AEROSPACE PROPULSION (R15A2103)

COURSE FILE

II B. Tech II Semester

(2017-2018)

Prepared By
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Department of Aeronautical Engineering

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

Affiliated to JNTU, Hyderabad, Approved by AICTE - Accredited by NBA & NAAC – 'A' Grade - ISO 9001:2015 Certified)
Maisammaguda, Dhulapally (Post Via. Kompally), Secunderabad – 500100, Telangana State, India.

MRCET VISION

To become a model institution in the fields of Engineering, Technology and Management.

To have a perfect synchronization of the ideologies of MRCET with challenging demands of International Pioneering Organizations.

MRCET MISSION

To establish a pedestal for the integral innovation, team spirit, originality and competence in the students, expose them to face the global challenges and become pioneers of Indian vision of modern society

MRCET QUALITY POLICY

To pursue continual improvement of teaching learning process of Undergraduate and Post Graduate programs in Engineering & Management vigorously.

To provide state of art infrastructure and expertise to impart the quality education.

PROGRAM OUTCOMES
(PO's)

Engineering Graduates will be able to:

1. **Engineering knowledge**: Apply the knowledge of mathematics, science, engineeringfundamentals, and an engineering specialization to the solution of complex engineering problems.
2. **Problem analysis**: Identify, formulate, review research literature, and analyzecomplex engineering problems reaching substantiated conclusions using first principles of mathematics, natural sciences, and engineering sciences.
3. **Design / development of solutions**: Design solutions for complex engineeringproblems and design system components or processes that meet the specified needs with appropriate consideration for the public health and safety, and the cultural, societal, and environmental considerations.
4. **Conduct investigations of complex problems**: Use research-based knowledge andresearch methods including design of experiments, analysis and interpretation of data, and synthesis of the information to provide valid conclusions.
5. **Modern tool usage**: Create, select, and apply appropriate techniques, resources, andmodern engineering and IT tools including prediction and modeling to complex engineering activities with an understanding of the limitations.
6. **The engineer and society**: Apply reasoning informed by the contextual knowledge toassess societal, health, safety, legal and cultural issues and the consequent responsibilities relevant to the professional engineering practice.
7. **Environment and sustainability**: Understand the impact of the professionalengineering solutions in societal and environmental contexts, and demonstrate the knowledge of, and need for sustainable development.
8. **Ethics**: Apply ethical principles and commit to professional ethics and responsibilitiesand norms of the engineering practice.
9. **Individual and team work**: Function effectively as an individual, and as a member orleader in diverse teams, and in multidisciplinary settings.
10. **Communication**: Communicate effectively on complex engineering activities with theengineering community and with society at large, such as, being able to comprehend and write effective reports and design documentation, make effective presentations, and give and receive clear instructions.
11. **Project management and finance**: Demonstrate knowledge and understanding ofthe engineering and management principles and apply these to one's own work, as a member and leader in a team, to manage projects and in multi disciplinary environments.
12. **Life- long learning**: Recognize the need for, and have the preparation and ability toengage in independent and life-long learning in the broadest context of technological change.

DEPARTMENT OF AERONAUTICAL ENGINEERING

VISION

Department of Aeronautical Engineering aims to be indispensable source in Aeronautical Engineering which has a zeal to provide the value driven platform for the students to acquire knowledge and empower themselves to shoulder higher responsibility in building a strong nation.

MISSION

The primary mission of the department is to promote engineering education and research. To strive consistently to provide quality education, keeping in pace with time and technology. Department passions to integrate the intellectual, spiritual, ethical and social development of the students for shaping them into dynamic engineers.

QUALITY POLICY STATEMENT

Impart up-to-date knowledge to the students in Aeronautical area to make them quality engineers. Make the students experience the applications on quality equipment and tools. Provide systems, resources and training opportunities to achieve continuous improvement. Maintain global standards in education, training and services.

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

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II Year B. Tech, ANE-II Sem	4	1/-/-	3

(R15A2103)AEROSPACE PROPULSION

Objectives:

- Students can focus on various propulsion systems available in aerospace industry and also understand the future scenario.
- Understand the performance aspects at the design point and off design operations.
- To provide an exposure with reference to numerical calculations and design limitations.

UNIT I

FUNDAMENTALS OF PROPULSION: Evolution of flight propulsion, types of aerospace propulsion, working principles, advantages, disadvantages, applications – reciprocating engines, propellers, jet engine, turboprop, turbofan, turbo-shaft, ramjet, scramjet, pulsejet. Engine components-performance requirements, thermodynamic processes- change of state- representation by T-sand p-v diagrams - pressure ratios, temperature ratios. Energy transfer, losses- entropy generation-mechanisms. Performance- polytropic, stage and component efficiencies, burning efficiency
Station numbering in engine, thrust generation, momentum equations, equation of thrust for installed and uninstalled cases, factors affecting thrust, Role of propulsion in aircraft performance.

UNIT II

ANATOMY OF JET ENGINE-I

INLETS: Locations, Types of inlets, operating principle, functions, geometry, operating conditions, flow field, capture area, sizing, flow distortion, drag, and diffuser loses, methods of mitigation, performance.

COMPRESSOR & TURBINE: types, construction, stage, cascade, blade geometry, velocity triangles, Euler equation, types of flow analysis, diffusion factor, stage loading, Variable stator, limits on compressor performance, typical blade profiles. Axial flow turbines-, similarities and differences with compressors, Velocity diagram analysis, no exit swirl condition, flow losses, causes tangential stresses, repeating stages, Computation of stage parameters for ideal and real turbine of given cascade, blade geometry and initial flow conditions and turbine speed- procedure. Typical turbine blade profiles, turbine performance maps, Thermal limits of blades, cooling, materials, construction, methods of production, Limits on stage pressure ratio of turbines- multistage, multi-spooled turbines. Range of axial flow turbine, design parameters, Typical turbine blade profiles.

UNIT III

ANATOMY OF JET ENGINE-II

BURNER: Burners- types, components- function, schematic diagram, airflow distribution, cooling-types, cooling effectiveness, performance parameters, combustion efficiency, overall total pressure loss, exit temperature profile, ignition relight envelope- effect of combustor design, Fuel injection, atomisation, vaporisation, recirculation- flame stabilisation, flame holders. Afterburners, function, components, design requirements, design parameters, bypass duct, total pressure losses, Mixing process pressure losses, fuels composition, specifications of commonly used fuels.

NOZZLE: Exhaust nozzles- primary nozzle, fan nozzle- governing equations of flow- choking, engine back pressure control, nozzle-area ratio, thrust reversal, vectoring mechanisms. Afterburner functions and its components, design requirements and parameters. Performance gross thrust coefficient, discharge, coefficient, velocity coefficient, angularity coefficient, performance maps.

UNIT IV

RAMJET & SCRAMJET ENGINE: components, Performance of turbojets, ramjets at high speeds- limitations. Need for supersonic combustion, Implications criticality of efficient diffusion and acceleration, problems of combustion in high speed flow, The scramjet engine- construction, flow process- description, control volume analysis spill-over drag, plume drag, Component performance analysis- isolator, combustor- flow detachment and reattachment, thermal throat, scheduled, distributed fuel injection, Nozzle flow, losses- failure to recombination, viscous losses, plume losses. Scramjet performance applications, Combined cycle engines- turbo-ramjet, Air turbo-rocket (ATR), ejector ramjet, Liquid-air collection engine (LACE)- need, principle, construction, operation, performance

UNIT V

ROCKET ENGINE:

CHEMICAL ROCKET: Classification of rocket engine, chemical rocket engine types, working principle, schematic diagram, applications, types, advantages and disadvantages solid, liquid and hybrid propellant rocket engine, propellants types used, injectors, nozzles, igniters, storage, TVC, combustion instabilities, combustion chamber, pulse detonation engine, rotary rocket engine

NUCLEAR: Power, thrust, energy. Nuclear fission- basics, sustainable chain reaction, calculating criticality, reactor dimensions, neutron leakage, control, reflection, prompt and delayed neutrons, thermal stability. Nuclear propulsion, history, principles, fuel elements, exhausts velocity, operating temperature, The nuclear thermal rocket engine, radiation and management, propellant flow and cooling, control, start-up and shutdown, nozzle, thrust generation. Potential applications of nuclear engines- operational issues, interplanetary transfer manoeuvres, faster interplanetary journey. Development status of nuclear engines, alternative reactor types, safety issues, nuclear propelled missions.

ELECTRICAL: Limitations of chemical rocket engines. Electric propulsion systems- structure, types, generation of thrust. Electrostatic thrusters, electro-magnetic thrusters, applications to space missions, pulsed plasma thrusters (PPT) for micro-spacecraft, solar electric propulsion.

ADVANCED: Micro-propulsion, micro-propulsion options, application of MEMS, chemical, electric micro-thrusters, principle, description, Propellantless propulsion, tethers, momentum exchange, electro-dynamic Photon rocket, beamed energy propulsion, solar, magnetic sails.

Text Books:

1. V Ganeshan Gas Turbines, Mc Graw-Hill Third Edition 2014.
2. Mattingly, J.D., Elements of Gas Turbine Propulsion, McGraw-Hill, 1996, ISBN0-07-912196-9.
3. Flack, R.D., Fundamentals of Jet Propulsion with applications, Cambridge University Press, 2005, ISBN0-521-81983-0.

Reference Books:

1. Ahmed F.EL. Sayeed., Aircraft Propulsion and Gas Turbine Engines, CRC Press, ISBN 978-0-8493-9196-5
2. Sutton, Rocket propulsion elements, Wiley Interscience publications, 7edition, ISBN- 0-471-32642-9

Outcomes:

- Students attain knowledge of all propulsion techniques being employed.
- Students will be able to configure the engine required for specific need.
- Students can able to design the engine requirements.

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II B.Tech II SEMESTER – AERONAUTICAL ENGINEERING

AEROSPACE PROPULSION (R15)

MODEL PAPER – I

MAXIMUM MARKS: 75

PART A **Marks:25**

All questions in this section are compulsory

Answer in two to four sentences.

1. **(a)** Draw the P-V and T-S diagrams of a brayton cycle and mention the processes in them.
   (b) What are the functions of air intake in a gas turbine engine?
   (c) Draw a neat sketch of a mixed compression inlet and label parts.
   (d) Define polytropic efficiency.
   (e) Differentiate between under-expanded and over-expanded nozzle.
   (f) Discuss with a neat sketch any two types of cooling in turbine blades
   (g) Define and explain Degree of Reaction in case of axial flow compressors.
   (h) Explain the relevance of C*and C_F of a solid rocket engine
   (i) State the advantages of scramjet engines in military and civil applications
   (j) Explain the formation of thermal throat in a scram jet engine

Part B **Marks: 50**

Answer all questions

2. Derive the expression for installed and uninstalled thrust for the aircraft.

or

3. Explain the turbojet engine operation, advantages and disadvantages with a neat sketch.

4. Derive an expression for the propulsive efficiency of a gas turbine engine.

or

5. State and explain four fundamental laws used in design and operation of gas turbine engines.
6. Derive the Euler's equation for turbine and pump with a neat sketch

or

7. Explain the problems of combustion in high speed flow.

8. What are the challenges faced by trans-atmospheric air-breathing engines.

Or

9. Explain briefly the operating principles of different types of electrical thrusters. With a neat diagram, explain the operation of a Hall Effect Thruster

10. What is the function feed system in a liquid propellant rocket system? Explain working of different feed systems with a neat diagram.

Or

11. Write short notes on the following:
    (a) Types of space tethers
    (b) MEMS technology

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AEROSPACE PROPULSION (R15)

MODEL PAPER – II

MAXIMUM MARKS: 75

PART A **Marks:25**

1. (a) State five differences between the compressor and turbine blade profiles
   (c) Explain the purpose of thrust augmentation in gas turbine engines.
   (d) Explain flammability limits of combustor with a neat graph
   (e) Explain the purpose of using dimensionless and corrected parameters in component performance maps.
   (f) Define and explain degree of reaction in case of an axial flow compressor.
   (g) Draw schematic diagram of a ramjet engine and mention the station numbering of the components
   (h) Explain the effect of ambient temperature on the take-off thrust of a jet aircraft.
   (i) Explain the need for a variable geometry air intake.
   (j) What are different types of combustion chamber geometries used in gas turbine engines?
   (k) What is purpose of cascade analysis of compressor stage

PART B **Marks: 50**

All questions carry equal marks

2. Explain different methods used for thrust augmentation in gas turbine Engines.

or

3. With a neat schematic diagram, explain the operation of a Ram jet engine.
4. Explain different types of supersonic air inlets with neat diagrams

or

5. State the differences between Centrifugal flow and Axial flow compressors.
6. Explain different methods of cooling used in axial flow turbine.

or

7. Explain the compressor and turbine operating curves both in design and off-design conditions
8. With a neat schematic diagram, explain the operation of scramjet engine.

or

9. What are the advantages of liquid propellant rockets over solid propellant rockets?
10. What are the advantages of electrical propulsion engines (thrusters) over chemical rocket engines?

Or

11. Explain the operation of a nuclear fission propulsion system and the problems associated with the design of such engine.

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II B.Tech II SEMESTER – AERONAUTICAL ENGINEERING

AEROSPACE PROPULSION (R15)

MODEL PAPER –III

MAXIMUM MARKS: 75

PART A Marks: 25

Answer all questions

1. (a) Draw schematic diagram of a turbojet engine and mention the station numbering of the components
   (b) Explain the effect of ambient temperature on the take-off thrust of a jet aircraft.
   (c) Explain the need for a variable geometry air intake.
   (d) What are different types of combustion chamber geometries used in gas turbine engines?
   e) What is purpose of cascade analysis of compressor stage?
   (f) Define and explain work done factor in case of an axial flow compressor
   (g) What are the functions of the nozzle in a gas turbine engine?
   (h) Explain the operating principle of nuclear fission process.
   (j) With a neat diagram, indicate different types of drag in a scramjet engine
   (k) What is TVC? Explain briefly.

PART B Marks: 50

2. Explain the operation of turbofan engine with a neat sketch, and discuss its advantages and disadvantages.

Or

3. A gas turbine rotates at 1000 rpm. At entry to the nozzle guide vanes, the total temperature and pressure are 700^{*} C and 4.5 bar. At the outlet to the nozzle guide vanes, the static pressure is 2.6 bar. At the turbine outlet, the static pressure is 1.5 bar. Mach number at the outlet is 0.5. Gas leaves the turbine in an axial flow direction. The outlet nozzle angle is 70°. Nozzle friction loss is 0.3%.

Calculate the gas angles at entry and outlet from the rotor and the output power developed by the turbine. Assume C_p as 1.147 KJ/kg K and γ as 1.33.

4. Draw the velocity triangles at the inlet and outlet of the rotor and derive an expression for the work done per stage.

Or

5. Draw compressor operating map and explain compressor operation through different off design conditions.
6. Explain the construction and operation of a combustor in a gas turbine engine with a neat diagram

Or

7. Write short notes on the following:
   (a) Thermal Throat
   (b) Function of Isolator in scramjet engine
8. Define specific impulse, total impulse, mass ratio, propellant mass fraction and the effective exhaust velocity of a rocket vehicle

Or

9. Write short notes on:
   (a) Application of Electric Propulsion
   (b) Break through propulsion
   (c) Ensuring Sustainable Chain Reaction in Nuclear Propulsion
10. Explain the process through which engine back pressure control is excercised by the nozzle.

Or

11. Explain the operation of Resisto-jet and Arc-jet engines with neat diagrams. What are limitations of electro-thermal thrusters?

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AEROSPACE PROPULSION (R15)

MODEL PAPER –IV

MAXIMUM MARKS: 75

PART A Marks: 25

Answer all questions

1. (a) Explain how turbofan with high bypass, is able to achieve better fuel economy compared to a turbojet generating same thrust levels.
   (b)Define and explain the relevance of propulsive efficiency of a gas turbine engine.
   ©Draw the flow patterns for a subsonic air intake at different operating conditions.
   (d) Explain the need for variable area nozzle in a turbojet engine.
   (e) Explain the through-flow field analysis of an axial flow compressor.

(f) Explain the purpose of primary, secondary and mixing air in the main burner of a gas turbine engine combustor
(g) Explain the operating principle of Electro-magnetic propulsion.
(h) What is a thermal throat in a scram jet engine
(i) Explain the concept of propellant-less propulsion.
(j) What is the need for distributed fuel injection in a scramjet engine?

PART B Marks: 50

All questions carry equal marks

2. Explain the effect of altitude and forward speed on the performance of a jet engine.

Or

3. With a neat schematic diagram, explain the function of different components of a turbo-jet engine.
4. Explain the use of velocity triangles in analysing the stage pressure rise in an axial flow compressor.

or

5. Explain the relevance of turbine inlet temperature for the gas turbine operation. Discuss various turbine blade cooling methods used with neat sketches.
6. In a gas turbine engine working on Brayton cycle with a regenerator effectiveness of 75%, the air at the inlet to the compressor is at 0.1 Mpa and 30° C. The pressure ratio of the compressor is 6 and the maximum cycle temperature is 900° C. If the turbine and compressor have an efficiency of 80%, find the percentage increase in the cycle efficiency due to regeneration.

or

7. Explain the differences between axial flow compressors and turbines.
8. Derive the expression Euler's equation for pump and turbine.

or

9. With a neat schematic diagram, explain the operation of scramjet engine.
10. What are the advantages of liquid propellant rockets over solid propellant rockets?

or

11. What are the advantages of electrical propulsion engines (thrusters) over chemical rocket engines?

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AEROSPACE PROPULSION (R15)

MODEL PAPER –V

MAXIMUM MARKS: 75

PART A Marks: 25

1. (a) Write the equation of thrust of a turbojet engine and explain each term in it.

   (b) Explain the term flat rating of thrust in a turbojet engine

   (c) Write the Euler's equation for a centrifugal compressor and explain different terms in the equation

   (d) Define work done factor in case of an axial flow compressor

   e) Explain the applications of a ramjet engine

   (f) Define the diffusion factor for an axial flow compressor.

   (g) What are the applications of Scramjet aircraft?

   (h) Explain the operation of a Pitot type inlet.

   (j) Briefly explain different types of Electric thrusters.

   (k) Explain the potential applications of nuclear propulsion.

PART B Marks: 50

All questions carry equal marks

2. Explain the classification of air breathing propulsion systems.

Or

3. Explain the various types of drag associated with inlets? State different types of air intakes used in gas turbine engines and their applications.

4. The pressure and temperature at the entry to convergent divergent nozzle are 5 kgf/cm^2 and 550° C respectively. The pressure at the exit of the nozzle is 1.4 kgf/cm^2. The efficiency of the nozzle is 90%. Assume ⍰ as 1.4 and R as 29.27 kgf/kg K. Find the area of the nozzle at throat and exit per unit mass flow rate.

Or

5. Explain the different nozzle coefficients that indicate the performance of the exhaust nozzle.

6. Draw a neat diagram of a combustor used in a gas turbine engine and explain the function of different components.

or

7. Explain the limitations of chemical rocket engines

8. What do you understand by multiphase flow in the nozzle? How does it affect nozzle performance?

Or

9. Explain the concept and use of Dual-mode engines. Explain the operation of the dual-mode Ram/Scram jet engine.
10. With neat diagrams of the shock pattern in the isolator.

Or

11. Write short notes of the following:
    - Calculating criticality of a nuclear fission reactor
    - Operating principle of LACE
    - Use of Reflector in nuclear fission rocket

Unit I-Fundamentals of Propulsion

UNIT I

FUNDAMENTALS OF PROPULSION: Evolution of flight propulsion, types of aerospace propulsion, working principles, advantages, disadvantages, applications – reciprocating engines, propellers, jet engine, turboprop, turbofan, turbo-shaft, ramjet, scramjet, pulsejet. Engine components-performance requirements, thermodynamic processes- change of state- representation by T-sand p-v diagrams - pressure ratios, temperature ratios. Energy transfer, losses- entropy generation-mechanisms. Performance- polytropic, stage and component efficiencies, burning efficiency
Station numbering in engine, thrust generation, momentum equations, equation of thrust for installed and uninstalled cases, factors affecting thrust, Role of propulsion in aircraft performance.

1.1 Evolution of Flight Propulsion:

Classes of Aircraft:

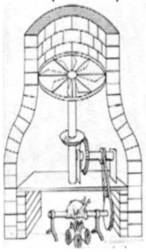
- Lighter than air category-Airships; Free balloons; Captive balloons
- Heavier than air category-Power driven; non-power driven
    - Power driven category-Aeroplane; Rotorcraft; ornithopters
    - Aeroplanes-Landplanes; Seaplanes & Amphibians

History of flight Propulsion:
Earliest known propulsive device: Hero's Aeoipile in Year 250 B.C

The Aeolipile is a steam reaction turbine, invented by Egyptian inventor, Hero of Alexandria, in the year 250 BC. The Aeolipile is a steam reaction turbine.
Hero mounted a sphere on top of a water kettle. A fire below the kettle turned the water into steam, and the gas traveled through the pipes to the sphere. Two L-shaped tubes on opposite sides of the sphere allowed the gas to escape. This produced a thrust to the sphere that caused it to rotate almost silently .
The aeolipile achieved spin speeds of at 1500 RPM.
Chinese used rockets with gunpowder, around AD 1000. They attached these rocket (bamboo) tubes to arrows and launched them with bows. Soon they discovered that these gunpowder tubes could launch themselves just by the power produced from the escaping gas. The true rocket was born.
Gunpowder changed the methods of war forever.

Da Vinci visualized flight vehicles as early as 1500 AD

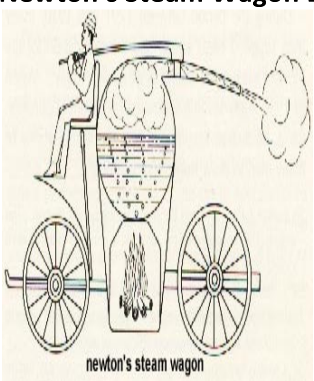
Da Vinci's Chimney Jack (1500 AD)



Da Vinci' Ornithopter

In 1629 an Italian engineer, Giovanni Branca, was probably the first to invent an actual **impulse turbine**. This device, a stamping mill, was generated by a steam-powered turbine.

Newton's Steam Wagon 1687:



In 1687, Jacob Govesand, a Dutchman designed and built a carriage driven by steam power. Sir Isaac Newton was believed to have supplied the idea in an attempt to put his laws of motion to test.

The first Gas Turbine: In 1791 John Barber, an Englishman, was the first to patent a design that used the thermodynamic cycle of the modem gas turbine.

Wright Brothers first Airplane "Triumph": First Flight 1903 Dec

Concept of Jet Propulsion:

Newton's Laws of Motion — Glenn Research Center

"Every object persists in its state of rest or uniform motion in a straight line unless it is compelled to change that state by forces impressed on it."

"Force is equal to the change in momentum (mV) per change in time. For a constant mass, force equals mass times acceleration."

$$F = m\ a$$

"For every action, there is an equal and opposite re-action."

Newton's Laws of Motion

Newton's first law.

An object at rest will remain at rest unless acted on by an external force. An object in motion continues in motion with the same speed and in the same direction unless acted upon by an external force.
This law is often called "the law of inertia" as it establishes the Newtonian frame of reference.

Newton's law I

This law states that if the vector sum of all the forces acting on an object is zero, then the velocity of the object is constant.
Consequently:
• An object that is at rest will stay at rest unless an unbalancing force acts upon it.
• An object that is in motion will not change its velocity (magnitude and/or direction) unless an unbalancing force acts upon it.

Newton's Laws of Motion

Newton's second law

Acceleration is produced when a force acts on a mass. The greater the mass (of the object) being accelerated the greater the amount of force needed to accelerate the object.

$$F = MA$$

From Newton's 2nd law of motion

The second law states that the net force on a body is equal to the time rate of change of its linear momentum **M**t in a specified reference frame for the inertial motion under interest:

$$F=\frac{dM_t}{dt}=\frac{d(mV)}{dt}=m\frac{dv}{dt}$$

For a constant mass system

Any mass that is gained or lost by the system will cause a change in momentum that is not the result of an external force. A different equation is necessary for a variable-mass systems

Newton's Laws of Motion

Newton's third law

For every action there is an equal and opposite re-action.

While the Newton's 3rd law allows us to comprehend the mechanics of action of the propulsive force (Thrust) acting on a flying body, the production of thrust is actually facilitated by the Newton's 2nd law, active on the engine body. Hence it is not only the jet coming out at the exhaust that creates thrust, but the entire body of the engine participates in creation of thrust.

History of Internal Combustion (I.C) Engines:

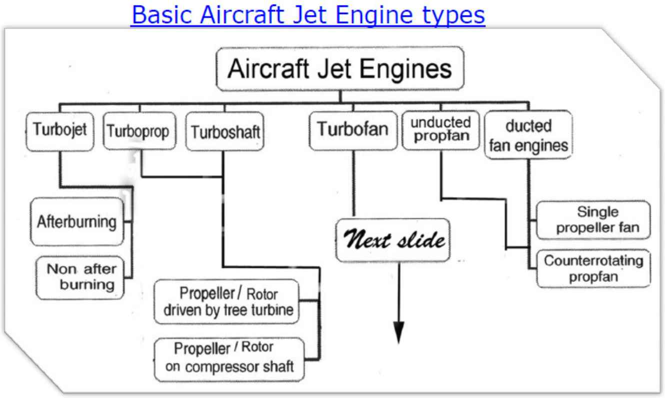
The first 4 stroke engine was built by the Germans, August Otto and Evgen Langer in 1876. As a result, the 4 stroke engine cycle are always called Otto Cycle engines.
George Brayton of the USA, also built a gasoline engine in 1876. Gottieb Daimler has built most successful 4 stroke engine in 1885. The first 4 stroke engine was built by the Germans, August Otto and Evgen Langer in 1876. As a result, the 4 stroke engine cycle are always called Otto Cycle engines.
Same year, Karl Benz, has built a similar engine. These two engines were extensively used in automobiles.
Wright brothers used 4 stroke four cylinder IC Engine in 1903.

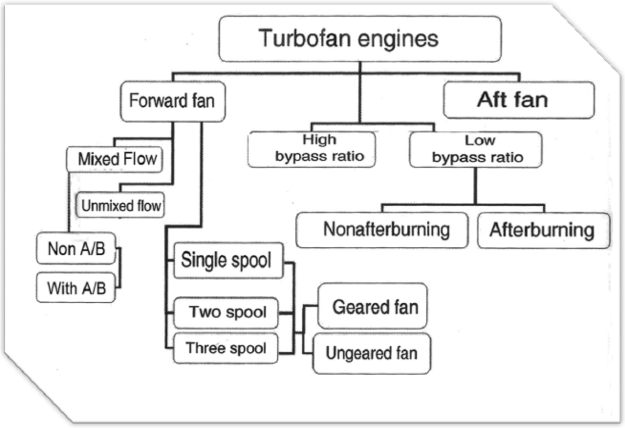
1.2.1: Types of Aerospace Propulsion:
Air Breathing Systems: Broadly grouped as - Reciprocating& Jet Propulsion Engines.

- Reciprocating Engines
- Gas Turbine Engines
- Ram Jets, Pulse Jets & Scram Jets

Non Air Breathing Systems
- Rockets

Basic Aircraft Jet Engine types





1.2.2. Working Principles, Advantages/Disadvantages & Applications:

1.2.2.1: Reciprocating Engines (I.C Engine): Working Principle: The four strokes of anInternal Combustion (I.C)engine are Intake, Compression, Power and Exhaust strokes.

During intake stroke, the piston moves downwards and the mixture of fuel and air (charge) is admitted in to the cylinder. At the completion of intake stroke, the inlet valve closes.

During the compression stroke, the piston moves up, compressing the charge. At the end of compression stroke, the electric spark ignites the charge.

On ignition, combustion of air fuel mixture releases thermal energy, exerting high force on the piston. This commences the power stroke.

During the power stroke, the piston is driven downwards.

Once the power stroke is completed, the exhaust valve opens. While the piston is moving up, the combustion gases are driven out of the cylinder through the exhaust valve. This creates a suction in the cylinder, that initiates the next cycle of operations.

The reciprocating movement of piston is transmitted to the crankshaft and converted into rotary motion. The crankshaft is connected to the propeller, which produces the forward thrust force for the aircraft.

The rotating output shaft of the I.C engine can be connected to a propeller, ducted fan, or helicopter rotor.

The propeller displaces a large mass of air rearwards, accelerating it in the process.

Reciprocating engines can produce up to 4000 KW power. Power to weight ratio (P/W) of up to 1.4 is produced.

The power produced by an I.C engine is given by

P = $\frac{KNV_c\rho_{air}fQ_f\eta_o}{60}$ where

K = constant; either 1.0 for 2 stroke engine or 0.5 for 4 stroke engine

N = rpm (around 5000-9000 rpm)

V_c = Volume of the cylinder

ρ_{air} = density of air

f = fuel air ratio (usually 13 to 15 ie one part fuel to 15 parts of air to burn the fuel completely)

Q_f = Calorific value of fuel (kerosene- 42 MJ/kg)

η_o = overall efficiency (usually 0.25 to 0.35)

$KNV_c\rho_{air}$ is the mass flow rate ingested in to the engine

- Multiplying mass flow rate with f gives the amount of fuel
- Multiplying with Q_f gives the heat energy released

To increase the power of the I.C engine, we need to

- Increase N –increases P
- As altitude increases ρ decreases, and P reduces. To offset this, turbo superchargers are used.

Advantages of Reciprocating Engines:

- Reciprocating engines provide excellent fuel economy and good take-off characteristics within their range of operations
- Highly suitable for small aircraft flying up to 500 km/hr and operating at low altitudes
- Components of reciprocating engines are subjected less thermal stresses than gas turbine-propeller combination
- Aircraft fitted with reciprocating engines need short runways
- Mainly used for business travel, farming & agriculture, air-taxi/ambulance, pilot training and unmanned aerial vehicles

Disadvantages of Reciprocating Engines:

- Reciprocating engines suffer drop in power at altitudes
- Difficulty in cooling and lubrication
- Low Power/Weight ratios compared to gas turbine engines
- Need high octane fuels to improve power output
- Increase in power output require larger number of cylinders, thereby increasing the frontal area and weight
- Use of reciprocating engines is limited to low speeds and altitudes
- Development reached a saturation stage as far as maximum power is concerned
- Maintenance requirement of piston-prop engines is more than turbojet aircraft
- Exhaust gases have less impurities in turbojet engines

1.2.2.2: Aircraft gas turbine Engines

All modern aircraft are fitted with gas turbine engines. Gas turbine engines can be classified into the following:-

(a) Turbojet engines

(b) Turbofan engines

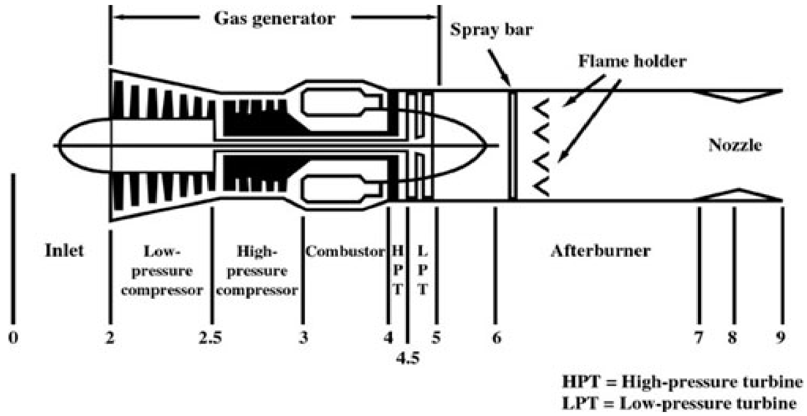
(c) Turbo-shaft engines

(d) Turboprop engines

Taken in the above order they provide propulsive jets of increasing mass flow and decreasing jet velocity. Therefore, in that order, it will be seen that the turbojet engines can be used for highest cruising speed whereas the turboprop engine will be useful for the lower cruising speed at low altitudes.

In practice the choice of power plant will depend on the required cruising speed, desired range of the aircraft and maximum rate of climb.

