 1 for the bypass stream, and the core numbering is as per sections A.2.1A.2.3. For a turbofan with separate jets, common bypass duct station numbers include:

12	Fan tip front face, if conditions are different from the fan root front (station 2)
13	Fan exit
17	Cold propelling nozzle inlet
18	Cold propelling nozzle throat

In the more complicated instance of mixed streams and an afterburner the following numbers are usually used through these components:

16	Cold mixer inlet
6	Hot mixer inlet
65	Mixer outlet/afterburner inlet
7	Afterburner outlet/propelling nozzle inlet

For a three spool turbofan common additional stations are 24 for the second compressor entry, and 26 for the third.

3.1.5 Spool rotational speeds and inertia

These are numbered as per that at the **inlet to the first compressor** on the given spool. For example, for a two spool turbojet the polar moments of inertia are XJ2 for the first spool and XJ26 for the second.

3.2 Classification of engine components

The engine, in general can be divided into multiple sections. They are listed down with a short introduction about each of them.

- **Air intake (Inlet):**
  For subsonic aircraft, the inlet is a duct which is required to ensure smooth airflow into the engine despite air approaching the inlet from directions other than straight ahead. This occurs on the ground from cross winds and in flight with aircraft pitch and yaw motions. The duct length is minimized to reduce drag and weight. Air enters the compressor at about half the speed of sound so at flight speeds lower than this the flow will accelerate along the inlet and at higher flight speeds it will slow down. Thus the internal profile of the inlet has to accommodate both accelerating and diffusing flow without undue losses. For supersonic aircraft, the inlet has features such as cones and ramps to produce the most efficient series of shockwaves which form when supersonic flow slows down. The air slows down from the flight speed to subsonic velocity through the shockwaves, then to about half the speed of sound at the compressor through the subsonic part of the inlet. The particular system of shockwaves is chosen, with regard to many constraints such as cost and operational needs, to minimize losses which in turn maximizes the pressure recovery at the compressor.

- **Compressor or fan:**
  The compressor is made up of stages. Each stage consists of rotating blades and stationary stators or vanes. As the air moves through the

compressor, its pressure and temperature increase. The power to drive the compressor comes from the turbine (see below), as shaft torque and speed.

- **Bypass ducts:**
  These ducts deliver the flow from the fan with minimum losses to the bypass propelling nozzle. Alternatively the fan flow may be mixed with the turbine exhaust before entering a single propelling nozzle. In another arrangement an afterburner may be installed between the mixer and nozzle.

- **Shaft:**
  The shaft connects the turbine to the compressor, and runs most of the length of the engine. There may be as many as three concentric shafts, rotating at independent speeds, with as many sets of turbines and compressors. Cooling air for the turbines may flow through the shaft from the compressor.

- **Diffuser section:**
  The diffuser slows down the compressor delivery air to reduce flow losses in the combustor. Slower air is also required to help stabilize the combustion flame and the higher static pressure improves the combustion efficiency.

- **Combustor or combustion chamber:**
  Fuel is burned continuously after initially being ignited during the engine start.

- **Turbine:**
  The turbine is a series of bladed discs that act like a windmill, extracting energy from the hot gases leaving the combustor. Some of this energy is used to drive the compressor. Turboprop, turboshaft and turbofan engines have additional turbine stages to drive a propeller, bypass fan or helicopter rotor. In a free turbine the turbine driving the compressor rotates independently of that which powers the propeller or helicopter rotor. Cooling air, bled from the compressor, may be used to cool the turbine blades, vanes and discs to allow higher turbine entry gas temperatures for the same turbine material temperatures.

- **Afterburner:**
  Produces extra thrust by burning fuel in the jet pipe. This reheating of the turbine exhaust gas raises the propelling nozzle entry temperature and exhaust velocity. The nozzle area is increased to accommodate the higher specific volume of the exhaust gas. This maintains the same airflow through the engine to ensure no change in its operating characteristics.

- **Exhaust or nozzle:**
  Turbine exhaust gases pass through the propelling nozzle to produce a high velocity jet. The nozzle is usually convergent with a fixed flow area.

- **Supersonic nozzle:**
  For high nozzle pressure ratios (Nozzle Entry Pressure/Ambient Pressure)

a convergent-divergent (de Laval) nozzle is used. The expansion to atmospheric pressure and supersonic gas velocity continues downstream of the throat and produces more thrust.

The various components named above have **constraints** on how they are put together to generate the most efficiency or performance. **The performance and efficiency of an engine can never be taken in isolation**; for example fuel/distance efficiency of a supersonic jet engine maximizes at about Mach 2, whereas the drag for the vehicle carrying it is increasing as a square law and has much extra drag in the transonic region. The highest fuel efficiency for the overall vehicle is thus typically at Mach 0.85.

3.3 Inlet

3.3.1 Air Intakes

- The air intake can be designed to be part of the fuselage of the aircraft (Corsair A-7, A-8, Dassault Mirage III, General Dynamics F-16 Fighting Falcon, and Mikoyan-Gurevich MiG-21) or part of the nacelle.
- **Subsonic Inlets:**

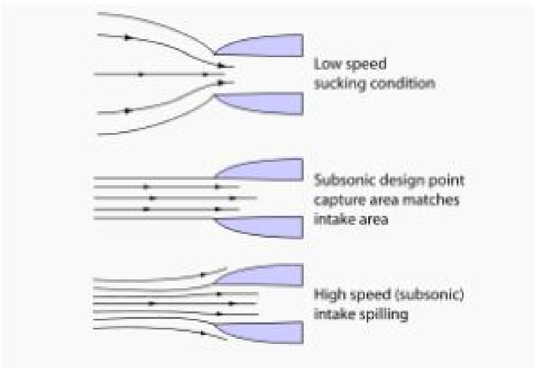


Figure 3: **Various intake conditions.**

- **Pitot intake operating modes:**
  Pitot intakes are the dominant type for **subsonic applications**. A subsonic pitot inlet is little more than a tube with an aerodynamic fairing around it.

  At **zero airspeed** (i.e., rest), air approaches the intake from a multitude of directions: directly ahead, radially, or even from behind the plane of the intake lip.

  At **low airspeeds**, the stream tube approaching the lip is larger in cross-section than the lip flow area, whereas at the intake design flight Mach number the two flow areas are equal.

At **high airspeeds** the stream tube is smaller, with excess air spilling over the lip.

Beginning around **Mach 0.85**, **shock waves** can occur as the air accelerates through the intake throat.

Careful radiusing of the lip region is required to optimize intake pressure recovery (and distortion) throughout the flight envelope.

- **Supersonic Inlets:**
  Supersonic intakes **exploit shock waves to decelerate the airflow** to a subsonic condition at compressor entry.
  Following is the detailed descreption for supersonic inlets.

3.3.2 Shock Waves

There are two forms of shock waves:

- **Normal Shock waves:**
  Normal shock waves lie **perpendicular** to the direction of the flow. These form sharp fronts and shock the flow to subsonic speeds.
  Microscopically the air molecules smash into the subsonic crowd of molecules like alpha rays. Normal shock waves tend to cause a large drop in stagnation pressure. Basically, the higher the supersonic entry Mach number to a normal shock wave, the lower the subsonic exit Mach number and the stronger the shock (i.e. the greater the loss in stagnation pressure across the shock wave).

- **Conical and Oblique Shock waves:**
  Conical (3D) and **oblique** shock waves (2D) are angled rearwards and radiate from a flow disturbance such as a cone or a ramp. For a given inlet Mach number, they are weaker than the equivalent normal shock wave and, although the flow slows down, it remains supersonic throughout.

  Conical and oblique shock waves **turn the flow**, which continues in the new direction, until another flow disturbance is encountered downstream. The same applies to the 2D shock waves.

A **sharp-lipped version** of the pitot intake, described above for subsonic applications, performs quite well at moderate supersonic flight speeds.

A detached normal shock wave forms just ahead of the intake lip and 'shocks' the flow down to a subsonic velocity. However, as flight speed increases, the shock wave becomes stronger, causing a larger percentage decrease in stagnation pressure or poorer pressure recovery).

3.4 Individual components of the engine

3.4.1 Compressors

Compressors are broadly classified into two categories:

- Radial Compressors.

- Axial Compressors.

We focus mainly on the axial compressors in the following descreption.

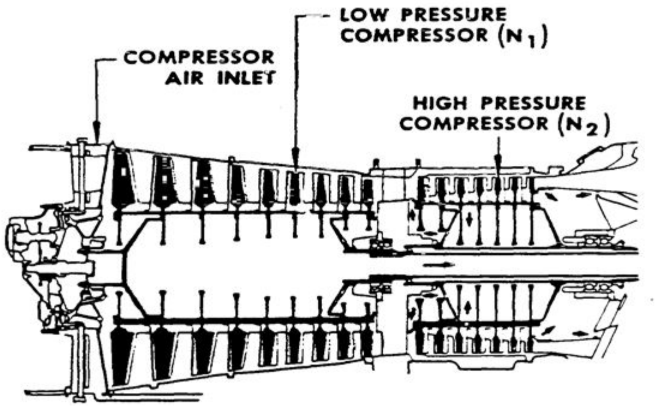


Figure 4: **Axial Compressor schematic.**

Axial compressors rely on spinning blades that have aerofoil sections, similar to aeroplane wings. As with aeroplane wings in some conditions the blades can stall. If this happens, the airflow around the stalled compressor can reverse direction violently. Each design of a compressor has an associated **operating map of airflow versus rotational speed** for characteristics peculiar to that type.

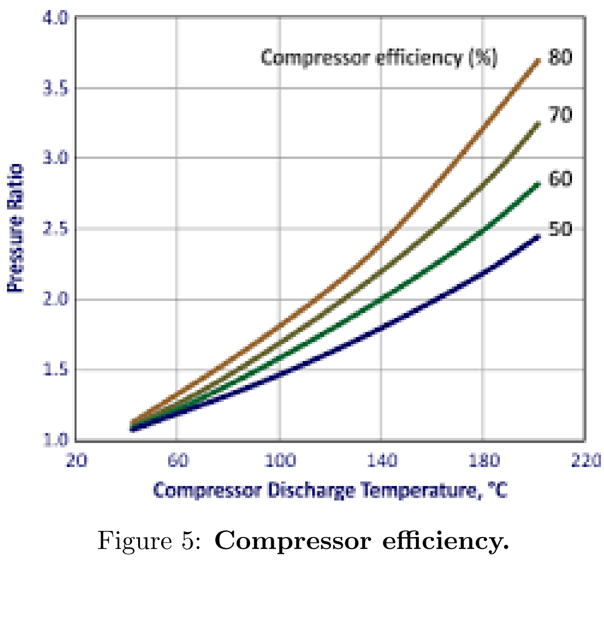


Figure 5: **Compressor efficiency.**

At a given throttle condition, the compressor operates somewhere along the steady state running line. Unfortunately, this operating line is displaced during transients. Many compressors are fitted with **anti-stall systems** in the form of

bleed bands or variable geometry stators to decrease the likelihood of surge. Another method is to split the compressor into two or more units, operating on separate concentric shafts.

Another design consideration is the **average stage loading**. This can be kept at a sensible level either by **increasing the number of compression stages** (more weight/cost) or the **mean blade speed** (more blade/disc stress).

Although large flow compressors are usually all-axial, the rear stages on smaller units are too small to be robust. Consequently, these stages are often replaced by a **single centrifugal unit**. Very small flow compressors often employ two centrifugal compressors, connected in series. Although in isolation centrifugal compressors are capable of running at quite high **pressure ratios** (e.g. 10:1), impeller stress considerations limit the pressure ratio that can be employed in high overall pressure ratio engine cycles.

Increasing overall pressure ratio implies raising the high pressure compressor exit temperature. This implies a higher high pressure shaft speed, to maintain the datum blade tip Mach number on the rear compressor stage. Stress considerations, however, may limit the shaft speed increase, causing the original compressor to throttle-back aerodynamically to a lower pressure ratio than datum.

