Chapter 2

Basic Aerothermodynamics

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2. Aerothermodynamics in Gas Turbine Engine

- First law of thermodynamics
- Second law of thermodynamics
- Aerodynamics fundamental equations
  - Continuity equation
  - Energy equation
  - Bernoulli's equation
  - Sound speed and Mach number
  - Stagnation parameters of flow and aerodynamic functions
  - Momentum equation
  - Shockwaves and expansion waves

1. Basic Aerodynamics

- The conservation equations for the continuity, momentum and energy of the flow can be expressed in the following form:

$$\frac{\partial U}{\partial t}+\frac{\partial F(U)}{\partial x}+\frac{\partial G(U)}{\partial y}=S(U)$$

- The conservation equations for the continuity, momentum and energy of the flow can be expressed in the following form:

Scalar Equation $\longrightarrow$

$$U=\begin{bmatrix}\rho\\ \rho u\\ \rho v\\ \rho E\\ \rho Y_1\end{bmatrix},F(U)=\begin{bmatrix}\rho u\\ \rho u^2+p\\ \rho uv\\ u(\rho E+p)\\ \rho uY_1\end{bmatrix},G(U)=\begin{bmatrix}\rho v\\ \rho uv\\ \rho v^2+p\\ v(\rho E+p)\\ \rho vY_1\end{bmatrix},S(U)=\begin{bmatrix}0\\0\\0\\0\\R_1\end{bmatrix}$$

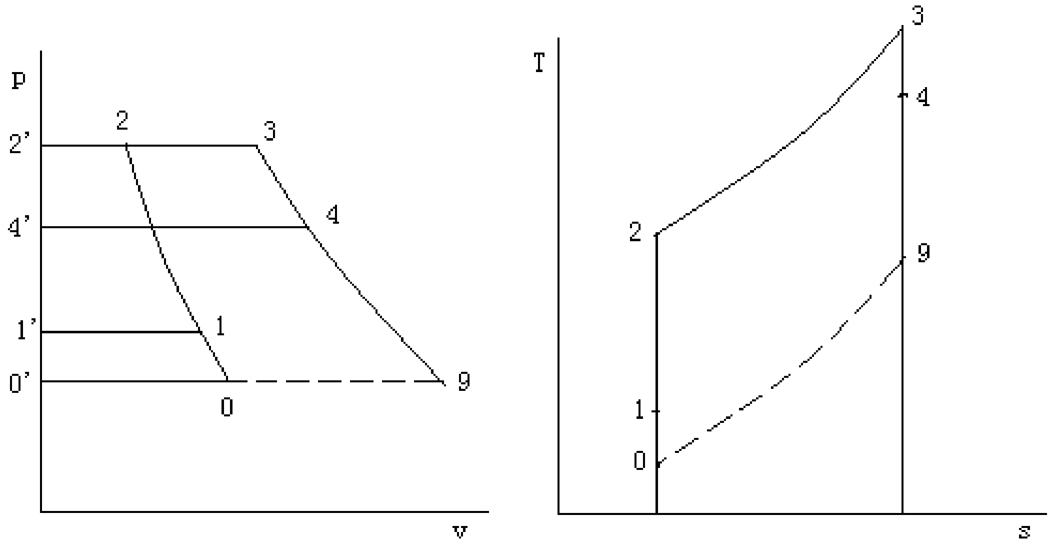
1. Basic Aerodynamics

Solution Methods

- Semi-Implicit Method for Pressure-Linked Equations (SIMPLE)
- Semi-Implicit Method for Pressure-Linked Equations-Consistent (SIMPLEC)
- Pressure Implicit with Splitting of Operator (PISO)
- Approximate Riemann Solvers; e.g. Harten Lax and van Leer- Contact (HLLC), HLL, etc.
- Advection Upwind Splitting Method (AUSM)
- Advection Upwind Splitting Method- Plus (AUSM-plus)

2. Aerothermodynamics in Gas Turbine Engine

Gas Turbine Cycle



2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Equation of state of ideal gas
  - Ideal gas
    - Gas molecule has only a mass but not volume
    - There is no attractive force between the molecules
    - Ideal gas does NOT exist
    - Idealization for real gas