Turbojet Engine: Schematic diagram of a turbojet engine with station numbering is given below:



Working Principle:

1. The thrust of a turbojet engine is developed by compressing the free stream air in the diffuser or inlet and compressor. The diffuser converts the kinetic energy of the entering air into pressure rise which is achieved by ram effect. Diffusion in the inlet occurs due the geometric shaping of the inlet.
2. The compressor is driven by the turbine. It rotates at high speed, adding energy to the airflow and at the same time squeezing (compressing) it into a smaller space. Compressing the air increases its pressure and temperature
3. Compressor types used in turbojets were typically axial or centrifugal.
4. Use of axial flow compressors enable high pressure ratios. Modern axial compressors are split into low pressure and high pressure spools, driven by corresponding two stage turbine. High compressor ratios of 15:1 or more can be achieved while improving stability of operation at off-design conditions.The high pressure air is then mixed with fuel and burnt in the combustion chamber under constant pressure condition.
5. The combustion gasses at high temperature and pressure are expanded in the turbine and the exhaust nozzle. The expansion of gasses in the turbine provides power to drive the compressor while the exhaust nozzle expands the gasses to atmospheric pressure, thereby producing propulsive force, thrust.
6. The net thrust delivered by the engine is the result of converting internal energy to kinetic energy.
7. The exhaust products downstream of the turbine still contain adequate amount of oxygen. Additional thrust augmentation can be achieved by providing an afterburner in the jet pipe in which additional amounts of fuel can be burnt.
8. Turbojet engines are most suitable for speeds above 800 km/hr and up to 3.0 mach number

Advantages of Turbojet:

1. Power to Weight ratio is about 4 times that of Piston-Prop combination

2. Simple, easy to maintain, requires lower lubricating oil consumption. Complete absence of liquid cooling reduces frontal area
3. Turbojets allow faster supersonic speeds up to 3.0 M
4. There is no limit to power output while piston engines reached their peak power, beyond which any increase will result in high complexity and greater weight/frontal area.
5. Speed of turbojet is not limited by the propeller.
6. Turbojets can attain higher speeds than turboprop aircraft

Disadvantages-Turbojet:

- Fuel economy at low operational speeds is very poor
- It has low take-off thrust and hence poor starting characteristics
- High operating temperatures and engine parts are subjected to thermal stresses

Application:Turbojet engine is highly suited for aircraft at speeds above 800 km/hr.
Advantages of Gas turbines over Reciprocating Engines:

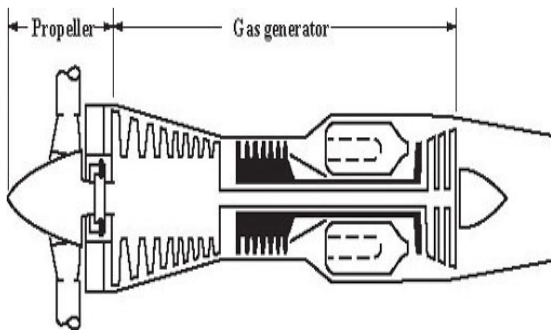
- **Mechanical Efficiency**: Mechanical efficiency of gas turbine engines is higher than reciprocating engines. This is mainly due to high friction losses in reciprocating engines.
- **Balancing**: Due to absence of reciprocating mass in gas turbine engines, balancing can be near perfect. Torsional vibrations are absent because gas turbine is a flow machine.
- **Smooth & Vibration-free operation**: Turboprop engines have fewer moving parts than piston-prop engines, offering greater reliability and time-between-overhaul (TBO).
- **Power**: The higher power of a turbo-prop engine allows it to fly at higher speeds and altitudes.
- **Shape**: Gas turbine engines have streamlined shape suitable from aerodynamic point of view.
- **Fuel**: Aviation turbine fuel is much cheaper than the high octane fuels used by reciprocating engines.
- **Lower Cost**: For a given power, gas turbine engine has lower cost and can be built faster
- **Weight:** Gas turbine engineshave higher power-to-weight ratios. This means, for a given weight, gas turbine engines develop more power.
- **Lubrication:** Lubrication in gas turbine engines is much simpler than reciprocating engines. The requirement is chiefly to lubricate the main bearing, compressor shaft and bearing auxiliaries.
- **High operational speed:** Turbine canbe run at much higher speed than reciprocating engine. Turbine can also be made lighter than the reciprocating engine of similar output. Therefore, for a given output, and higher speed, the torque can be lower. Gas turbine engines have better torque characteristics.

- **Silent Operation:** Since exhaust from gas turbine engines occurs under practically constant pressure conditions unlike the pulsating nature of the reciprocating engine exhaust, the usual vibrational noises will be absent in gas turbine engines.
- **Maintenance:** Relatively simpler in case of gas turbine engines.

Advantages of Reciprocating Engines over Gas turbine Engines:

- **Efficiency**: The overall efficiency of gas turbine engines is much less than the reciprocating engines.
- **Temperature Limitation**: The turbine blades in gas turbine engine are exposed to high temperature gasses continuously, and hence cannot exceed 1500 K.
- **Cooling**: We can achieve very good results by cooling the cylinder walls effectively. Cooling of turbine blades is complicated.
- **Ease of Starting**: It is more difficult to start a gas turbine than a a reciprocating engine.
- **Complexity**: Reciprocating engines are far less complex than their turbo-prop counter parts, from engineering considerations. This is primarily because of the high temperatures and forces unique to turbo-prop engine operation, which must be accommodated from materials and engine design.

1.2.2.3: Turboprop Engine: Schematic diagram is given below:



Working Principle: Turboprop engine is an intermediate between a pure jet engine and a propeller engine.

Turboprop engine provides high thrust per unit mass flow of fuel burnt by increasing mass flow of air. It offers better fuel economy. The propeller displaces a large mass of air rearwards, thereby increasing the net thrust.

The turbine extracts more power from the combustion gasses to drive the propeller. A small remaining energy is extracted by expansion in the jet nozzle.

The propeller and the compressor may be mounted on a single shaft or on separate shafts with a free turbine driving the propeller.

Advantages:

Turboprop engines have a higher thrust at take-off and better fuel economy.

The engine can operate economically over a wide range of speeds ranging from low speeds, where turbojet is uneconomical, to high speeds of about 800 km/hr where piston-prop engine cannot operate efficiently

It is easy to maintain and has lower vibration levels than piston-prop engine. The frontal area is much less than corresponding piston-prop engine.

Disadvantages:

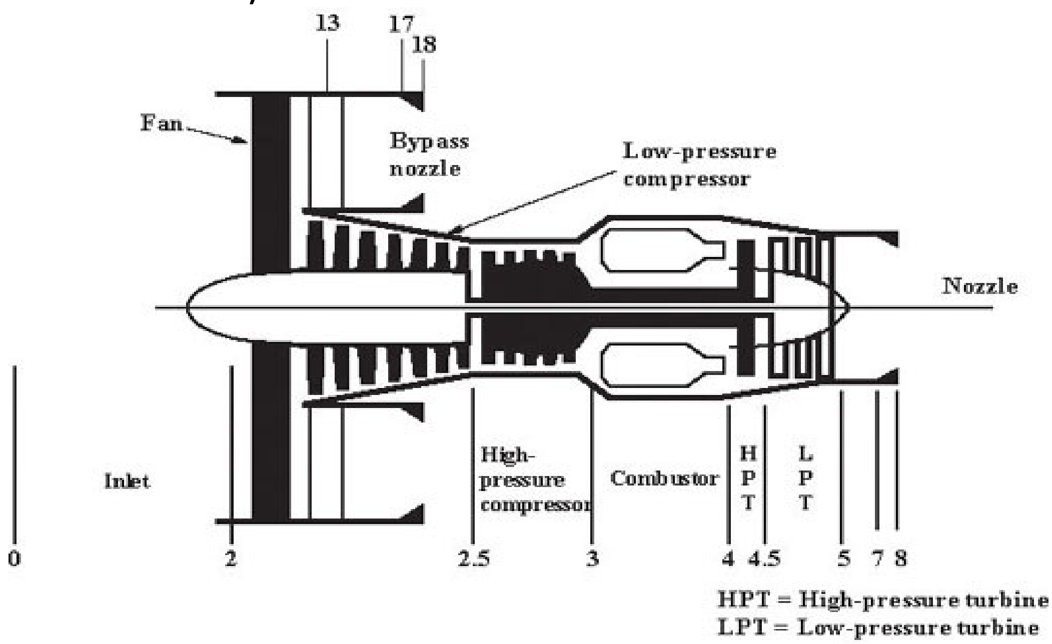
The main disadvantage is that the propeller efficiency decreases greatly at high speeds due flow separation and shocks. The maximum speed is thus limited.
The turbine speeds need to be reduced through a suitable reduction gearing so that propeller runs at lower speeds, which adds to weight.

Applications:

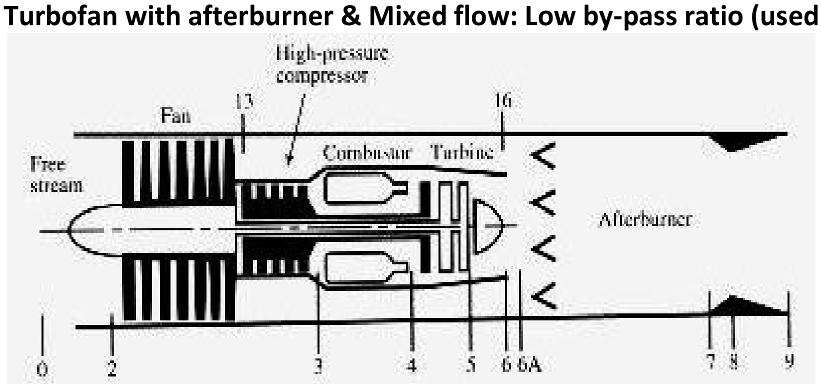
The turboprop engine is widely used in commercial and military aircraft due to its flexibility of operation and good fuel economy.

Turbofan Engine

Schematic Diagram of Turbofan (with station numbering): High by-pass ratio (used for commercial aircraft)



Turbofan with afterburner & Mixed flow: Low by-pass ratio (used for military aircraft)



Turbofan engine is designed as a compromise betweenturbojet and turbopropengines. The turbofan engine consists of a fan larger in diameter than the compressor, driven by the turbine. The fan displaces/bypasses free stream air around the primary engine. Two streams of air flow through the engine, primary airstream pass through the compressor and is delivered to the combustion chamber at high pressure to mix with fuel, while the other stream bypasses the primary engine to be expanded in the nozzle as a cold stream. The **hot and cold streams may be expanded through separate nozzles or combined together through a single nozzle**. The ratio of mass of cold air to the hot air is the by-pass ratio.

Thus the turbofan accelerates a larger mass of air at lower velocity than turbojet for a higher propulsive efficiency. Turbofan engines can also employ afterburner for higher thrust.

Turbofan engines can be aft-fan or forward fan (position of the fan), mixed or unmixed(hot and cold air streams) and high and low bypass ratio configuration

Advantages:

Fan is not as large as the propeller, therefore higher aircraft speeds can be attained without facing flow separation problems.

Turbofan engines do not encounter vibration problems associated with propellers. The fan could be encased in a duct/cowling to provide better aerodynamic shape.

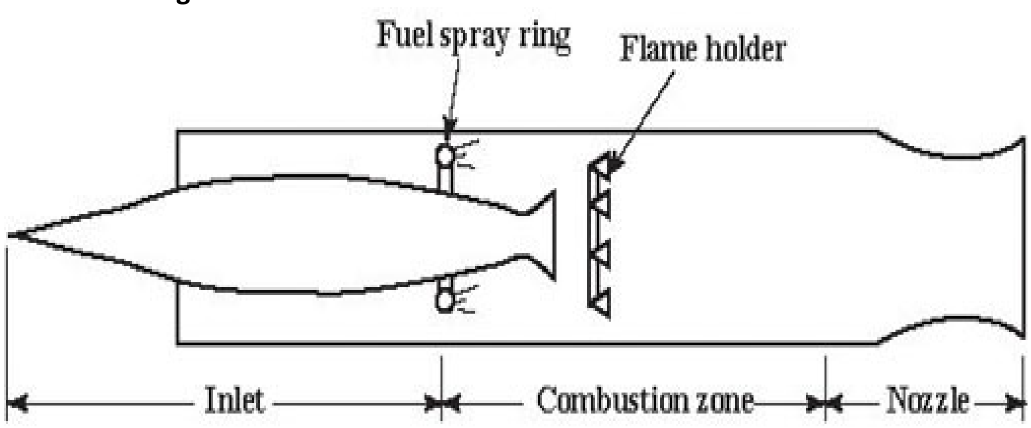
A **geared fan** connected to turbine reduces power consumed by the fan. It also produces low sound.

Turbofan is fuel efficient than turbojet, offers better propulsive efficiencies.

Lowers the sound levels of the exhaust gasses

Ramjet Engine

Schematic Diagram:



Operating Principle:

Ramjet Engine consists of supersonic diffuser, subsonic diffuser, combustion chamber and nozzle section

Air from atmosphere enters the supersonic diffuser at a very high speed. The air velocity gets reduced in the supersonic diffuser through normal and oblique shock waves.

Air velocity is furthur reduced in the subsonic diffuser.

The diffuser converts the kinetic energy of the entering air into static pressure and temperature rise which is achieved by ram effect. Diffusion in the inlet also occurs due the geometric shaping of the diffuser. The diffuser thus slows down the air enabling combustion.

Fuel is injected into the combustor through suitable injectors causing mixing of fuel with the air and the mixture is burnt

Combustion gases attain a temperature of around 1500-2000 k by continuous combustion of fuel air mixture

Fresh air supply continuously will not allow gasses towards the diffuser. Instead, gases are made to expand towards the tail pipe and nozzle, which expands the gases completely.

The gases leave the engine with a speed much higher than the air entering the engine.

The rate of increase of momentum of the working fluid produces thrust F in the direction of flight

Distinguishing Features:

Air enters the engine at supersonic speeds, must be slowed down to subsonic value, to prevent blow out of the flame in the combustor
Velocity must be low enough (approximately around 0.2-0.4 mach number) to allow mixing of fuel and stable combustion
Cycle pressure ratio depends on the diffusion pressure ratio. Very high pressure ratios of about 8 to 10 through ram compression is possible, therefore, a mechanical compressor is not required
Slowing down speeds from mach 3.0 to 0.3 will result in a pressure ratio of more than 30
As the ram pressure increases, a condition is reached where the nozzle gets choked.
Thereafter, the nozzle operates at Mach 1 condition at throat
Advantages:
Ramjet is a simple machine and does not have any moving parts
Since turbine is not used, maximum temperature allowed is very high, around 2000 C, as compared to around 900 C in turbojets.
We can burn air/fuel ratios of 13:1which gives greater thrust
Specific fuel consumption is much better than other gas turbine engines, at high speeds and altitudes
Wide range of fuels can be used
It is very cheap to produce; adoptable for mass production
It is not possible to start a ramjet engine without an external launching device
The engine heavily relies on the diffuser and it is very difficult to design a diffuser which gives good pressure recovery over a wide range of speeds
Due to high air speed, the combustion chamber requires flame holders to stabilize the combustion
At very high temperatures of about 2000 C, dissociation of combustion products take place, reducing the efficiency of the plant
High fuel consumption at low speeds
Applications:
Highly suitable for propelling missiles.
Used in high speed military aircraft, in a combined cycle engine (Turbojet-Ramjet combination).
Development is in progress for a hypersonic aircraft system using turbojet-ram-scramjet combined cycle.
Subsonic ramjets are used as target weapons in conjunction with turbojet aircraft.
Pulsejet Engine:
Schematic Diagram:

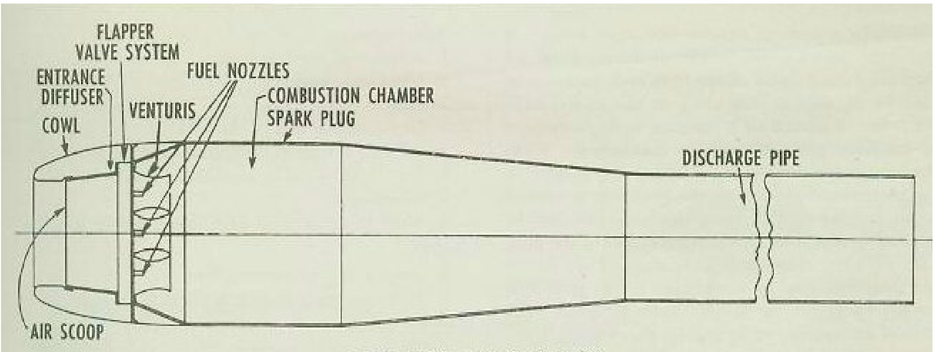


FIGURE 11D3.—A pulse-jet engine.

Basic Components are diffuser, Valve grid with spring loaded flapper valves, Combustion chamber with spark plug, tail pipe and discharge nozzle

Operation:

The diffuser converts the kinetic energy of the entering air into static pressure rise and slows down the air. Ram action also builds pressure in the diffuser.

The pressure differential opens the flapper valves which are spring loaded and the high pressure air enters the combustion chamber.

Fuel is injected and ignited by the spark plug

Combustion proceeds at constant volume with sudden explosion.

There is a sudden pressure rises in the combustion chamber which closes the flapper valves

The combustion gasses expand in the nozzle and escape to the atmosphere at high velocity

As combustion products leave the combustion chamber, a low pressure is created which causes the valves to open and a new charge of air enters the chamber

Distinguishing feature: Since the combustion chamber builds pressure, the engine can operate in static condition also. Proper design makes the duct to fire at a given pulse rate which can be as high as 500 cycles/sec

Advantages:

1. Simple to construct and hence cheap.
2. Can be mass produced in a short time.
3. Since it does not have any moving parts like compressor of turbine, it can be used in high temperatures.
4. Can be used for military applications.

Disadvantages:

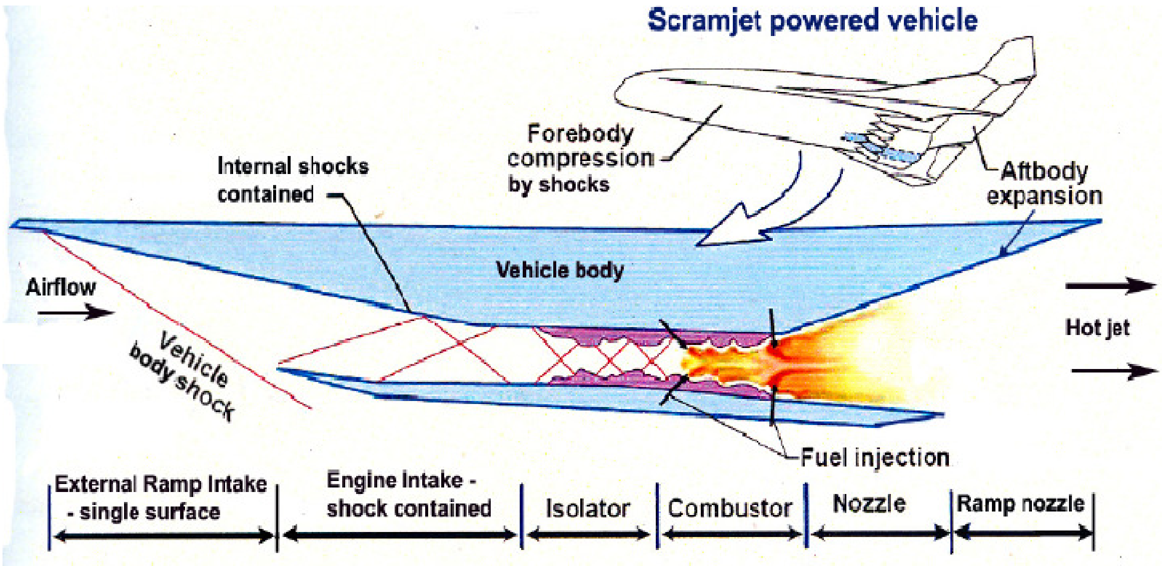
1. It is having limited flight speed only.
2. Limited flying altitude.
3. High vibration and noise due to the pulses of thrust produced

Scramjet Engine:

- Scramjet engine stands for supersonic combustion ramjet engine.
- The flow speed in the combustion chamber is supersonic
- Scramjet engine is characterized by high flow speeds ie low residence times in the engine.
- The engine needs larger combustion volumes; leading to integrated design of airframe and engine.
- In scramjet aircraft, the entire lower body of the aircraft is occupied by the engine. The front portion of the underside operates as external/internal diffuser, with rear portion providing expansion surface.

The scramjet consists of

- Diffuser (compression component) consisting of external ramp intake and engine intake
- Isolator
- Supersonic combustor
- Exhaust nozzle or aft body expansion component



Scramjet Engine- Construction:Scramjet engine is characterized by slow reaction times and high flow speeds ie low residence times in the engine. The engine needs larger combustion volumes; leading to integrated design of airframe and engine. In scramjet aircraft, the entire lower body of the aircraft is occupied by the engine. The front (fore) portion of the underside operates as external/internal diffuser, with rear (aft) portion providing expansion surface.

The scramjet consists of

- Diffuser (compression component) consisting of external ramp intake and engine intake
- Isolator
- Supersonic combustor
- Exhaust nozzle or aft body expansion component

Diffuser

- It consists of fore-body external intake and internal intake
- The fore-body provides the initial external compression and contributes to the drag and moments of the vehicle.
- The internal inlet compression provides the final compression of the propulsion cycle.

Since the flow upstream is supersonic, the geometry of the diffuser is entirely convergent.

Isolator: Isolator is constant area diffuser containing the internal shock structure, swallowed during supercritical operation of the inlet (or during operation after the inlet "started"). The isolatoris inserted before thecombustor to diffuse the flow further, through a shock train, producing desired flow speeds in the combustors. The function of the isolator is:

- The shock train provides a mechanism for the supersonic flow to adjust to a static back pressure higher than its inlet static pressure
- The isolator cross-sectional area may be constant or slightly divergent to accommodate boundary layer separation.
- When the combustion process begins to separate the boundary layer in the combustor, a pre-combustion shock train forms.
- The shock structure allows the required pressure rise, thus isolating the combustion process from the inlet compression process. Thus the isolator functions to prevent inlet surge or "unstart".

Combustor: Main features include:

- Avoidance of hot pockets near the walls implies that the fuel be injected from centrally located struts.
- The usual circular configuration for combustors can be sacrificed in favor of a rectangular configuration.
- Typical velocities in the combustion chamber are about 1 to 1.5 km/s and the Mach numbers will be 1.4 to 2.3 for a typical combustor entry Mach number of 2.5

Combustion limits: Two limits are very critical for the operation

- First, since when a supersonic flow is compressed, it slows down, the level of compression must be low enough (or the initial speed high enough) not to slow the gas below Mach 1. If the gas within a scramjet goes below Mach 1 the engine will "choke", transitioning to subsonic flow in the combustion chamber. Additionally, the sudden increase in pressure and temperature in the engine can lead to an acceleration of the combustion, leading to the combustion chamber exploding.
- Second, the heating of the gas by combustion causes the speed of sound in the gas to increase (through increase of $\sqrt{t}$and hence cause Mach number to decrease) even though the gas is still travelling at the same speed. Forcing the speed of air flow in the combustion chamber under Mach 1 in this way is called "**thermal choking**".
- A thermal throat results when the flow is slowed through tailored heat for causing dual-mode operation.
- There are engine designs where a ramjet transforms into a scramjet over the Mach 3-6 range, known as dual-mode scramjets.

Expansion System:

- The expansion system, consists of
  - a. Internal nozzle
  - b. Vehicle aft body
- It completes the propulsion flow path and controls the expansion of the high pressure and temperature gas mixture to produce net thrust.

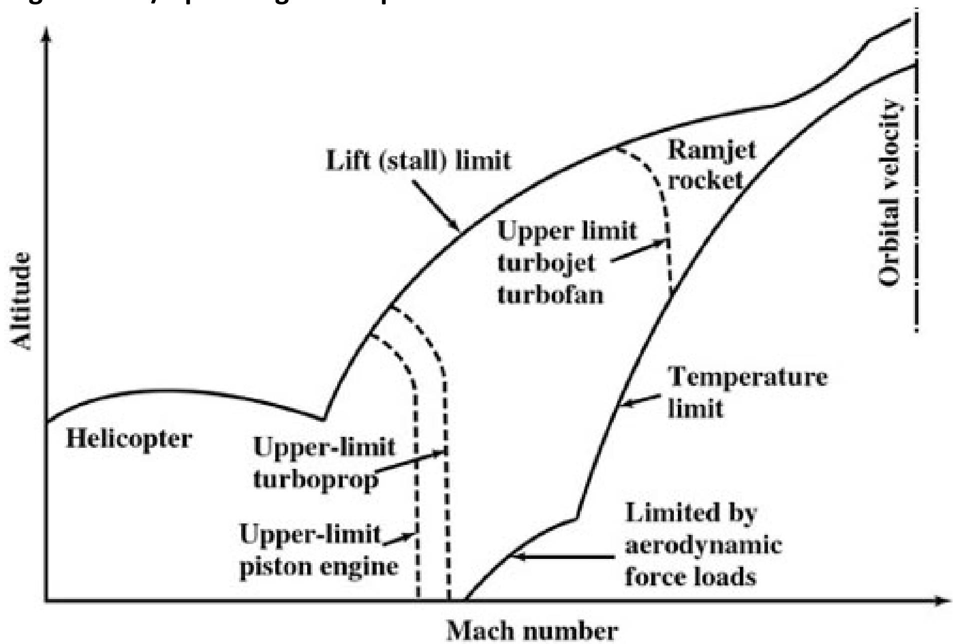
Applications of Scramjets:

- Weapons systems -hypersonic cruise missiles
- Aircraft systems - global strike / reconnaissance
- Space access systems that will take off and land horizontally like commercial Airplanes
- Using these Scramjet technologies, along with additional ground-and flight-test experiments, will pave the way for affordable and reusable air-breathing hypersonic propulsion systems such as missiles, long range aircraft and space-access vehicles

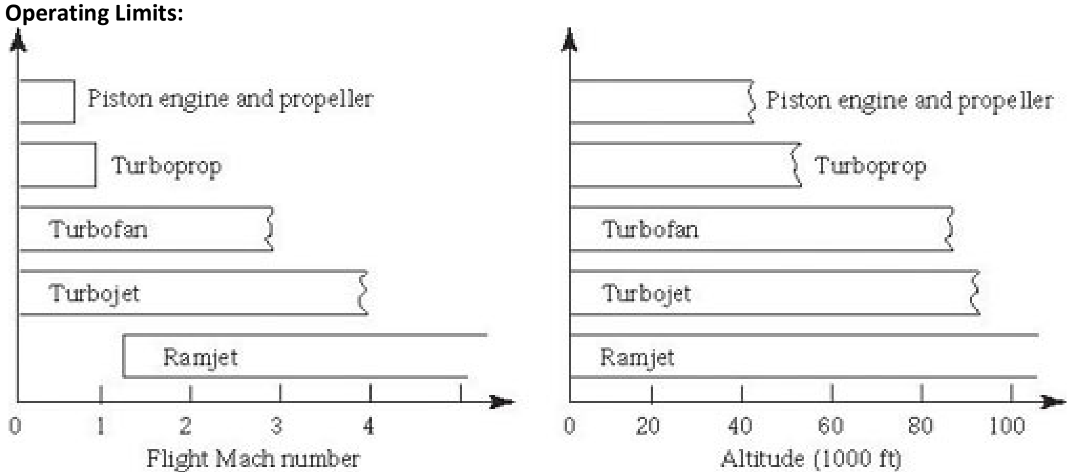
Advantages:

1. Need not carry oxygen on board
2. No rotating parts makes it easier to manufacture than a turbojet
3. Has a higher specific impulse (change in momentum per unit of propellant) than a rocket engine; could provide between 1000 and 4000 seconds, while a rocket only provides 450 seconds or less
4. Higher speed could mean cheaper access to outer space in the future

Flight Limits/Operating Envelope:

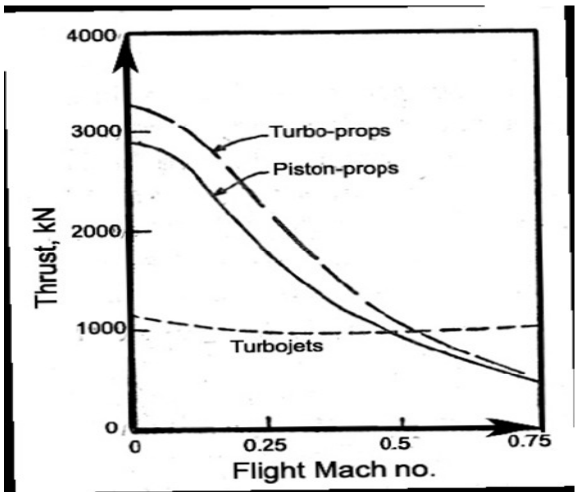


Operating Limits:

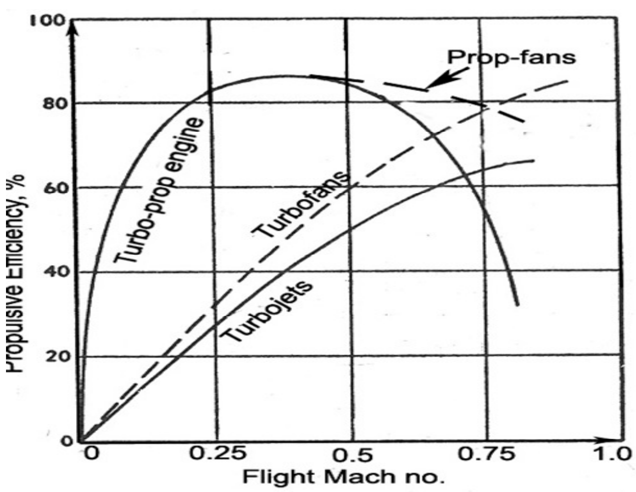


Performance Characteristics:

Thrust generation at Low Speeds:



Propulsive Efficiency at Low Speeds



Engine Components-Function:
Brief function of engine components:
Function of Components:

1. **Diffuser (or air inlet)**: The thrust of a turbojet engine is developed by compressing the free stream air in the diffuser (or air inlet) and compressor. The diffuser converts the kinetic energy of the entering air into pressure rise.

   Diffuser provides the air required by the engine from free stream conditions to the conditions required by the compressor entrance with minimum pressure loss. It reduces/supplies air to the compressor at a low velocity of around 0.4 Mach.

   Diffusion (conversion of velocity of air in to pressure) in the inlet occurs due the geometric shaping of the inlet. Design and geometric shaping of the diffuser (or air inlet) depends on whether air entering the diffuser is subsonic or supersonic.

   Performance Requirements: The air velocity is reduced through a diffusion process which increases the air pressure. The inlet must supply mass flow of air to the compressor at uniform speeds at all off-design conditions. The operation and design of an inlet depend on whether the air entering the inlet is subsonic or supersonic. As the aircraft approaches the speed of sound, the air at the entry to the inlet tends to be compressed more and at mach 1, shock waves will occur. Shock waves are compression waves, with high pressure loss across the shock wave. At higher mach numbers, the shock waves get stronger.

Thermodynamic Processes:

Inlet losses arise because of the presence of wall friction and shock waves (in a supersonic inlet). Both wall friction and shock losses result in a reduction in total pressure so that $\pi_d < 1$. Inlets are adiabatic to a very high degree of approximation, and so we have $\tau_d = 1$. The inlet's figure of merit is defined simply as π_d.

The *isentropic efficiency* η_d of the diffuser is defined as (refer to Fig. 6.3)

$$\eta_d = \frac{h_{t2s} - h_0}{h_{t0} - h_0} \cong \frac{T_{t2s} - T_0}{T_{t0} - T_0} \tag{6.3}$$

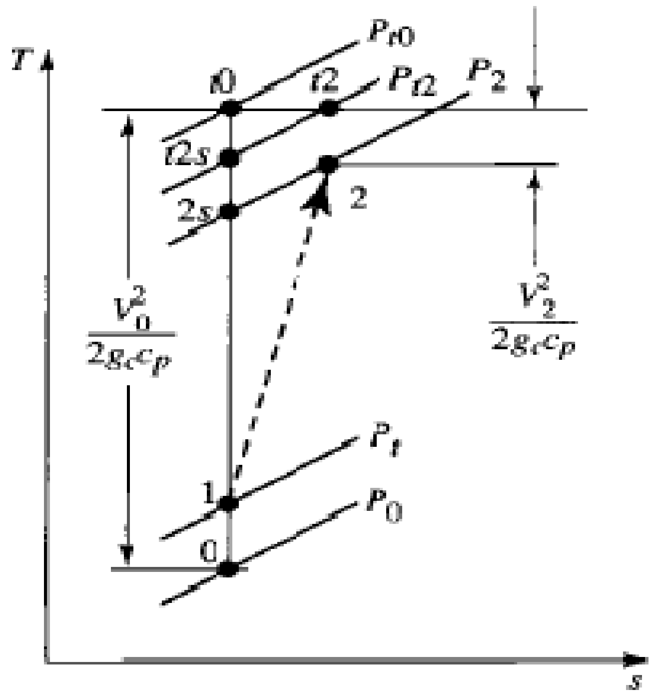


Fig. 6.3 Definition of inlet states.

“s” denotes isentropic condition. “t” denotes total or stagnation condition. 0 denotes free stream condition

The static pressure rises from P_o to P_2. Since, it is the stagnation pressure at compressor inlet (P_{t2}) which is required for cycle calculations, we obtain (P_{t2}) by adding ${V_2^2}/{2C_p}$ to P_2.

The pressure rise ($P_{t2} - P_1$) is called ram pressure rise. At subsonic speeds, it is due to subsonic diffusion and at supersonic speeds, it comprises of pressure rise across a system of shock waves at the inlet followed by that due to subsonic diffusion.

T_{t2s} is the temperature which would have reached after an isentropic ram compression to P_{t2}.

$P_{t0}$is the total (stagnation) pressure at exit of diffuser if all the dynamic pressure (${V_0^2}/{2C_p}$) is captured without losses. (wall friction, non-isentropicity, & shock). T_{t0} is the temperature corresponding to P_{t0}.