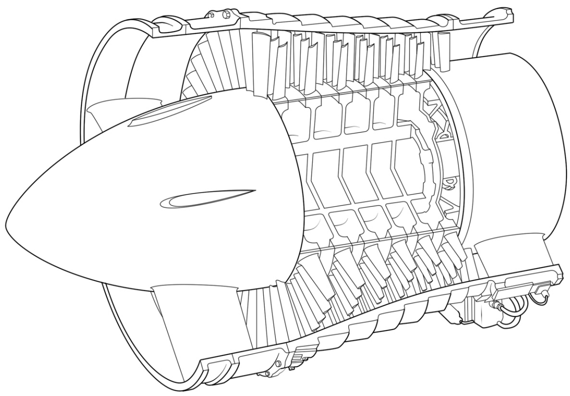


Figure 6: **The cut section of a 17 stage axial compressor.**

3.4.2 Combustors

Flame fronts generally travel at just Mach 0.05, whereas airflows through jet engines are considerably faster than this. Combustors typically employ structures to give a sheltered combustion zone called a flame holder.The three major categories of combustors are

- Tubular (single can)
- Tuboannular
- Annular

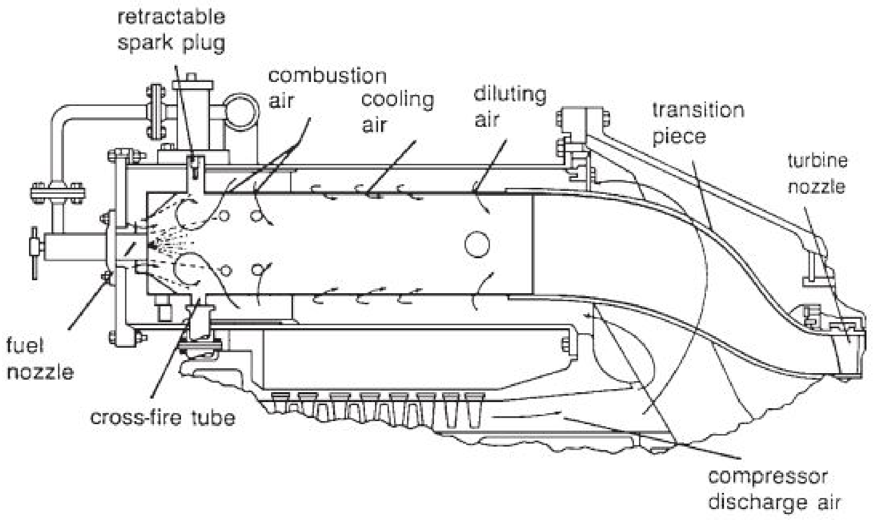


Figure 7: **A can annular reverse flow combustor.**

Great care must be taken to keep the flame burning in a moderately fast moving airstream, at all throttle conditions, as efficiently as possible. Since the turbine cannot withstand stoichiometric temperatures (a mixture ratio of around 15:1), some of the compressor air is used to quench the exit temperature of the combustor to an acceptable level (an overall mixture ratio of between 45:1 and 130:1 is used). Air used for combustion is considered to be primary airflow, while excess air used for cooling is called secondary airflow. The secondary airflow is ported through many small holes in the burner cans to create a blanket of cooler air to insulate the metal surfaces of the combustion can from the flame. If the metal were subjected to the direct flame for any length of time, it would eventually burn through.

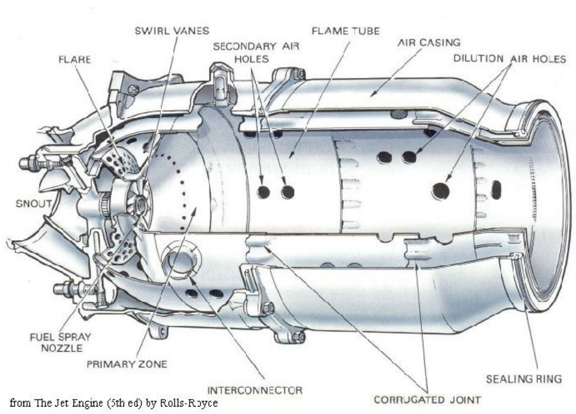


Figure 8: **The combustion chamber in RR engine.**

Rocket engines, being a non 'duct engine'; have quite different combustor systems, and the mixture ratio is usually much closer to being stoichiometric in the main chamber. These engines generally lack flame holders and combustion occurs at much higher temperatures, there being no turbine downstream. However, liquid rocket engines frequently employ separate burners to power turbo pumps, and these burners usually run far off stoichiometric so as to lower turbine temperatures in the pump.

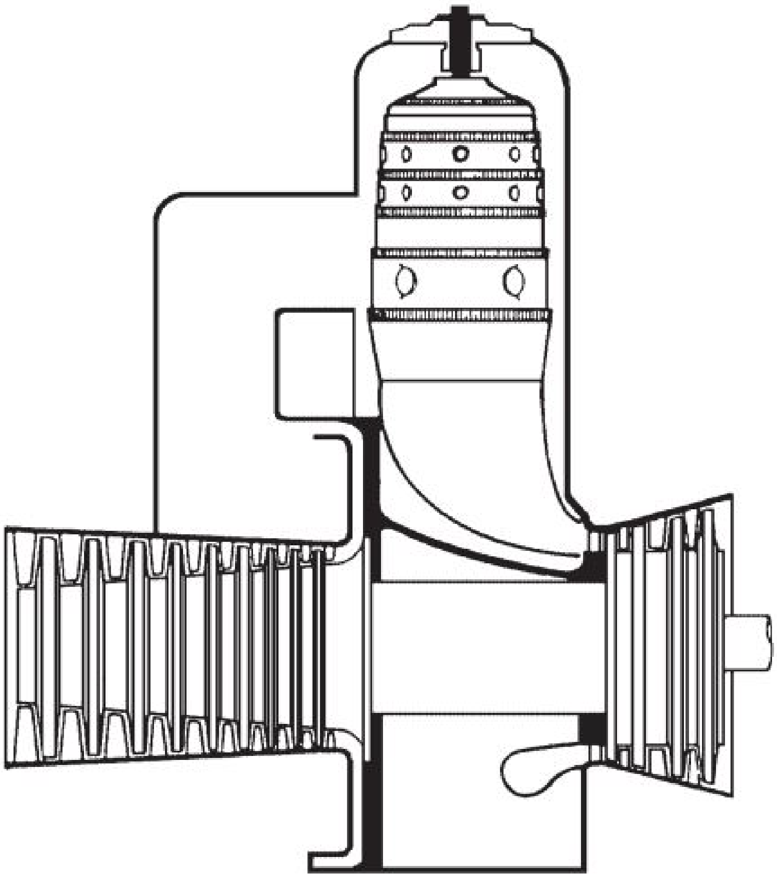


Figure 9: **Industrial engine featuring single tubular combustor.**

3.4.3 Turbines

Because a turbine expands from high to low pressure, there is no such thing as turbine surge or stall. **The turbine needs fewer stages than the compressor**, mainly because the higher inlet temperature reduces the DeltaT/T (and thereby the pressure ratio) of the expansion process. The blades have more curvature and the gas stream velocities are higher.

Designers must, however, prevent the turbine blades and vanes from melting in a very high temperature and stress environment. Consequently bleed air extracted from the compression system is often used to cool the turbine blades/vanes internally. Other solutions are improved materials and/or special insulating coatings. The discs must be specially shaped to withstand the huge stresses imposed by the rotating blades. They take the form of impulse, reaction, or combination impulse-reaction shapes. Improved materials help to keep disc weight down.

3.4.4 Afterburners

Due to temperature limitations with the gas turbines, jet engines do not consume all the oxygen in the air. **Afterburners burn the remaining oxygen after exiting the turbines,** but usually do so inefficiently due to the low pressures typically found at this part of the jet engine make the subsequent nozzle inefficient at extracting the heat energy; however afterburners still gain significant thrust, which can be useful. Engines intended for extended use with afterburners often have variable nozzles and other details.

3.4.5 Nozzle

The propelling nozzle converts a gas turbine or gas generator into a jet engine. Power available in the gas turbine exhaust is converted into a high speed propelling jet by the nozzle. The power is defined by typical gauge pressure and temperature values.

3.4.6 Thrust Reversers

These either consist of cups that swing across the end of the exhaust nozzle and deflect the jet thrust forwards, or they are two panels behind the cowling that slide backward and reverse only the fan thrust (the fan produces the majority of the thrust). Fan air redirection is performed by devices called "blocker doors" and "cascade vanes".

If you are on an aircraft and you hear the engines increasing in power after landing, it is usually because the thrust reversers are deployed. The engines are not actually spinning in reverse. The term is a misnomer. **The reversers are used to slow the aircraft more quickly and reduce wear on the wheel brakes.**

3.5 Cooling systems

3.5.1 Air Cooling Systems

A complex air system is built into most turbine based jet engines, primarily to cool the turbine blades, vanes and discs.

Air, bled from the compressor exit, passes around the combustor and is injected into the rim of the rotating turbine disc. The cooling air then passes through complex passages within the turbine blades. After removing heat from the blade material, the air (now fairly hot) is vented, via cooling holes, into the main gas stream. Cooling air for the turbine vanes undergoes a similar process.

Cooling the leading edge of the blade can be difficult, because the pressure of the cooling air just inside the cooling hole may not be much different from that of the oncoming gas stream. One solution is to incorporate a cover plate on the disc. This acts as a centrifugal compressor to pressurize the cooling air before it enters the blade. Another solution is to use an ultra-efficient turbine rim seal to pressurize the area where the cooling air passes across to the rotating disc.

Seals are used to prevent oil leakage, control air for cooling and prevent stray air flows into turbine cavities. Series of (e.g. labyrinth) seals allow a small flow of bleed air to wash the turbine disc to extract heat and, at the same time, pressurize the turbine rim seal, to prevent hot gases entering the inner part of the engine. Other types of seals are hydraulic, brush, carbon etc.

Small quantities of compressor bleed air are also used to cool the shaft, turbine shrouds, etc. Some air is also used to keep the temperature of the combustion chamber walls below critical. This is done using primary and secondary airholes which allow a thin layer of air to cover the inner walls of the chamber preventing excessive heating.

3.5.2 Cowl Flaps

Cool air is taken in at the front of the engine and after cooling the cylinders, the warm (and expanded) air needs to be exhausted. This is done through openings in the **lower cowling**, controlled by **cowl flaps**. These pilot operated flaps are open during high power/ low speed operations (letting more air through during climb and taxi), they will also increase the **parasite drag** of the aircraft when in the open position. During normal cruise and descent the cowl flaps should be closed.

3.5.3 Liquid cooling

This type of cooling has a **weight penalty** but this is offset by the advantage that all cylinders are even in temperature, they cannot be shock cooled during high speed/ low power descends and the coolant can be thermostatically controlled so that the engine is quicker to warm up and remains on a constant operating temperature at all times. Which extends into more reliability, lower fuel consumption and longer engine life, to name but a few advantages.

3.5.4 Spinner

The **propeller spinner** is part of the cooling system as it guides the incoming ram air to the intakes, usually to the right and left of the spinner. These intakes are square/ rectangle of shape and the more modern ones are round. These have **lower drag**, thus more effective by reducing the total aircraft drag.

4 TURBINE BLADES

4.1 Introduction

A **turbine blade** is the individual component which makes up the turbine section of a gas turbine. The blades are responsible for extracting energy from the high temperature, high pressure gas produced by the combustor. The turbine blades are often the limiting component of gas turbines. To survive in this difficult environment, turbine blades often use exotic materials like super alloys and many different methods of cooling, such as internal air channels, boundary layer cooling, and thermal barrier coatings. The blade fatigue failure is one of the major source of outages in any steam engines and gas turbines which is due to high dynamic stresses caused by blade vibration and resonance within the operating range of machinery. To protect blades from these high dynamic stresses, friction dampers are used.

The number of turbine stages varies in different types of engines, with **high bypass ratio** engines tending to have the most turbine stages. Many gas turbine engines are twin spool designs, meaning that there is a high pressure spool and a low pressure spool. The high pressure turbine is exposed to the hottest, highest pressure air, and the low pressure turbine is subjected to cooler, lower pressure air. Steam turbine blades are critical components in power plants which convert the linear motion of high temperature and high pressure steam flowing down a pressure gradient into a rotary motion of the turbine shaft.

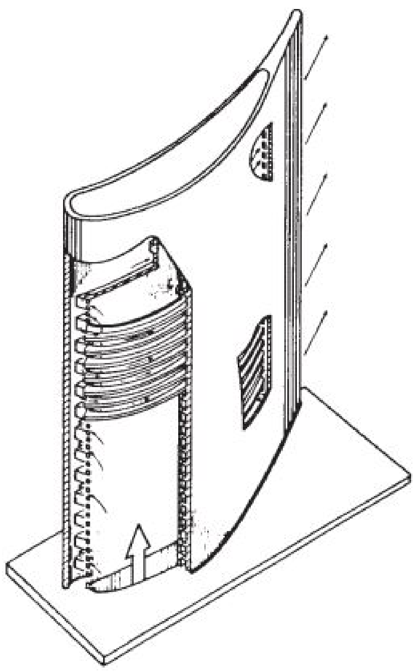


Figure 10: **Strut insert blade.**

Turbine blades are subjected to very strenuous environments inside a gas turbine. They face high temperatures, high stresses, and a potential environment of high vibration. All three of these factors can lead to blade failures, potentially destroying the engine, therefore turbine blades are carefully designed to resist these conditions. Turbine blades are subjected to stress from **centrifugal force** (turbine stages can rotate at tens of thousands of revolutions per minute (RPM)) and fluid forces that can cause **fracture, yielding, or creep failures**. Additionally, the first stage (the stage directly following the combustor) of a modern turbine faces temperatures around 2,500 F (1,370 C), up from temperatures around 1,500 F (820 C) in early gas turbines.