2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Equation of state of ideal gas

$pv = RT$, *or* $p=\rho RT$

  - p ——*Pressure*，N/m^2
  - v ——*Specific volume*，m^3/kg
  - T ——*Temperature*， K
  - R ——*Gas constant*，*J/kgK (287.1 for air and 288 for gas)*

2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

Heat

- Heat is a form of energy exchange

Substance mass (kg)

Temperature after heat exchange (K)

Heat (J)

$$Q = mc(T_2 - T_1)$$

Specific Heat，J/kgK

2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Specific heat
  - Heat needed to increase *T* of 1K for 1kg of substance
  - Dependent on substance, temperature and process
- Specific heat at constant volume

$$Q_v = mc_v(T_2 - T_1)$$

2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Specific heat at constant pressure

$$Q_p = mc_p(T_2 - T_1)$$

- **In reality in general, and in particular for certain substances, *Cv* and *Cp* are functions of temperature.**

- These temperature defined properties are defined in terms of polynomial functions; e.g. **Janaf Table**.

2. Aerothermodynamics in Gas Turbine Engine

$$\frac{C_p(T)}{R} = a[1] + a[2]T + a[3]T^2 + a[4]T^3 + a[5]T^4$$

$$\frac{H(T)}{R} = a[1]T + \frac{a[2]T^2}{2} + \frac{a[3]T^3}{3} + \frac{a[4]T^4}{4} + \frac{a[5]T^5}{5} + a[6]$$

$$\frac{S(T)}{R} = a[1]\ln T + a[2]T + \frac{a[3]T^2}{2} + \frac{a[4]T^3}{3} + \frac{a[5]T^4}{4} + a[7]$$

2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Ratio of specific heats

$$\gamma = c_p / c_v$$

  - For air 1.4;
  - For a mixture of gases ~ 1.33;
  - The value of γ can vary based on composition.
  - In addition, the ratio is also temperature dependent when the specific heats are calculated as temperature-dependent functions.
  - Hypersonic limit: $\gamma \rightarrow 1$.

2. Aerothermodynamics in Gas Turbine Engine

- Specific internal energy
  - Summation of molecule average kinetic and potential energy
  - Potential energy for ideal gas = 0 (no attractive forces between the molecules).
  - Specific internal energy of ideal gas is function of temperature

$$u = u(T) \quad \text{J/kg}$$

  - Since ‘T’ represents molecular average kinetic energy of the gas.

2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Heating gas in container of constant volume

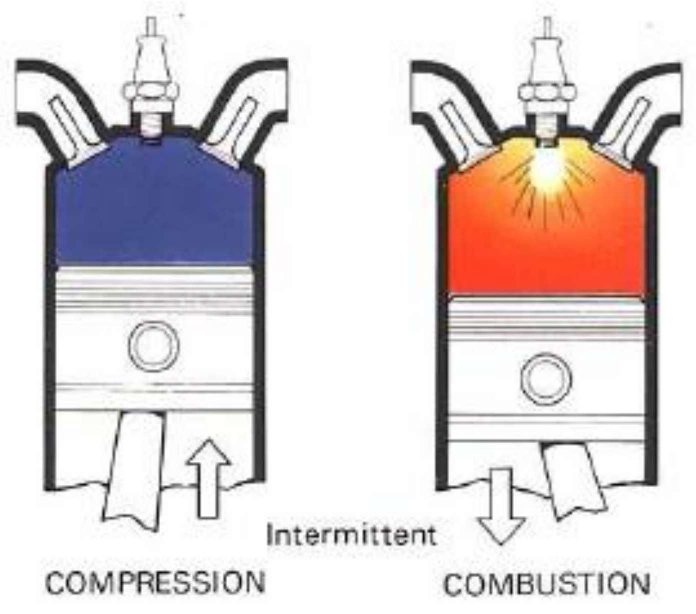
$$q_v = \int_{T_1}^{T_2} c_v(T)dT = c_v(T_2 - T_1)$$

  - *Cv* average specific heat (volume)
- Internal energy increase
- $\Delta u = q_v = c_v(T_2 - T_1)$
  - No sense for absolute value of internal energy

2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Work
  - Mechanical work
  - In thermal process, gas expansion does work on objects
  - Or work can compress gas



2. Aerothermodynamics in Gas Turbine Engine

First law of thermodynamics

- Volume change for a unit mass gas *dv*， then work:
  - $dW = pdv$
- By integration

  P constant: $$W = \int_{v_1}^{v_2} pdv$$

$$W = p(v_2 - v_1)$$

- *If V=constant*, $W = 0$.