- Pressure recovery, π_d is the figure of merit

The isentropic efficiency of the diffuser is defined as

$$\frac{difference\ in\ enthalpy\ for\ actual\ process}{difference\ in\ enthalpy\ for\ ideal\ process}$$

$$\eta_d = \frac{h_{t2s} - h_0}{h_{t0} - h_0} \cong \frac{T_{t2s} - T_0}{T_{t0} - T_0}$$

Variation of pressure recovery ratio and isentropic efficiency of the diffuser with Mach number in the subsonic speed range is shown below:

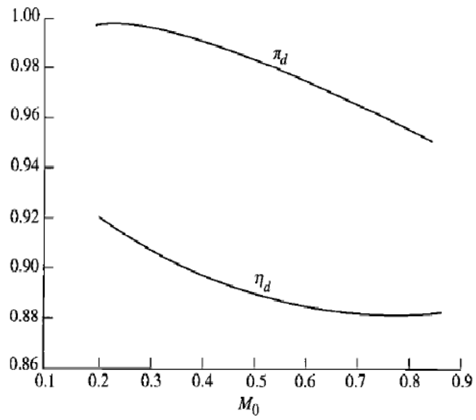


Fig. 6.4 Typical subsonic inlet π_d and η_d.

2. **Compressor**: The compressor is driven by the turbine. It rotates at high speed, adding energy to the airflow and at the same time squeezing (compressing) it into a smaller space. Compressing the air increases its pressure and temperature.

Function: The function of the compressor is to increase the pressure of the incoming air so that the combustion process and the expansion process after combustion can be carried out more efficiently.

By increasing the pressure of the air, volume of the air is reduced and the combustion of fuel/air mixture will occur in a smaller volume.

Two types of compressors are used in turbojet engines; axial flow compressor or centrifugal flow compressor. Centrifugal flow compressor can provide pressure ratios of up to 4.0, whereas axial flow compressor provides up to 1.2 pressure ratio. However, a number of stages (multi-staging) of axial flow compressor can provide much higher pressure ratios above 8.0.

Use of axial flow compressors enable high pressure ratios. Modern axial compressors are split into low pressure and high pressure spools (twin-spooling), driven by corresponding two stages of turbine. High compressor ratios of 15:1 or more can be achieved while improving stability of operation at off-design conditions.

Requirements : The basic requirements of compressors for gas turbine engine are

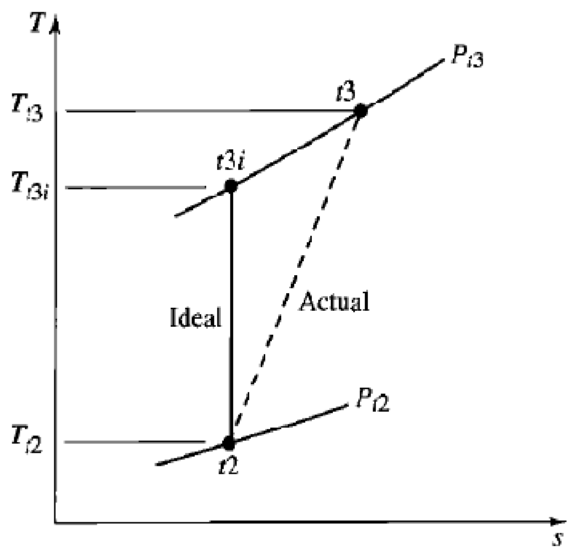
- High air flow capacity per unit frontal area
- High pressure ratio per stage
- High efficiency
- Stable off-design performance
- Discharge direction suitable for multi-staging

Because of the demand for rapid engine acceleration and for operation over a wide range of flight conditions, a high level of aerodynamic performance must be maintained over a wide range of mass flow rates and speeds.

The compressor must be designed in such a way to have minimum length and low weight. The structure must be mechanically rugged and have high reliability.

Thermodynamic Process:

Actual &ideal compression process:



Compression process is an adiabatic process. The performance of a compressor is measured by the isentropic efficiency η_c . If π_c is the pressure recovery ratio across the compressor, then,

$$\eta_c = \frac{ideal\ work\ of\ compression}{actual\ work\ of\ compression}$$

The actual work per unit mass in the T-S diagram is $C_p(T_{t3} - T_{t2})$

The ideal work per unit mass is $C_p\ (T_{t3i} - T_{t2})$

Therefore, $\eta_c = \dfrac{Cp\ (T_{t3i} - T_{t2})}{Cp(T_{t3} - T_{t2})}$

Turbine:

The turbine extracts kinetic energy from the high pressure/high temperature gases which flow from the combustion chamber. The kinetic energy is converted to shaft horsepower to drive the compressor and the fan. Nearly three fourth of the available energy is used to drive the compressor.

Like axial compressor, the axial turbine is usually multi-staged. There are generally fewer stages than the compressor, since in the turbine pressure is decreasing (expansion process), whereas in the compressor, the pressure is increasing (compression process).In both the processes, the blades act as aerofoils.

Operating Principle:

There are two types of turbines, impulse type and reaction type.

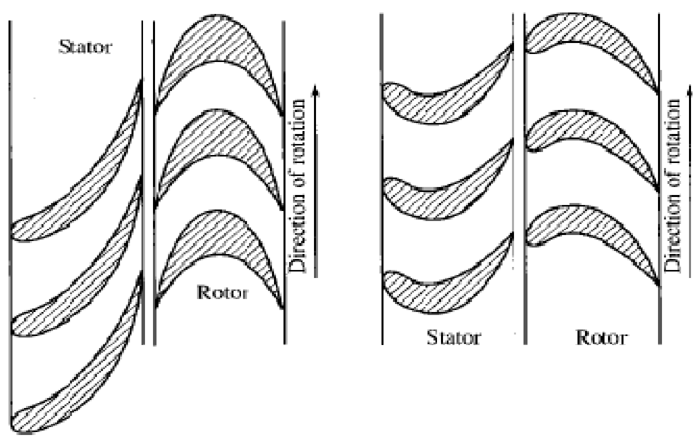
In impulse turbine, there is no change in the gas pressure in the rotor and the relative velocity of gases at rotor entry and exit remains same. The stator nozzles are shaped to form passages which increase the velocity and decrease the pressure of the escaping gases.

In a reaction type turbine, the relative discharge velocity of the gases increases and the pressure decreases in the rotor passages. The stator nozzle passages merely alter the direction of flow.

Most turbines in jet engines are a combination of impulse and reaction turbines.

Construction: Two types turbines are in use; Axial flow turbine and radial flow turbine.

The axial flow turbine consists of a rotor and set of stationary vanes (nozzles) stator. Each stage of turbine consists of a set of stationary vanes that form a series of nozzles which discharge of gases on to the rotor blades. The discharge of the hot gases allows the kinetic energy of the gases to be transformed to mechanical shaft energy.



a) Impulse   b) Reaction

Fig. 4.17 Impulse and reaction stages.

Thermodynamic Cycle:

The isentropic efficiency of turbine is given by

$$\eta_t = \frac{h_{t4} - h_{t5}}{h_{t4} - h_{t5i}} = \frac{T_{t4} - T_{t5}}{T_{t4} - T_{t5i}}$$

$$\eta_t = \frac{1 - \tau_t}{1 - \pi_t^{(\gamma-1)/\gamma}}$$

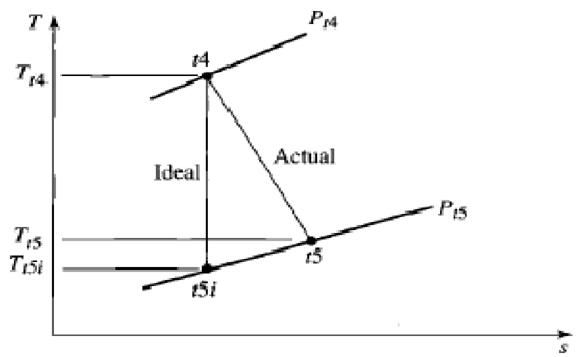
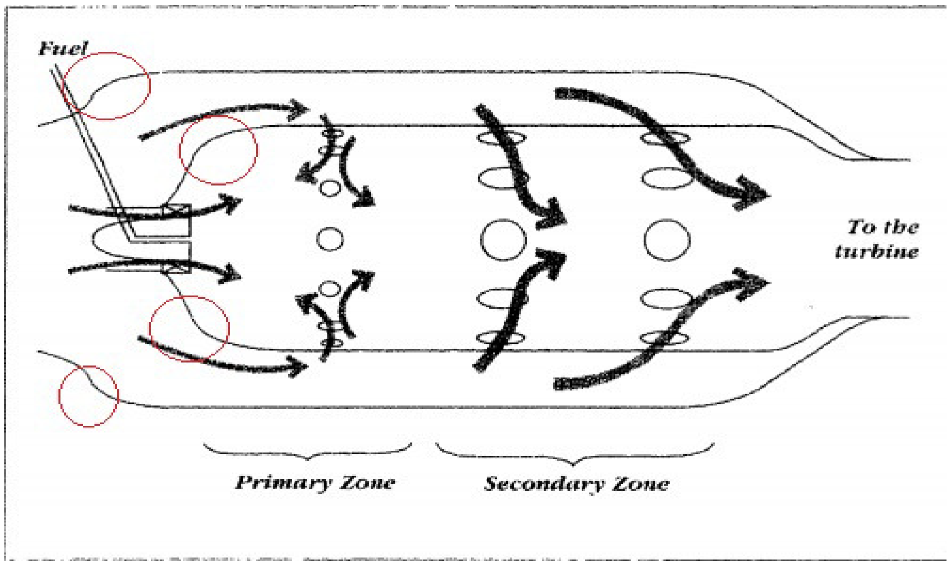


Fig. 6.10 Actual and ideal turbine processes.

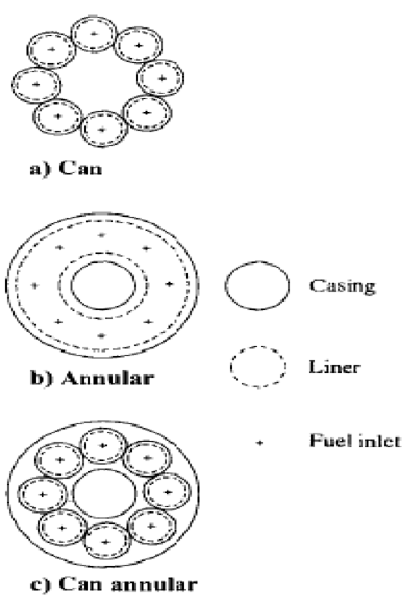
Combustion Chamber: The high pressure air is then mixed with fuel and burnt in the combustion chamber under constant pressure conditions.

The combustion chamber is designed to burn a fuel/air mixture and to deliver the hot gasses to the turbine at uniform temperature. The gas temperature must not exceed the allowable structural temperature of the turbine.
The high pressure air from the compressor enters the combustion chamber. Of this, less than half of the total volume of air mixes with fuel and burns. The rest of the air, known as secondary air is used as cooling the products of combustion or the burner walls. The ratio of total air to fuel varies between 30 to 60 parts of air to 1 part of fuel by weight.

The pressure loss as the gasses pass through the burner must be minimum and the combustion efficiency must be high. There should be no tendency for burner to flame-out.



Combustion chambers are of three types; can, annular and can-annular types. Typical arrangement is as follows:



Nozzle: The combustion gasses at high temperature and pressure are expanded in the turbine and the exhaust nozzle. The expansion of gasses in the turbine provides power to drive the compressor while the exhaust nozzle expands the gasses to atmospheric pressure, thereby producing propulsive force, thrust.

The net thrust delivered by the engine is the result of converting internal energy to kinetic energy.

The exhaust products downstream of the turbine still contain adequate amount of oxygen. Additional thrust augmentation can be achieved by providing an afterburner in the jet pipe in which additional amounts of fuel can be burnt.

Thrust Augmentation

If the thrust of an engine has to be increased above the original design value several alternatives are available. Increase of turbine inlet temperature, for example, will increase the specific thrust and hence the thrust for a given engine size. Alternatively the mass flow through the engine could be increased without altering the cycle parameters. Both of these methods imply some redesign of the engine, and either or both may be used to up-rate an existing engine.

Frequently, however, there will be a requirement for a temporary increase in thrust, e.g. for take-off, for acceleration from subsonic to supersonic speed or during combat maneuvers; the problem then becomes one of thrust augmentation. Numerous schemes for thrust augmentation have been proposed, but the two methods most widely used are liquid injection and afterburning (or reheat).

Liquid injection (Water-methanol/alcohol) is primarily useful for increasing take-off thrust. Substantial quantities of liquid are required, but if the liquid is consumed during take-off and initial climb the weight penalty is not significant.

Spraying water into the compressor inlet causes evaporation of the water droplets, resulting in extraction of heat from the air; the effect of this is equivalent to a drop in compressor inlet temperature. Reducing the temperature at entry to a compressor will increase the thrust, due to the increase in pressure ratio and mass flow.

In practice a mixture of water and methanol is used; the methanol lowers the freezing point of water, and in addition it will burn when it reaches the combustion chamber. Liquid injection into the compressor, however, has corrosive effect on the blades.

Liquid is sometimes injected directly into the combustion chamber. In both cases the mass of liquid injected adds to the useful mass flow, but this is a secondary effect.

Water injection on a hot day can increase the take-off thrust by 50% because the original mass of air entering the engine is low on a hot day.

Liquid injection is now seldom used in aircraft engines.

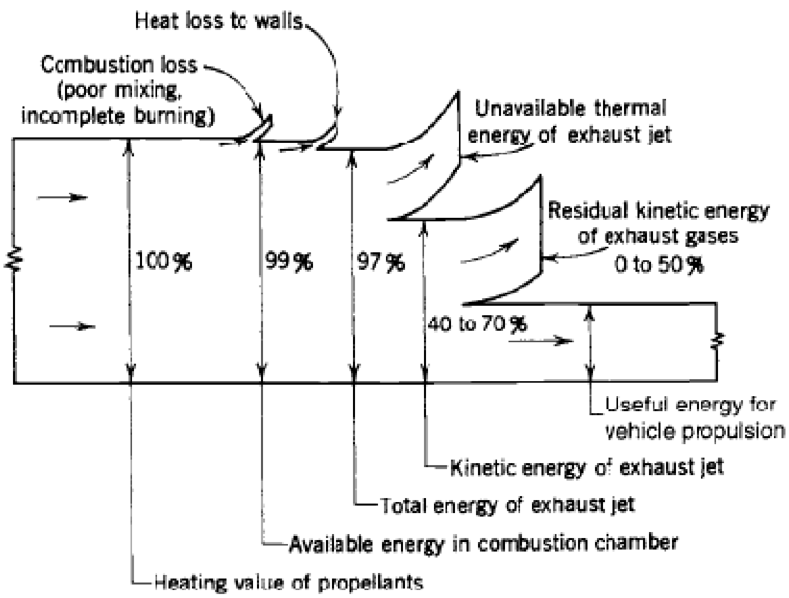
Afterburning, as the name implies, involves burning additional fuel in the jet pipe. In the absence of highly stressed rotating blades the temperature allowable following afterburning is much higher than the turbine inlet temperature. The effect of afterburning is to increase the temperature of the exhaust gases which in turn will result in higher thrust through expansion in the exhaust nozzle. The afterburning produces high thrust at the expense of fuel economy.

The temperatures of around 2000 K are possible through afterburning.

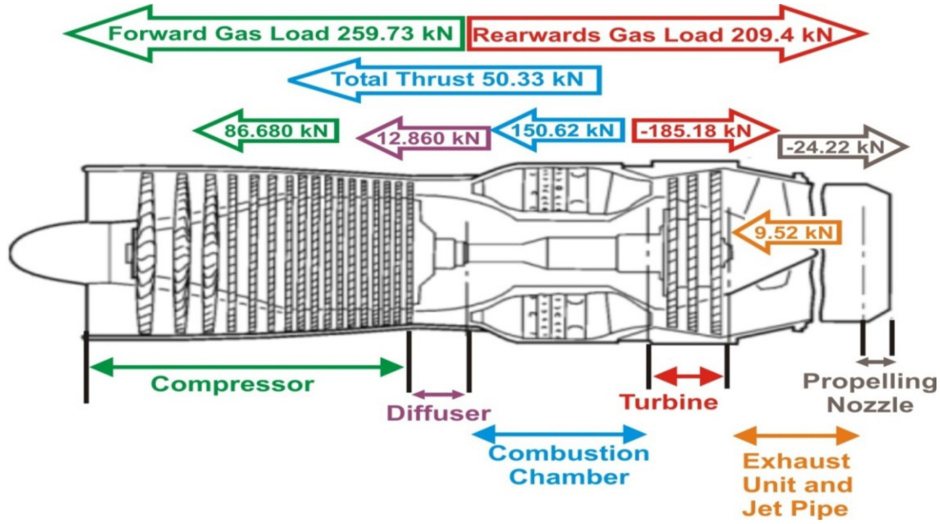
Energy Transfer/Loses:

Two types ofenergy conversion takesplace in the propulsion system. One is generation of energy which is conversion of stored energy in to available thermal energy. The other is transforming the thermal and pressure energy in to kinetic energy. The kinetic energy of the exhaust gases is the form of energy useful for propulsion.

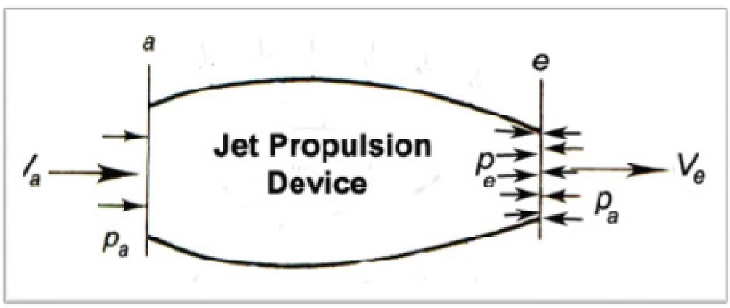
Theenergy balance diagram indicating different loses is given below:



Thrust Generation:



The jet engine can be considered as a single device developing velocities and pressures at entry and exit as shown below:



$$\text{Thrust} = \dot{m}(V_e - V_a)$$

Thrust equation from first principles is derived by writing the force balance ΣF_x and equating the same to the change of momentum based on Newton's II law.

Uninstalled engine thrust F is defined as the force F_{int} acting on the internal surface of the propulsion system from 1 to 9 plus the force F'_{int} acting on the internal surface of the stream tube 0 to 1 that contains the air flowing into the engine. It will be shown that F is independent of the nacelle.

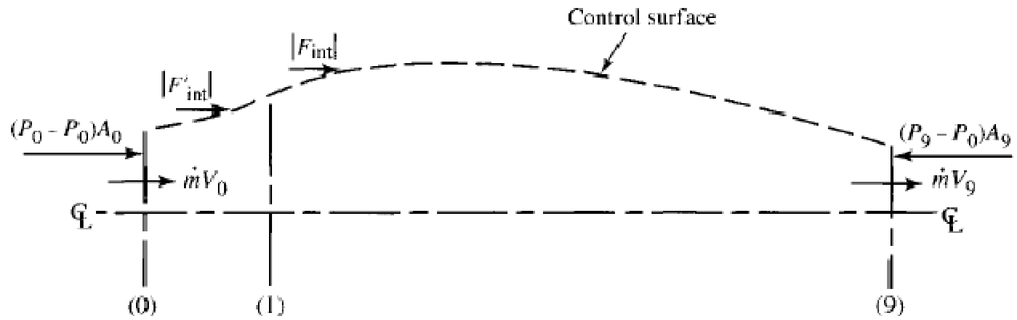


Fig. 4.3 Control surface forces and momentum fluxes for evaluating F (pressure referred to P_0).

Referring to Fig. 4.3 and equating forces to the change in momentum flux, we get

$$F'_{int} - F_{int} + (P_0 - P_0)A_0 - (P_9 - P_0)A_9 = \frac{\dot{m}_9 V_9 - \dot{m}_0 V_0}{g_c}$$

$$F + 0 - (P_9 - P_0)A_9 = \frac{\dot{m}_9 V_9 - \dot{m} V_0}{g_c}$$

$$\text{Uninstalled engine thrust} \quad F = \frac{\dot{m}_9 V_9 - \dot{m}_0 V_0}{g_c} + (P_9 - P_0)A_9 \qquad (4.1)$$

1.5.2: Evaluating Installed Thrust:

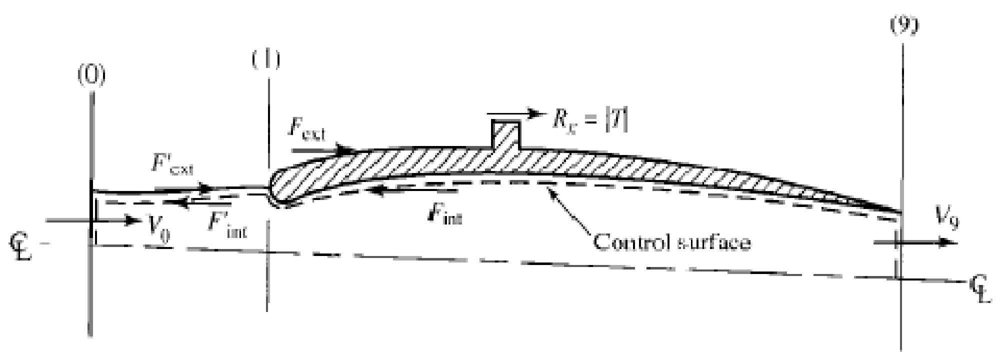


Fig. 4.2 Forces on propulsion system.

To obtain the installed thrust, the drag forces (F'_{ext} and F_{ext}) must be deducted from the uninstalled thrust.

Installed engine thrust $T = F - D$

where T is the installed thrust, F is the uninstalled thrust and D is the drag force created in flight due to the nacelle.

Referring to above diagram,

F'_{ext} = Pressure forces acting on the external stream tube between stations 0 and 1, which is called “additive drag”.

F_{ext} = External pressure force acting on the nacelle’s outer surface, which is called “nacelle drag”

Usually, the Drag forces are viscous forces and the pressure forces contribute to the engine thrust.

The forces on the stream tube between stations 0 to 1, F'_{int} and F'_{ext} are equal in magnitude and cancel each other.

Therefore, the shear force on the strut = T = $F_{int} - F_{ext}$ = T – D

1.6: Engine performance parameters:

1.6.1: Installed thrust T: The first performance parameter is the thrust T.

Thrust T = $(m_a + m_f)\, V_e - m_a V_a + A_e(\, p_e - p_a)$,

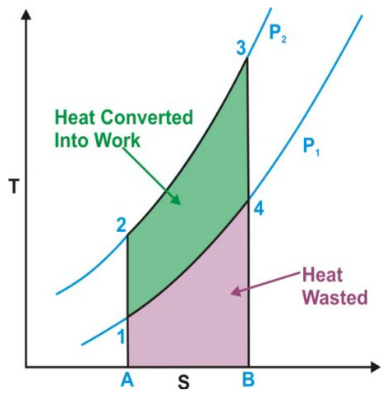
The installed thrust T, uninstalled thrust F and drag D are linked by the equation

1.6.2: Thrust specific fuel consumption Tsfc:

Tsfc $= \frac{\dot{m}_f}{T}$, where $\dot{m}_f$ is the mass flow rate of fuel burnt.

The effect of altitude and Mach number (speed) on thrust and Tsfc is shown in 1.7 under “combined effect of altitude and speed”.

1.6.3: **Brayton Cycle**:



The Brayton cycle consists of four internally reversible processes:

- 1-2 Isentropic compression (in a compressor)
- 2-3 Constant-pressure heat addition
- 3-4 Isentropic expansion (in a turbine)
- 4-1 Constant-pressure heat rejection

Thermal Efficiency: The thermal efficiency η_T is another useful engine performance parameter. It is defined as net rate of energy out of the engine (kinetic energy) divided by the rate of thermal energy available from the fuel. The fuel's available thermal energy is equal to mass flow rate of fuel $\dot{m}_f$ multiplied by the heating value of fuel.

$$\eta_T = \frac{\dot{W}_{\text{out}}}{\dot{Q}_{\text{in}}}$$

where

η_T = thermal efficiency of engine
$\dot{W}_{\text{out}}$ = net power out of engine
$\dot{Q}_{\text{in}}$ = rate of thermal energy released ($\dot{m}_f h_{PR}$)

- The thermal efficiency of the ideal Brayton cycle under the cold air standard assumptions becomes:

$$\eta_{th,Brayton} = \frac{w_{net}}{q_{in}} = 1 - \frac{q_{out}}{q_{in}} = 1 - \frac{T_4 - T_1}{T_3 - T_2} = 1 - \frac{T_1(T_4/T_1 - 1)}{T_2(T_3/T_2 - 1)}$$

Processes 1-2 and 3-4 are isentropic and $P_2 = P_3$ and $P_4 = P_1$.

Therefore, $\frac{T_2}{T_1} = \left(\frac{P_2}{P_1}\right)^{(\gamma-1)/\gamma} = \left(\frac{P_3}{P_4}\right)^{(\gamma-1)/\gamma} = \frac{T_3}{T_4}$

If $\frac{T_2}{T_1} = \frac{T_3}{T_4}$, then $\frac{T_4}{T_1} = \frac{T_3}{T_2}$

Therefore, $\eta_{th.Brayton} = 1 - \frac{T_1}{T_2} = 1 - \frac{1}{T_2/T_1}$

Substituting these equations into the thermal efficiency relation and simplifying:

$$\eta_{th,Brayton} = 1 - \frac{1}{r_p^{(\gamma-1)/\gamma}}$$

where, $r_p = \frac{P_2}{P_1}$ is the pressure ratio.

The thermal efficiency of a Brayton cycle is therefore a function of the cycle pressure ratio and the ratio of specific heats.

1.6.4: Propulsive Efficiency: The propulsive efficiency η_p is a measure of how effectively the engine power output $\dot{W}_{out}$ is used to power the aircraft.

$$\eta_P = \frac{TV_0}{\dot{W}_{out}}$$

where

η_P = propulsive efficiency of engine
T = thrust of propulsion system

V_0 = velocity of aircraft
$\dot{W}_{out}$ = net power out of engine

In terms of vehicle and exit velocities, the propulsive efficiency is expressed as

$$\eta_P = \frac{2}{V_e/V_0 + 1}$$

1.6.5: Overall efficiency: The thermal and propulsive efficiencies are combined to give the overall efficiency η_o.

$$\eta_O = \eta_P \eta_T$$

$$\eta_O = \frac{TV_0}{\dot{Q}_{\text{in}}}$$

Since $\dot{Q}_{in}$ = $\dot{m}_f h_{PR}$, overall efficiency is written as

$$\eta_O = \frac{TV_0}{\dot{m}_f h_{PR}}$$

$$\eta_O = \frac{V_0}{\text{TSFC} \cdot h_{PR}}$$

Above relations give the expression for TSFC as

$$\text{TSFC} = \frac{V_0}{\eta_P \eta_T h_{PR}}$$

1.7: Effect of Flight conditions:

Factors affecting the thrust of gas turbine:

The thrust of the engine, F is given by

$$F = (m_a + m_f)\, V_e - m_a V_a + A_e(\, p_e - p_a),$$

Where m_a is the mass flow rate of air, m_f is mass flow rate of fuel, V_a is the aircraft speed, p_e and p_a are the pressures at the nozzle exit and inlet of the engine and A_e is the nozzle exit area.

The first term, [$(m_a+m_f)V_e$] in the thrust equation is the **momentum thrust**, the second term, [$m_a V_a$] is the **ram drag** and the third term, [$A_e(\, p_e - p_a)$] is the **pressure thrust.**

The quantity [$(m_a+m_f)V_e$] + [$A_e(\, p_e - p_a)$] is also called **gross thrust**.

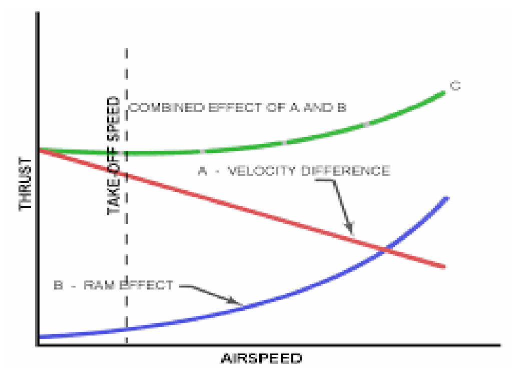
- **Air Speed & Ram Effect**: Incoming air velocity affects the thrust in two different and opposite ways. When the aircraft is static, as when the engine is being run before take-off, momentum/ram drag [$m_a V_a$] is zero, because V_a = 0.

  However, as the aircraft commences to move, the velocity of air entering the engine begins to increase because of the speed of the aircraft.

  Therefore, the difference between V_e and V_a becomes less as airspeed increases. This tends to reduce the thrust.

If the air mass flow and fuel flow are assumed constant, then the increase of air speed will cause a linear reduction in net thrust.
On the other hand, as the aircraft gains speed on the runway, the movement of aircraft relative to the outside air causes air to be rammed in to the engine inlet duct.

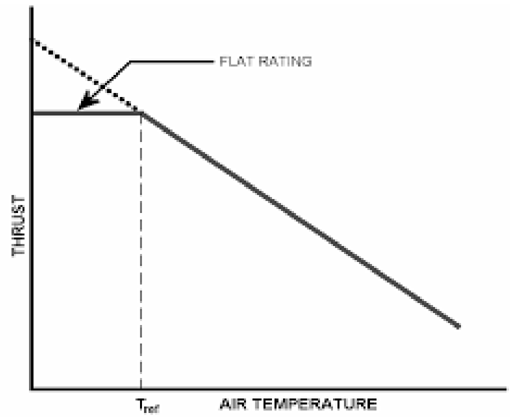
This compression of air in the engine inlet duct arising from the forward movement of the aircraft is called ram pressure or ram effect.
The ram effect both increases the air mass flow to the engine and the intake pressure rise, and therefore increases the thrust.
The combined thrust variation due to ram effect & velocity difference is shown in the graph below:



However, ram pressure rise is not significant at lower speeds and thus it cannot offset the loss of thrust due to reduced difference velocities of air and exhaust jet,$(V_e - V_a)$. The thrust decreases slightly as the aircraft speeds up during take-off.
The increase of thrust due to ram becomes significant as the air speed increases, which will compensate for the loss of thrust due to reduced pressure at high altitude. Ram effect is thus important for high speed fighter aircraft. Also, modern subsonic jet powered aircraft flying at high subsonic speeds and higher altitudes make use of ram effect.

Ambient Temperature: The thrust generated by a jet engine is inversely proportional to the ambient air temperature, thus the **thrust decreases as the**

air temperature increases. The effect of ambient temperature on thrust is shown below:



However, this also means an increase in thrust when the temperature decreases, so that the engine may generate higher thrust than its design rating at lower ambient temperatures. Higher thrust above the design rating can harm the engine.

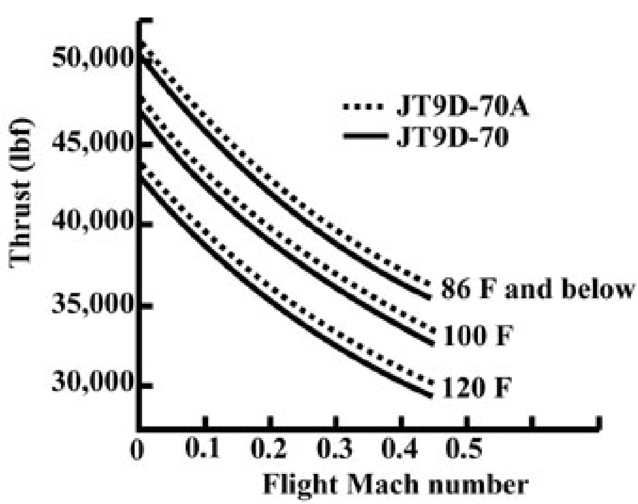
A 10,000 lbs thrust engine might generate only 8000 lbs of thrust on a hot day. The same engine may generate12,000 lbs of thrust on a cold day.

For this reason, engines are restricted to a maximum thrust. This thrust restriction is called "flat rating".

At a given pressure altitude, temperature has no influence on engine take-off thrust, below the flat rating temperature, called T_{ref}.

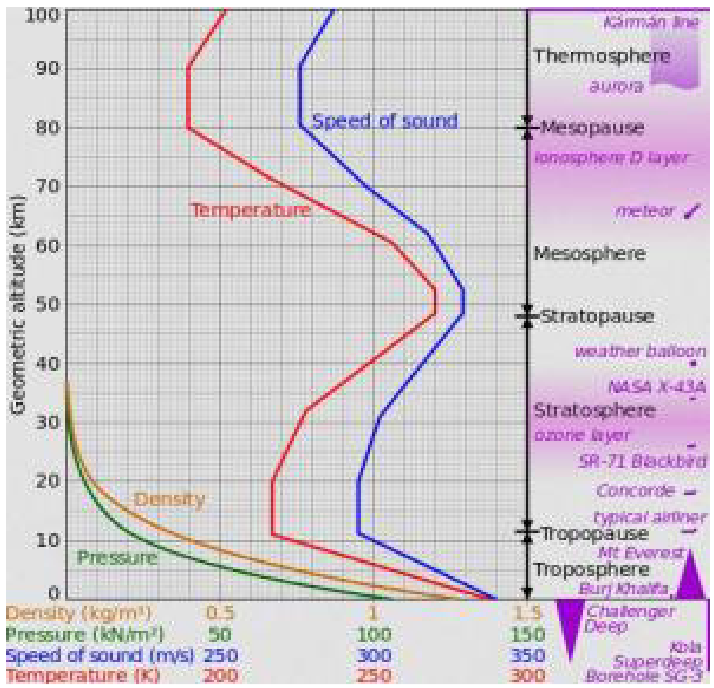
The available thrust decreases as the temperature increases as shown the above graph.