4.2 Materials and Processes

One of the earliest of these materials was **Nimonic**, used in the British Whittle engines. The development of super alloys in the 1940s and new processing methods such as **vacuum induction melting** (Vacuum induction melting (VIM) utilizes electric currents to melt metal within a vacuum. One of the only ways to induce a current within a conductor is through electromagnetic induction. Electromagnetic induction induces eddy currents within conductors by changing the magnetic field. Eddy currents create heating effects to melt the metal) in the 1950s greatly increased the temperature capability of turbine blades.

Modern turbine blades often use **nickel-based** super alloys that incorporate chromium, cobalt and rhenium. Aside from alloy improvements, a major breakthrough was the development of directional solidification (DS) and single crystal (SC) production methods. These methods help greatly increase strength against fatigue and creep by aligning grain boundaries in one direction (DS) or by eliminating grain boundaries altogether (SC). Where DS and SC developments improved creep and fatigue resistance, TBCs (**Thermal barrier coatings** using aluminide and ceramic) improved corrosion and oxidation resistance, both of which become greater concerns as temperatures increased.

Most turbine blades are manufactured by **investment casting** (or lost-wax processing). This process involves making a precise negative die of the blade shape that is filled with wax to form the blade shape. If the blade is hollow (i.e., it has internal cooling passages), a ceramic core in the shape of the passage is inserted into the middle. The wax blade is coated with a heat-resistant material to make a shell, and then that shell is filled with the blade alloy. This step can be more complicated for DS or SC materials, but the process is similar. If there is a ceramic core in the middle of the blade, it is dissolved in a solution that leaves the blade hollow. The blades are coated with the TBC they will have, and then cooling holes are machined as needed, creating a complete turbine blade.

C**eramic matrix composites (CMC)** are being developed for use in turbine blades. These are materials consisting of fiber embedded in a ceramic matrix. The main advantage of CMC materials over conventional super alloys is their light weight and high temperature capability. **SiC/SiC composites** consisting of silicon matrix reinforced by silicon carbide fibers have been shown to have operating temperatures that are 200-300F higher than for the nickel super alloys they replace and used in F414 by GE.

List of turbine blade materials:

- **U-500**
  This material was used as a first stage (the most demanding stage) material in the 1960s, and is now used in later, less demanding, stages.
- **Rene 77**
- **Rene N5**
- **Rene N6**
- **PWA1484**
- **CMSX-4**
- **CMSX-10**
- **Inconel**
  IN-738:
  GE used IN-738 as a first stage blade material from 1971 until 1984, when it designed for land-based turbines rather than aircraft gas turbines.
- **GTD-111**
  Blades made from directionally solidified GTD-111 are being used in many GE. Energy gas turbines in the first stage. Blades made from equiaxed GTD-111 are being used in later stages.
- **Nimonic 80a:**
  Was used for the turbine blades on the Rolls-Royce.
- **Nimonic 90:**
  Was used on the Bristol Proteus.
- **Nimonic 263:**
  Was used in the combustion chambers of the Bristol Olympus used on the Concorde.

4.3 Single Crystal Turbine Blades

The **thermal efficiency of gas turbine** can be greatly increased with components that are able to withstand higher working temperatures. The use of **single crystal super alloy turbine blades** allows for this to be possible. The single crystal turbine blades are able to operate at a higher working temperature than crystalline turbine blade and thus are able to **increase the thermal efficiency** of the gas turbine cycle.

4.3.1 Advantage of single crystal turbine blade

Single crystal turbine blades have the **mechanical advantage** of being able to operate at a much higher temperature than crystalline turbine blades. Given the ability to increase turbine efficiency with higher temperatures, the development of these blades is very beneficial. The turbine blades are able to operate at these high temperatures due to the single crystal structure and the composition of the nickel based superalloy.

Creep is a common cause of failure in turbine blades and is in fact the life limiting factor. When temperatures of a material under high stress are raised to a critical point, the creep rate quickly increases. The single crystal structure has the ability to withstand creep at higher temperatures than crystalline turbine blades due to the lack of grain boundaries present. Grain boundaries are an area of the microstructure where many defects and failure mechanisms start which leads to creep occurring. The lack of these grain boundaries inhibits creep from occurring in this way. Creep will still occur in single crystal turbine blades but due to different mechanisms that occur at higher temperatures. The single crystal turbine blade does not have grain boundaries along directions of axial stress which crystalline turbine blades do. This also works to increases the creep strength.

4.3.2 Nickel based super alloy

There have been several superalloys that have been used in attempting to create a single crystal turbine blade that is able to withstand the highest operating temperatures possible. These superalloys are generally **nickel based** and contains several other elements that all contribute to optimizing the mechanical properties of the turbine blade under high temperature conditions. The composition of each element added is constantly being tested to allow for this optimization. An example of a superalloy used for the purpose of single crystal turbine blades is **CMSX6**. The composition of this super alloy is shown in the table below.

CMSX-6								
Element	Ni	Cr	Co	Ti	Al	Mo	Ta	Hf
Wt %	70.4	10	5	4.7	4.8	3	2	0.1

Figure 11: **The composition of CMSX-6 alloy.**

Within the single crystal of the superalloy, there are two phases present, a **gamma matrix** and a **gamma prime precipitate**. The gamma prime phase needs to be greater than 50 percent volume fraction in the superalloy to provide the increase in creep resistance. The presence of the gamma prime phase increases the mechanical strength of the turbine blade by preventing dislocation motion. The gamma prime phase has the unusual property of increasing strength as temperature increases.

This is true up to 973 degrees Celsius. This increase in strength cause by an increased in temperature results in the superalloy being able to operate under higher temperatures. The **lack of grain boundaries** in the turbine blade allows for the superalloy being used to reduce the presence of elements that are usually used to strengthen grain boundaries, such as **carbon and boron**. These elements reduce the creep strength and the melting temperature of the alloy when found in more significant compositions.

Without the need for significant concentrations of these elements, the single crystal turbine blade is able to maintain its strength and use at higher temperatures.

4.3.3 Manufacturing process and crystal growth

There are several different manufacturing methods that are used in practice to create single crystal turbine blades. The manufacturing methods all use the idea of **directional solidification**, or **autonomous direction solidification** where the direction of solidification is controlled. A common method is the **Bridgman method** to grow single crystals.

In this method a **casting furnace** is used for crystal growth. In this process, a mould must first be made of the blade. Molten wax is injected into a metallic mould of the desired turbine blade and left to set and take the form of the turbine blade. The wax model is then used to create a ceramic mould to use for production of the single crystal turbine blades. When the ceramic mould is created, it is heated to raise the strength of the mould. Once the mould is sufficient for use, the wax is melted out from the inside of the mould. The mould is now filled with the molten form of the nickel based superalloy. The molten superalloy contained within the mould is placed in some type casting furnace, often a vacuum induction melting furnace, which uses **Bridgman techniques.**

4.3.4 Crystal growth in Bridgman Furnace

The furnace is set up with an area of high temperature which is above the melting temperature, controlled by heaters, and low temperature below the melting zone, with a gradient zone where the solid-liquid interface occurs. The superalloy is initially entirely within the high temperature zone in molten form. The superalloy is then lowered extremely slowly, at rates of about a few inches per hour, so that the solid liquid interface rises slowly up the mould. The superalloy solidifies from the base up. The slow rate of solidification causes grains to grow as dendrites in the direction in which the mould is pulled from the furnace. The dendrites form only as columns in the one direction because of the effect of **constitutional undercooling**.

As the solid begins to form, a varying solute concentration is found just ahead of the solid-liquid interface. The variance in solute throughout the liquid causes a change in the **equilibrium solidification temperature**. At this point the temperature of the liquid is lower than the equilibrium solidification temperature causing an **undercooling effect**.

Undercooling causes heat to be transferred **from solid protrusions to the liquid promoting dendritic growth**. The rate at which dendrites grow is directly related to the amount of undercooling present. Dendrites that are aligned at an angle have to grow faster to keep up with the dendrites taking a more direct, vertical direction. To grow faster, a greater amount of undercooling is needed which means these angled dendrites grow further back from the solid-liquid interface. Eventually the more favorable vertical dendrites overtake the angled dendrites that are further back. To remove grain boundaries from the turbine blade, a grain selector is attached to the bottom of the wax mould.

The grain selector is a spiral shaped tube that is not much larger than a single dendrite grain. As the vertical dendrites grow at the base of the mould, only one dendrite will be able to fit through the spiral and eventually into the turbine blade mould.

Thus once the solidification is complete, the turbine blade is created **entirely from one grain** and becomes a single crystal turbine blade.

5 DESIGN PROCESS OF AN ENGINE

To design an engine from the scratch, the basic details required are the following:

- Weight
- Thrust generated
- Torque produced
- Dimensions of the engine
- Inlet and exhaust conditions of the engine

These are the various departments in HAL which constitute the complete design process at HAL:

5.1 Engine process or cycle

In this first and basic stage the **complete cycle analysis** for the engine is carried out. The basis for these calculations are the available raw **input parameters** and the **expected output efficiency** from the cycle.

This stage gives a fair idea about what can be expected from the engine pertaining to efficiency, dimensions, engine cycle et. all.

5.2 Component Designer

The Component designer is responsible for the **3D modelling** of all the parts of design given by the Detail design block. **Miniature models** are also made to analyse the parts further. The required materials for the manufacturing of the parts and modules are chosen and after **thorough testing** and research, the **final full scale prototype** parts are manufactured. Then the parts and the designs are sent for CFD analysis.

5.3 CFD Analysis

Computational fluid dynamics, usually abbreviated as CFD, is a branch of fluid mechanics that uses **numerical analysis** and **algorithms** to solve and analyse problems that involve fluid flows. Computers are used to perform the calculations required to simulate the interaction of **liquids** and **gases** with surfaces defined by boundary conditions. This is necessary to analyse the fluid flow inside the engine. The flow of air and its interaction with the various elements of the engine is considered from its entry into the engine to the exhaust system of the engine. All the permutations of various parameters are tried and the possibilities are looked into, following which the design is sent to the design block and then again in the **loop for refinement**.

5.4 Design Block

This is where the engine is **conceptualized**. This group of engineers/designers build up the foundations of the basic structure of the engine to be later developed upon in the rest of the process. **The engine structure** is roughly determined considering the above mentioned factors and then improved assuming more variables and dropping assumptions. The internally occurring processes are sketched out to accept the inlet and exhaust conditions of the engine as the input and output of the system. We know that engines can be used to mainly produce either thrust/torque. Knowing the exact requirement of thrust/torque for this engine, the output of each **cycle** occurring inside the engine can be designed accordingly. An approximate **cycle analysis** is done and then refined until some of the constraints are matched. Once the **efficiencies** of various cycles involved and the **isentropic efficiency** of the engine as a whole is computed within a desired range, the design is sent to the Detail design group.

5.5 Stress Analysis

Once the structure of the engine is complete with all the modules fit into it, the engine is sent for **structural analysis**. The aim of this part of the process is to test and ensure the **structural capabilities** of the engine are suited to its needs and **functions**. The technique used for the test purposes is called **FEM** or **Finite Element Method/Analysis**. Different parts of the engine undergo different amounts of stresses and strains and thus need to be **tested to their limits**. The performance of the engine, even after these test procedures, should be within the prescribed limits to pass these tests. If the engine fails any of the tests, its design is again sent to the Design block in the loop for the required corrections. The general **factor of safety is around 2.5** for the designing of the emgine.

5.6 Vibration Analysis

Once the engine design is completed as per the above departments consents, it is sent for **Vibration analysis**. In this set of experiments, the amount of vibration produced by the engine at various performance levels is noted and it is ensured that these values are within the prescribed limits. If allowed to go out of bounds, these vibrations can lead to catastrophic failures because of **resonance.**

5.7 Detailed Design Group

In this part of the design process, the design received from the Design block is **further refined** here. This group deals with the **detailed design** of each of the cycles, parts and modules involved. All the designs are made in accordance with the initial designs given by the Design block. Then the designs are sent to the **Component designer** for the next stage of the process.