2. Aerothermodynamics in Gas Turbine Engine

Enthalpy

- $h = u + pv = u + p/\rho \; (J/kg)$
- u is internal energy
- pv considered as pressure potential energy
- $h(T) = u(T) + p/\rho = u(T) + RT$
- Enthalpy, a state parameter, is a single value function of T.

$$q_p = \int_{T_1}^{T_2} c_p(T)dT = c_p(T_2 - T_1) = c_v(T_2 - T_1) + p(v_2 - v_1)$$

- Since, Heat = U + W,

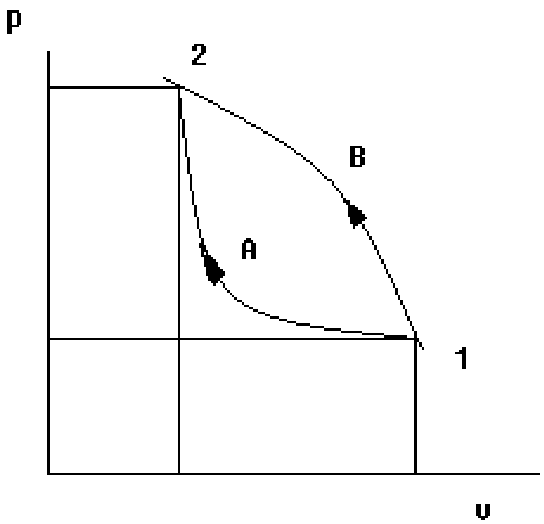
$$= (c_v + R)(T_2 - T_1) = \Delta h$$

$$\therefore \Delta h = c_p(T_2 - T_1)$$

2. Aerothermodynamics in Gas Turbine Engine

Enthalpy

- $dq = du + pdv$ = Relation of heat, internal energy and work (transformations and conservaton)
- $u = h - pv$
- $du = dh - d(pv) = dh - pdv - vdp$
- $dq = dh - vdp$ (pressure work)
- $or,\ dh = dq + vdp$
- *For no heat addition, $dh = vdp$)*



$$q = h_2 - h_1 - \int_{p_1}^{p_2} vdp = c_p(T_2 - T_1) - \int_{p_1}^{p_2} vdp$$

- *For constant pressure combustion, $dq = dh$*

2. Aerothermodynamics in Gas Turbine Engine

Adiabatic Process

$$c_v dT = \frac{c_v}{R}(vdp + pdv) \qquad pv = RT$$

$$c_v dT + pdv = 0$$

$$\frac{c_v}{c_p - c_v}(vdp + pdv) + pdv = 0$$

$$\frac{dp}{p} + \gamma \frac{dv}{v} = 0$$

$$\frac{dp}{d\rho} = \gamma \frac{p}{\rho} \qquad \left(\frac{dp}{d\rho}\right)_s = a^2$$

2. Aerothermodynamics in Gas Turbine Engine

- $pv^{\gamma} = constant$

$$\frac{p_2}{p_1} = \left(\frac{v_1}{v_2}\right)^{\gamma} = \left(\frac{T_2}{T_1}\right)^{\frac{\gamma}{\gamma-1}}$$

Adiabatic Process

$$W = -\Delta u = Cv(T_1 - T_2) = \frac{R}{\gamma - 1}(T_1 - T_2) = \frac{RT_1}{\gamma - 1}\left[1 - \left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}}\right]$$

- I.E. =>Work

2. Aerothermodynamics in Gas Turbine Engine

- Reversible process
  - Path 1=>2 is the same as 2=>1
  - Uniform, no friction——gas, machine, gas/machine
- Irreversible process
  - Friction exists
- Polytropic process
  - pv^n=constant

Polytropic Process

$$\frac{p_2}{p_1} = \left(\frac{v_1}{v_2}\right)^n = \left(\frac{T_2}{T_1}\right)^{\frac{n}{n-1}}$$