JT9D-70/-70A Engines Takeoff Thrust at Sea Level



Effect of Altitude:

Plot showing P,T & ρ variation with Altitude:



As the altitude increases, the pressure and density decreases so does the thrust. However, as altitude increases, temperature decreases, the thrust increases. The pressure and density decreases faster than the temperature, so the net effect on thrust is to reduce up to an altitude of 11000 (troposphere). After 11000 mt, the temperature stops falling, but pressure continues to drop with altitude. Consequently, above 11000 mt, thrust will drop off more rapidly. This makes 11000 mt as optimum altitude for long range cruise.

Factor Affecting Thrust

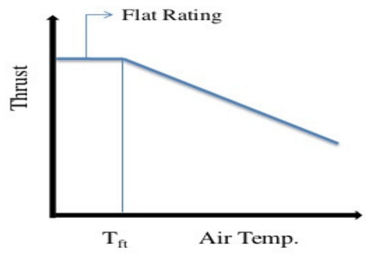


Fig.16. Temp. effect on thrust [1]

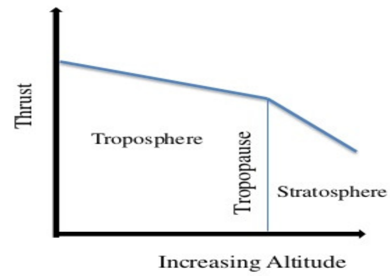
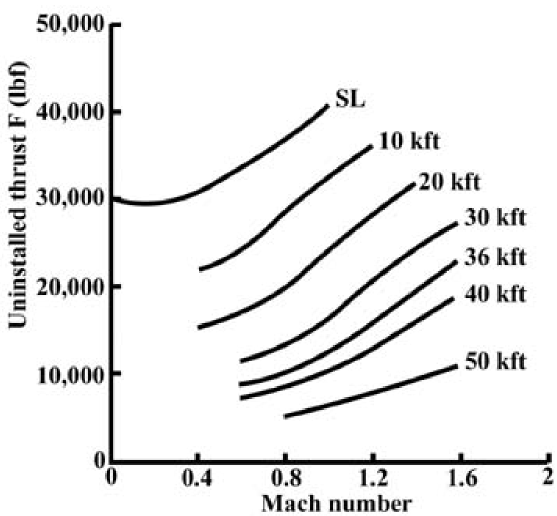
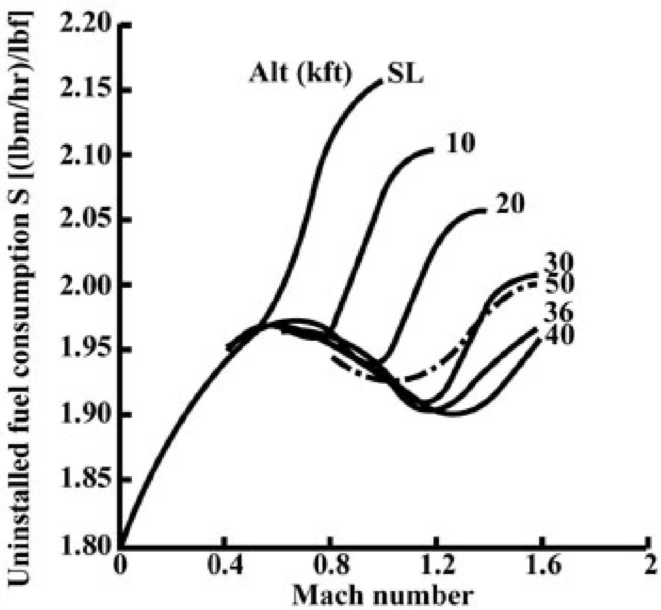


Fig.17. Altitude effect on thrust [1]

- **Combined Effect of Altitude & Speed**: Variation of uninstalled engine thrust and thrust specific fuel consumption with altitude and Mach number (speed) is shown below:

  The thrust (*F*) decreases with altitude and the fuel consumption (*S*) also decreases with altitude until 36k ft (the start of the isothermal layer of the atmosphere). Also note that the fuel consumption increases with Mach number and the thrust varies considerably with Mach number.





Air mass flow: The mass of air flow is the most significant of the thrust equation. It depends on the ambient air temperature and pressure as both determine the density of air entering the engine. In free stream air, a rise in temperature will decrease the density. Thus, as ambient or inlet temperature increases, the mass flow rate and thrust decreases.

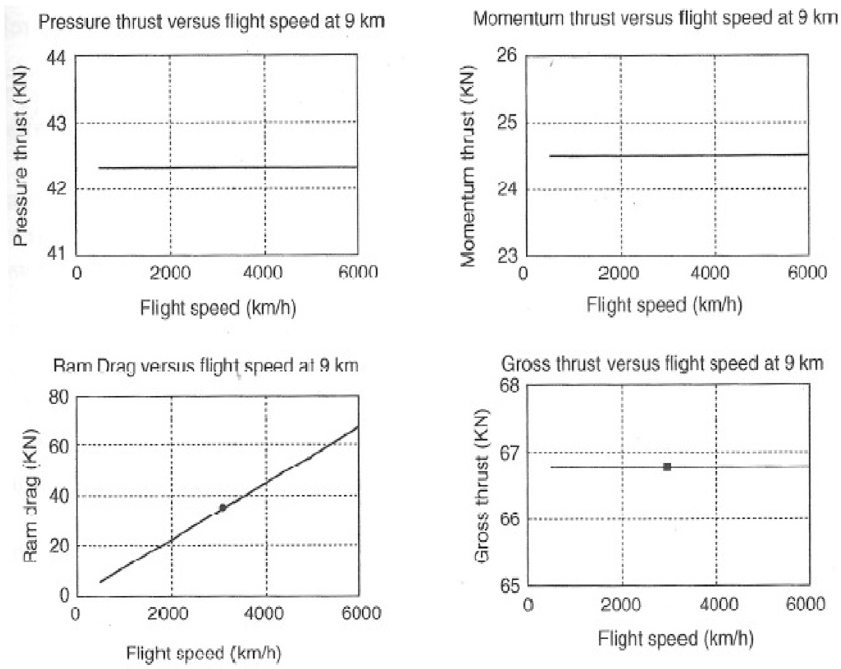
On the contrary, an increase in the pressure of free stream air increases the density and consequently the mass flow rate and thrust increases.

In brief, the density affects the inlet air mass flow and it directly affects the thrust.

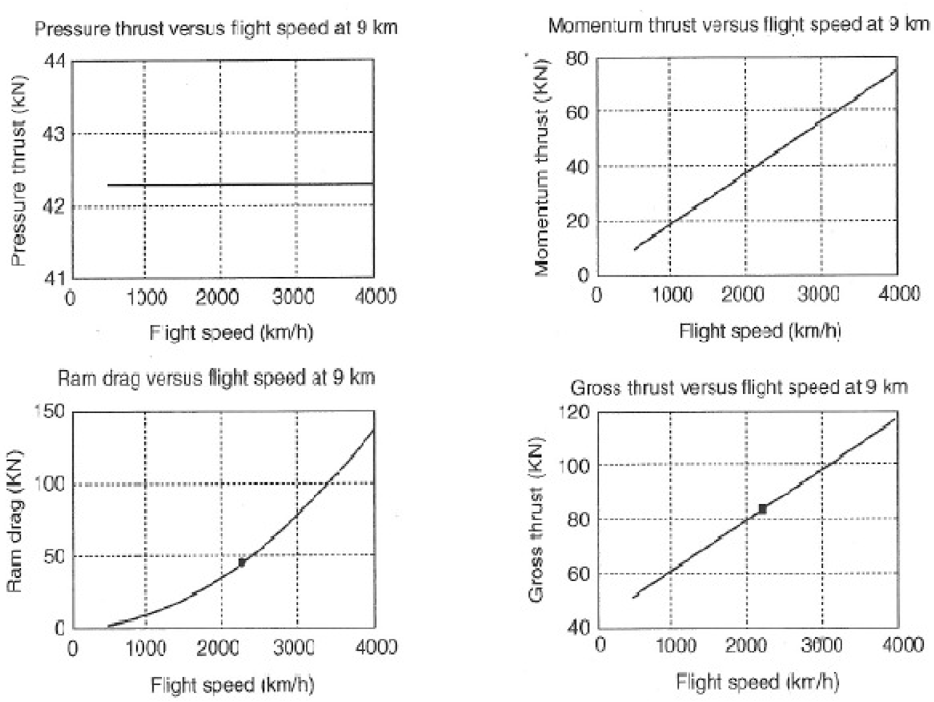
- The near-constant thrust characteristics at any altitude a desirable and attractive feature of jet engines (flat rated engines)
- The basic thrust equation indicates that as forward speed V_a increases it is necessary to increase either the mass flow, or exit velocity V_{ex}, or both, in order to hold the thrust, F, constant.

Effect of variation of flight Speed V_a on thrust, at two different conditions, one with mass flow of air constant and the other with mass flow rate varying are shown below:

Typical thrust variation at constant air mass flow



Typical thrust variation with variable air mass flow



- **Effect of Turbine Entry Temperature/Mass of fuel**: Operating Condition of the jet nozzle and TET &m_f have considerable effect on the thrust produced by the engine.
  The pressure thrust is the product of nozzle exit area and difference of nozzle exit pressure and ambient pressure. Similarly, the momentum thrust is dependent on the nozzle exit velocity.
  The nozzle can either be convergent or convergent-divergent.
  Convergent nozzles may be choked or un-choked during the flight. For a choked convergent nozzle, the speed of exhaust gasses V_e is equal to sonic speed. The sonic speed is directly proportional the local temperature T_e. Therefore, the momentum thrust will be affected by the sonic speed which is mainly influenced by the exhaust gas temperature. The exhaust pressure for a choked nozzle is greater than ambient pressure and thus the pressure thrust has a non- zero value.
  If the nozzle is un-choked, then the jet velocity varies with the atmospheric pressure. The pressure thrust will be zero when the exhaust pressure is equal to the ambient pressure.

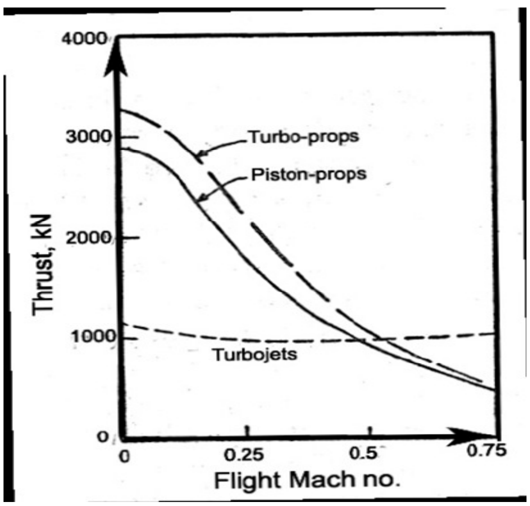
Role of Propulsion in aircraft performance:

Propulsion systems have bearing on two parameters of performance, altitude & forward speed of the aircraft.

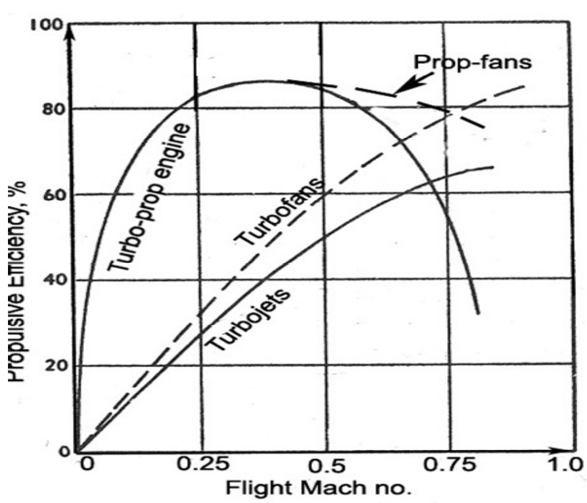
The spectrum of aircraft performance versus the type of propulsion is depicted below:

Performance Characteristics:

Thrust generation at Low Speeds:



Propulsive Efficiency at Low Speeds



Unit II – Anatomy of Jet Engine-I

UNIT II

ANATOMY OF JET ENGINE-I

INLETS: Locations, Types of inlets, operating principle, functions, geometry, operating conditions, flow field, capture area, sizing, flow distortion, drag, and diffuser loses, methods of mitigation, performance.

COMPRESSOR & TURBINE: types, construction, stage, cascade, blade geometry, velocity triangles, Euler equation, types of flow analysis, diffusion factor, stage loading, Variable stator, limits on compressor performance, typical blade profiles. Axial flow turbines-, similarities and differences with compressors, Velocity diagram analysis, no exit swirl condition, flow losses, causes tangential stresses, repeating stages, Computation of stage parameters for ideal and real turbine of given cascade, blade geometry and initial flow conditions and turbine speed- procedure. Typical turbine blade profiles, turbine performance maps, Thermal limits of blades, cooling, materials, construction, methods of production, Limits on stage pressure ratio of turbines- multistage, multi-spooled turbines. Range of axial flow turbine, design parameters, Typical turbine blade profiles.

2.1: Inlets-Types, geometry, drag and diffuser loses, methods of mitigation and performance

2.1.1: Functions of Inlets:

(a) Provide the engine with the amount of air which it demands.

(b) Provide the air over the full range of Mach numbers and engine throttle settings.

(c) Provide the air at all operating altitudes.

(d) Provide the air evenly distributed over the compressor face.

(e) Accelerate or decelerate the air so that it arrives at the compressor face at the required velocity [normally about M 0.5 (160-220 m/s)].

(f) Provide optimum initial air compression to augment compressor pressure rise.

(g) Minimize external drag.

2.1.2: Types of inlets: Broadly, intakes can be classified as

- Subsonic Intakes
- Supersonic intakes

<u>Intake Designs</u>

Some typical intake designs are:

- (a) Simple Pitot Intake: (Divergent duct)
- (b) Supersonic Intake with central conical spike: mounted in the fuselage center
- (c) Wedge type intake; Split intake mounted on the side of the fuselege

(a) The simplest form is a divergent duct known as a pitot intake (Fig 2-4). The air stream in the duct follows the usual compressibility laws and the design can be very efficient provided that there are no sharp corners or irregularities. Care must be taken with the design of the intake lip since this can determine the critical mach number of the intake. Rounded lips assist in preventing boundary layer separation, but lower the critical Mach number.

(b) Conical Spike Intake: Supersonic intake with a variable area, controlled by a movable central body,

(c) Wedge type side (box) intake: The side intake puts the entry in a region of thick boundary layer on the fuselage and it is usually necessary to bleed this air away. This entails some loss of engine mass flow but ensures a large reduction in pressure losses.

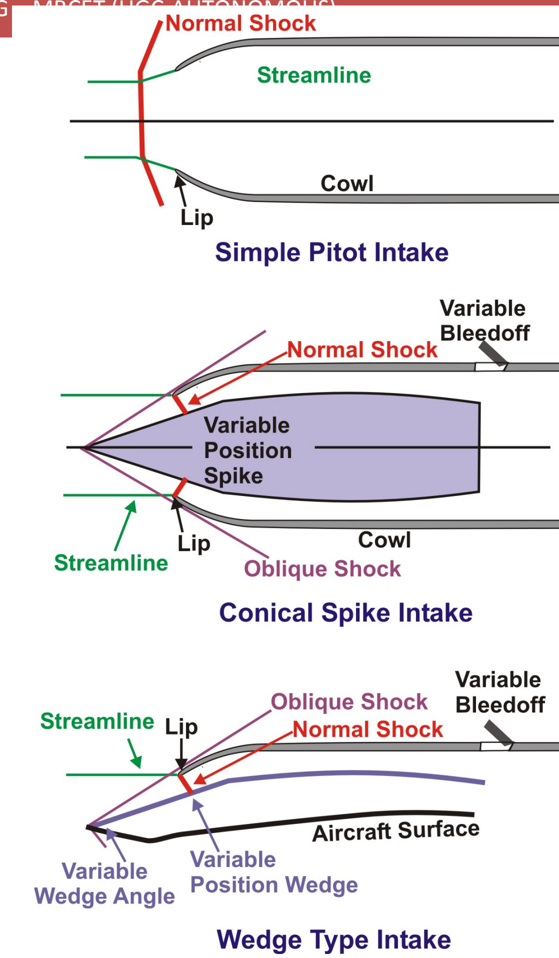
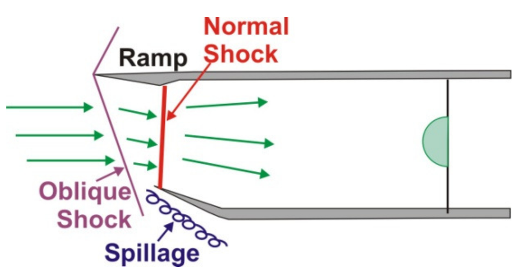
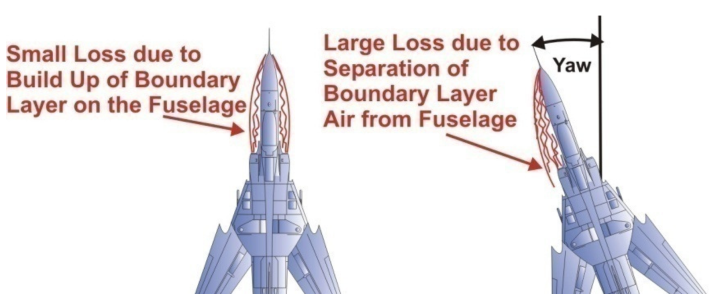


Fig 2-4: Three Types of Intake Designs



Locations of Inlets:

The divided type of intake also suffers to some extent from boundary layer problems in that when the aircraft yaws, a loss of ram pressure occurs on one side of the intake, causing an uneven distribution of airflow to the compressor.

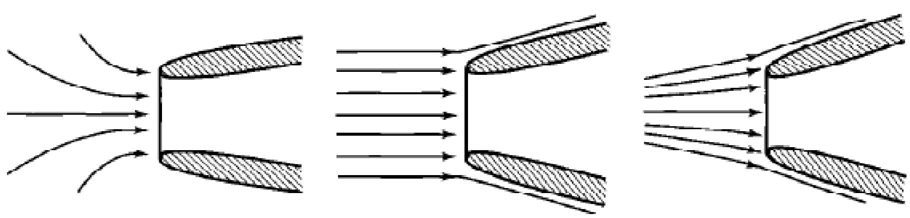


2.1.3: Operating Principle:

- The inlet interchanges the kinetic energy andthermal energy of the gas in an adiabatic process. The perfect inlet would follow an isentropic process.
- The primary purpose of the inlet is to bring the air by the engine at the entrance of fan orcompressor from free stream conditions with minimum total pressure loss.
- While the subsonic intake slows down the flow by virtue of it's divergent shape.
- The supersonic intake uses series of shock waves in reducing the speed gradually with minimum pressure loss.

Subsonic Intakes:

2.1.4: Operating Conditions: The operating conditions at the inlet depend on flight Mach number and mass flow demanded by the engine. Streamline pattern of three conditions are shown below:



a) Static operation (or insufficient air for engine)

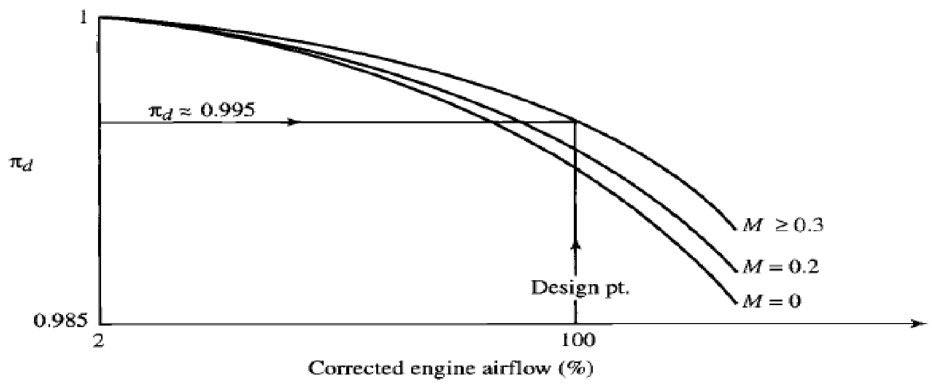
b) Low-speed operation (correct engine air)

c) High-speed operation (more air than the engine needs)

Fig a) shows acceleration of fluid external to the inlet that will occur when the inlet operates at a velocity lower than the design value or at a mass flow higher than the designed value.

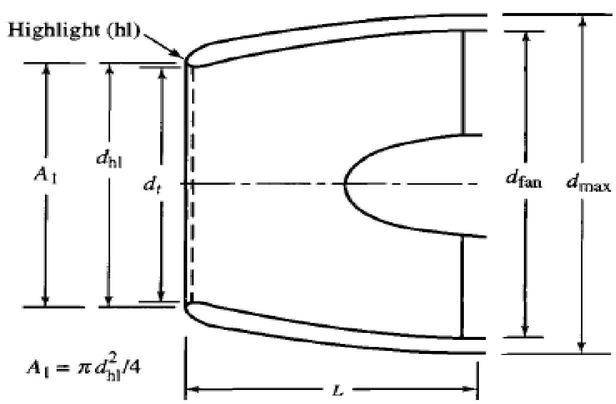
Fig c) shows deceleration of fluid external to the inlet that will occur at a velocity higher than design or mass flow lower than design.

2.1.5: Inlet Total Pressure Ratio π_d: The inlet pressure recovery is usually assumed to be constant for subsonic inlets. However, $\pi_d$varies withMach number and mass flow rate, as shown below:



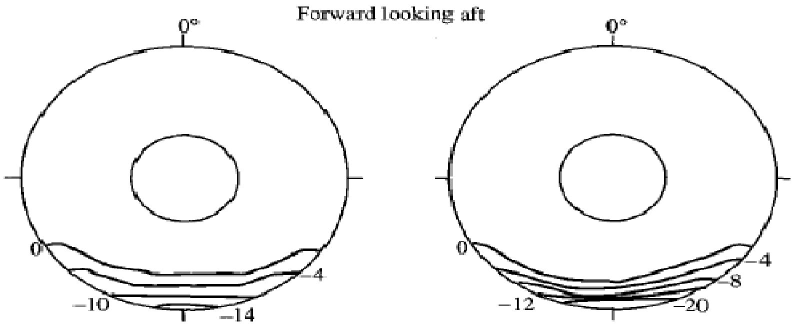
Subsonic Inlets-nomenclature

- The inlet area A_1 is based on the flow cross section at the inlet "highlight"(hl).
- The subsonic inlet can draw in airflow whose free stream area A_0 is larger than the inlet area A_1, variable inlet geometry is not required.
- Sometimes, blow-in doors or auxiliary inlets are used to reduce installation drag during take-off.
- Capture area A_1 is the inlet area based on the geometry of the inlet and cross section diameter at inlet highlight d_{hl}. The least diameter is the throat diameter d_t



2.1.6: Inlet Sizing-Throat diameter d_t: The diameter at the throat of the subsonic inlet is sized such that the mach number at this location does not exceed 0.8. This design calculation is based on 1-Dimensional (1-D) flow analysis. This would usually correspond to actual Mach number of 0.9.

2.1.7: Inlet Flow Distortion:



Steady-state pattern $(\Delta P/P)_{max} = 0.167$ Dynamic pattern $(\Delta P/P)_{max} = 0.227$

Inlets when operated in an attitude of high angles of flow incidence have flow separation from the inside lower contour. This flow separation causes large regions of low total pressure, as shown below:

The magnitude of this distortion from the desired uniform flow is measured by a term called ”inlet distortion” given by

$$\text{Inlet distortion} = \frac{P_{t\,\max} - P_{t\,\min}}{P_{t\,av}}$$

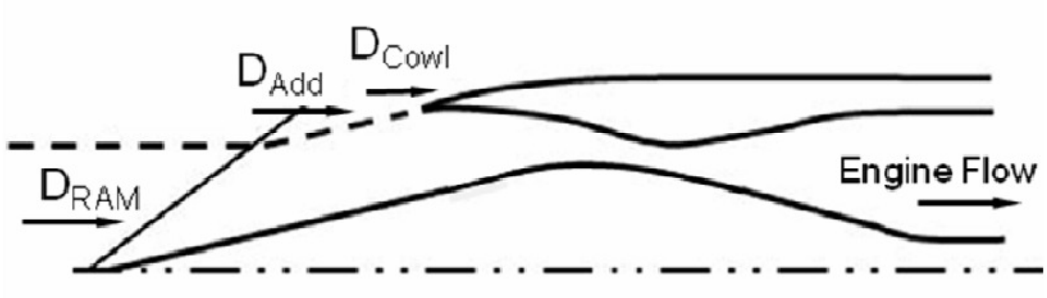
2.1.8: Inlet Drag: Whilederiving equation for installed thrust, two types of drag caused by engine in installed condition. They are

- Additive drag or pre-entry drag D_{add}
- Nacelle drag, D_{nac}

The additive drag is high at low flight Mach numbers. The lip of subsonic intake is well rounded to avoid separation of flow as the streamlines negotiate the intake lip.
To reduce the additive drag at low flight Mach number, some subsonic intakes are provided with “blow-indoors” or auxiliary inlets.
At the engine inlet, flow separation can occur on the external surface of the nacelle due to high local velocities and subsequent deceleration. Flow separation can also occur on the internal surface of the inlet due to flow deceleration due to the adverse pressure gradient.
At high subsonic Mach numbers, flow around the inlet lip, especially on the outside may become supersonic locally, causing shock waves to appear.
Nacelle and Interference Drag: The nacelle drag and interference drag will change with flight Mach number. The engine location on the wing that

provides the best integration of engine and airframe depends on the nacelle design, wing design and resulting interference. The optimum value of the ratio $[{}^{d_{hl}}/_{d_{max}}]$ is found experimentally.

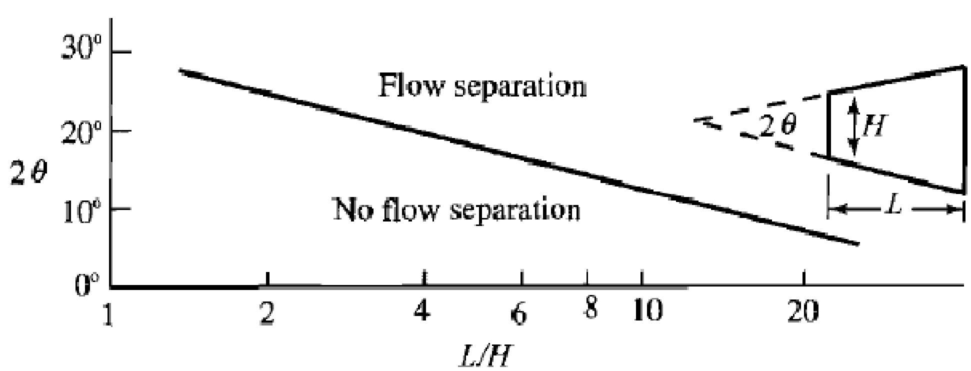
When air is taken aboard a flight vehicle, it is accelerated from zero velocity to approximately the velocity of the vehicle. Accelerating a mass of air forward requires energy and is felt as an impeding or downstream force, or drag, on the airplane. The term ramrepresents the drag of taking air on board. The drag associated with accelerating the flow is ram drag or D_{RAM}, as shown below:



- The dashed line represents the boundary of the airflow stream tube that is taken aboard.
- Air on the outside of this boundary is spilled around the inlet and the drag associated with that spillage is additive drag, Dadd.
- The drag associated with flow over the outer lip or cowl is termed as D_{COWL} (or nacelle drag, D_{nac})

2.1.9: Diffuser losses- Mitigation by use of Vortex Generators:

By using vortex generators, it is possible to have lower total pressure loss. The boundary layer gets reenergized by vortex generators, avoiding flow separation. This also enables use of short length diffusers reducing the weight. Flow separation limits in two-dimensional straight-walled diffusers is shown below:



2.1.10: Supersonic Inlets

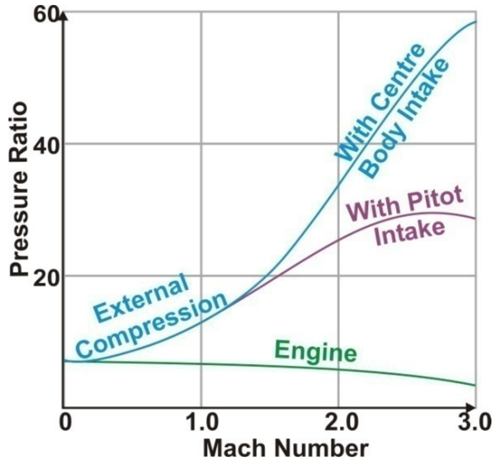
The supersonic inlet is required to provide the proper quantity and uniformity of air to the engine over a wider range of flight conditions than the subsonic inlet is.

In supersonic inlet, the flow is decelerated by shock waves which can produce total pressure loss much greater than the boundary layer losses.

The engine overall compression ratio is a product of engine's ram, diffuser and compressor pressure ratios.

Cycle Pressure ratio = $\pi_r \pi_d \pi_c$

By suitable design it is possible to augment the compressor pressure rise by compressing the air in the intake. This additional pressure rise is known as ram effect and increases with an increase in forward speed. Figure below indicates contribution of intake pressure rise, ram pressure rise to the engine compressor design pressure rise.



The green line in Fig shows the pressure ratio of a jet engine varying with Mach number. The design pressure ratio is maintained up to about 1.4 M, but beyond this speed the ratio falls off because the air temperature increases with increasing Mach number. At 1.0 M the external compression caused by ram effect in the engine intake is approximately equal to that of the engine. At higher mach numbers the contribution of the intake increases markedly, whilst that from the engine decreases.

The pressure rise due to ram effect and diffuser action, $\pi_r \pi_d$ is a major fraction of cycle compression ratio at high supersonic Mach numbers. The engine specific thrust and Tsfc are very sensitive to the diffuser pressure ratio. Also, since the mass or weight of air moved through an engine directly affects the thrust, an increase in intake pressure will increase the weight of air available and therefore, the thrust.

Normal Shock Wave: In the above example, we considered flow through a station 1, slowing down to M=1 and further to M=0. But we did not consider shock wave formation at sonic state.

Consider a perfect gas flow through a normal shock wave, which causes entropy rise across the shock wave. Considering states before and after the normal shock as x and y respectively, variation of properties across a normal shock are:

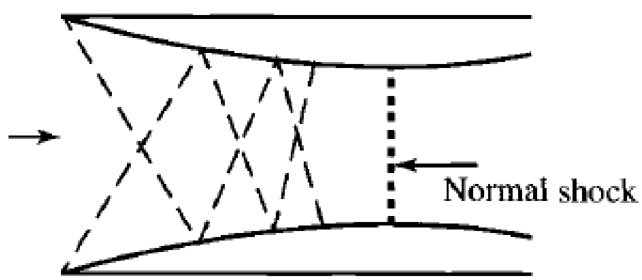
1. $S_y > S_x$
2. Flow through normal shock is irreversible and adiabatic at constant T_t
3. $P_{ty} < P_{tx}$

2.1.9: Supersonic Inlet Types: Supersonic inlets are classified based on the location of supersonic compression wave (shock wave) system. The three types of inlets are

- Internal Compression Inlet
- External Compression Inlet
- Mixed Compression Inlet

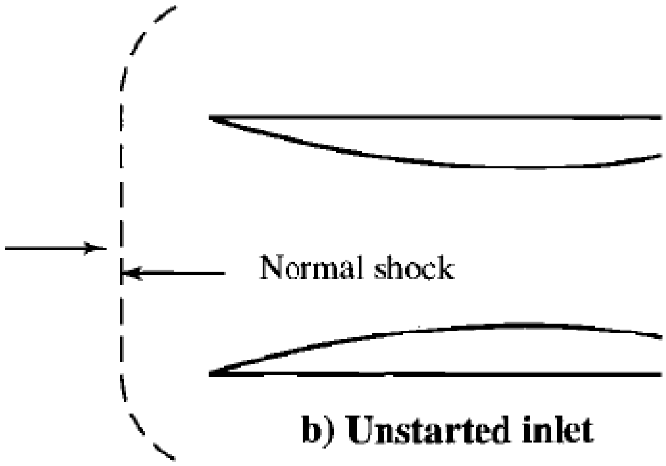
Internal Compression Inlets: The internal compression inlet achieves compression through a series of internal oblique shock waves followed by a terminal normal shock positioned downstream of the throat (its stable location). This type of inlet requires variable throat area to allow the inlet to swallow the normal shock (during starting). Fast reaction bypass doors are also required downstream of the throat to permit proper positioning of normal shock under varying flight and engine conditions.

Normal Operation:



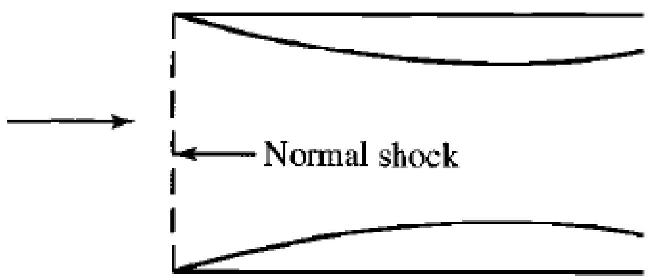
Above diagram shows normal operation of internal compression inlet at design condition of 2.5 Mach. The normal shock is positioned slightly downstream of the throat and efficiency of 0.9794.