- **Note:**

It is important to note that all the above departments work together simultaneously, in coordination with each other, so as to obtain the optimum design after numerous iterations of the above said design process.

6 ELEMENTARY THEORY OF AXIAL FLOW TURBINES

As with compressors, there are two types of turbines - axial and radial. Axial turbines find more industrial applications than radial. The turbine consists of two stages:

- **Row of rotor blades.**
- **Row of stator blades.**

The rotor is in motion while the stator isn't. The absolute velocity of the air is increased by the rotor but the relative velocity is decreased. This is diffusion. A single compressor provides only a small pressure ratio, too less for one turbine. Thus, a single turbine can run multiple compressors. An **Inlet Guide Vane (IGV)** guides the flow to the first stage. This is an optional feature which need not be in all engines. As the size of the turbines increase, the tip speed shoots up. To prevent this, **biconvex blading** is incorporated for supersonic /transonic flow.

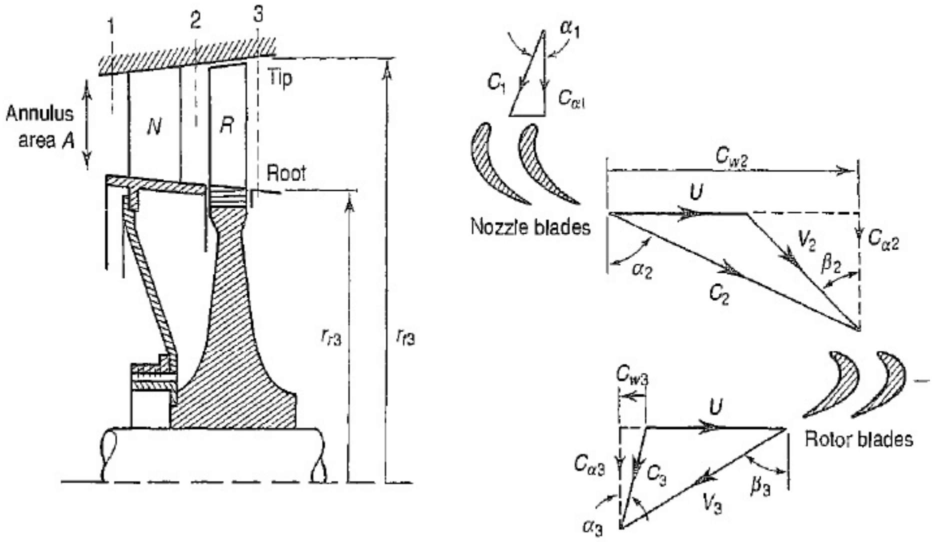


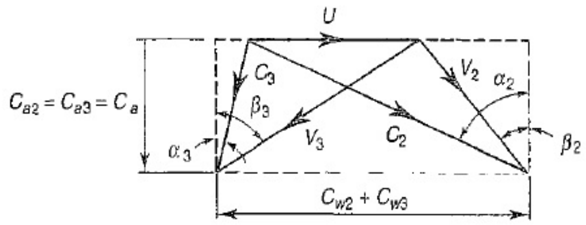
Figure 12: **Turbine stages and velocity triangles.**

Air enters the nozzle blades at velocity C_1 and angle α_1 and leaves at an increased velcoity C_2 and angele α_2. The rotor blade inlet angle will be chosen to suit the direction β_2 of the gas velocity V_2 relative to the blade inlet. Vectorial subtraction of U from absolute velociyt C_2 gives β_2 and V_2. The gas leaves the rotor blades at relative velocity V_3 and angle β_3. The absolute exit velocity is C_3 at an angle α_3, also called the swirl angle.

Velocity triangles vary from root to tip and the ones shown are at the mean diameter. This is particularly called the **Mean diameter approach**. This method is suitable for low tip to root radius.

The change in the whirl component is $(C_{w2} + C_{w3}\ )$. The axial component $(C_{a2} - C_{a3})$ produces thrust on the rotor. For a single stage turbine, we assume $\alpha_1 = 0$ and $C_1 = C_a$ and for multiple stages, for each stage to have the same blade design, $\alpha_1 = \alpha_3$ and $C_1 = C_3$.

For our analysis, we assume a constant axial component C_a throughout the rotor. On superimposing the velocity triangles, the geometry gives the following relations.



$$\frac{U}{C_a} = tan\alpha_2 - tan\beta_2 = tan\beta_3 - tan\alpha_3 \tag{1}$$

Angular momentum conservation gives,

$$W_s = U(C_{w2} + C_{w3}) = UC_a(tan\alpha_2 + tan\alpha_3) = UC_a(tan\beta_2 + tan\beta_3) \tag{2}$$

Steady flow energy equation gives,

$$W_s = c_p \Delta T_{0s} \tag{3}$$

where ΔT_{0s} is the stage stagnation temperature drop. If $C_1 = C_2$, then the static temperature drop is given by the same equation.

The pressure ratio relation is as follows,

$$\Delta T_{0s} = \eta_s T_{01}[1 - (\frac{1}{\frac{p_{01}}{p_{03}}})^{\frac{\gamma-1}{\gamma}}] \tag{4}$$

where η_s is the isentropic efficiency or the total stage efficiency.

Alternatively,

$$\eta_s = \frac{T_{01} - T_{03}}{T_{01} - T_3'} \tag{5}$$

where T_3' is the temperature reached after an isentropic expansion from p_{01} to p_3.

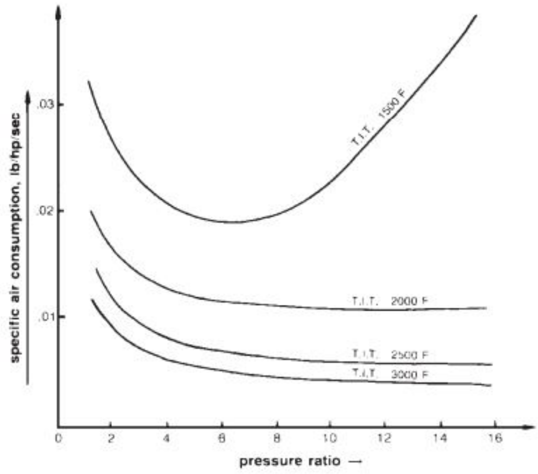


Figure 13: **Specific air consumption v/s pressure ratio.**

6.1 Turbine design parameters

The design of turbines is simplified with the help of three non-dimensionless parameters.

- **Blade loading co-efficient/ Temperature drop co-efficient:**
  This expresses the work capacity of a stage.

$$\psi = \frac{c_p \Delta T_{0s}}{\frac{1}{2}U^2} \tag{6}$$

- **Degree of Reaction:**
  This expresses the fraction of the stage expansion that happens in the rotor.

$$\Lambda = \frac{T_2 - T_3}{T_1 - T_3} \tag{7}$$

From our assumptions, $C_{a2} = C_{a3} = C_a$ amd $C_3 = C_1$,

$$=> c_p(T_1 - T_3) = c_p(T_{01} - T_{03}) = UC_a(tan\beta_2 + tan\beta_3) \tag{8}$$

Relative to rotor blade, flow does no work, thus,

$$c_p(T_2 - T_3) = \frac{1}{2}(V_3^2 - V_2^2) = \frac{1}{2}C_a^2(sec^2\beta_3 - sec^2\beta_2) = \frac{1}{2}C_a^2(tan^2\beta_3 - tan^2\beta_2) \tag{9}$$

$$\Lambda = \frac{C_a}{2U}(tan\beta_3 - tan\beta_2) \tag{10}$$

- **Flow co-efficient:**
  This is defined as

$$\phi = \frac{C_a}{U} \tag{11}$$

The three parameters and the gas angles can be related as follows:

$$\psi = 2\phi(tan\beta_2 + tan\beta_3) \tag{12}$$

$$\Lambda = \frac{\phi}{2}(tan\beta_3 - tan\beta_2) \tag{13}$$

$$tan\beta_3 = \frac{1}{2\phi}(\frac{\psi}{2} + 2\Lambda) \tag{14}$$

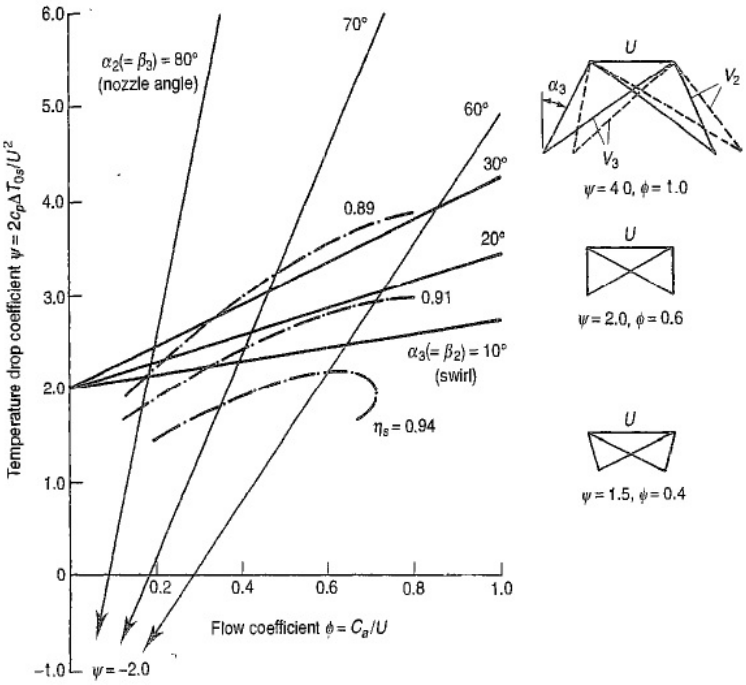
$$tan\beta_2 = \frac{1}{2\phi}(\frac{\psi}{2} - 2\Lambda) \tag{15}$$

$$tan\alpha_3 = tan\beta_3 - \frac{1}{\phi} \tag{16}$$

$$tan\alpha_2 = tan\beta_2 + \frac{1}{\phi} \tag{17}$$

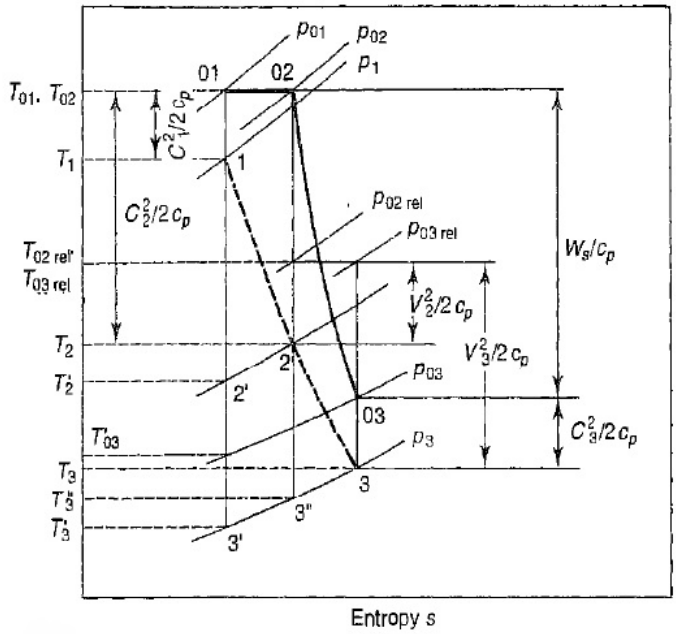
Ideally we divide the expansion process between the rotor and the stator equally at the mean diameter, thus $\Lambda = 0.5$. But it must be noted that Λ varies from root to tip of the blade.

This gives $C_3 = C_1$ and $\alpha_1 = \alpha_3 = \beta_2$,i.e., stator and rotor blades have the same inlet and outlet angles. Low values of ψ and ϕ yield the best stage efficiencies but increases the weight. Optimum range of ψ and ϕ are 3 to 5 and 0.8 to 1 respectively.



50 per cent reaction designs

6.2 Blade losses



T–s **diagram for a reaction stage**

The loss co-efficient for the nozzle blades can be defined in terms of temperature or pressure as below,

$$\lambda_N = \frac{T_2 - T_2'}{\frac{C_2^2}{2c_p}} \tag{18}$$

or,

$$Y_N = \frac{p_{01} - p_{02}}{p_{02} - p_2} \tag{19}$$

The latter one is easily measured by cascade tests and it is numerically almost equal to the former co-efficient.