2. Aerothermodynamics in Gas Turbine Engine

Irreversible & Polytropic Process

$$\frac{p_2}{p_1} = \left(\frac{v_1}{v_2}\right)^n = \left(\frac{T_2}{T_1}\right)^{\frac{n}{n-1}}$$

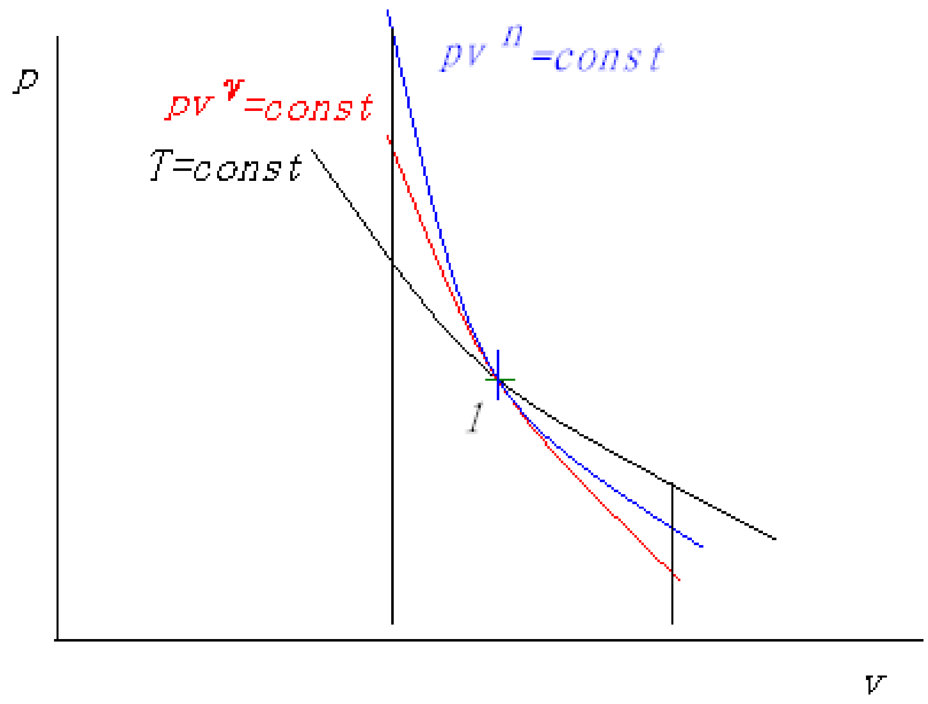
n polytropic index

Const P	n= 0	*p*= const
Const V	n= ∞	*v*= const
Const T	n= 1	*T*= const
Adiabatic	n= γ	pv^{γ}= const
Irreversible	Compression n> g Expansion 1<n<g	

2. Aerothermodynamics in Gas Turbine Engine

Irreversible & Polytropic Process

- pv^n = const, intersect at ‘1’
- Compression
  - $W = W_n + q_{in}$
  - *Add more work for compression*
- Expansion
  - $W = W_n - q_{in}$
  - Get less work done at expansion



2. Aerothermodynamics in Gas Turbine Engine

Irreversible & Polytropic Process

$$W_n = \frac{R}{n-1}(T_1 - T_2) = \frac{RT_1}{n-1}\left(1 - \frac{1}{\left(\frac{v_2}{v_1}\right)^{n-1}}\right) = \frac{RT_1}{n-1}\left(1 - \left(\frac{p_2}{p_1}\right)^{\frac{n-1}{n}}\right)$$