As the Mach number reduces, the normal shock is thrown out and the inlet operates under "un-start condition", and the total pressure recovery suffers with efficiency dropping to 0.52. The un-start inlet is shown below:

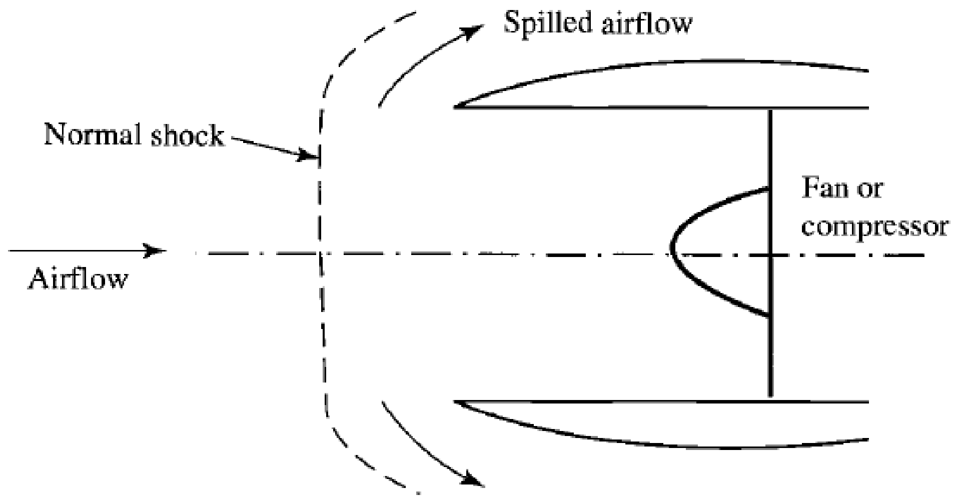


b) Unstarted inlet

Starting of the inlet can be achieved when the area of the throat is made large enough for the normal shock wave to move back and touch the inlet tip (critical operation) as shown below:

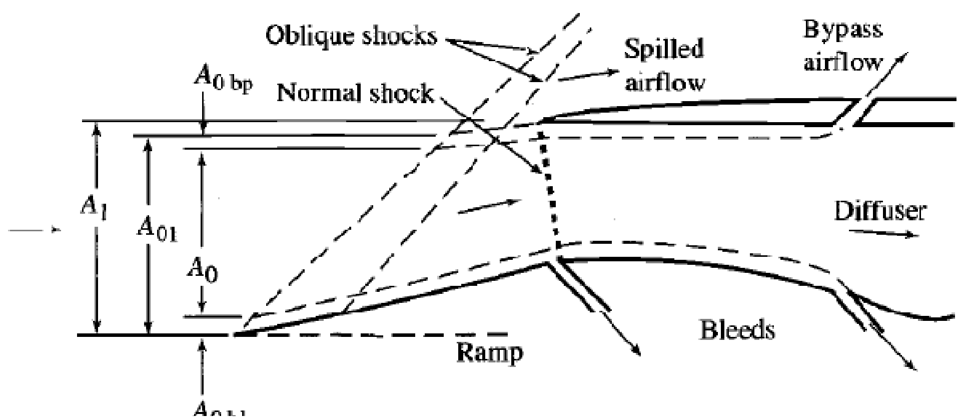


2.1.10: External Compression Inlet: The compression of the external compression inlet is achieved through either one or a series of oblique shocks followed by a normal shock or simply through one normal shock, as shown below:



The compression is achieved through one normal shock, which is called **pitot inlet** or normal shock inlet. The pitot inlet is simple, short and inexpensive. The total pressure recovery is satisfactory up to a free stream Mach number of 1.6.

Above this Mach number, the total pressure recovery is very low, and a more efficient design is used, as shown below:

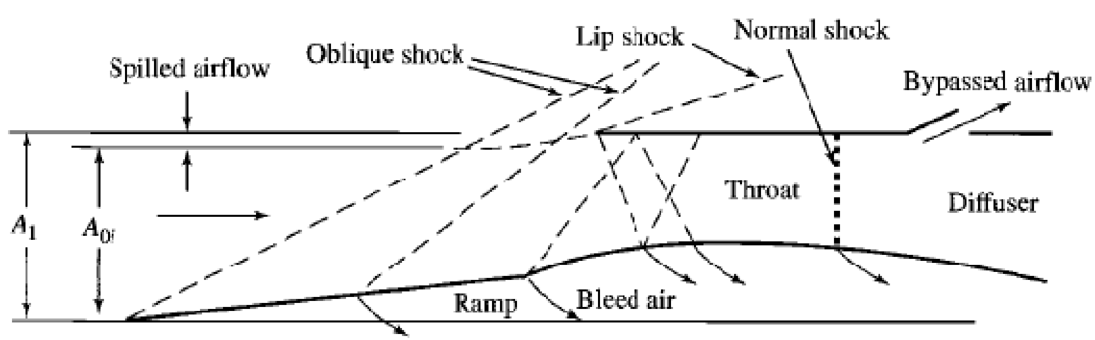


The external compression inlet needs variable throat area and bypass doors to ensure anchoring of normal shock at the throat (lip). The external compression inlet with one or more oblique shocks has its inlet throat at or very near the cowl leading edge. The normal shock anchored at the cowl lip is the desired design condition (critical operation).

2.1.11: Mixed Compression Inlet: At flight Mach numbers above 2.5 M, mixed compression inlet is used to obtain acceptable pressure recovery and low cowl drag.

Mixed compression inlet achieves compression through external oblique shocks, internal reflected oblique shocks followed by the terminal normal shock. The inlet is heavy, complex to design and costly.

The mixed compression inlet also requires both bypass doors and variable throat area.



2.1.12.Rectangular/Circular Inlets: Supersonic inlets can also be classified as two dimensional (rectangular) and axisymmetric (Circular or portion of a circle).

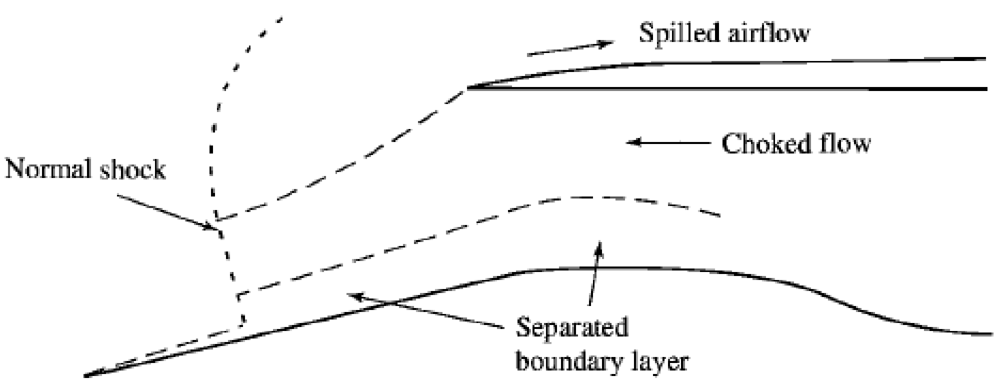
Circular intakes have a slight advantage over rectangular intakes with respect to weight and total pressure ratio. Rectangular intakes are simple to design.

An **intake ramp,** a rectangular, plate-like device within the rectangular air intake, is designed to generate the required number of shock waves to aid the inlet compression process at supersonic speeds. The ramp sits at an acute angle to deflect the intake air from the longitudinal direction. At supersonic flight speeds, the deflection of the air stream creates a number of oblique shock waves at each change of gradient along at the ramp. Air crossing each shock wave suddenly slows to a lower Mach number, increasing pressure. F-15 aircraft with rectangular ramp controlled intake:l

2.1.13: Air Intake Buzz:

Buzz is an unsteady flow phenomenon associated with the External and mixed compression inlets.

Buzz is a low-frequency, high amplitude pressure oscillation that is linked to shock/boundary layer or shock/shock interaction caused during sub-critical operation of inlets as shown below:



Buzz causes flow separation and choking of air intake. To avoid buzz, the external compression inlets are often designed to operate in slightly super-critical condition.

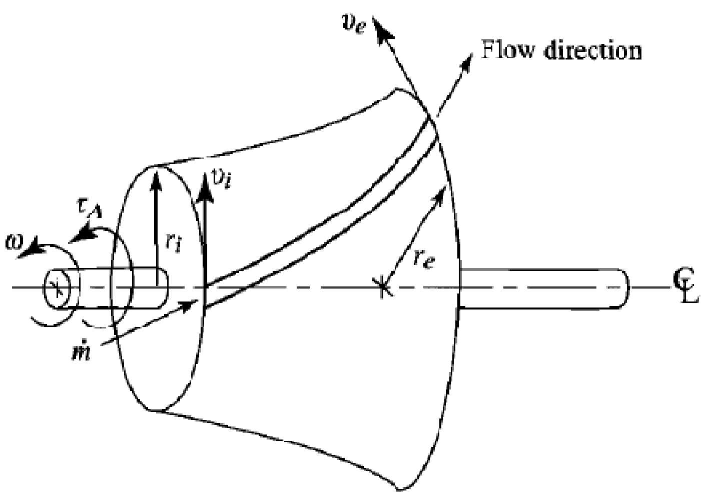
When buzz occurs in a mixed compression inlet, the inlet will unstart and engine flameout is possible.

Compressors & Turbines:

2.2:Euler's Turbo-machinery Equations:

In turbo-machinery, power is added to (as in compressors & pumps) or removed (as in turbines) from the fluid by the rotating components. These rotating components exert forces on the fluid which change both the energy and the tangential momentum of the fluid.

Euler's Pump Equation: Consider an adiabatic flow of fluid in to a compressor or pump, as shown below:



The fluid in a stream tube enters the control volume at radius r_i with tangential velocity v_i and exits at radius r_e with tangential velocity v_e. For a compressor or pump with steady flow, the applied torque τ_A is equal to the change in the angular momentum of the fluid, which is

$$\tau_A = \frac{\dot{m}}{g_c}\,[r_e v_e -- r_i v_i]$$

The input power is $\dot{W}_c = \omega\ \tau_A = \frac{\dot{m}w}{g_c}\,[r_e v_e -- r_i v_i]$

This equation is known as Euler's pump equation. Application of the first law of thermodynamics to the flow through the control volume gives

$$\dot{W}_c = \dot{m}(h_{te} -- h_{ti})$$

Combining above two equations, we get,

$$(h_{te} -- h_{ti}) = \frac{\omega}{g_c}\,[r_e v_e -- r_i v_i]$$

Similarly, for a steady flow turbine, the output torque τ_o is equal to the change in angular momentum of the fluid, or

$$\tau_o = \frac{\dot{m}}{g_c}[r_i v_i -- r_e v_e]$$

The output power $\dot{W}_t = \omega\ \tau_o$, or

$$\dot{W}_t = \frac{\dot{m}\omega}{g_c}[r_i v_i -- r_e v_e]$$

This equation is often referred to as Euler's turbine equation. Application of first law of thermodynamics to the flow through the control volume in a turbine gives,

$$\dot{W}_t = \dot{m}[h_{ti} -- h_{te}]$$

Combining above two equations give,

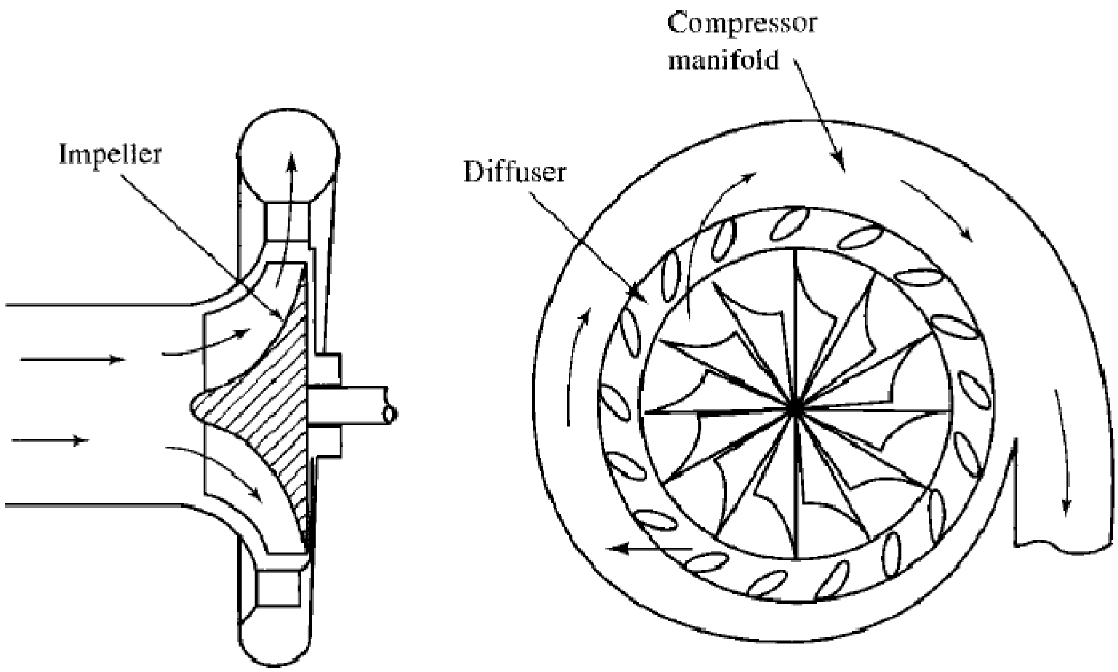
$$h_{ti} -- h_{te} = \frac{\omega}{g_c}[r_i v_i -- r_e v_e]$$

2.2.1: Compressors:

Types of Compressors: Two types of compressors are used in gas turbine engines, the centrifugal compressor and the axial flow compressor. The axial flow compressor allows multi-staging and is more popularly used in present day engines.

Centrifugal Compressor-Construction: : The centrifugal compressor consists of four main parts.

- **The inlet casing with converging nozzle**: The incoming air from air inlet is accelerated by the converging nozzle and is guided in to the impellor inlet. The outlet of the inlet casing is known as the eye.
- **Impellor**: Energy transfer takes place in the impellor (rotor) which rotates at high speeds. The kinetic energy and static pressure of the air rises due to the rotational motion of the impellor
- **Diffuser**: Diffuser (stator) receives the high energy air coming out of the impellor. Diffuser constitute a number of diverging passages, where the kinetic energy of the air is transformed into static pressure
- **The outlet manifold**: It comprises of a fluid collector known as volute, which guides the air from the outlet of impellor in to the combustion chamber.



Operation:

- Air enters the compressor near the hub of the impellor and is then compressed by the rotational motion of the impellor.
- The compression occurs by first increasing the velocity of the air (through rotation) in the impellor. The rotating impellor imparts high velocity to the air. The flow also experiences a centripetal acceleration due to the pressure head. Hence, the static pressure of the air increases from the eye to the tip of the impellor.
- The diffuser has a divergent passage which transforms the high kinetic energy of the air at the outlet of the impellor in to static pressure.

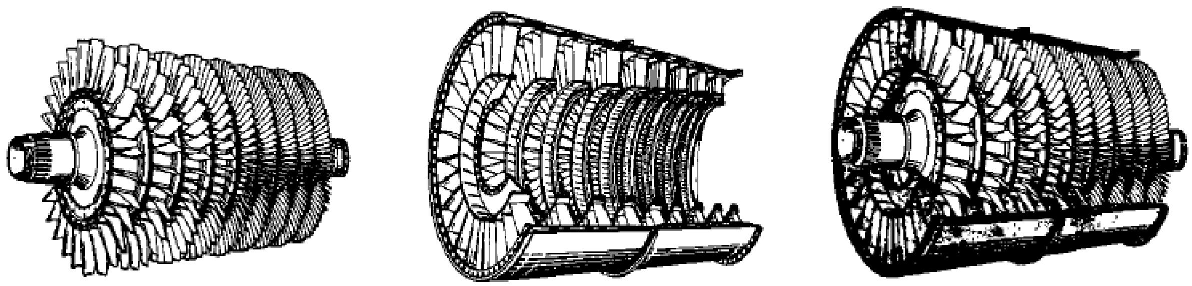
Thus the rotating impellor imparts high velocity and increases the static pressure to the air, while the diffuser slows down the air converting velocity in to static pressure.

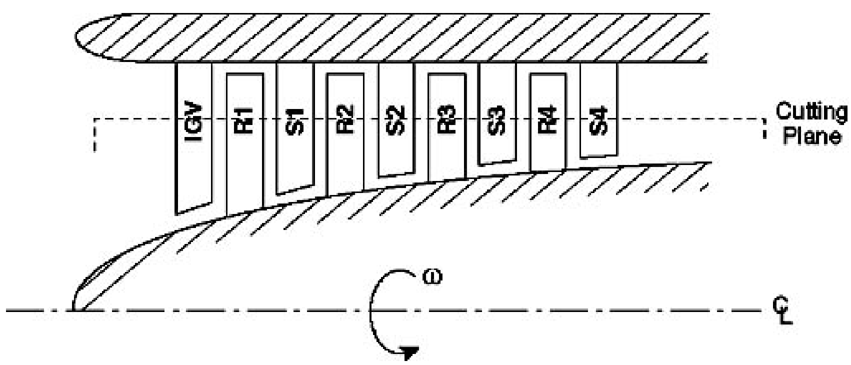
A pressure ratio of 4:1 can be achieved in a single stage centrifugal compressor.

Axial Flow Compressors:

Construction: The axial flow compressor consists of alternating sequence of fixed and moving blades. The fixed blades are attached to the outer casing and are called the stator. Moving blades are attached to the spindle and are called the stator. One set of stator blades and one set rotor blades are called a stage.

A set of stator blades called the inlet guide vanes (IGV) are fitted ahead of the first stage. The function of IGV is to guide the air correctly into the first stage rotor. Similarly, one to three rows of stator or straightener blades are installed after the last stage to straighten and slow down the air before it enters the combustion chamber. The IGV passages are slightly convergent and the velocity increases slightly.





Operation:

The rotating blades of the rotor impart kinetic energy to the air by doing work on the air. The static pressure also rises due to the divergent passages of the rotor. The high kinetic energy of the air is converted in to static pressure rise in the divergent passages of the stator.

Each stage of axial flow compressor produces a small pressure ratio of 1.1:1 or 1.2:1, at a high efficiency. For achieving high pressure ratios of around 12:1, multiple stages are used. For a single rotational speed, there is a limit in balance of operation between the first and the last stage. To obtain more flexibility and uniform loading of each stage, a dual compressor with two different rotational speeds is generally used.

The annulus area decreases along the axial axis in the direction of flow. Since the mass flow rate $\dot{m}$= ρAV, the density ρ increases in the direction of flow as pressure rises. Hence to keep V constant, we need to reduce the annulus area, as the flow progresses to the high pressure stages.

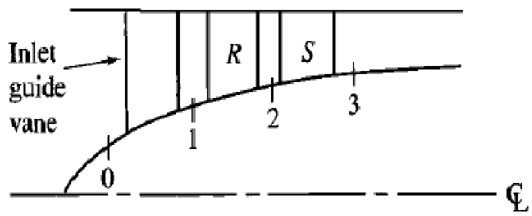
Comparison between Centrifugal & Axial flow compressors

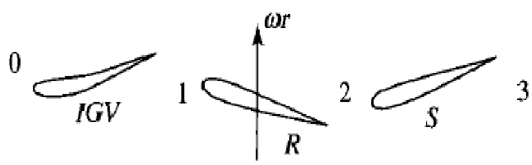
- Centrifugal compressor offers high pressure ratios of up to 4: 1 in a single stage. The axial flow compressor offers much smaller pressure ratios per stage, up to 1.2:1. Centrifugal compressors are more suited for smaller gas turbine engines.
- Axial flow compressors have air flowing through axially, therefore multi staging is possible. High pressure ratios of 12 and above are possible with use of axial flow compressors. Multi-staging is not feasible with centrifugal compressors since the air is turned and discharged radially outwards. Therefore, multi-staging increases the frontal area, hence not feasible for aircraft application.
- Centrifugal compressors are rugged in construction. They can operate efficiently over a wide range of mass flow rates and speeds. Axial flow compressors are sensitive to off design conditions.
- Centrifugal compressors are simple to manufacture at low cost. Axial flow compressors need accurate manufacturing and design specifications. For the same pressure ratio, centrifugal compressors have low weight. They have low starting power requirements.
- Centrifugal compressors are bulky and have large frontal area for given mass flow.
- Axial flow compressors offer high peak efficiency at design point.
- Axial flow compressors offer high ram efficiency since the air flows parallel to engine axis.

2.2.2: Stage & Blade geometry:

One set of stator blades and one set rotor blades are called a stage.

A set of stator blades called the inlet guide vanes (IGV) are fitted ahead of the first stage. The function of IGV is to guide the air correctly into the first stage rotor. Similarly, one to three rows of stator or straightener blades are installed after the last stage to straighten and slow down the air before it enters the combustion chamber. The IGV passages are slightly convergent and the velocity increases slightly.
A cross section and top view of a axial flow compressor stage is shown below:



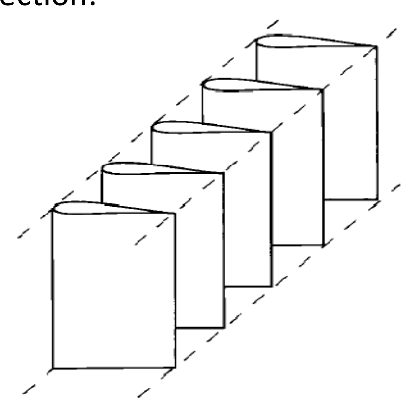


A row of inlet guide vanes is used deflect the incoming airflow to a pre-determined angle towards the direction of rotation of the rotor. The rotor increases the angular velocity of the fluid, resulting in increases in total temperature, total pressure and static pressure. The following stator decreases the angular velocity of the fluid, resulting in an increase of static pressure, and sets the flow up for the following rotor. A compressor stage is made up of a rotor and stator.

2.3.1: Cascade:

The basic building block of aerodynamic design of axial flow compressors is the **cascade.** The cascade is an endless repeating array of airfoils, that results from "unwrapping" of the stators and rotor airfoils. Each cascade passage acts as a diffuser, and the changes in the fluid velocity induced in the blade rows of the stator and rotor are same as that taking place in the cascade sections, upstream and downstream.

A cascade Section:



The cascade is mountedon a turntable so that its angular direction relative to the inlet can be varied in the wind tunnel. Pressure, velocity and flow angles downstream are measured.

2.3.2: Velocity Triangles:

Following notation is used for representing the velocity of flow through the stage:

$$V = V_R + U$$

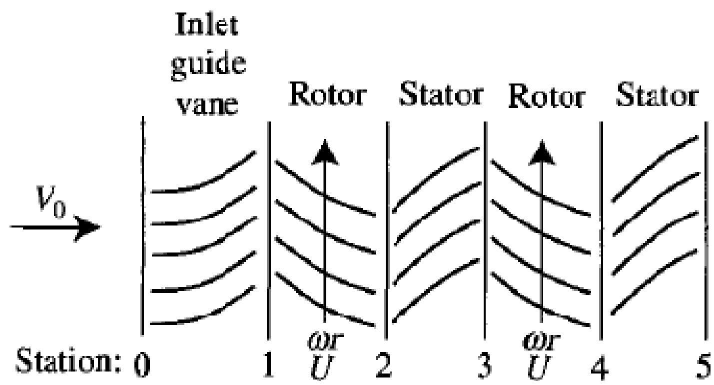
Where

V = velocity of flow in a stationary coordinate system, or **absolute velocity**

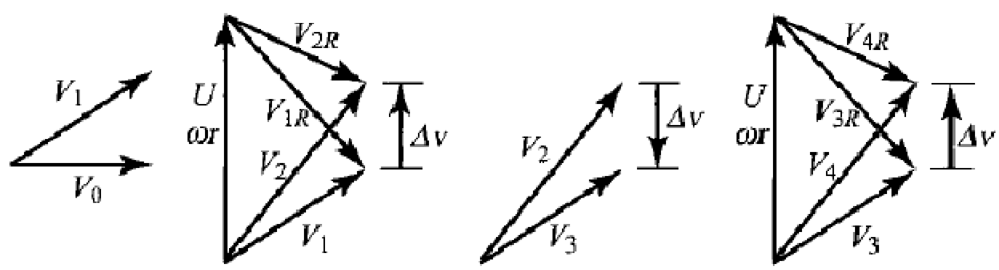
V_R = velocity in a moving coordinate system, or **relative velocity**

U = velocity of a moving coordinate system, **rotational velocity (= ωr)**

Two repeating stages of compressor with a set of IGVs are shown below, along with station numbers:



The velocity diagrams for above repeating stages are



V_1 is the **absolute velocity** entering the rotor at station 1.

V_{1R} is the **relative velocity** of flow entering rotor at station 1, obtained by subtracting the rotor speed ωr from V_1 vectorially.

The **rotor blade passages** act as diffusers and **reduce** the **relative velocity from V_{1R} to V_{2R}**. The static pressure increases from P_1 to P_2.

The absolute velocity of flow at station 2, $\mathbf{V_2}$ is obtained by vectorial addition of $V_{2R}$and ωr. The absolute velocity increases in the rotor.

The stator diffuses the velocity to V_3, increasing the static pressure from P_2 to P_3.

The stator is designed such that velocity is diffused such that V_3 is equal to V_1. Therefore, the velocity triangle of the second stage is a repeat of first stage.

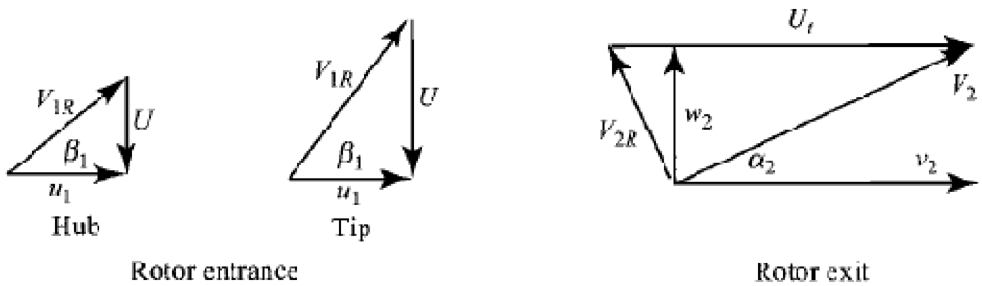


Fig. 9.44 Velocity diagrams for radial vaned centrifugal compressor.

2.3.3: Stage Performance:

- ΔV is the change in the tangential velocity
- Based on Euler's formula, work done by compressor Wc is given by
- Wc = U ΔV
- flow angles at rotor inlet are $\alpha 1$ & $\beta 1$ and at rotor outlet $\alpha 2$ & $\beta 2$
- From the velocity triangles, work done& temperature rise per stage is equal to

Wc = U Va(tan α_2 ? tan α_1) = Cp ΔT
The input energy will reveal itself as rise in stagnation temperature of the air.
The work done above is also equal to rise in stagnation enthalpy of the air.

$$h_{02} - h_{01} = U\Delta C_w$$

$$T_{02} - T_{01} = \frac{U\Delta C_w}{c_p} \Rightarrow \frac{\Delta T_o}{T_{01}} = \frac{U\Delta C_w}{c_p T_{01}}$$

Using isentropic relationship, we obtain the pressure and temperature rise per stage
in terms of whirl velocity change
Stage pressure ratio is given by

In terms of pressure ratio,

$$\frac{P_{03}}{P_{01}} = \left[1 + \eta_{st}\frac{\Delta T_o}{T_{01}}\right]^{\gamma/(\gamma-1)}$$

This can be combined with the earlier equation to give,

$$\frac{P_{03}}{P_{01}} = \left[1 + \eta_{st}\frac{U\Delta C_w}{c_p T_{01}}\right]^{\gamma/(\gamma-1)}$$

High pressure ratio per stage can be obtained by

- High blade speed U – limited by blade stresses
- High axial velocity
- High fluid deflection (change in whirl velocity component) -- limited by aerodynamic considerations & adverse pressure gradient
- Fluid deflection can be increased by increasing the difference $\boldsymbol{\beta}2$? $\boldsymbol{\beta}1$

2.3.4: Degree of Reaction:

The degree of reaction is defined as

$$R_x = \frac{staticenthalpyriseintherotor}{staticenthalpyriseinthestage} = (h_2 -- h_1)/(h_3 -- h_1)$$

For a calorifically perfect gas, the static enthalpy rise is equal to the static temperature rise. Since variation of C_p over the relevant temperature range is negligible, the degree of reaction can also be expressed in terms of temperature rise as

$$\boldsymbol{R_x = {T2 - T1}/{T3 - T1}}$$

Diffusion takes place both in the rotor as well as the stator and the static pressure rises in both rotor and stator. Degree of reaction provides a measure of the extent to which the rotor contributes to the overall pressure rise of the stage.

The degree of reaction is a useful concept in compressor design and it is possible to obtain a formula for it in terms of the various velocities and angles associated with the stage. This will be done in most common case, in which it is assumed as

(a) the absolute axial velocity V_a is constant through the stage and

(b) the air leaves the stage with the same absolute velocity with which it enters ie $V_3 = V_1$.

It can also be shownthat $R_x = \frac{1}{2} - \frac{V_a}{2U}(\tan\alpha_1 - \tan\alpha_2)$

Special Case: When Rx= 0.5: It gives α_1=β_2 and α_2=β_1, the velocity triangles are symmetric, equal pressure rise in rotor and stator. Also the velocities V_1=V_{2R} and V_2=V_{1R}

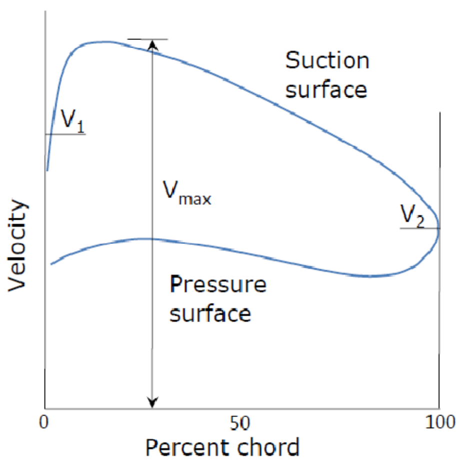
In general, it is desirable to have degree of reaction in the vicinity of 0.5.

The entry and exit triangles will be identical, with α_1 = β_2 and α_2 = β_1

2.3.5: Diffusion Factor:

Fluid deflection ($_2$--$_1$) is a parameter that affects the stage pressure rise. Excessive deflection ie high rate of deflection leads to blade stall. Diffusion factor associates blade stall with deceleration on the suction side of the aerofoil section. Diffusion factor is measured on the suction side of the blades, and is expressed as below:

Diffusion Factor D^* = (V_{max}--V_2)/V_1, where V_{max} is the ideal surface velocity at the minimum pressure point and V_2 is the ideal velocity at the trailing edge and V_1 is the velocity at leading edge.



2.3.6: Stage Loading Coefficient: The ratio of stage work to the square of rotor speed is called the stage loading coefficient.

$$\psi = \frac{g_c c_p \Delta T_t}{(\omega r)^2} = \frac{g_c c_p \Delta T_t}{U^2}$$

Flow Coefficient: The ratioof the axial velocity to the rotor speed is called the flow coefficient and is defined as

$$\phi = C_a / U$$

The flow coefficient for modern axial flow compressor of aircraft gas turbine engines are in the range of 0.45-0.55.

The flow coefficient variation will cause changes in the incidence of the flow over the blade

Work Done Factor λ (Loss due to blockage in compressor annulus area): Because of the adverse pressure gradient in the compressors, the boundary layers along the annulus walls thicken as the flow progresses. The main effect is to reduce the area available for the flow below the geometric area of the annulus.

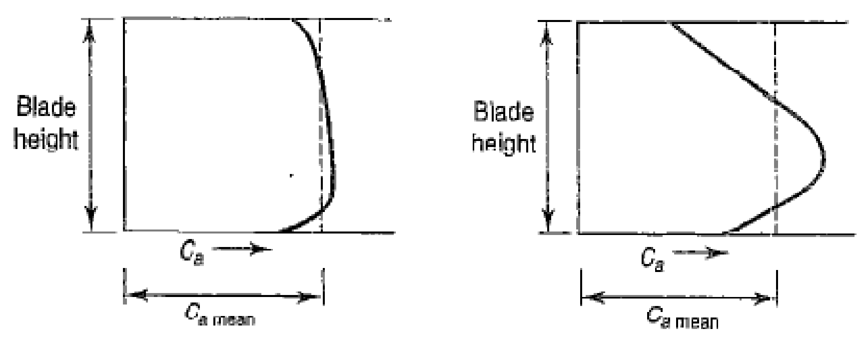
The stage temperature rise is always less than the design value. The reason for this is that the radial distribution of the axial velocity is not constant across the annulus, but becomes increasingly peaky as the flow proceeds, settling down to a fixed profile at about the fourth stage. The change in the axial velocity affects the work absorbing capacity of the stage.