The rotor blade loss co-efficient is also defined in two ways as below,

$$\lambda_R = \frac{T_3 - T_3''}{\frac{V_3^2}{2c_p}} \tag{20}$$

$$Y_R = \frac{p_{02rel} - p_{03rel}}{p_{03rel} - p_3} \tag{21}$$

Even here, the former and latter co-efficients are numerically almost equal.

The relations between λ_N, λ_R and η_s are as below,

$$\eta_s = \frac{T_{01} - T_{03}}{T_{01} - T_{03}'} \tag{22}$$

It can be shown that,

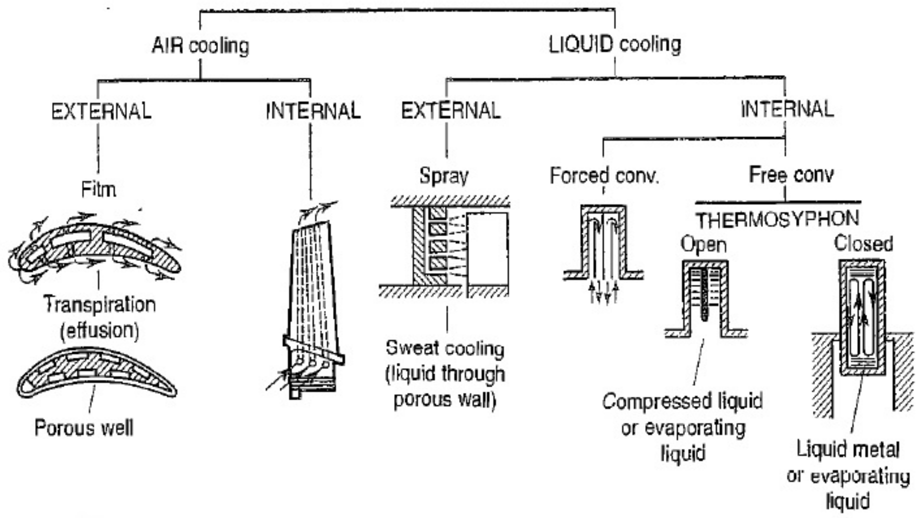
$$\eta_s \quad \frac{1}{1 + \frac{C_a}{2U}\left[\frac{\lambda_R sec^2\beta_3 + (\frac{T_3}{T_2})\lambda_N sec^2\alpha_2}{tan\beta_3 + tan\alpha_2 - \frac{U}{C_a}}\right]} \tag{23}$$

Blade losses are of many types based on the loss in pressure, temperature, enthalpy or the gain in entropy. Some of the types are :

- **Profile loss** is the aerodynamic loss due the mechanical build or the shape of the blades itself. They can be made optimum with an appropriate aerodynamic design.
- **Tip clearance loss** concerns the tip leakage flow between a blade and a casing wall has a strong impact on compressor pressure rise capability, efficiency, and stability. In axial compressors, clearance gaps between the rotating blades and the stationary casing are necessary to prevent physical rubbing between them. Due to sudden discharge of blade loading at the tip, a large pressure difference remains across the tip clearance. A tip leakage flow driven by the pressure difference between the pressure and suction sides of a blade rolls up into a concentrated-tip leakage vortex, lying in the corner of the endwall and the suction surface and moving along the mainstream direction. There are two aspects of the tip leakage flow: one is blockage, which is a fluid dynamic effect, and the other is loss, which is a thermodynamic effect. Consequently, there is a strong motivation to look for methods to reduce the strength of the tip leakage flow and desensitize compressors to the changes in tip clearance.
- **Trailing edge thickness loss**: The experimental results show increased efficiency loss for increased trailing-edge thickness for all trailing-edge geometries. The blade with round trailing edge, equal to about 11 percent of the blade throat width, had 60 percent more loss than the sharp-edged blade. For the same trailing-edge thickness, square trailing edges caused more loss than round trailing edges, and the tapered trailing edges caused about the same loss as the round trailing edges.

6.3 The cooled turbine

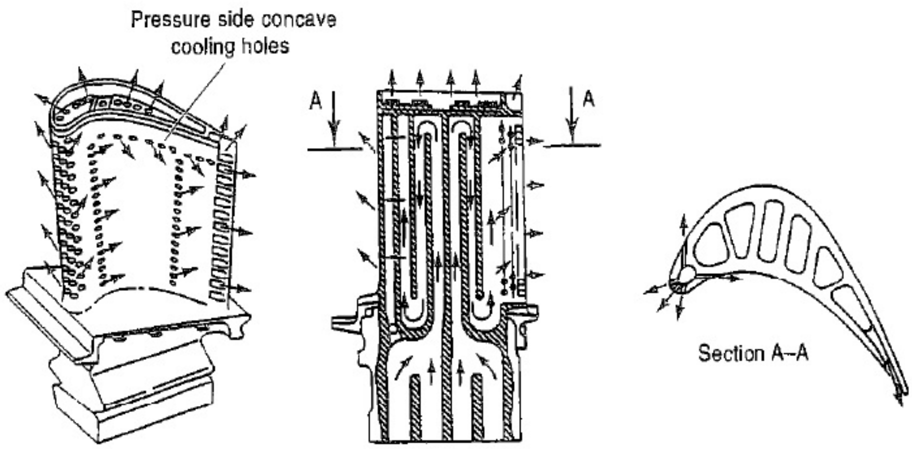
It has always been a practice to pass a quantity of cooling air over the turbine disc and blade roots. When speaking of *coole turbine*, however, we mean passing a substantial quantity of coolant to the nozzle and rotor blades themselves. Increase in permissible turbine inlet temperature facilitates reduced SFC and increased specific power output. The benefits remain substantial even with additional losses due to introduction of the cooling system.



Methods of blade cooling

Liquid systems have not proved to be practicable due to difficulties in channeling the liquid to and from the blades. Corrosion and formation of deposits are other associated problems. The only successful method in production engines has been **internal, forced convection, air cooling**. With 1.5 - 2 percent of the air flow used for cooling per blade row,the blade temperature can be reduced by between 200^o and 300^oC. The next technology is **transpiration cooling**, where the cooling air is forced throufh a porous blade wall.

Cooling of rotor blade is of primary importance. Nevertheless, with high gas temperatures, oxidation becomes as significant a limiting factor as creep, and it is equally important to cool even relatively unstressed components such as nozzle blades and annulus walls.



Cooled turbine rotor blade [courtesy General Electric]

The above picture shows the principal features of nozzle cooling. The air is introduced in such a way as to provide jet impingement cooling of the inside surface of the very hot leading edge.

There are two distinct aspects of cooled turbine design.

1. The aerodynamuc design must be chosen such that a given cooling performance can be achieved by the least amount of cooling air. A common cooling performance parameter is blade relative temperature defined by,

$$BladeRelativetemperature = \frac{T_b - T_{cr}}{T_g - T_{cr}} \tag{24}$$

   where, T_b= mean blade temperature, T_{cr} = coolant temperature at the inlet, T_g = mean effective gas temperature relative to the blade.

2. Effect on the cycle efficiency of losses incurred by the cooling process. The following sources of losses must be kept in mind while designing,

   - Reduction in turbine mass flow directly reduces turbine work.
   - Expansion is no more adiabatic, and furthermore there will be a negative reheat effect in multi-stage turbines.
   - There is a pressure loss, and a reduction in enthalpy, due to the mixing of spent air with the main gas stream at the blade tips.
   - Some pumping work is done by the blades on the cooling air as it passes radially outwards through the cooling passages.
   - When considering cooled turbines for cycles with heat-exchange, account must be taken of the reduced temperature of the gas leaving the turbine which makes the heat exchanger less effective.

6.4 An example of turbine design

The following example incorporates the 'mean diameter' design for a High pressure turbine whose specifications are proposed as follows based on cycle calculations.

N.G.V. gas mass flow	m_s	25.99 kg/s
N.G.V. cooling	c_s	0.067
Rotor gas mass flow	m_r	27.92 kg/s
Rotor blade cooling	c_r	0.038
Pressure ratio	$\frac{P_{01}}{P_{03}}$	3.13
Isentropic efficiency	η_t	0.88
Temperature drop	ΔT_{0s}	349.59 K
Inlet total pressure	P_{01}	1953.73 kPa
Exit total pressure	P_{03}	623.4 kPa
Inlet total temperature	T_{01}	1450 K
Exit total temperature	T_{03}	1100.41 K
HP spool mechanical efficiency	η_m	0.99
Power	P	9919 kW

In addition to this, the rotational speed N is fixed by the compressor design. Also, we may choose to set the limit on the turbine blade tip speed U. Let's assume:

N = 580 rev/s
U = 589 m/s
$\lambda_N = 0.022$

Assuming single stage turbine analysis for simplicity, the flow is purely axial giving (i) $C_{a2} = C_{a3}$ (ii) $\alpha_3 = 0$. The temperature drop co-efficient $\psi = \frac{2c_p \Delta T_{0s}}{U^2}$ $= 2.27$

Next lets assume a flow co-efficient 0f $\phi = 0.58$ and a modest swirl angle of $\alpha_3 = 15^o$

$\tan\alpha_3 = 0.27$, which gives $\tan\beta_3 = 1.99$.
This gives a fairly acceptable reaction $\Lambda = 0.57$
The gas angles can now be established.
$\alpha_3 = 15^o$, $\beta_3 = 63.37^o$, $\beta_2 = 0.28^o$, $\alpha_2 = 59.99^o$

From the velocity diagrams, $C_{a2} = U\phi = 341.62m/s$, $C_2 = \frac{C_{a2}}{cos\alpha_2} = 682.36m/s$
$T_2 = T_{02} - \frac{C_2^2}{2c_p} = 1247.21$ K $(T_{02} = T_{01} = 1450K)$
$T_2' = T_2 - \frac{\lambda_N C_2^2}{2c_p} = 1242.75$ K
Pressure ratio $\frac{P_{01}}{P_2} = \frac{T_{01}}{T_2'}^{\frac{\gamma}{\gamma-1}} = 1.853$. This is the critical pressure indicating choking. Hence, this is the design for maximum mass flow rate.
Pressure in the plane of the throat $p_2 = 1.0541$ bar
$\rho_2 = \frac{p_2}{RT_2} = 0.2945kg/m^3$ Annulus area at plane 2,
$A_2 = \frac{m_2}{\rho_2 C_a 2} = 0.2583m^2$
Throat area of nozzles required is $A_{2N} = A_2 cos\alpha_2 = 0.1293m^2$

Similarly we calculate the annulus dimensions at 1 and 3. As its a repeating stage, our previous assumptions give, $C_{a1} = C_1 = C_3 = \frac{C_{a3}}{cos\alpha_3} = 353.66m/s$
$T_1 = T_{01} - \frac{C_1^2}{2C_P} = 1395.52K$
$p_1 = p_{01}(\frac{T_1}{T_{01}})^{\frac{\gamma-1}{\gamma}} = 1.676bar$
$\rho_1 = \frac{p_1}{RT_1} = 0.4185kg/m^3$
$A_1 = \frac{m_1}{\rho_1 C_{a1}} = 0.17559m^2$

At the outlet of the stage, we have
$T_3 = T_{03} - \frac{C_3^2}{2c_p} = T_{01} - T_{0s} - \frac{C_3^2}{2c_p} = 1045.52K$
$p_3 = p_{03}[\frac{T_3}{T_{03}}]^{\frac{\gamma-1}{\gamma}} = 0.6155bar$
$\rho_3 = \frac{p_3}{RT_3} = 0.2051kg/m^3$
$A_3 = \frac{m_3}{\rho_3 C_{a3}} = 0.3984m^2$

Now we calculate the turbine stage efficiency, assuming a rotor loss $\lambda_R = 0.08$,

$\eta_s \frac{1}{1+\frac{C_a}{2U}[\frac{\lambda_R sec^2\beta_3+(\frac{T_3}{T_2})\lambda_N sec^2\alpha_2}{tan\beta_3+tan\alpha_2-\frac{U}{C_a}}]} = 0.935$

This is better than the required efficiency of 0.88, but this doesn't include cooling effects. The efficiency drops due to rotor and NGV are below,