The reduction in the work capacity is accounted for by use of the work done factor λ, which is less than unity. The actual stage temperature rise is given by

$$\Delta T_{0S} = \frac{\lambda}{C_p} UC_a(\tan{}_1 -- \tan{}_2)$$

The mean work done factor will vary across the compressor stages due to the variation in the axial velocity. The axial velocity distribution along the blade height in the first stage and the fourth stage is shown below. The variation of mean work done factor across the stages is also shown.

Axial Velocity Distributions: a) 1st Stage and b) 4th Stage



(a) (b)

2.3.7: Types of Flow Analysis:

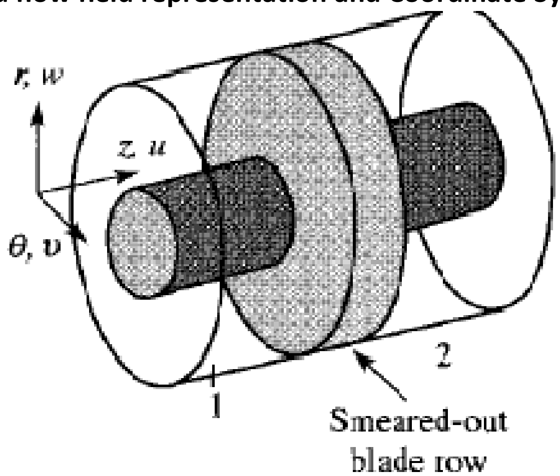
Flow Analysis: The flow of working fluid through the compressor is inherently three-dimensional. This complex flow is analysed by dividing the flow in to three two dimensional fields. The complete flow field is the sum of these less complex two dimensional flow fields.

The two dimensional flow fields are called the **through-flow field**, the **cascade field** (or the blade to blade field), and the **secondary flow field**.

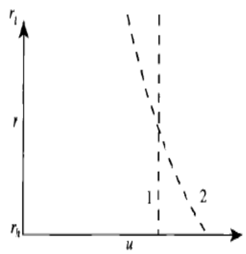
The Through-flow Field: The through-flow field is concerned with the variation in the fluid properties in only the radial r and axial z direction. As a result of through-flow analysis, we obtain the axial, tangential and radial velocities as a function of z and r.

When axial velocity changes between successive stages as the flow proceeds, conservation of mass requires that a downward flow of fluid occur between stations 1 and 2.

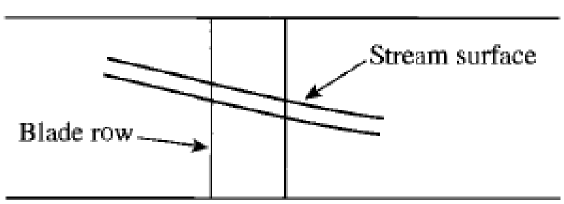
Through-field flow field representation and Coordinate System:



As a result of the flow field, axial velocity along the height of the blade will follow a profile as shown below:



This change in the axial velocity along the height of the blade, causes the flow to turn downwards along the axial direction, as shown below:

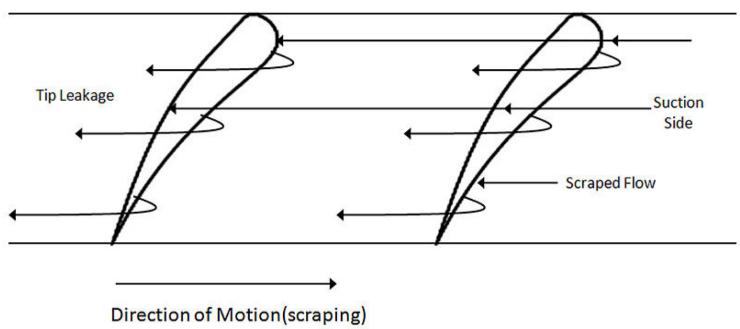


Cascade Flow Fields: The cascade field considers the flow behavior along **stream surfaces and tangentially through blade rows. (in the direction of zand θ**). The most common method of obtaining performance data for different blade profiles is to run cascade tests.

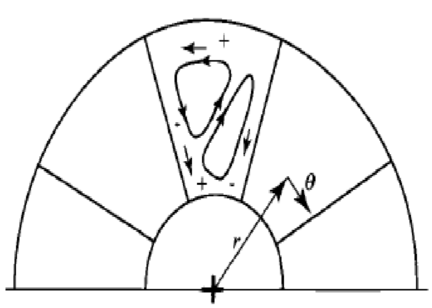
A cascade is a stationary array of blades. A flow through cascade is a row of blades representing the blade ring of the compressor. These blades can be arranged in straight line or annular, thus representing an actual blade row, these arrangements are known as "**rectilinear cascade**" and "**annular cascade**" respectively. The "annular cascade" is more towards a real-life situation.

Cascade has porous end walls to remove boundary layer for a two dimensional flow. The three dimensional flow is reduced to two dimensional plane flow in which variations occur only in pitch-wise and stream-wise directions only. Radial variations (along blade height) in the velocity field are therefore excluded.

Measurement usually consists of pressures, velocities and flow angles downstream of the cascade. C_p distribution is measured and plotted against $\frac{X}{C}$, the chord-wise distribution.



Secondary Flow Fields: The secondary flow field exists because the **fluid near the solid surfaces (in the boundary layer)**, ie the blade surfaces and passage walls **has a lower velocity than that in the free stream** (external to the boundary layer). The pressure gradients imposed by the free stream will cause the fluid in the boundary layer to flow from regions of higher pressure to regions of low pressure.



2.4: Dimensionless and Corrected Component Performance Parameters:

Purpose: Dimensional analysis identifies correlating parameters that allow data taken under one set of conditions to be extended to another set of conditions. These parameters are useful and necessary because it is always impractical to accumulate experimental data for a number of possible operating conditions.
The quantities of pressure and temperature are made dimensionless by dividing them by the respective sea-level static conditions. These are called the corrected parameters. The dimensionless pressure and temperature are represented by δ and θ respectively.

$$\delta_i = \frac{P_{ti}}{P_{ref}}$$

$$\theta_i = \frac{T_{ti}}{T_{ref}}$$

The corrected mass flow rate at engine station "i" , used in the performance analysis is defined as

$$\dot{m}_{ci} = \frac{\dot{m}_i}{\delta_i}\sqrt{\theta_i}, \text{ and}$$

The corrected mass flow rate is a function of Mach number alone. A reduction in the engine power (throttle) setting will lower the Mach number and the corrected mass flow ratein to the engine compressor or fan. Since the entrance condition to turbine and the exhaust nozzle is choked, the corrected mass flow entering these stations is constant. However, when the afterburner is engaged in turbojet or turbofan engine, the nozzle throat area needs to be increased to maintain the corrected mass flow rate increase.
The corrected engine speed at station "i' is defined as

$$N_{ci} = \frac{N}{\sqrt{\theta_i}}$$

The corrected engine speed is related to the blade speed and hence the blade Mach number.
Three additional corrected parameters are used the performance analysis. They are the corrected thrust, corrected thrust specific fuel consumption and corrected fuel mass flow rate.
For gas turbine engines operating at maximum turbine entry temperature, the corrected thrust is a function of only the corrected free stream total temperature. Similarly, the corrected thrust specific fuel consumption and the corrected fuel mass flow rate depend on the flight condition and throttle setting.

2.6: Multistage Operation-Off design conditions:

Let us consider the inlet and outlet stations of a multistage compressor as 1 & 2 respectively

The overall pressure ratio of the compressor will be $\frac{P_{02}}{P_{01}}$

The compressor outlet pressure P_{02} and isentropic efficiency η_c depend upon several physical variables

P_{02}, η_c = f [$\dot{m}$, P_{01}, T_{01}, γ, Ω, R, v, D]

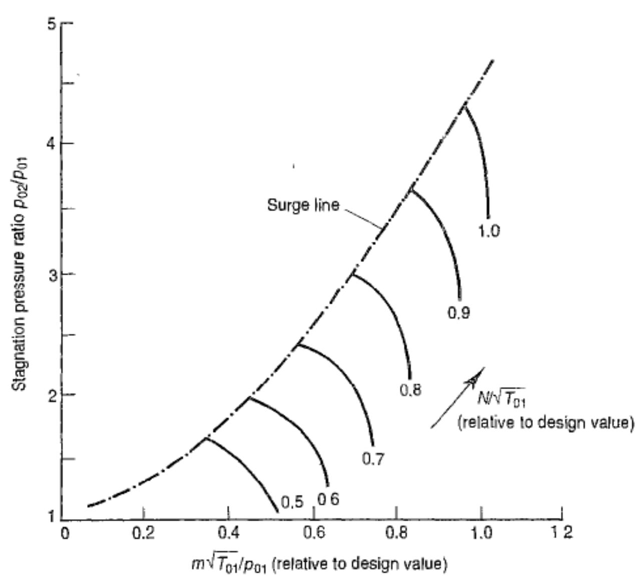
The above variables are grouped in to non dimensional parameters,

$\frac{P_{02}}{P_{01}}, \eta_c$ = f [$\frac{\dot{m}\sqrt{T_{01}}}{P_{01}}, \frac{N}{\sqrt{T_{01}}}$], for a given design, assuming D, R are fixed

Then $\frac{P_{02}}{P_{01}}, \eta_c$ = f [$\frac{\dot{m}\sqrt{T_{01}}}{P_{01}}, N/\sqrt{T_{01}}$]

Of the above non dimensional parameters, the first one denotes the mass flow rate and the second the speed. Compressor overall pressure ratio and isentropic efficiency are plotted against the non dimensional parameters, as below:

Plot of $\frac{\dot{m}\sqrt{T_{01}}}{P_{01}}$ vs $\frac{P_{02}}{P_{01}}$ and η for varying speeds $N/\sqrt{T_{01}}$

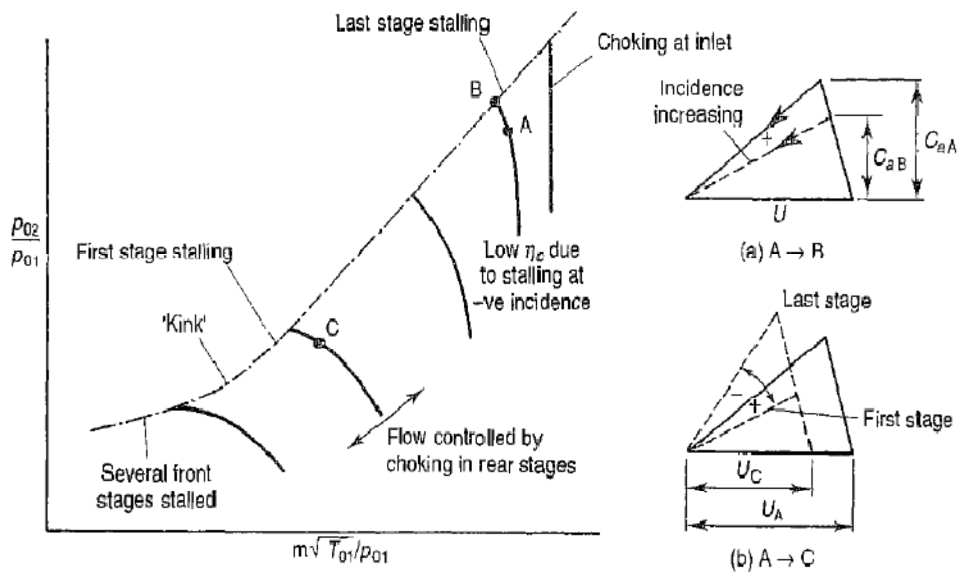


2.6.1: Limits on Compressor performance:

- For a given speed, the range of mass flow for stable operation is very narrow.
- At high rotational speeds, the constant speed lines become very steep, almost vertical.
- The limitations at either end of speed lines are surging and choking.

- The surge line denotes the locus of unstable operation of the compressor.
- Surge is characterized by violent, periodic oscillations in the flow. Surge may lead to flame blow-out in the combustion chamber. Surge can lead to substantial damage to compressors and must be avoided. The operating line of the compressor is therefore kept slightly away from the surge line, thereby maintaining surge margin.

Off-Design Working of Compressor:



When the speed is reduced from point A to C, the mass flow falls off more rapidly than the speed (N&U), and the effect is to decrease the axial velocity at the inlet. This causes the incidence angle of the front stages to increase leading to stalling.

The effect on rear stages will be different. The speed reduction below design speeds, the temperature rise and pressure rise will be lower than the design value. The density will reduce, increasing the axial velocity (to compensate for drop in $\boldsymbol{\rho}$ in quantity $\boldsymbol{\rho}$ACa). This will cause choking of the rear stages.

Thus at low speeds, mass flow rate is limited by the choking of rear stages.

As speeds are increased, the density increase, causing the rear stages to un-choke, but eventually, the choking will occur at the inlet.

If we consider moving from the design point A to point B on the surge line at the design speed, the mass flow rate is slightly reduced (although there is a marginal increase in the pressure ratio), causing axial velocity to reduce. This increases the incidence, leading to rotor blades stall. This effect is severe on the rear stages. Surge at high speeds is caused by stalling of rear stages.

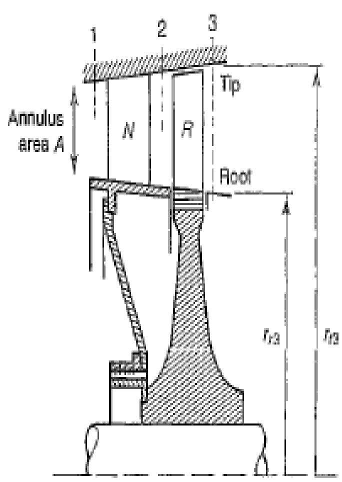
2.6.2: Solutions to increase stability of operation during off-design conditions:

Twin Spool Arrangement: Reduction in speed increases the incidence of the front stages, while decreasing the incidence of rear stages. The incidence could be maintained by running the rear stages at higher speed than the front stages These conflicting requirements of speeds could be met by splitting the compressor in to two or more sections. The common twin spool arrangement allows more stable operation at off-design conditions.

Variable Geometry Compressor: An alternate approach is to use several rows of variable-stator at the front of the compressor, permitting pressure ratios of up to 16:1, using a single spool arrangement.

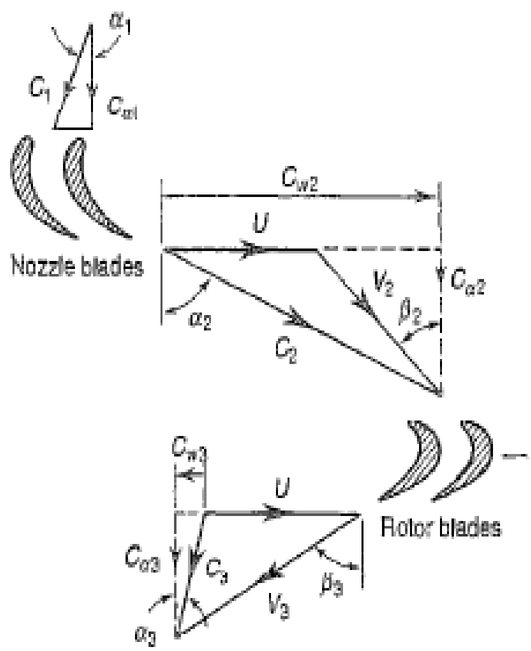
2.6: Working Principle of Axial Flow Turbine: A stage of axial flow turbine consists of a stator nozzle and a rotor. The flow of gas comes from the combustion chamber with high internal energy (T_{01}, P_{01} and V_1) and is made to pass through the stator where a large part of its internal energy is converted to kinetic energy.

The transfer of energy occurs in the rotor as the high speed gas flow impinges on the rotor blade, and as the flow is made to turn while flowing through the passage between the blades. The turning of the gas produces a change in the momentum of the gas which creates an impulse force causing the rotation of the rotor.



Turbines using the fundamental principle of impulse force for making the blades rotate are called impulse turbines. The amount of energy given up by the gas is decided by the energy level of the incoming gas, and by the amount of turning executed by the gas in the blade passages. The energy transfer occurs as per the Newton's laws of Motion, based on the rate of change of momentum of the gas in the direction of blade rotation.
Based on the principle of energy transfer, we have two types of axial flow turbines.
Impulse Turbines: High energy flow is first accelerated in the stator nozzle and made to impinge on the rotor blade with high momentum and then made to take a huge turn through the passage between the rotor blades.
Reaction Turbines: The flow is accelerated in the rotor blades by making a converging blade passage in addition to the large turning. Jet effect creates a reaction force as per Newton's third law.
All the gas turbines used in aircraft are reaction turbines.

2.6.2:Velocity Diagrams: The velocity diagrams are given below:



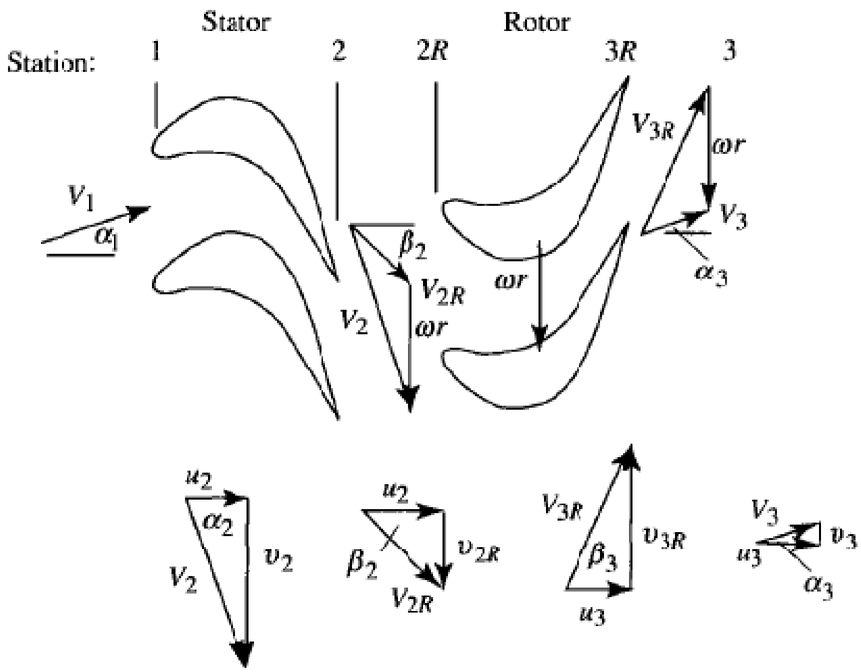


Fig. 9.49 Velocity triangles for a typical turbine stage.

The stator nozzle accelerates the flow where the absolute velocity increases. Rotor blades turn the flow, reducing the absolute velocity, while accelerating the flow by increasing the relative velocity.

- Gas enters the stator nozzle blades at an angle $\alpha_1$with an absolute velocity C_1. The absolute velocity C_1 is increased to C_2 in the stator nozzle blades.
- The rotor blades turn the gas reducing the absolute velocity to C_3 leaving the rotor at an angle α_3. The change of momentum produces the impulse force.
- The relative velocity V_2 increases to V_3 in the converging passages of the rotor in the reaction turbines. However, in the impulse turbines V_3=V_2.
- The rotor blades are designed such that the flow is turned near axial direction for entering the next stage.
- The first stage stator nozzle blades are designed such that gas leaves the stator with a local mach number near to unity. M_2 is therefore equal to unity.

- However, gas acceleration in the rotor is such that V_3 is subsonic. ie M_3= $\frac{V_3}{a_3}$< 1.0. This condition is necessary to avoid high stresses due to shock formation in rotating blades.
- The whirl components of absolute and relative velocities have major role in the work extraction.
- The work done in a gas turbine may be increased by increasing the turbine entry temperature T_{01}.

The following points are important:
In a single stage turbine, C_1 will be axial, ie α_1=0 and C_1=C_{a1}.
If on the other hand, in case of a multi-stage axial turbine, then C_1 and α_1 are designed to be equal to C_3 and α_3 so that the same blade shapes can be used for successive stages.
Because the blade speed increases with increase in radius, the shape of velocity triangles vary from root of the blade to the tip. The above velocity triangles are drawn for a mean diameter.
The quantity (C_{w2}+C_{w3}) represent a change of whirl (or tangential) component of the absolute velocity and represents the change of momentum per unit mass flow of the fluid.
The annulus of the turbine is flared to accommodate the decrease in density as the gas expands through the turbine. This will keep the axial flow velocity $C_a$is kept constant through the turbine.

The geometry of velocity triangles gives the following:
$\frac{U}{C_a}$ = tan α_2 -- tan β_2 = tan β_3--tan α_3,
Alternatively, (tan α_2+ tan α_3)=(tan β_2+ tan β_3)
Applying the principle of angular momentum to the rotor, the stage work output per unit mass flow of fluid is
W_s = U(C_{w2}+C_{w3}) = UC_a(tan α_2+ tan α_3) = UC_a(tan β_2+ tan β_3)
The work-done factor as we applied in compressor is not necessary in case of turbine because the flow is accelerating and there is no adverse pressure gradient in turbine. Therefore the growth of boundary layer along the annulus walls is very much less. /
For a steady flow energy state, W_s=$C_p\Delta T_{0S}$, where ΔT_{0S} is the stagnation temperature drop across the stage.
Hence, $C_p\Delta T_{0S}$ = UC_a(tan β_2+ tan β_3)

We usually use C_p=1.148 kJ/kg K and γ = 1.333 and $\gamma/(\gamma—1)$ = 4., R= 0.287 kJ/kg K

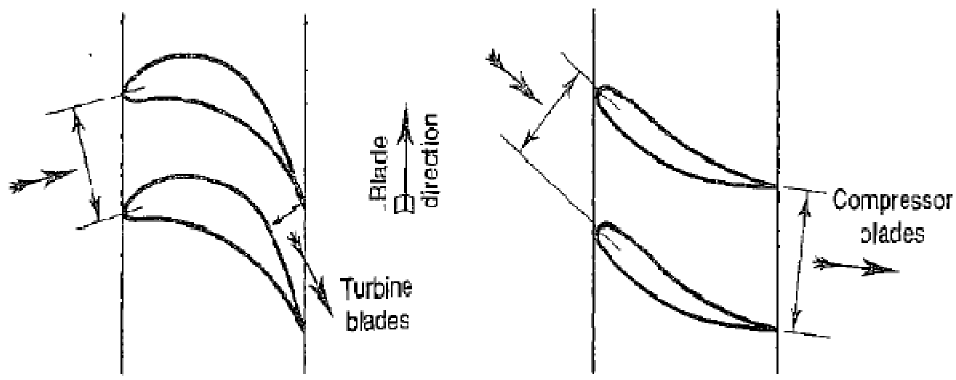
$\Delta T_{0S}=\eta_s T_{01}[1-(\frac{1}{p_{01}/p_{03}})^{\frac{\gamma-1}{\gamma}}]$ where η_s is the isentropic stage efficiency based on stagnation or total temperature of the stage and is referred to as **total-to-total stage efficiency**.

2.6.4: Similarities & Differences- Axial Flow Compressors and Turbines:

1. The gas flow is accelerated through the turbine while the flow is decelerated through the compressor.
2. The blade-to-blade passage is **convergent in the turbine** while the passage is **divergent in the compressor**.
3. The gas flow is **speeded up in stator nozzle of turbine** while the flow is **diffused in the stator of a compressor**.
4. The flow faces an **adverse pressure gradient in the compressor** while the flow encounters a **pressure drop in the turbine.**
5. **Work is done on the gas by the compressor** while **work is extracted from the gas in the turbine**.
6. Due to the adverse pressure gradient, the **number of stages in the compressor are higher** while the **turbine** extracts the work, as required to drive the compressor, **in lesser number of stages**. Therefore, usually the axial flow compressor has 3-20stages while the turbine has 1-4stages.
7. The **blade height is higher in the compressor** than in the turbine.
8. The **annulus area in a multi-stage axial flow compressor is decreasing** while it is **increasing in the turbine**.
9. The **per stage temperature is increasing in small increments in compressor** while the per stage temperature reduces by a **large decrease in turbine**.
10. The **compressor operates at a lower temperature** (between 500-1000 R) while the **turbine operates at a much higher temperature** (1000-3500 R)
11. The **flow tends to separate due to adverse pressure gradient in the compressor**, while the **turbine flow negotiates a reducing pressure gradient**. The turbine efficiencies are higher (usually >0.9), while compressors operate with efficiencies between 0.8-0.9.
12. The **flow deflections are large in turbine** than in the compressor.

13. The stagnation properties of enthalpy, temperature and pressure increase in the compressor while they decrease in the turbine.

2.6.5: Blade Profiles-Turbines & Compressors:



1. Both turbine and compressor blades are aerofoil sections.
2. The turbine blades are thicker than the compressor blades to withstand very high temperatures. Turbine blades need to be thicker to allow for cooling passages.
3. The aerofoil blade sections are defined usually by a set of 11 parameters, including aerofoil radius, axial and tangential chords, inlet and exit blade angles, leading and trailing edge radii together with number of blades and throat area between blade passages.
4. Each stage is defined by three radii, at the hub, mean and tip sections.
5. Initially the blade sections at hub, mean and tip are decided and then the blade shape along the height is defined using at least 10 sections generated through two-dimensional design computations.
6. Once the blade sections are generated, cascade testing is used to calculate velocity and Mach number distributions over pressure and suction surfaces.
7. The sections so generated are stacked to get the blade shape.
8. The blades are designed withstand steady and unsteady stresses. The **steady stresses arise out of centrifugal and pressure loading and**

thermal stresses. Unsteady stresses arise out from interaction between rotating blades and stationary blades, thermal gradients of gas flowing into the turbine from combustion chamber.

9. There will also be mechanical stresses arising out of residual imbalance or bearing wear-out.
10. Turbine blades are also designed for creep.
11. In general, there are types of mechanical stresses for turbo-machines. They are:
    - Centrifugal stresses
    - Gas bending stresses
    - Thermal stresses

2.7.1: Cycle Analysis:
Studiesthermodynamic changes of working fluid. Two types of cycle analysis are conducted while design/manufacturing of gas turbine Engines.

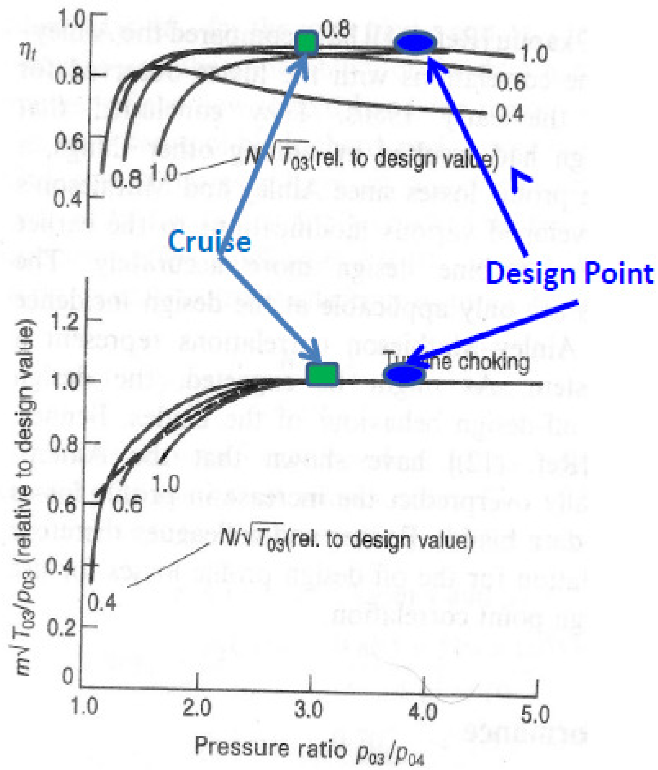
1. Parametric Cycle Analysis
2. Engine Performance Analysis

Parametric Cycle Analysis (PCA) is a design Point or "On-Design" analysis
Engine Performance Analysis is an Off-Design Analysis
Engine Performance Analysis determines:

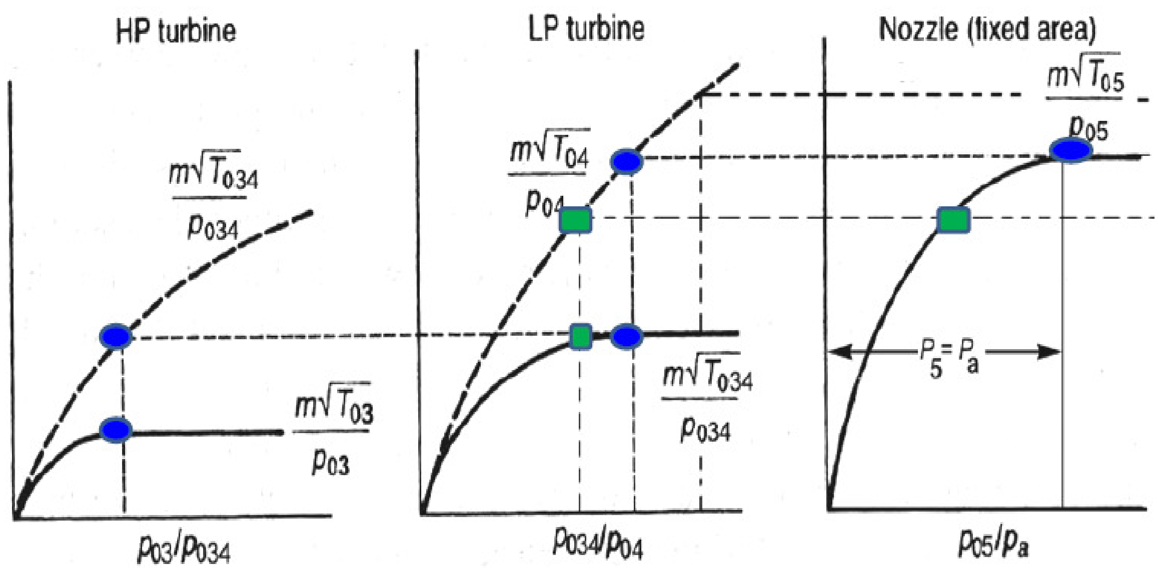
- performance of a Specific engine
- At all flight conditions
- At all throttle settings

2.8: Turbine Maps: To obtain high power/weight ratio from the turbine, the flow entering the first stage rotor is usually supersonic and the sonic condition is reached in the minimum passage area in the stator nozzle. The corrected mass flow rate is based on this minimum throat area.

The performance map of the turbine is drawn between the total pressure ratio and corrected mass flow rate for different corrected speeds and component efficiencies.



Design point matching of turbine and nozzle

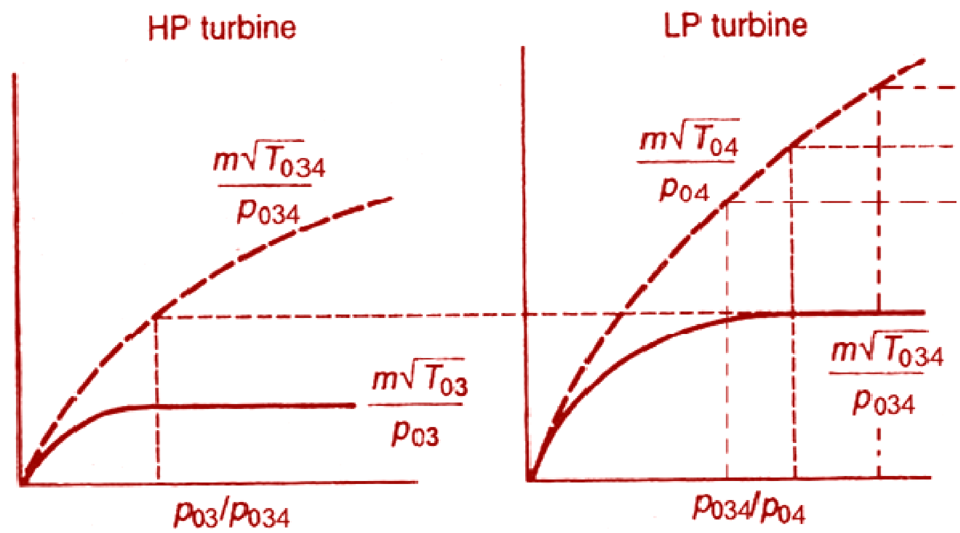


Multi-Spooling:

- Multi-staging of turbine is done extract more energy for mechanical power
- To restrict size and number of stages each stage does more work

- Multi-spooling is done to make the spools rotate at different speeds
- Multi-spooling is needed to connect fan/propeller/rotor

Multi stage operating maps:



A matched LP + HP turbine operation, HP turbine may be choked all the time, as the pressure ratio across the LP turbine change.