$\Delta\eta_r = \frac{W_r}{W_g}\eta = 0.067 * 0.935 = 0.063$

$\Delta\eta_s = \frac{W_s}{W_g}\eta = 0.037 * 0.935 = 0.034$

Effective cooled turbine efficiency is $\eta_C = 0.935 - 0.063 - 0.034 = 0.838$

7 ENGINE TESTING

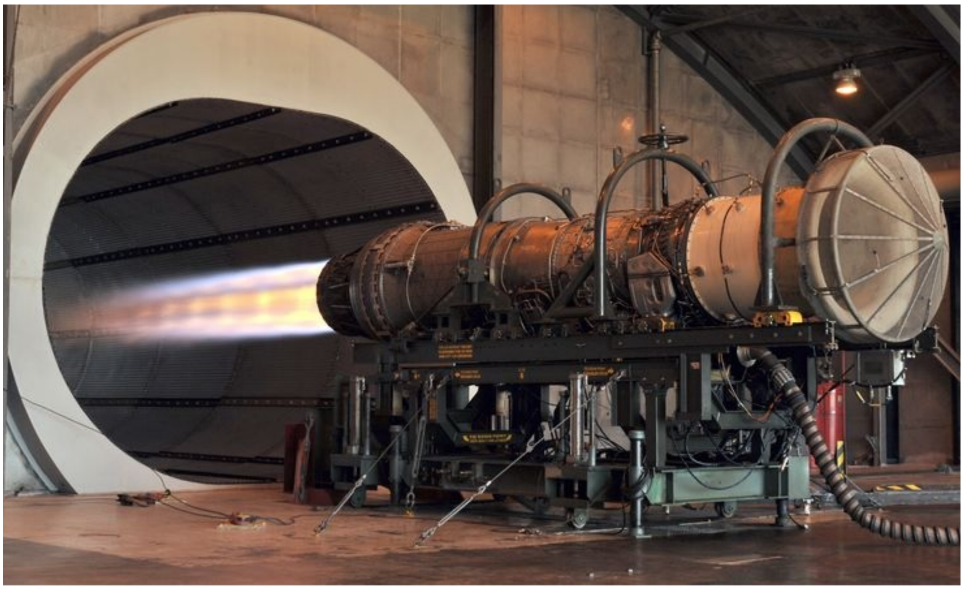


Figure 14: **An engine test rig**

For obvious reasons, an engine in development cannot be run directly and needs a facility to test it. A sophisticated engine test stand houses several sensors (or transducers) data acquisition features and actuators to control the engine state. The sensors would measure several physical variables of interest which typically include:

- Crankshaft torque and angular velocity.
- Intake air and fuel consumption rates, often detected using volumetric and/or gravimetric measurement methods.
- Air fuel ratio for the intake mixture, often detected using an exhaust gas oxygen sensor.
- Environment pollutant concentrations in the exhaust gas such as carbon monoxide, different configurations of hydrocarbons and nitrogen oxides, sulphur dioxide , and particulate matter.
- Temperatures and gas pressures at several locations on the engine body such as engine oil temperature, spark plug temperature, exhaust gas temperature, intake manifold pressure.
- Vibrations of different components.
- Atmospheric conditions such as temperature, pressure, and humidity.

Information gathered through the sensors is often processed and logged through data acquisition systems. Actuators allow for attaining a desired engine state (often characterized as a unique combination of engine torque and speed). For gasoline engines, the actuators may include an intake throttle actuator, a loading device for the engine such as an induction motor, The engine test stands are often custompackaged considering requirements of the OEM customer. They often include microcontroller-based feedback control systems with following features:

- Closed loop desired speed operation (useful towards characterization of steady- state or transient engine performance).
- Closedloop desired torque operation (useful towards emulation of invehicle, onroad scenarios, thereby enabling an alternate way of characterization of steady-state or transient engine performance.

Test rigs can either be placed outdoors or indoors depending on the location. Noise levels must be kept as low as possible. So indoor test rigs are suitable for reducing noise level to the maximum. The exhaust guide in such cases must be designed to absorb maximum noise of the exhaust.

7.1 Engine Test Rig Applications

- Research and Development of engines, typically at an OEM laboratory.
- Tuning of inuse engines, typically at service centers or for racing applications.
- End of production line at an OEM factory. The changing of the engines to be tested takes place automatically, and fluid, electrical and exhaust gas lines are connected to the test stand and engine and disconnected from them by means of docking systems. When the engine docks in the test stand the mechanical drive shaft is automatically connected to it.

7.2 Engine Test Beds

- Engine calibration process is more and more complex.
- With increasing complexities of the complete automobile, it is becoming difficult to isolate the engine development from the vehicle development.
- Need to increase the quality of tests performed on engine testbed.
- The current financial context requires reducing engine calibration time and cost.
- Urgent need to make engine tests as efficient as possible.

Simulation on engine test bed allows:

- To reduce engine development costs by performing some new tasks on engine testbed with reduced costs compare to roller testbed
- To reduce project time by reducing the need to wait for vehicle prototypes availability.

- To increase engine testing quality on engine test bed.
- All data concerning the test, the engine, and the test bed are centralized into the automation system, models can inherit of them automatically.
- Models independent of the engine test bed hardware communication

7.2.1 Test bed calibrations

Engine performance measured on indoor test beds will differ from that in an "infinite atmosphere". Test bed calibrations enables data from an indoor test bed to be adjusted to reflect the infinite atmosphere conditions. In addition, engines of a given mark may be tested on more than one test beds, each with a different configuration and hence different calibration factors.

For shaft power engines where air is ducted from ambient as per typical installations and the measurment made at duct entry, the test bed configuration does not normally need to be calibrated for. All that remains is a usual requirement for all instrumentation to be functioning satisfactorily.

Test bed definitions: Test bed approval is a formal process, involving regulation by the appropriate authority for aircraft engines and high customer involvement for other engine types. The following basic definitions apply, though other similar terms are also used:

- **'Gold' standard test bed:**
  This is the datum for cross calibration of all other beds for that engine type. Such calibration may be performed directly or via calibration of an intermediate 'silver' standard test bed.
- **'Silver' standard test bed:**
  This has directly been cross calibrated against the 'gold' standard test bed. It may be used for cross calibration of other beds for that engine type.
- **'Bronze' standard test bed:**
  This has been calibrated against a 'silver' test bed. It may be used for performance testing, but not for calibrating other test beds.
- **Functional test bed:**
  This test bed has not been calibrated for performance purposes and provides results that are indicative in nature. Such a test bed may be useful for endurance testing and demonstrating the functionality of a production engine, but not its performance.

7.3 Engine Testing for Research and Development

Research and Development (R&D) activities on engines at automobile OEMs have necessitated sophisticated engine test stands. Automobile OEMs are usually interested in developing engines that meet the following threefold objectives:

- To provide high fuel efficiency.

- To improve drivability and durability.
- To be in compliance to relevant emission legislation.

Consequently, an R&D engine test stands allow for a fullfledged engine development exercise through measurement, control and record of several relevant engine variables.

7.3.1 Objectives

- Determine fuel efficiency and drivability: torquespeed performance test under steadystate and transient condition.
- Determine durability: ageing tests, oil and lubrication tests.
- Determine compliance to relevant emission legislations: volumetric and mass emission tests over stated emission test cycles.
- Gain further knowledge about the engine itself: engine mapping exercise or development of multidimensional inputoutput maps among different engine variables. e.g. a map from intake manifold pressure and engine speed to intake air flow rate.
- Tests under ideal conditions of intake.
- Tests with aircraft with facility to simulate intake distortions.
- Test in altitude chamber to simulate altitude.
- Test with preheated air and pressurization of air to simulate high speed flight.
- Altitude speed characteristics.
- Determination of stall margin of compressor.
- Mechanical integrity of engine.
- Determination of engine performance under different climatic and weather conditions.

7.3.2 Test Beds at HAL

HAL demonstrated the use of test beds for the first time in 1965 for its maiden jet engine HJE2500. The engine division of HAL houses the largest number of test beds in India. They are:

- TM 3332B2 / Shakti Engine Test Bed
- Adour Engine Test Bed
- Pegasus Engine Test Bed
- Garrett Engine Test Bed
- R29B Engine Test Bed

- LM 2500 Engine Test Bed

The space enclosing the test rig is usually constructed with **sound proofing** in mind. The **aft region** of the building must have air inlet guides and the rear section must have an exhaust guide pipe. The test rig consists of a **fixture** for the engine to be mounted. These fixtures must be able to withstand the large thrust the engine produces while testing. It must be able to withstand the loading and unloading due to powering on and off of the engines, and must also bear with long testing hours. Depending on the size of the engine, it can be **floor mounted** or it can **hang** on to the fixture without ground support. Small engines like **TM333-Shakti** use the former, while **ADOUR** is tested on the latter.

The **control room** is situated adjacent to the rig with a small window made of tough glass. This is meant for **visual inspection** during the testing. The glass must be able to withstand turbine blades impinging it at high velocities in an unlikely event. The engine noise is too high for anyone to be present in the test rig itself. To prevent the engine from sucking in debris in the testing room, a **filter** is fixed to the front of the engine. It also acts a **inlet guide.**

Some **other ventures** by HAL:

- g **loading** for which HAL(EDB) designed a special **test rig** (centrifuge) to simulate the g loads while the engine was run at a specified low rpm by air impingement on the blades.
- Due to special mission requirement that the engine on sea water dunking should be reusable, a **special rigs** was developed by HAL to **simulate the sea dunking process.** In this rig, the engine, after engine kill, is immersed in salt water within 60 to 90 seconds in a nose down altitude and remains at a depth of 3 meters for 3 hours. The engine is checked on the test bed after a laid down decontamination and refurbishment procedure.

Figure 15: **An engine undergoing water ingestion test.**

The engines are designed to suck in air, but they have to be capable of taking on everything else they may encounter in the sky, most notably birds and bad weather. To ensure thats the case, **manufacturers** run tests that are as straightforward as they are brilliant: the jets are turned on and things are thrown in there. The teams job is to make sure that engines keep working when they run into bad thunderstorms or a stray seagull. They expose the machines to hail and monsoon rain, hit them with bird carcasses, and even set off small explosions inside to simulate blade failure. Following are the various types of extreme testings carried out

7.3.3 Water injestion testing

The engines need to handle **water**, so a hose is set up and blast a running engine with hundreds of gallons of water per minute. When the test goes as planned, all that rain flies through the chamber and out back without diminishing thrust.

7.3.4 Testing against ice formation

Ice is slightly more problematic, so much that the FAA requires engines to be able to handle several specific varieties of ice formation and ensure they can recover quickly. Besides starting the engines in freezing conditions, testers shoot huge ice balls inside a running engine.

7.3.5 Bird injestion testing

Bird strike is another serious problem that needs attention. The big ones can actually bend back the blades at the front of the engine, making it stall or explode. A special chicken gun fires (already dead) birds into running engines. The goal is for the blades to hold their form after the collision.

7.3.6 Blade-out testing

The most violent test of all is the **blade out procedure**. This simulates an event where a single blade at the front of the engine, due to wear, snaps off from the shaft while spinning at **over 3,000 RPM**. At that engine speed, the blade can quickly become shrapnel and tear through the rest of the plane if the fracture isnt contained.

To make sure its **containment**, engineers rig a small explosive to the base of the blade, which separates it from the shaft. When the test goes well, the blade stays within the engine chamber, while the casing diffuses the energy from the impact.

7.3.7 Flame out testing

Flameout refers to the **rundown of an engine** caused by the **extinction of the flame in the combustion chamber**. It can be caused by a number of factors, including fuel starvation, compressor stall, insufficient oxygen (at high altitudes), foreign object damage (such as caused by birds, hail or volcanic ash), severe weather conditions, mechanical failure and very cold ambient temperatures.

A flameout is most likely to occur when flying through certain **weather conditions** at a **low power setting** such as flight idle (e.g. during the descent). These conditions include flight through moderate to heavy turbulence, rain, hail or sleet. The **FADEC engine controller** will select ignition automatically if it detects specific changes in engine parameters. It will also perform a **relight if necessary.**

Thus in a flameout test, the aim of the experiment is to **starve the engine of fuel and/or air** for a **certain amount of time** and ensure it does not flameout and can **selfsustain** in spite of such condition.

This is done by **suddenly cuttingoff the fuel supply** to the engine. In cases where the engine flames out, it has to be **manually started** and hence it fails this test.

8 HTFE 25

Figure 16: **The HTFE 25**

Aero Engine Research & Design Centre (AERDC), HAL, Bengaluru had taken up the Design and Development of a **25 kN thrust class turbofan engine** for **Basic Military Trainer Aircraft**. HAL used its internal resources with an aim of producing indigenously designed and developed aero engines in a time frame of six years beginning from 2013. In the first phase of the programme, the design of the full engine and manufacture and testing of the technology demonstrator of the core engine are the subsequent phases the manufacture and testing of the full engine will follow.

The **HTSE120** can be used for helicopters of **3.5 ton class** in the single engine configuration (e.g. LUH) and for **5 to 8 ton class in twin engine** configuration (e.g. ALH, LCH). The engine develops a **power of 1200 kW** at sea level and can operate up to an altitude of 7 km. This project too is undertaken by HAL with its internal resources with an aim of developing the capability of indigenous design and development of turboshaft engines.

8.1 Engine Specifications

- **25kN** thrust engine.
- Engine **length** :1730mm
- Engine **diameter** : 590mm
- Net **weight** : 350 kg
- **Low bypass twin spool** mixed flow turbofan engine with:
  - **Wide chord** fan blades.
  - High efficiency compressor with **variable IGV.**
  - Advanced combustor.
- Engine control is through **FADEC.**
- Useful for business jet and combat trainers in the **5 ton class.**

8.2 Salient Features

- **Multidisciplinary approach** for design.
- **Systems Engineering** Simulation/Concept.
- **Digital Manufacturing**.
- Planned **concept** is to develop core engine first and then develop full engine based on the platform it would be fitted to.
- Emphasize on **MD design iterations** with quick iteration churning.
- **Secondary flows**:
  First **3D simulations** to capture flow characteristics and then use **1D models** calibrated using the 3D simulations. 1D for **unsteady simulations**. Verification using existing engine data done as a whole.
- **Idle** = 70 % RPM
- **3D printing technology** adapted right from concept stage. Main idea behind this was to reduce lead time to manufacture critical components in order to reduce risk of program. (Since only CAD input needs to be changed to change component produced).
- **Casting** to be replaced by 3D printing while forging, sheet metal processes retained.
- **Nozzle GV** 3D printed with **Inconel material** intricate cooling passages with additional which would not be possible with conventional techniques are easily made with 3D printing. The blades are perforated to incorporate fill cooling. The **ATS** provides 30-40% of the flow for cooling.
- **CC part** (flame tube section), Turbine blades, HPC all stator vane sectors, HPC tandem stator, gear box parts. Pros of 3D printing:
  - Practically **zero lead time** for implementing design change in manufacturing.
  - Full **freedom to designer**.
  - Possibility of a lot of **weight reduction** by having internal cavities at noncritical location.

9 ARTOUSTE TURBOSHAFT ENGINE

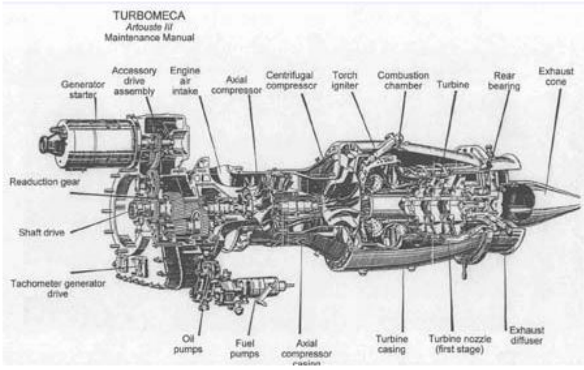


Figure 17: **The Artouste**

Artouste III B Engine powers both **Cheetah** and **Chetak** Helicopters. Manufacture of the Engine commenced in 1962 under licence from **Turbomeca**, France. The Engine has **side air intake**, **one axial** and **one centrifugal compressor** connected to a **threestage Turbine**. The **power output** is 550 SHP at **33,500 RPM**. The reduction Gearbox in the front transfers the power to Helicopter. More than 665 Engines have been manufactured and 2950 Engines overhauled and repaired for various customers.

9.1 Engine Specifications

- Maximum **Diameter**: 0.50 (m)
- **Length**: 1.815 (m)
- **Weight**: 182 (kg)
- **Power**: 550 kW
- **Specific Fuel Consumption**: 0.346 (kg / shphr)

9.2 Salient Features

- **EFC (Electric Fuel Control):**
  Gas turbine engines are primarily controlled by the **amount of fuel supplied** to the combustion chambers. With this in mind, the very simplest fuel control for a turbine engine is a **fuel valve operated by the pilot**. Many preproduction models of early turbojet engines featured just that, but it was soon found that this kind of control was difficult and dangerous in actual use. Closing the valve too quickly while trying to reduce power output could cause a **lean dieout**, where the airflow through the engine blows the flame out of the combustion chamber and extinguishes it. Adding fuel too quickly to increase power can damage the turbines due to excessive heat, or the sudden rise in combustion chamber pressure

may cause a compressor stall. Another danger of too much fuel is a **rich blowout**, where soaking the fire with fuel displaces the oxygen and lowers the temperature enough to extinguish the flame. The excess fuel may then be heated on the hot tailpipe and ignite, possibly causing **damage to the aircraft**.

The operator has a **power lever** which only controls the engine's potential, not the actual fuel flow. The fuel control unit acts as a computer to determine the amount of fuel needed to deliver the power requested by the operator.

An **EFC** is essentially a **hydromechanical fuel control** but with added **electrical components** to prevent overheating or overspeeding the engine. If the electrical part of the control should fail, an **EEC** will revert to a standard hydromechanical fuel control.

- The **fuel pump** is designed to run from the shaft power with the help of the gear box.
- A **speed governor** is employed to prevent overspeeding of the engine.
- An in built **AID (Altitude Indicator Device)** helps to run the engine at the required conditions of air flow and fuel supply at different altitudes.
- The fuel pumps employ **SSV Solenoidal valves**. They are compact, general service, two way **guide type** solenoid valves for air, gas, water, and other liquid applications. They are available in stainless steel with a normally open design and can be oriented in any position. The solenoid enclosure provides protection against dust, while also protecting against seepage of oil and noncorrosive coolants.
- The turbine use **Wiekel IN718 blades**. The engine contains both directionally solidified as well as **single crystal blades.**

10 SHAKTI ENGINE

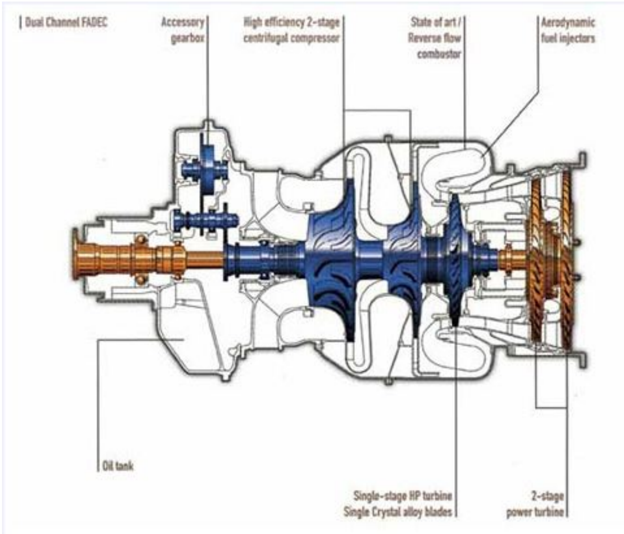


Figure 18: **The Shakti engine**

The **Turbomeca Ardiden** is a family of turboshaft engines featuring **simple, modular and compact design**. They are built around a gas generator with two centrifugal compressor stages, coupled to a singlestage highpressure turbine. The**power turbine** comprises two stages. The engine is controlled by a dualchannel **Engine Electronic Control Unit (EECU)**. The Ardiden engines offer very low cost of maintenance and ownership. Developing from **1,200 to 2,000 shp of maximum power**, the engine is suitable to power helicopters in the five to eight tons class. Besides, the Ardiden engines satisfy the most demanding mission requirements, while retaining full performance under high altitude and hot temperature conditions.

The **Shakti, also known as Ardiden 1H1**, is a turboshaft engine jointly developed by Turbomeca (France) and Hindustan Aeronautics Limited (HAL) based on the Ardiden turboshaft to power weaponized variants of the **Dhruv Advanced Light Helicopter (ALH)**. It made its maiden flight powering the Dhruv helicopter on **August 16, 2007**. The first Ardiden 1H1 reengined Dhruv helicopter took to the skies on January 12, 2009.

The Shakti features **three modules** for easy maintenance:

- A reduction / accessory gearbox module.
- A gas generator module.
- A power turbine module.

The time between overhaul (TBO) ranges from **3,000 to 6,000 hours**. The Shakti engine delivers 30 percent more power than TM3332B2 power plant, which also powers the Dhruv, and features 19 percent of components developed in India. The Shakti is the prime candidate for powering **Tactical Supporting**

Helicopters Dhruv and the future Light Combat Helicopter **(LCH)** also being developed by HAL.

10.1 Engine Specifications

- **Emergency Power**: 1204 kW
- **Max Continuous Power**: 880 kW
- Max power at **take off**: 1053 kW
- **OEI 2 minutes**: 1099 kW
- OEI **continuous**: 1024 kW
- Dry **weight**: 180 kg
- **Time between overhaul**: 3000 hours

The development of Shakti from Ardiden in India began in 2005 in multiple phases.

- **Phase Zero:**
  HAL shared 11% of the production while the rest was done by Turbomeca. 65 engines were imported from Turbomeca.
- **Phase One:**
  Semi assembled kits were imported from Turbomeca and the complete assembly was done by HAL itself.
- **Phase Two:**
  Some of the individual parts were imported, assembled and tested by HAL on its own.
- **Phase Three:**
  More individual parts were assembled. No semi assembled kits were imported. Some of the components were manufactured indigenously.
- **Phase Four:**
  In house manufacture of components at HAL upto 73% was achieved.

10.2 Salient Features

- The gearbox consists of two trains of gears attached to two different spools:
  (i) **Reduction train / Power train:** It comprises of two gears which reduces the rpm of from the main shaft to 6000 rev/s.
  (ii) **GG train:** This is the gear train attached to the Gas Generator
- A typical helicopter like Dhruv employs two engines where the rpm is reduced from **6000 rev/s** of the power turbine to **400 rev/s** of the rotor blades using a gear box. **27%** of the power turbine output is supplied to the tail rotor.

- The engine comprises of **two casings**. They were previously made of **magnesium.** Currently, aluminum has proved to be of higher strength with reduced weight.
- The power turbine has a built in Torque sensor which measures the output torque.
- There are three main **Subsystems**:
  - **Oil System:**
    This for lubrication and cooling of engine components.
  - **Electrical System:**
    This is for various aspects like controlling fuel flow, temperature and pressure measurements, RPM measurements etc. For example, a **fuel metering unit** controls fuel flow with th e help of a stepper motor. It has no mechanical linkages and tuning is done with the help of potentiometers. **Piezo Sensing mechanisms** are used in transducers of most of the actuators.
  - **Air Systems:**
    The air suction case is of the annular type with grid/filter. The air enters sideways through a sand screen and snow screen.
- For detecting blockage and cracks in the engine without its disassembly, a **Boroscope** is used.
- The engine has two **mountings front and rear.**
- The **gas generator** is of the **centrifugal type** as the amount of mass flow is low. The combustion chamber is a Reverse flow annular type combustor.
- The **G Rotor** oil pumps are three in number. One is a **pressure pump** while two others are **scavenging pumps** one each at the front and the rear section.
- Two mechanisms are employed to cool air **Air cooling and fuel cooling.**
- An oil filter processes the retrieved oil before recirculation in two stages, namely **Debris stage and Micronic stage.**
- There are two types of pumps:
  (i) **EDP Engine Driven Pumps**
  (ii) **Electrically Driven booster pump** which is LP pump.
  The latter one is a liquid ring type pump. The pressure pump has constant mesh and maintains a constant pressure.
- The **injectors** start the **combustion process.** This is required only on start up. Air is allowed to pass through them after fuel to avoid sudden ignition due to fuel residue. The injector is a **Differential Injector** that has no direct connection to the throttle. It provides a constant flow of air to **prevent flameout.**

- The combustion chamber has a simple burner which is supplied with fuel through a **Single Line Fuel system** controlled by **FADEC.** The burner is self centering which comprises of diluting holes for cooling as well as for air entry.
- For **frequency matching** and **vibration control**, a Phonic Wheel Generators are used instead of a **Tachometer.**
- The NGV in this engine doesnt have cooled blades, but the **surface area is increased** to compensate for the effects.
- The **directionally solidified Power turbine blades** are longer than the turbine blades which are primarily Single Crystal blades. The compressor turbine assembly consists of **two stages of directionally solidified** blades and **one stage of single crystal** blades. The blades are mounted on the disc using **firtree** type **mountings**. Each of the manufactured blades may vary in weight and their position is critical for their performance. Positioning of blades is usually done with the help of computer software.
- The **first stage** of the compressor is a **supersonic type.** The **tip speed** goes **above Mach 1.** They are not manufactured at HAL and are imported from Turbomeca. The **second stage** is as **subsonic compressor** which is manufactured in house at HAL.
- The air and oil that is circulated by the pumps is separated for recirculation with the help of a **Centrifugal breather**. A **chuck pipe** creates a low pressure for the same.
- A special type of **Self locking nuts** are used in the power shaft which are primarily imported. The movement on the threading happens only between specific minimum and maximum torques and it sits in place at a particular net torque.
- The **Power shaft** consists of **reference ring**, a **torque sensor** and a **reference shaft.** The space between the ring and the sensor gives the **torque equivalent**, which makes it easier for direct torque measurements.