Blade Cooling:

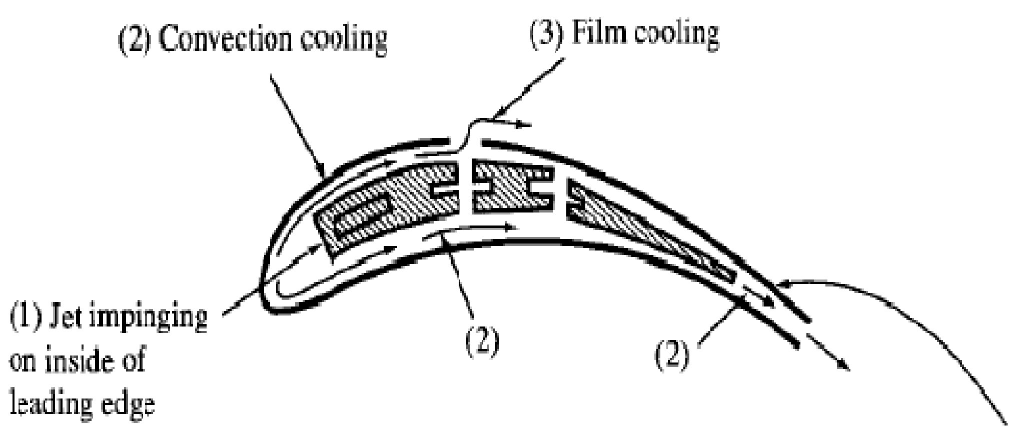
The turbine components are subjected to much higher temperatures in the modern gas turbine engines being designed and built today. This is due mainly due to improvements in metallurgy and cooling of turbine components. The cooling air to cool the turbine is taken from the compressor. The cooling air from compressor is routed through the inner passages provided in the turbine blades and the inner wall of turbine flow passage.
The first stage of the stator nozzle blades are exposed to highest turbine temperatures. The first stage rotor blades are exposed to somewhat lower temperatures because of dilution of the turbine gasses with the cooling air. The turbine temperature decreases gradually over second and later stages due to the lowering of temperatures due to expansion.
The cooling methods used in the turbine are as follows:

1. **Convection Cooling**: The compressor air is used to cool the turbine through forced internal convection cooling which reduces the blade temperature by 200 C. Using current alloys, this permits the turbine inlet temperatures of more than 1650 K. The blades are either cast using

cores to form the cooling passages or forged with holes of any desired shape through laser drilling. The turbine rotor/stator nozzle blades, disks and inner walls of turbine flow passage are cooled using cool air that is routed through inner passageways.

2. **Impingement cooling**: The HP stator nozzle blades are cooled by the cooling air introduced in such a way to produce a jet impingement cooling the inner surface of the very hot leading edge. The spent air after jet impingement leaves the blade through slots or holes to provide film cooling to the outer blade surface and the trailing edge.
3. **Film Cooling**: Slots or holes on the blade surface and the trailing edge allow cooling air to pass through forming a film over the surface reducing the blade temperature to provide film cooling to the outer blade surface and the trailing edge.
4. **Full coverage film cooling**: Slots provided at regular intervals provide full coverage film cooling.
5. **Transpiration Cooling** : A very economical method of cooling where cooling air is forced through a porous blade wall. This method removes the heat more uniformly and the cooling air also forms an effusing layer insulating the outer surface of the blade from the hot gas reducing the rate of heat transfer to the blade. This form of cooling is still in development stage of suitable porous materials and manufacturing methods.

At current levels of turbine inlet temperatures, three or four stages of turbine rotor may be cooled using air bled from compressor.



At current levels of turbine inlet temperatures, three or four stages of turbine rotor may be cooled using air bled from compressor.

To summarise

- Various blade cooling techniques provide various amounts of cooling.
- Maximum cooling is usually applied to first stage (HP) stator which faces highest temperature.
- Cooling is also applied to HP rotors , But the cooling technology applied to rotor is a bit complicated because the rotor is rotating at high speeds
- Modern LP stage stators are also cooled, however they operate at slightly lower temperatures
- Last stage blades do not need cooling since they operate at substantially lower temperatures.

Over the last fifty years more effort has been given to turbine cooling rather than turbine aerodynamics.
As the flow in turbine is always in a favourable pressure gradient, high turbine efficiency is easily obtained.
Amount of local cooling may vary from 50 degrees to nearly 500 degrees in modern blades. Cooling must cater for local temperature fields on the blade surface which differ substantially based on C_p distribution.
Coatings are also applied to the blade surfaces for saving the blades from high temperature.

Stresses developed in gas turbine engines(especially turbines):
The fundamental source of stresses in gas turbine engines is the centrifugal forces developed in the rotating parts.
The most important sources of stresses are:

- Stresses due to bending moments like those due to the lift on the aerofoils or pressure difference across the disc
- Vibratory stresses that occur as the aerofoils pass through non-uniform flows in the wakes of blades upstream. This can be most dangerous when the blade passing frequency coincides with one of the natural frequency of the aerofoils.
- Aerofoil or disc flutter, an aero-elastic phenomenon in which the natural frequency is excited. This is most often found in compressor andfan.
- Torsional stresses that result from transfer of power from turbine to the compressor.
- Temperature gradients occurring due to throttle variations when the engine moving from one power setting to another. These cyclic thermal variations are called thermal or low cycle fatigue.

- Foreign object damage (FOD) or domestic object damage (DOD) that result from external and internal objects need to factored in to the design

Engine Materials:

Several materials commonly used in critical parts of gas turbine engines are Aluminium alloys, titanium alloys, high strength nickel alloys and single crystal super-alloy. The measure of usefulness of these alloys is the strength/weight ratio which is obtained by dividing the material's creep strength by it's density. During it's life, a fan or compressor blade is subjected to billions of half-cycle fatigue cycles due to vibrations. Titanium alloys are usually employed in manufacture of fan and low-pressure compressor blades. Titanium's strength/weight ratio is severely reduced at temperatures beyond 900 F (480 C). Hence, nickel-based alloys are commonly used for high pressure compressor components.

The critical components for turbines are exposed to very high temperatures. Many of these parts require super materials, commonly called super-alloys. They also use compressor air for cooling. Certain materials and environment need protective coatings.

Typical examples of these materials are Mar-M 509, a high chromium carbide strengthened cobalt based super-alloy and Rene 80, a cast, precipitation hardenable nickel based super-alloy.

Many of the newer super-alloys for turbine rotor blades are cast and solidified in such a manner as to align the crystals in the radial direction, called directional solidification, or to produce a single crystal. The resulting turbine blades are capable of operating at temperatures 100 to 200 F above those of conventionally cast blades.

In high-pressure turbines, the blades are typically made of super-alloys while high-strength, nickel based alloys are used for the disk and rim. Since the temperatures are much lower in in the low pressure turbine, the critical components are not cooled and are frequently made from high strength nickel based alloys.

Unit III – Anatomy of Jet Engine II

UNIT III

ANATOMY OF JET ENGINE-II

BURNER: Burners- types, components- function, schematic diagram, airflow distribution, cooling-types, cooling effectiveness, performance parameters, combustion efficiency, overall total pressure loss, exit temperature profile, ignition relight envelope- effect of combustor design, Fuel injection, atomisation, vaporisation, recirculation- flame stabilisation, flame holders. Afterburners, function,

components, design requirements, design parameters, bypass duct, total pressure losses, Mixing process- pressure losses, fuels- composition, specifications of commonly used fuels.

NOZZLE: Exhaust nozzles- primary nozzle, fan nozzle- governing equations of flow- choking, engine back pressure control, nozzle-area ratio, thrust reversal, vectoring mechanisms. Afterburner functions and its components, design requirements and parameters. Performance gross thrust coefficient, discharge, coefficient, velocity coefficient, angularity coefficient, performance maps.

Combustors/Burners:

The thermal energy of the air/fuel mixture flowing through the gas turbine engine is increased by the combustion process. The components of the gas turbine engine where thermal energy is added are

- Main burners (also called burners or combustors)
- Afterburners (also called thrust augmenters or re-heaters)

Combustion process needs thorough vaporizing (atomization) of fuel and mixing of fuel with air before combustion takes place. The air fuel mixture continues to flow through the burners as the combustion is taking place. For complete combustion at given reaction rates of fuel with air, time and space are needed. Therefore, the design of burner, especially the length is very critical. The desirable properties/requirements of combustors are

- Complete combustion
- Low total pressure loss
- Stability of combustion process at all flight conditions (different altitudes and mach numbers)
- Proper temperature distribution at the exit section of the burner, without hot spots
- Short length and small cross section compatible with engine geometry
- Freedom from flameout or flame extinction while operating over wide range of speeds, mass flow rates and pressures and temperatures
- Ability to utilize broader range of fuels
- Durability and re-lighting capability
- Ease of maintenance

Many of the above properties are in competition with each other. For example, small size and complete combustion contradict each other. Low total pressure loss in the given size and turbulence also are contradicting each other. Hence, the design of combustor is a compromise.

Combustion Process: The thermal energy of the air/fuel mixture flowing through the gas turbine engine is increased by the combustion process.

The combustion process occurs due to the chemical reactions between the vaporized fuel and air mixed at a molecular level.

The objective of combustion process is to introduce and burn fuel in the compressed air flowing through combustor with minimum pressure loss and with as complete utilization of fuel as possible.

Pre-Combustion Process: The injection of liquid fuel in the form of a fine atomized spray of droplets enables mixing of fuel with the primary air entering the combustor. This mixing and vaporization of fuel reduces the temperature of the working fluid before combustion since both enthalpy used in raising the temperature of the fuel to boiling point and the latent heat of evaporation of fuel droplets are absorbed from the enthalpy of the incoming air.

Combustion Process:

Compressed air from compressor enters the combustion chamber at a velocity range of 140-150 m/sec. This speed of air is far too high for stable combustion to take place. The first step in the combustion process is to slow down (diffuse) the air to around 20-30 m/sec and raise the static pressure (since the rate of reaction increases with static pressure).

The aviation turbine fuel (kerosene) needs a region of low axial flow velocity and fuel air ratio of around 15:1. However, the actual fuel air ratios prevailing in the combustor are around 50:1 to 150:1, which is not suitable for stable combustion.

So only around 20% of the air at the exit of compressor is admitted in to the combustion zone of the combustor. This zone is called the primary combustion zone. The entry section of the combustor is called the "snout". Immediately after the snout, swirl vanes and a perforated flare are located, through which air passes into the primary combustion zone. The swirling air promotes desired circulation. The remaining part of the compressed air coming out of compressor will pass through annular space between flame tube and casing. This part of the air is called secondary air. The wall of the flame tube, adjacent to the combustion zone, is provided with selected number of secondary holes, through which another 20% of the remaining secondary enters the primary zone. This air interacts with the mixture of vaporized fuel droplets and primary air and creates a region of low velocity recirculation zone. The resulting flow takes the form of a torroidal vortex which has the effect of stabilizing and anchoring the flame.

The re-circulating gases hasten the burning of freshly injected fuel droplets, by rapidly bringing them to ignition temperature. It is arranged such that the conical spray of fuel from the nozzle intersects the re-circulation vortex at it's centre, which assists in breaking up of fuel droplets and mixing it with incoming air.

The temperature of the gases released through the combustion process is about 1800^0- 2000^0 C, which is very high for entry in to the nozzle guide vanes of the turbine.

The air that is not used in the combustion process, around 60% of the total airflow, is therefore introduced progressively in to the flame tube. This zone is called the dilution zone. This air is used to lower the gas temperature in the dilution zone before it enters the turbine and also for providing film cooling of the walls of the flame tube.

An electric spark from an ignitor plug initiates the combustion and the the flame is then self-sustaining throughout the flight.

Chemistry of Combustion:

Characteristics of Combustion:

The **rate of reaction** depends on the static pressure P and temperature T, as given below:

$$\text{Reaction rate} \propto P^n f(\mathrm{T}) \exp \frac{-E}{RT}, \text{ where}$$

n is an exponent that depends on the number molecules involved in the collision/reaction

f(T) is a function that relates to the forms of energy that the molecules have (translation, rotation or vibration)

The term (exp $\frac{-E}{RT}$) accounts for the number of molecular collisions in which the energy of one molecule relative to another exceeds the activation energy E; and R is the universal gas constant.

At low pressures, the reaction rate becomes slow and can become limiting for aircraft engines at very high altitudes.

During most of the operating conditions, the rate of combustion is limited by the rate at which the fuel is vaporized and mixed with air. In most of the combustors, the fuel is injected as an atomized liquid-droplet spray into the hot reaction zone where it mixes with air and hot combustion gases. The atomized fuel vaporizes, and mixes with air. If the temperature and pressure are sufficiently high, then the reaction rate will be fast and the fuel vapor will react as it comes in contact with sufficient oxygen.

The **Equivalence Ratio ϕ** isthe actual fuel/air ratio divided by the fuel/air ratio required for complete combustion (stoichiometric ratio) or it is the ratio of fuel-air ratio of consideration divided by the stoichiometric fuel-air ratio.

$$\Phi = \frac{f}{f_{stoich}} \text{ ie,}$$

$$\phi = (f/a)/(f/a)_{\text{stoichiometric}}$$

Values of ϕ less than unity correspond to lean operation, while those greater than unity correspond to rich combustion.

If the equivalence ratio is greater than 1, indicates rich fuel/air ratio, and less than 1 indicates a lean fuel/air ratio.

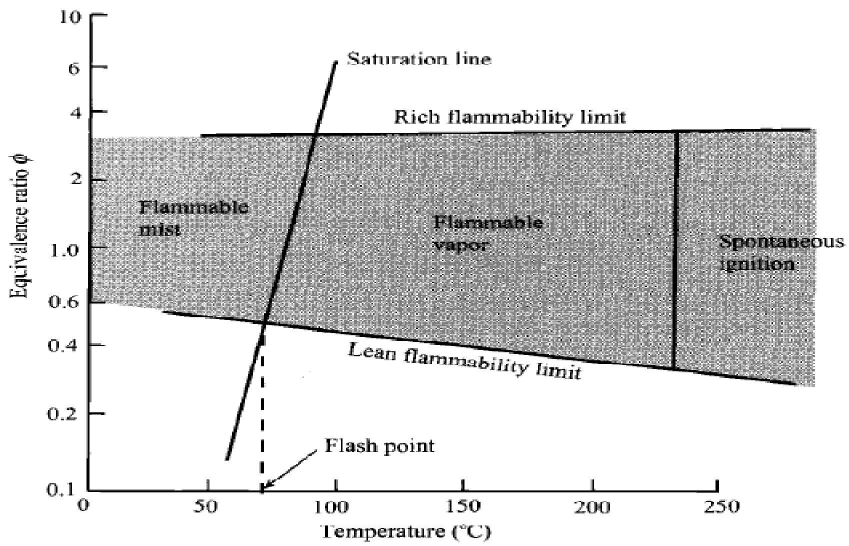
The overall fuel/air ratio must be less than the stoichiometric ratio with $\phi <$ 1.0, to prevent excessive temperatures at the exit of the main burner or afterburner and protect it's walls.

Effect of fuel/air ratio and mass flow rate:

Flammability Characteristics

For a fuel like kerosene, the f_{stoich} is 0.0667, the equivalence ratio keeps varying with flight conditions.

The flammability limits are shown in the diagram below:



The above graph drawn for a fuel like kerosene indicates that

- The hydrocarbon fuels have a narrow range of equivalence ratio between 0.5-3.0.
- The standard atmospheric pressure of below 0.2 atmis not acceptable for reaction rates to be favorable.
- Special fuels are needed for extending altitude limits for engine operation.
- The lean flammability limit of around 0.5 presents a design problem for values of ϕ corresponding to full throttle position.

Velocity of mixture: For stable flame, the velocity of mixture must be maintained within certain limits. If the velocity is too high, the flame will be blown out; if the velocity is too low, the flame will travel upstream and be extinguished.

The problem of holding the combustion flame within the combustion system is solved by establishing regions of recirculation at the front of the main burner, or in case of afterburner, by introducing a bluff body "flame-holder" in the mixture flow path.

Ignition:

Ignition of fuel/air mixture in the combustion system requires inlet air and fuel quantities with in flammability limit, sufficient residence times of the combustible mixture and effective ignition source.

The flammability region shown in the above graph is subdivided in to two sub-regions separated by the "spontaneous ignition temperature (SIT)". SIT is the lowest temperature below which an ignition source is required to bring the local temperature above spontaneous ignition temperature. Value of SIT varies with different fuels.

Once the flammability limits and SIT requirements are met, the ignition delay time becomes the key combustion characteristic. The ignition delay time t_{ign} is related to the initial temperature T and energy E by

$$t_{ign} \propto \exp \frac{E}{\mathcal{R}T}$$

The variation of ignition delay time with pressure is observed experimentally to follow the relation

$$t_{ign} \propto 1/\text{P}$$

Combustion Stability: Combustion stability is the ability of the combustion process to sustain itself in a continuous manner.

While the flight goes through different operating conditions, fuel/air mixture becoming too lean or too rich make the temperatures and reaction times to drop below desired levels necessary to effectively heat and vaporize the incoming fuel and air. This variation can upset stable and efficient combustion. Such a situation can cause blow out of the combustion process.

Combustor Loading Parameter (CLP) is defined to express the effects of mass flow rate, combustion volume and pressure on the stability of combustion process, as follows:

CLP = $\frac{\dot{m}}{P^n \times (Combustion\ Volume)}$, value of n is 2 for bimolecular reactions; can also be taken as 1.8.

Length Scaling: The cross sectional size of the combustor is determined from the 1-D gas dynamics. But the length requires application of scaling laws. Length of combustor is primarily based on the distance required for complete combustion. The relation between length and pressure & temperature is given below:

$$L \propto P_{t3}^{-r} / \sqrt{T_{t4}}$$

Thus the length of main burners varies with the pressure and temperature and is not affected by the size of the engine.

As the compressor pressure ratio is increased, the combustor gets shortened.

Overall Total Pressure Loss: The overall total pressure loss of the main burner is the sum of inlet diffuser, burner dome and liner loss. It is normally expressed as % of compressor discharge pressure. Total pressure losses of 4-5% are typically encountered in current combustion systems. Main burner pressure loss is identified as necessary to achieve design objectives of exit temperature profile (profile factor) and complete combustion. Total pressure loss impacts the total thrust and Tsfc of the engine.

Exit Temperature Profile: The performance parameters are related the temperature uniformity of the combustion gases as they entered the turbine. Combustion gases at the exit are measured using high-temperature thermocouples.

Pattern factor and profile factor are two important main burner design requirements. They describe the thermal impact on the turbine and are critical in matching the main burner and turbine components. Failure to achieve desired pattern factor or profile factor will result in shorter turbine life or may require redesign of the main burner or the turbine.

The **pattern factorPF** is defined as

$$PF = \frac{T_{t\,\max\,av} - T_{t\,av}}{T_{t\,av} - T_{t\,in}}$$

Where

$T_{t\,\max}$ = maximum measured exit temperature (local)

$T_{t\,av}$ = average of all temperatures at exit plane

$T_{t\,in}$ = average of all temperatures at inlet plane

Contemporary burners have pattern factor ranges from 0.25 to 0.45.

The **profile factor** $\boldsymbol{P_f}$ characterizes the main burner average exit temperature profile and is defined as

$$P_f = \frac{T_{t\,\max\,av} - T_{t\,in}}{T_{t\,av} - T_{t\,in}}$$

Where $T_{t\,max\,av}$ is the maximum circumferential average temperature. Main burner exit profile factors range from 1.04 to 1.08.

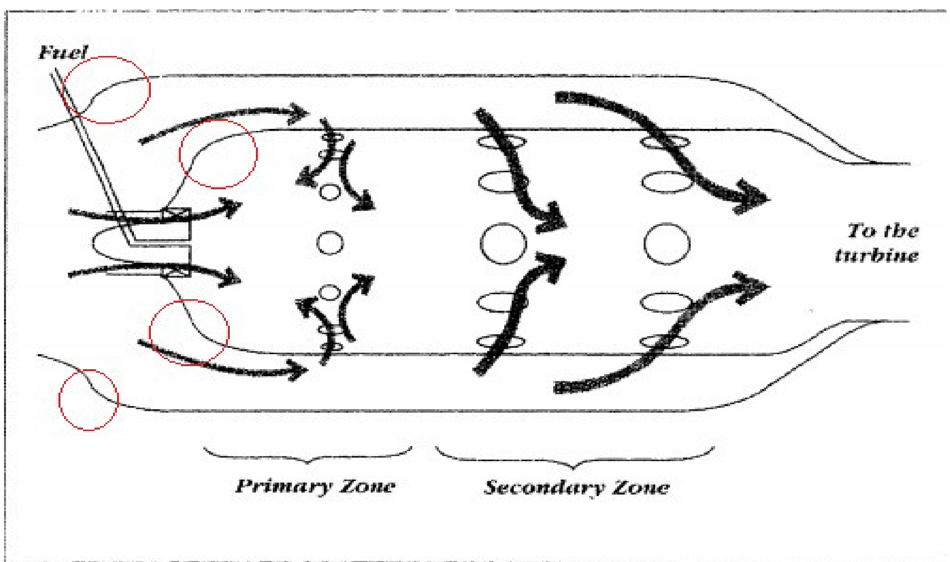
Profile factor is a critical design requirement for the first stage turbine rotors.

Combustion Chamber – Construction:

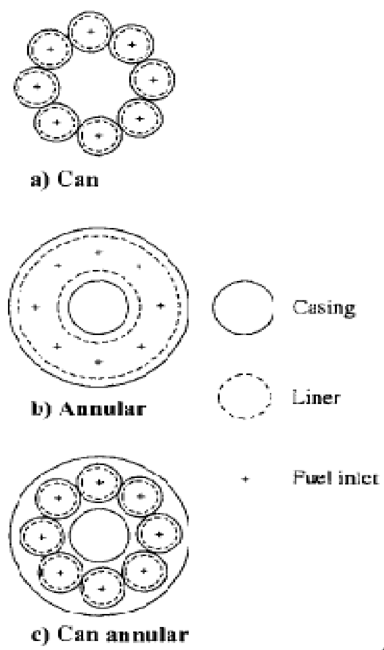
The combustion chamber is designed to burn a fuel/air mixture and to deliver the hot gasses to the turbine at uniform temperature. The gas temperature must not exceed the allowable structural temperature of the turbine.
The high pressure air from the compressor enters the combustion chamber. Of this, less than half of the total volume of air mixes with fuel and burns. The rest of the air, known as secondary air is used as cooling the products of combustion or the burner walls. The ratio of total air to fuel varies between 30 to 60 parts of air to 1 part of fuel by weight.

Combustion chambers are of three types; can, annular and can-annular types.

The pressure loss as the gasses pass through the burner must be minimum and the combustion efficiency must be high. There should be no tendency for burner to flame-out.



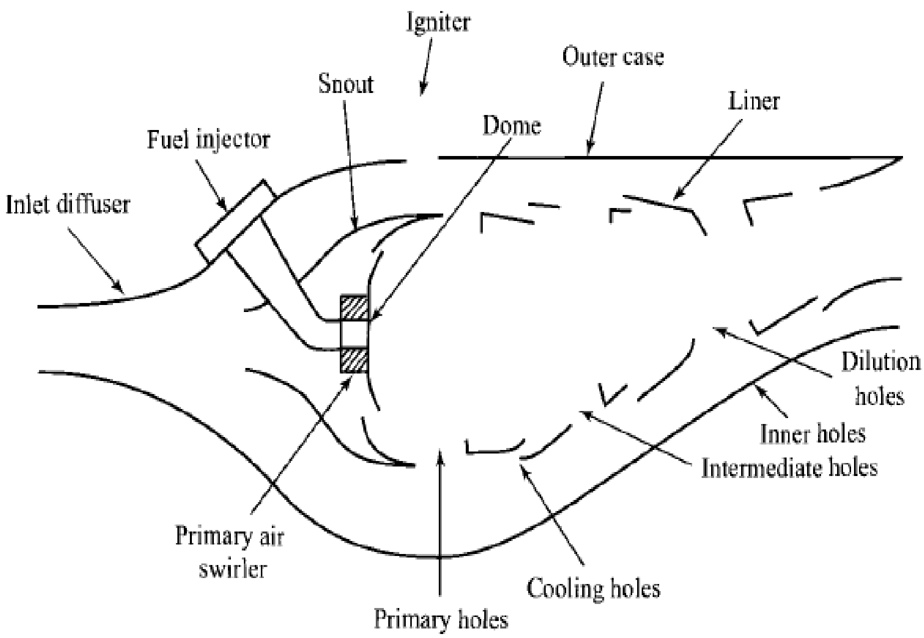
Combustion chambers (burners) -Classification: Combustion system burners are broadly classified as three types-**Can, Cannular & Annular** , as shown below:



A can system consists of one or more cylindrical burners, each contained in a burner casing.
The can-annular (cannular) system consists of a series of cylindrical burners arranged within a common annulus area. This type of burner is most commonly used design in gas turbine engines.
However, the modern engines employ the annular design wherein a single burner having an annular cross-section enclosed by an outer burner casing. This type of arrangement ensures improved combustion zone uniformity, design simplicity, reduced linear surface area and shorter length.

3.2.1: **Main Burner Components**: The main burner system consists of three main components:

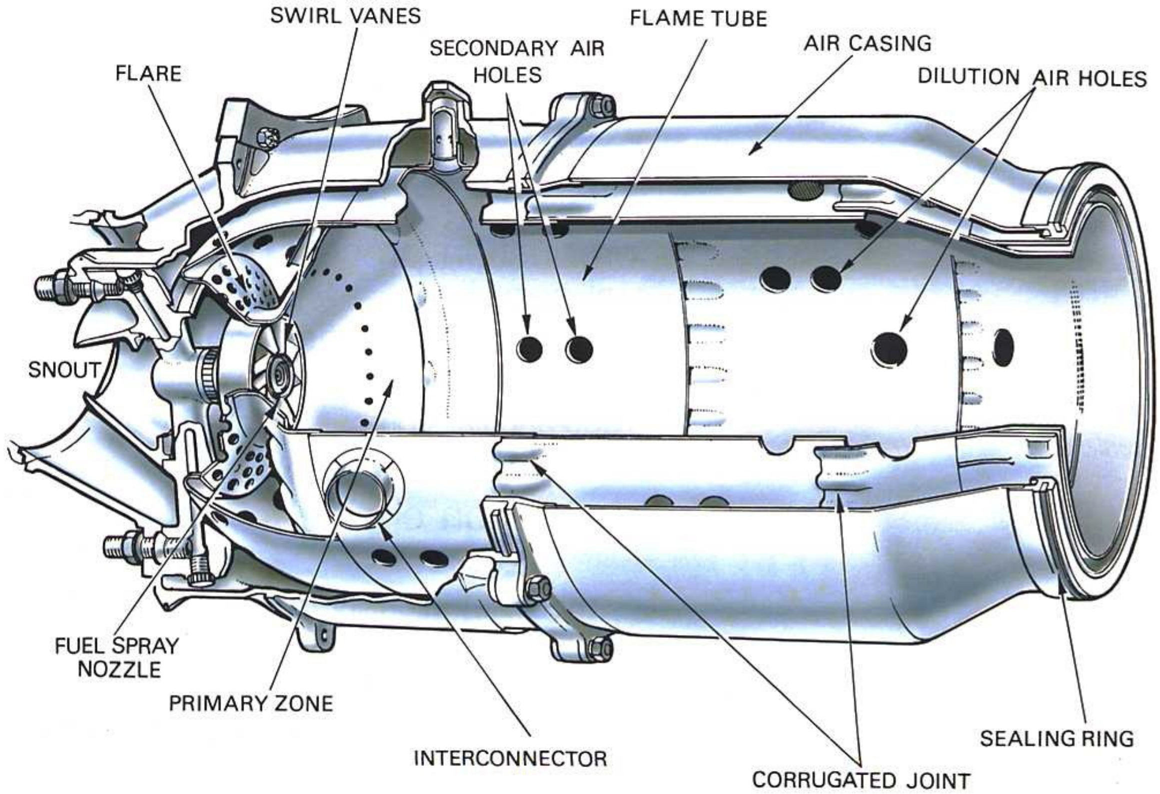
- The inlet Diffuser
- The Dome
- The Cowl or Snout



In addition to above components, following subsystems are used:

- Fuel Injector
- Ignitor
- Burner Casing
- Primary Swirler

The purpose of **inlet diffuser** is to reduce the velocity of the air exiting the compressor and deliver the air to the combustion zone as a stable, uniform flow field while recovering as much of the dynamic pressure as possible. Most of the size limitations need short diffusers, with curved walls and dump design. Criteria for design of diffuser are high pressure recovery and avoidance of flow separation.



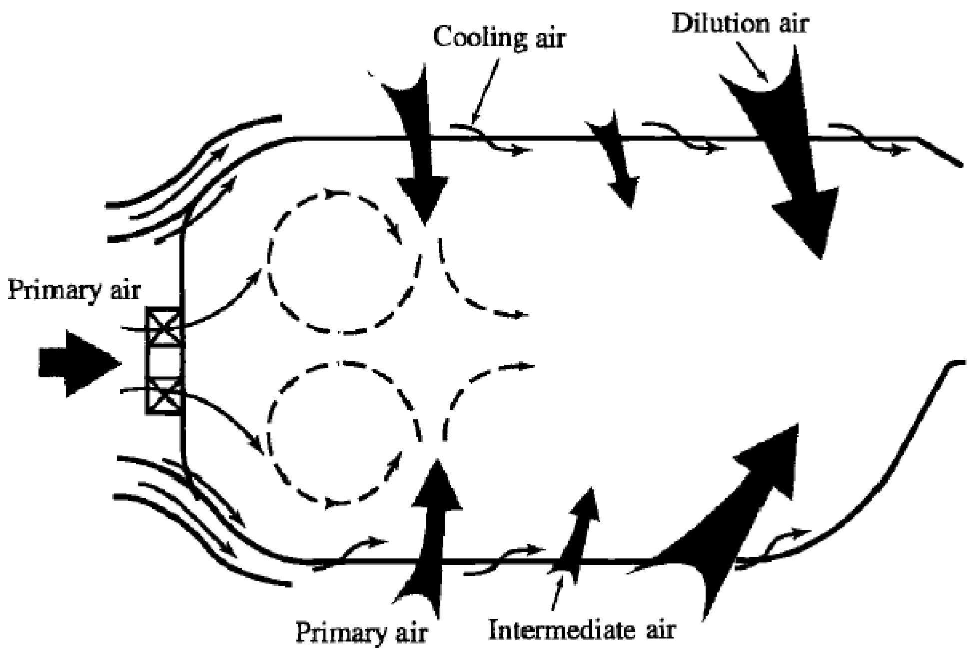
The **Snout** divides the incoming air into two primary air and secondary air (intermediate, dilution and cooling air). The snout streamlines the combustor dome and permits a larger diffuser divergence angle with shorter length.
The combustor **dome** is designed to produce an area of high turbulence and flow shear in the vicinity of fuel nozzle to finely atomize the fuel spray and promote rapid fuel/air mixing. There are two types of combustor domes-bluff body and swirl stabilized. The bluff body domes were used in the early designs, bur swirl-stabilized domes are used in most modern combustors.
The combustion process is controlled by the **liner**. The liner allows introduction of intermediate and dilution airflow and liner's cooling airflow. The liner must be designed to support the forces resulting from the pressure drop and must have high thermal resistance capable of continuous and cyclic high temperature operation.

3.2.2: Fuel Injectors: Fuel injectors can be classified into four basic types according the injection method utilized.
The methods of fuel injection are pressure-atomizing, air-blast, vaporizing and pre-mix/pre-vaporizing.
Pressure-atomizing needs pressure levels of around 500 psi above main burner pressure, and results in good fuel atomization. This system is liable to fuel leaks in the system.

Air-blast atomizing fuel injector achieves fuel atomization through the use air-blast created by the primary air momentum with a strong swirling motion. The air-blast atomizing fuel injector requires lower fuel pressures (around 50 to 200 psi above main burner pressure) than the pressure-atomizing type fuel injectors.

Spark ignitors, similar to automotive spark plugs, are used to ignite the cold, fuel/air mixture in main burners. Redundancy is provided by use ofatleast two spark igniters.

3.2.3: Air flow distribution and cooling air: For airflow regions are formed in the main burner area as shown below:

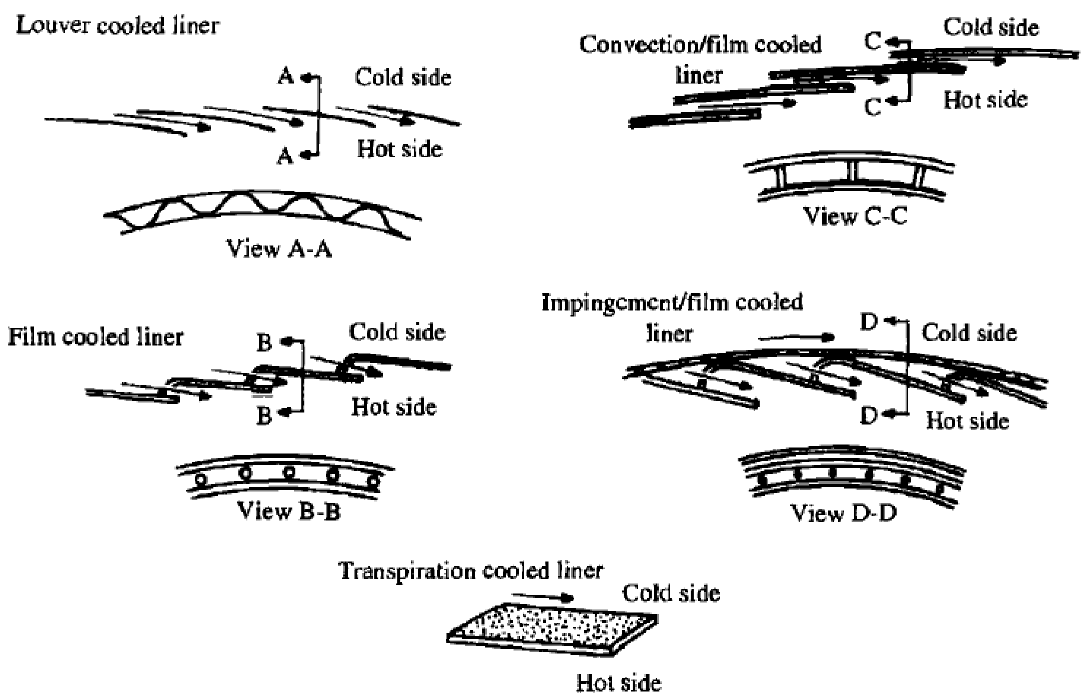


Primary airisthe combustion air introducedthrough the dome of the burner. This mixes with the incoming fuel, producing a mixture ready for combustion. The **secondary air** which is introduced through the liner holes completes the reaction process and consumes the unburnt fuel. The secondary air reduces the local concentration of temperature and reduces the mixture ratio close to stoichiometric ratio.