11 THE ADOUR ENGINE

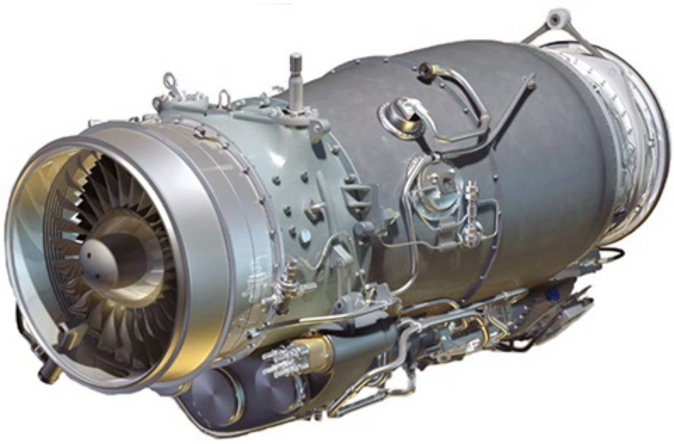


Figure 19: **Adour Mk 804**

The Rolls-Royce Turbomeca Adour is a two-shaft low bypass turbofan aircraft engine developed by Rolls-Royce Turbomeca Limited, a joint subsidiary of Rolls-Royce (UK) and Turbomeca (France). The engine is named after the Adour, a river in south western France. The Adour is a turbofan engine developed primarily to power the Anglo-French Jaguar fighter-bomber, achieving its first successful test run in 1968. It is produced in versions with or without reheat. The following Adour engines are being license built by HAL since 1981:

- **Adour Mk 804** - Licence-built by HAL for Indian Air Force phase 2 Jaguars
- **Adour Mk 811**- Licence-built by HAL for Indian Air Force phase 3 to 6 Jaguars
- **Adour Mk 821** - Engine upgrade of Mk804 and Mk811 engines, currently under development, for Indian Air Force Jaguar aircraft.

11.1 General characteristics

- Type: Turbofan
- Length: 114 inches (2.90 m)
- Diameter: 22.3 inches (0.57 m)
- Dry weight: 1,784 lb (809 kg)

11.2 Performance:

- Maximum thrust: 6,000 lb (27.0 KN) dry / 8,430 lb (37.5 KN) with reheat
- Overall pressure ratio: 10.4
- Bypass ratio: 0.75-0.8
- Fuel consumption: dry 0.81 lb/(lbfh) (23 g/(kNs))
- Thrust to weight ratio: 4.725:1

11.3 Adour modules

Adour employs a modular design which makes it easy to assemble, maintain and detect defects. **It consists of mainly 12 modules:**

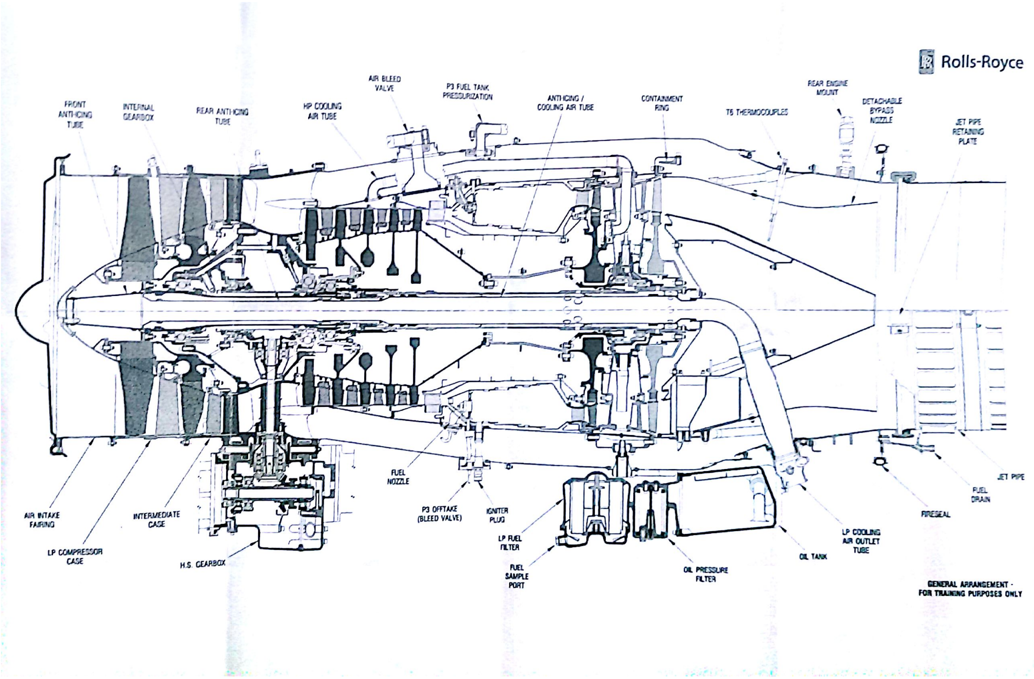


Figure 20: **Adour schematic**

11.3.1 Module 1: LP Compressor Stage 1

It comprises of a low pressure compressor with a rotor (27 blades), a stator (23 blades) and a rotor again (32 blades). The rotors in this stage rotate at a relatively low rpm as they are larger and the blade tip speeds must be nominally low.

11.3.2 Module 2: LP Compressor Stage 2

The stage 2 of LP compressor comprises of only 2 stators. The compressors are protected by an **LP Compressor case.** The Air Intake Fairing guides the incoming air into the compressor stages. Ice accumulation on the leading edge (lip) of engine inlets causes flow problems and can lead to ice ingestion. In turbofan engines, laminar airflow is required at the face of the fan. Because of this, there are two Anti-icing systems in the compressor stages, the **Front Anti-icing tube** and the **Rear Anti-icing Tube.** They are located just at the inlet and after the compressor stages respectively. The former prevents ice accumulation at the rotor blades as well as the immediate shaft locations, while the latter prevents ice accumulations at the gearbox and makes sure no ice enters the high pressure compressor.

11.3.3 Module 3: Intermediate casing, Front Engine Mount and Internal Gearbox Assembly

The compressor intermediate casing houses the stage 2 LP compressor and the internal gearbox. The **internal gearbox** consists of a bevel gear mounted in ball thrust and roller bearing assembly. It provides drive for the splined HP Compressor rotor shaft. It is driven by the High speed gear box. This module is comprises of 3 shafts. The LP shaft is connected to the M1 intershaft. An Internal Gear Box Assembly connects the shafts and the Stages 1 and 2 compressors. This module also contains the **Front Bypass air duct** to separate the flow into two streams, for by-pass and to the core. It trails with the **Combustion Outer Case Assembly.** The entire Compressor assembly consists of 5 stages of high pressure compressors and 2 stages of low pressure compressors.

11.3.4 Module 4: HP Compressor

The HP Compressor consists of two main assemblies, the **HP rotor** and **HP compressor case and vanes.** The rotor consists of 5 stages of rotor blades secured to a drum, case and vanes. Stator consists of 4 stages built up around the compressor rotor stages. The **Internal gearbox** is usually placed between the compressor outlet and the combustor. For two-shaft designs, an accessory drive will be taken from the high pressure shaft i.e. the outer and shorter of the two concentric shafts. The drive and accessory gearboxes may also be split in two, one driven from each engine shaft, so as to distribute their loads. The engine-critical systems, including the starter drive, are arranged on the high-pressure shaft, with aircraft systems on the low-pressure shaft. The high-pressure shaft also rotates faster than the low-pressure shaft.

11.3.5 Module 5: HP NGV

HP NGV refers to **high pressure nozzle guide vanes.** These vanes help in streamlining of the high pressure, turbulent air exiting from the HP compressor. This helps in preventing damage to the HP turbine caused due to turbulent air.

11.3.6 Module 6: HP Turbine

- The HP turbine uses the High Pressure air from the compressor to drive the HP shaft.

11.3.7 Module 7: LP NGV

The LP NGV has a similar function to HP NGV. It removes any turbulence in the air exiting from HP turbine.

11.3.8 Module 8: LP Turbine

The LP turbine uses the energy of the LP air to drive the LP shaft. This module is protected by a **Containment ring.** This casing ensures that in case of breakage of engine parts/blades, the damage is contained".

11.3.9 MODULE 9: Exhaust Mixer Section

The exhaust mixer section has two parts, the **annular exhaust mixture** and the **exhaust cone.** The exhaust mixer section mixes the cold bypass air with air coming through the compressors/turbine.

11.3.10 Module 10: HS Gearbox

The high speed gearbox is mounted outside the intermediate casing. It comprises of 32 gears and 28 bearings. The HSG is coupled to the internal gearbox and micro-turbo. It draws power from the micro-turbo and transmits it to the internal gearbox which in turn drives the HP rotor shaft. The HSG is coupled to the internal gearbox and micotube.

11.3.11 Module 11: Accessories Pack- Oil Tank Cooler and Filters Assembly

The accessories pack mainly has to do with the fuel and oil. It consists of fuel and oil filters. It also consists of a **low pressure fuel pump, high pressure fuel pump** and an **oil pump**. The **fuel control unit** and **Cambox** are also present here. The pack also has a fuel cooled oil cooler. Hot oil scavenged from the engine is pumped through a one way check valve into the oil cooler. **Oil cooler** blows air at ambient temperature is ducted through the **cooler/heat exchanger.** Cooled oil is returned to the oil tank. Oil drawn from the tank by the suction side of the **engine driven pump** is then distributed through the engine under pressure to cool and lubricate moving parts.

11.3.12 Module 12: Reheat Vapour Gutter and Manifold Assembly

This forms the **afterburner assembly** of the engine which is used for thrust boosting for a limited period of time. Due to reheating, the **turbine exit dryness fraction** increases so moisture decreases, so blade erosion becomes minimum, so life of the turbine will be increased.

Apart from the 12 modules, the engine also has a Micro turbo which is used to start the engine and the combustion section.

11.3.13 Combustion Module

The combustion section does not have any designated module. Major components are the **combustion liner, front inner combustion case** and **combustion outer case.** The combustion liner is circular in shape and is built up in section welded together to form 2 skins, the area between the 2 skins being the combustion area. The unit tapers towards the front and forms slots, each slot being lined with fuel supply nozzles. There are **18 nozzles** present. Some of the components are:

- **Air bleed valves** are used to unload the force of excess air in the compressor.
- The **Fuel tank pressurization unit** is required to increase the pressure and temperature of the fuel
- to maintain a desired rate of fuel combustion.
- The **Anti-icing/cooling air tube** provides pressurised air for anti-icing purposes.
- **Fuel nozzle** injects the fuel into the combustor and can be used to control the amount and timing of the fuel input. The fuel is sprayed into the combustor after it is atomised.
- **P3 off-take (bleed valve)** The hot air is "bled" off into tubes routed through wings, tail surfaces, and engine inlets.
- The **Igniter plug** is used to ignite the atomised fuel in the combustor to initiate the oxidation of fuel.

The **Bypass section** in the engine splits the air from the LP compressor into 2 main flows in the compressor intermediate casing. Approximately **50%** of the airflow is directed into the **high pressure compressor** and the remaining flow passes through the **annular bypass duct area** to the exhaust mixture. The front Bypass duct has liners riveted to soleplates which is used for fuel tank pressurization, fuel spray nozzle feed and air bleed valve. The **Rear Bypass duct** similarly is used for oil return and vent, feed oil to bearings, fuel drain and oil tank support.

Air pressures in the Adour engine are used as follows; **P2** is used for fuel tank pressurization, HSG air blown seal, anti-icing and cabin air. **P3** is used for cooling of HP NGV, HP Turbine (front). **P3** intermediate is used for cooling LP NGV, HP turbine rear and LP turbine cooling.

The major materials used are **Titanium, Magnesium, nimonic, steel alloys** and **Sermetal.** The combustion chamber is coated with thermal barrier coating. **Thermoseal coating** are also provided to certain part made of **Sermetal.**