The **dilution air**is used to ensure uniform temperature profile at the exit of combustor and increases the turbine durability and performance.

Cooling air is used to protect the combustor liner and dome from high heat loads. This air is introduced through the liner such that a protective blanket or film of air is formed between the combustion gases and the liner hardware.

3.2.4: The liner cooling techniques vary with liner geometry and introduction of cooling air and are shown below:



Methods employed for thrust augmentation:

Increase of thrust above the original value: Thrust of an engine can be achieved by two methods, both involving extensive redesign of the engine. They are

- **Increase of turbine inlet temperature**, for example, will increase the specific thrust and hence the thrust for a given engine size.
- Alternatively,**increasing mass flow rate through the engine** without altering the cycle parameters.
- Both of these methods imply some redesign of the engine, and either or both may be used to up-rate an existing engine.

Frequently, however, there will be a requirement for a **temporary increase in thrust**, e.g. for take-off, for acceleration from subsonic to supersonic speed or during combat manoeuvers; the problem then becomes one of thrust augmentation. Numerous schemes for thrust augmentation have been proposed, but the two methods most widely used **are liquid injection and afterburning (or reheat).**

Liquid injection (Water-methanol/alcohol) is primarily useful for increasing take-off thrust. Substantial quantities of liquid are required, but if the liquid is consumed during take-off and initial climb the weight penalty is not significant.
Spraying water into the compressor inlet causes evaporation of the water droplets, resulting in extraction of heat from the air; the effect of this is equivalent to a drop in compressor inlet temperature. Reducing the temperature at entry to a compressor will increase the thrust, due to the increase in pressure ratio and mass flow.
In practice a mixture of water and methanol is used; the methanol lowers the freezing point of water, and in addition it will burn when it reaches the combustion chamber. Liquid injection into the compressor, however, has corrosive effect on the blades.
Liquid is sometimes injected directly into the combustion chamber. In both cases the mass of liquid injected adds to the useful mass flow, but this is a secondary effect.

Water injection on a hot day can increase the take-off thrust by 50% because the original mass of air entering the engine is low on a hot day.

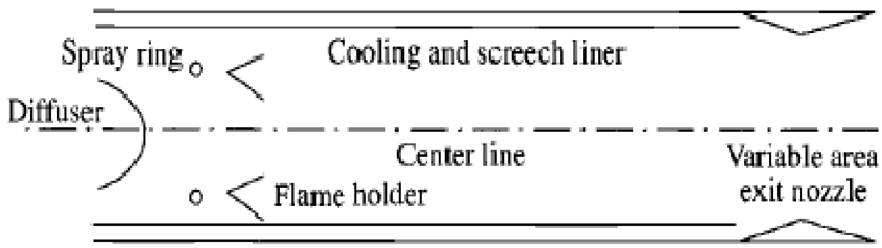
Liquid injection is now seldom used in aircraft engines.

Afterburning, as the name implies, **involves burning additional fuel in the jet pipe**. In the absence of highly stressed rotating blades the temperature allowable following afterburning is much higher than the turbine inlet temperature.

The effect of afterburning is to increase the temperature of the exhaust gases which in turn will result in higher thrust through expansion in the exhaust nozzle. The afterburning produces high thrust at the expense of fuel economy. Temperatures of around 2000 K are possible through afterburning.

The afterburner increases the thrust by adding thermal energy to gas stream leaving turbine in a turbojet engine. However, for a turbofan engine, thermal energy addition in the afterburner may be to the mixture of turbine exit air and bypass air.
At the afterburner inlet, there is still un-burnt oxygen in the gas stream. The higher inlet gas temperature and near-stoichiometric ratios of fuel/air of the afterburner enables a simpler design of the afterburner.
The resultant increase in temperature raises the exhaust velocity of the exhaust gases and boosts the engine thrust. Most afterburners produce an approximate 50% thrust increase, but with corresponding three-fold increase in the fuel flow.

Since the fuel consumption is very high during afterburning periods, these are limited to few occasions like take-off/climb and maximum speed bursts. The afterburning period is called wet operation and non-afterburning periods are called dry operation of the engine.
For a turbofan engine, afterburning can be used both in fan as well as core streams. Afterburning in separate fan stream is normally referred to as duct burning.
Typical afterburning components are shown below:



Operation: Gas leaving the turbine is de-swirled and diffused, fuel is added by fuel spray bars (tubes) or rings. The combustion process is initiated by igniter or pilot burner, in the wake created by a number of flame stabilizing devices (flame holders). The afterburning process causes screech or howl acoustic instabilities. These are controlled by providing liner for cooling as well as anti-howl purpose.
This liner can also serve as a passage for the cooling air meant to cool the nozzle.
All engines incorporating afterburner must also be equipped with variable-area throat exhaust nozzle to provide for proper operation under afterburning and non-afterburning conditions.
In addition to above components, an afterburner need the following components:

- Afterburner fuel pump
- Afterburner fuel control
- Pressurising valve, if multistage operation is involved
- Connections from main fuel pump control, throttle and engine.

Specific design requirements of afterburner are as follows:

1. **Large temperature increase**: The afterburner is not constrained by the physical and temperature limitations of the turbine. The temperature raise is mainly limited by availability of oxygen for combustion.

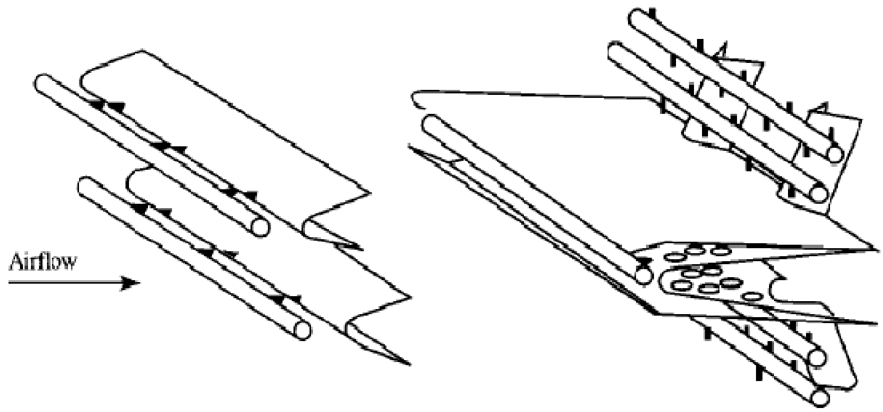
2. **Low dry loss**: The engine suffers very slight penalty in thrust during cold operation due to the drag caused by the flame holders, fuel spray bars and walls of the afterburner.
3. **Wide temperature modulation**: This is necessary to obtain degrees(also called zones or **stages**) of afterburning for better control of thrust.
4. **High combustion efficiency**
5. **Short length; light weight**
6. **Altitude light-off capability**
7. **No acoustic combustion instabilities**
8. **Long life, low cost, easy repair**

Afterburner Components:

- **Diffuser**: Flow entering the afterburner is first slowed to a lower Mach number (depends on the local pressure and diameter of afterburner). A shorter diffuser length is desired without producing flow separation and reduce weight.
- **Fuel injection, Atomization & Vaporization**: The fuel is introduced in a staged manner so that heat addition rate is can be increased gradually to the desired value. Successive annular fuel stream tubes are added to keep the fuel/air ratio close to stoichiometric ratio. Each stream tube has its own set of fuel injectors and control system and can be activated independently.

  Fuel is injected in to the gas through small diameter holes located in the side of the tubes so that liquid jet enters the gas stream in a direction perpendicular to the flow direction. The air stream tears the jet apart producing droplets of micron size diameters. Heat transfers from the gas then vaporizes the droplets. Remarkably thorough mixing of fuel with air can be achieved through this injection system.
- **Ignition**: Ignition of the fuel/air mixture is accomplished by using a spark plug or arc igniter or a pilot burner. Once ignited in the primary stream tube, combustion continues in the wake of flame stabilizer (a bluff body) and the process will spread to the rest of the flame stabilizers. The wakes of the flame stabilizers re arranged in a manner that they overlap and enable uniform spreading of the flame.
- **Flame Stabilizers**: Two general types of flame stabilizers are used as shown below:

Vee-gutter flame stabilizers: Have advantage of causing low flow blockage and low total pressure loss. They are simple, light weight'
Piloted-burner flame stabilizers: Can hold a small piloting heat source to ignite the main fuel flow.



a) Vee-gutter flame holders b) Piloted burner

Fuels-Composition & Specifications:

Jet fuel is refined from crude oil petroleum. The heating value h_{PR} of most jet fuels is around 42,800 kJ/kg.
Hydrogen is also considered for used in high speed aircraft due to its high h_{PR} of around 1,16,000 kJ/kg. Hydrogen also has capacity to absorb the thermal loads of high Mach number flight.

High speed aircraft prefer using fuels with high boiling point.

Types of Exhaust Nozzles: Two types of exhaust nozzles are used:

- Convergent Nozzle
- Convergent Divergent Nozzle (CD Nozzle)

Convergent Nozzle:
The convergent nozzle is a simple convergent duct.
When the nozzle pressure ratio (Total exit pressure/atmospheric pressure) is low (less than 4.0), the convergent nozzle is used.
The convergent nozzle is generally used in engines for subsonic aircraft.

Convergent divergent nozzle:

- Convergent duct followed by divergent duct

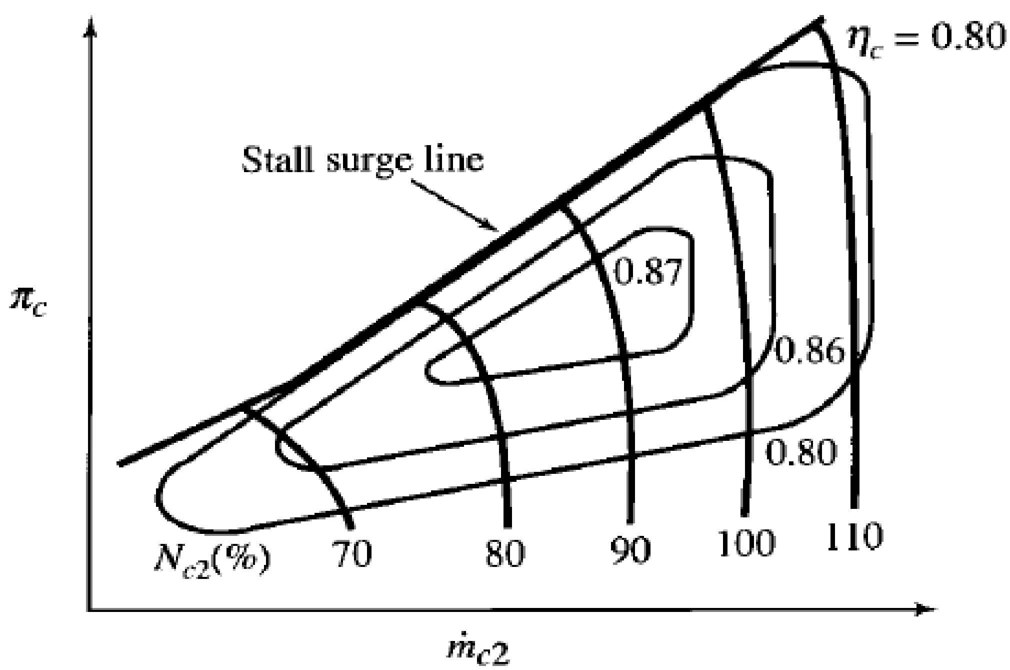
- Cross-sectional area minimum in between-called "Throat" of the nozzle
- Used for high performance engines with pressure ratio (Pt exit/P0)of greater than 6.0

- Nozzle throat area/exit area is varied to match off-design operating conditions, to produce maximum thrust
- During afterburner operation, nozzle throat area is changed to isolate upstream engine operation from the back pressure

Nozzle Functions:

- **Engine Back-pressure cont**rol: Throat area is the main means of control for optimising thrust and sfc
- During design stage, nozzle throat area is fixed to match design values of mass flow rate, thrust and sfc
- Changing the design value of throat area optimise performance parameters both on and off-design engine operation

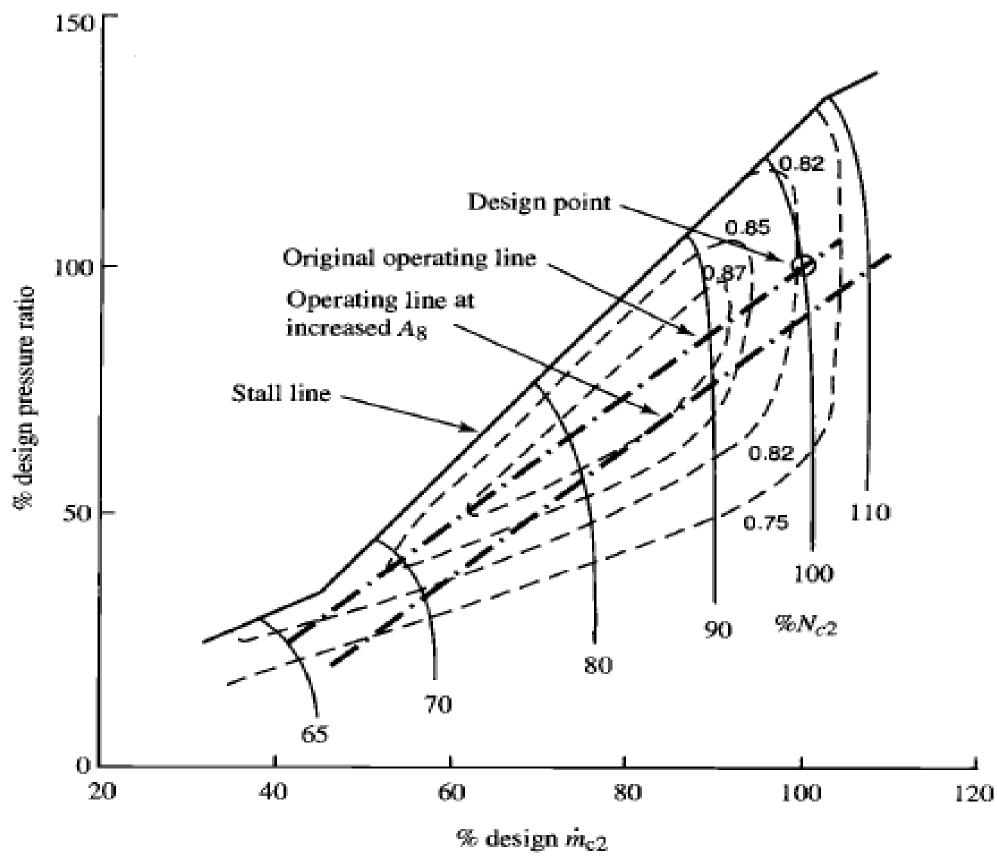
Engine Back-Pressure Control - Compressor Performance map:
Relates compressor pressure ratio, corrected mass flow rate, corrected engine speed and efficiency.



Key points:

- At reduced engine throttle settings, corrected mass flow rate reduces
- Compressor operating point moves closer to stall/surge line
- Increase in nozzle throat area, reduces engine back pressure and increases corrected mass flow rate
- This causes moving operating line away from stall/surge line

Compressor performance map with nozzle throat area change:



Throat area with afterburner:

- Large changes in throat area needed to compensate large increase in total temperature
- Throat area change is also scheduled to isolate upstream of afterburner from back pressure build up
- Normal engine will be unaware of afterburner operation

Improved Starting operation with variable area nozzle:

- Variable area nozzle improves starting operation
- During start-up, throat area is at maximum
- This reduces back pressure on turbine, improves expansion, achieving turbine power needed for starting at lower turbine inlet temperature
- Compressor can also easily achieve required pressure ratio

Exhaust nozzle area ratio:

- Max thrust achieved when nozzle exit pressure = ambient pressure
- At design point, nozzle throat is in choked condition, thereby causing supersonic acceleration in divergent section
- Over-expansion may cause regions of separated flow
- Slight under-expansion is preferred

Thrust Reversers:

- Thrust Reversers are used in commercial transport aircraft to supplement brakes
- In-flight thrust reversal has been shown to enhance combat effectiveness of fighter aircraft
- Two types of thrust reversers are used
- Cascade-Blocker type and
- Clamshell type

5.8. Thrust Reversers and Vectoring

5.8.1. *Reversers*

The difficult problem of stopping an aircraft after landing has become more pronounced with modern aircraft because of the large aircraft weights, high speeds, and existing runways. Wheel brakes alone are not an effective means to stop such an aircraft owing to brake pad and tire thermal limitations. A reversible pitch propeller is used for turboprops to reverse the thrust direction upon landing. Turbojet and turbofans do not have such an option, however. To provide a "brake" for such aircraft, a thrust reverser is usually used.

For this, the turbine exhaust, fan air, or both are diverted at a suitable angle in the reverse direction by the means of an inverted cone, half-sphere, turning vanes, or other shape introduced in the exhaust flow upon landing. Because the exhaust flow is turned by almost 180°, the linear momentum equation can be used to show that the thrust is nearly reversed. The clamshell and cascade reversers are two of the most common of these devices.

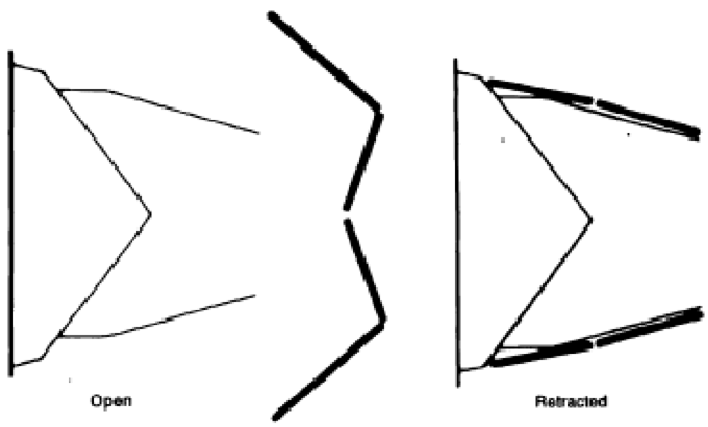


Figure 5.20a Clamshell thrust reverser.

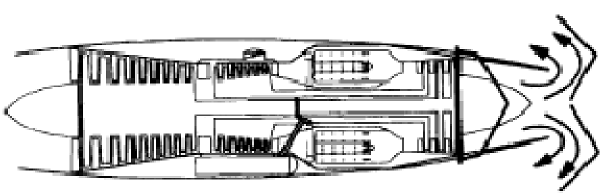


Figure 5.20b Clamshell thrust reverser in operation (courtesy of Pratt & Whitney).

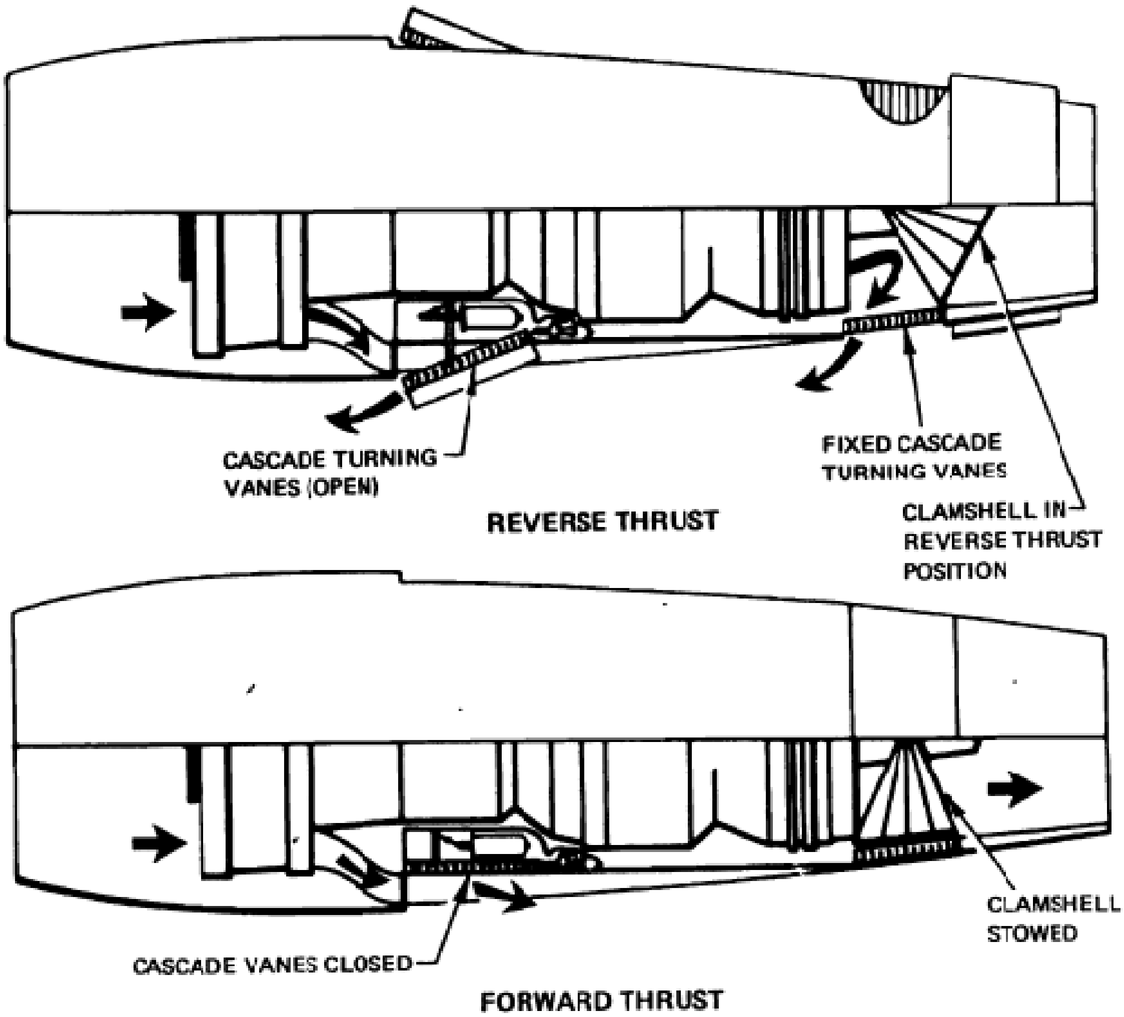


Figure 5.21 Thrust reversers for turbofan engine – exhausted fan (courtesy of Pratt & Whitney).

In Figure 5.20, the clamshell type is shown. For this geometry, the clamshell is opened upon landing and the shell is approximately one diameter downstream of the exhaust. When the reverser is not in use, the shell is retracted and stowed around the nozzle. Sometimes the retracted shell forms a part of the rear section of the nozzle nacelle.

A cascade-type reverser employs numerous vanes in the gas path to reverse the gas flow. Reversers for turbofans are often of this type. In Figures 5.21 and 5.22, the reversers for an exhausted turbofan are shown. For the fan ("cold air") itself, only turning vanes are used, as shown for the two positions. For forward thrust, the vanes are out of the gas path; however, for reverse thrust, they are moved into the flow path. For the primary flow (hot gas), a combination of clamshell and cascade reversers are used. For this geometry, the clamshell is directly downstream of the turbine and, when activated, diverts the flow to the fixed cascade of vanes. Hydraulic pistons move both the movable vanes and clamshell. It is interesting to note that the engine manufacturer is usually not totally responsible for the design of thrust reversers but that this is left to the airframe manufacturer or a joint effort between the airframe end engine manufacturer.

5.8.2. *Vectoring*

To gain more control of aircraft maneuverability in military fighter applications, manufacturers have begun to develop thrust vectoring (thrust direction control) as a part of the exhaust of the engine. Thus, instead of the thrust being along the engine centerline,

the thrust can be along another vector. Vectoring the thrust reduces response times. Such designs also allow for more rapid takeoffs. Applications include two-directional vectoring and multidirection vectoring. Thrust vectors can typically be varied from up to 20° from the nominal direction. Although the design is somewhat more complicated, weight and complexity of vectoring thrust nozzles are not much greater than for nozzles with variable areas. In fact, the all-directional vectoring nozzles use the iris variable area design as a basis for the vectoring. That is, the "flaps" are independently controlled and moved by independent amounts, thus allowing the thrust direction to be changed.

Performance Measures/Correction Factors:

The **energy conversion efficiency** e is defined as the ratio of the kinetic energy per unit of flow of the actual jet leaving the nozzle to the kinetic energy per unit of flow of a hypothetical ideal exhaust jet that is supplied with the same working substance at the same initial state and velocity and expands to the same exit pressure as the real nozzle.
The **velocity correction factor**ζ_v is defined as the square root of the energy conversion efficiency $\sqrt{e}$.
This factor is also approximately the ratio of the actual specific impulse to the ideal specific impulse.
The **discharge correction factor** ζ_d is the ratio of mass flow rate in the real rocket to that of an ideal rocket that expands an identical working fluid from the same initial conditions to the same exit pressure.
The **thrust correction factor** ζ_F is the ratio of actual thrust divided by ideal thrust.
Computing Rocket Engine Performance: Thrust, Specific impulse, Propellant flow and other performance parameters are used in different calculations/comparisions. It is important to specify the conditions under which these performance parameters are computed. There are four sets of

conditions under which the above performance parameters are specified. They are

- **Theoretical performance Values**: these values are applicable to ideal rockets, usually with some corrections. The nozzle flows are considered for two dimensional flow, use real gas properties for the chemical reactions and correct for divergence loss. Solid propellant rocket nozzle flows are corrected for nozzle erosion and multiphase flow. The computer programs simulate one dimensional flow with above corrections incorporated.
  Theoretical performance values are used for preliminary estimates and proposals.
- **Delivered (actually measured) Performance Values**: These values are obtained from static tests or flight tests of full scale propulsion systems. The measured values are corrected for instrument deviations, errors or calibration constants. Flight test data is corrected for aerodynamic effects like drag. Emperical correction factors like thrust correction factor, velocity correction factor or mass discharge correction factors to convert the measured performance values in to approximate actual values.
- **Performance Values at Standard Conditions**: These values are above values (theoretical or measured) corrected for the standard conditions specified by the customer. These are values corrected to standard conditions, like atmospheric pressure of 1000 psi; Optimum area ration corresponding to optimum expansion ie p_2=p_3; nozzle divergence half angle=15° etc.
  A rocket propulsion system is generally designed, built and tested in accordance with the pre-determined specifications.
- **Guaranteed Minimum Performance**: Rocket manufacturers are often required by their customers to deliver rocket propulsion systems with a guaranteed minimum performance, such as minimum F or I_{sp} or both. The determination of these values is based on theoretical or measured performance values corrected for losses. These losses include loss due to nozzle surface roughness, pressure drop in fuel (liquid) supply pipelines, propellant grain initial temperature etc.

Under and Over-expanded Nozzles:
An under-expanded nozzle discharges the fluid at an exit pressure greater than the external pressure because the exit area is too small for optimum expansion. The expansion is incomplete. This condition occurs at altitudes higher than the design altitude.
In an over-expanded nozzle, the fluid attains a lower exit pressure than the atmospheric pressure. The exit area at this condition is too large than the optimum area. This condition occurs when the nozzle operates at altitudes lower than the design altitude. Since the pressure inside the nozzle is lower than the outside pressure, there is possibility of flow separation due to adverse pressure gradient.
Different possible **flow conditions** are explained with reference to the diagram below:

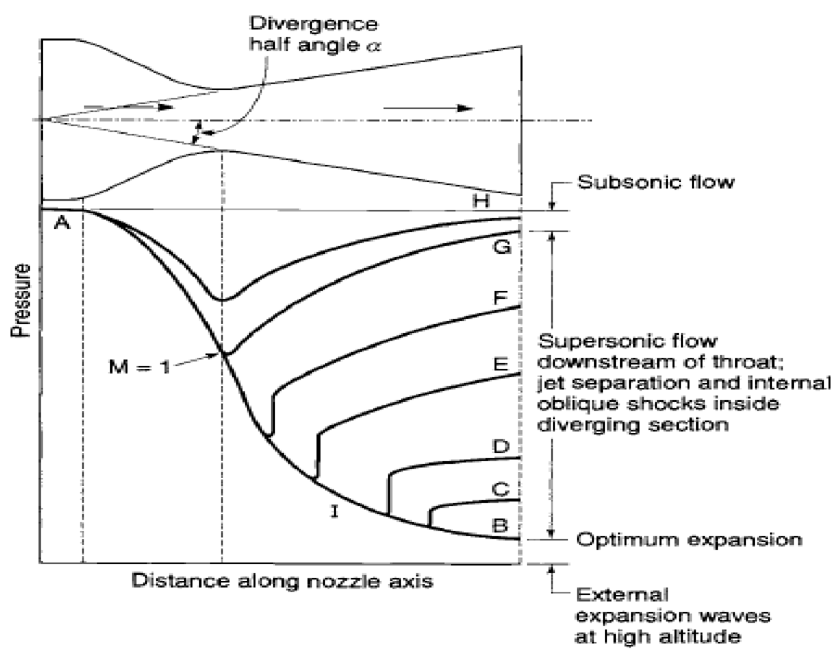


FIGURE 3–9. Distribution of pressures in a converging–diverging nozzle for different flow conditions. Inlet pressure is the same, but exit pressure changes. Based on experimental data from A. Stodala.

- Curve AB shows variation of pressure with optimum back pressure at the design area ratio.
- Curves AC and AD show variation of pressure along the axis for increasingly higher external pressure (over-expansion). At point I, on curve AD, the pressure is lower than the exit pressure and a sudden rise

in pressure takes place accompanied by separation of flow from the walls.

- The sudden pressure rise in the curve AD is a compression discontinuity accompanied by a compression wave.
- Expansion waves occur in cases where external pressure is lower than the exit pressure, ie below point B.
- When external pressure p_3 is below the nozzle exit pressure p_2, it signifies under-expansion. The expansion of gas inside the nozzle is incomplete and the value of C_F and I_s will be less than at optimum expansion.
- For external pressure p_3 is slightly higher than the nozzle exit pressure, the nozzle continues to flow full. This continues till p_2 reaches a value between about 25 and 40% of p_3. The expansion is inefficient, C_F and I_s values are lower than optimum.
- For higher external pressures, separation of flow takes place inside the divergent portion of the nozzle. The axial location of separation depends on the local pressure and wall contour. With steady flow, separation is also axially symmetric.
- On separation, at the nozzle exit plane, the center portion remains supersonic while the surrounding annular flow is subsonic.

UNIT IV

RAMJET & SCRAMJET ENGINE: components, Performance of turbojets, ramjets at high speeds- limitations. Need for supersonic combustion, Implications criticality of efficient diffusion and acceleration, problems of combustion in high speed flow, The scramjet engine- construction, flow process- description, control volume analysis spill-over drag, plume drag, Component performance analysis- isolator, combustor- flow detachment and reattachment, thermal throat, scheduled, distributed fuel injection, Nozzle flow, losses- failure to recombination, viscous losses, plume losses. Scramjet performance applications, Combined cycle engines- turbo-ramjet, Air turbo-rocket (ATR), ejector ramjet, Liquid-air collection engine (LACE)- need, principle, construction, operation, performance

UNIT V

ROCKET ENGINE:

CHEMICAL ROCKET: Classification of rocket engine, chemical rocket engine types, working principle, schematic diagram, applications, types, advantages and disadvantages- solid, liquid and hybrid propellant rocket engine, propellants types used, injectors, nozzles, igniters, storage, TVC, combustion instabilities, combustion chamber, pulse detonation engine, rotary rocket engine

NUCLEAR: Power, thrust, energy. Nuclear fission- basics, sustainable chain reaction, calculating criticality, reactor dimensions, neutron leakage, control, reflection, prompt and delayed neutrons, thermal stability. Nuclear propulsion, history, principles, fuel elements, exhausts velocity, operating temperature, The nuclear thermal rocket engine, radiation and management, propellant flow and cooling, control, start-up and shutdown, nozzle, thrust generation. Potential applications of nuclear engines- operational issues, interplanetary transfer manoeuvres, faster interplanetary journey. Development status of nuclear engines, alternative reactor types, safety issues, nuclear propelled missions.

ELECTRICAL: Limitations of chemical rocket engines. Electric propulsion systems- structure, types, generation of thrust. Electrostatic thrusters, electro-magnetic thrusters, applications to space missions, pulsed plasma thrusters (PPT) for micro-spacecraft, solar electric propulsion.

ADVANCED: Micro-propulsion, micro-propulsion options, application of MEMS, chemical, electric micro-thrusters, principle, description, Propellantless propulsion, tethers, momentum exchange, electro-dynamic Photon rocket, beamed energy propulsion, solar, magnetic sails.