2. Aerothermodynamics in Gas Turbine Engine

Irreversible & Polytropic Process

- Entropy

$$ds = \frac{dq}{T} \quad \text{J/kg.K}$$

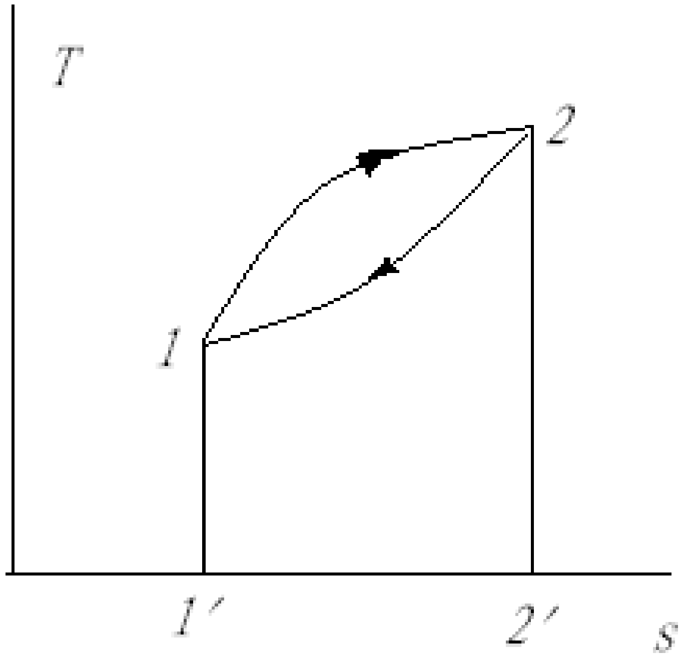
$$\left.\begin{aligned} \Delta s &= Cv \ln \frac{T_2}{T_1} + R \ln \frac{v_2}{v_1} \\ &= Cp \ln \frac{T_2}{T_1} - R \ln \frac{p_2}{p_1} \\ &= Cp \ln \frac{v_2}{v_1} + Cv \ln \frac{p_2}{p_1} \end{aligned}\right\} \qquad q = \int_{s_1}^{s_2} T ds$$

2. Aerothermodynamics in Gas Turbine Engine

Irreversible & Polytropic Process

$$q = \int_{S_1}^{S_2} T ds$$

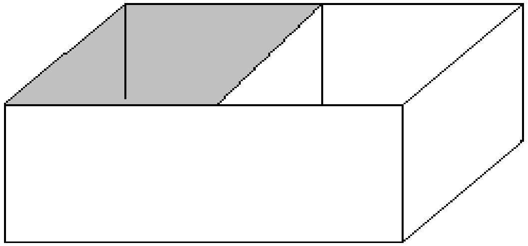
- Entropy is an abstract parameter for a useful thermodynamic concept ('entity')
- State parameter, that is a function of any two out of *p*， *v and T*
- Can be used to get heat
  - d*q* = *T*d*s* J/kg
  - d*q* and d*s* retain same sign
- When irreversible, *q* contains q_{in} (friction heat)



2. Aerothermodynamics in Gas Turbine Engine

Second law of thermodynamics

- Any natural thermal process is irreversible.
- Example:



$$\Delta s = Cp \ln \frac{v_2}{v_1} + Cv \ln \frac{p_2}{p_1} = (Cp - Cv) \ln 2$$

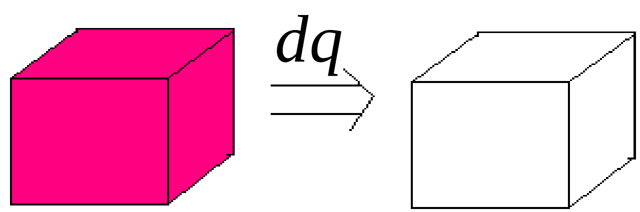
$\Delta s > 0$

2. Aerothermodynamics in Gas Turbine Engine

Second law of thermodynamics

- Or,
- *Without work, heat can NOT automatically go from LT to HT or it goes always naturally from HT to LT.*

$$\Delta s_1 = \int \frac{dq_1}{T_1}$$



$$\Delta s_2 = \int \frac{dq_2}{T_2}$$

$$\Delta s = \int \left(\frac{1}{T_2} - \frac{1}{T_1} \right) dq \qquad T_1 > T_2$$

2. Aerothermodynamics in Gas Turbine Engine

Second law of thermodynamics

- Unique heat source engine can not be made.
- Natural thermal process follows path of entropy increase.
- Naturally go **from ordered to unordered**.
- Notes
    - Naturally
    - Isolated system
    - Can be used in other domain, such as in social sciences

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

- From above equations, flow kinetic energy (speed), enthalpy and pressure potential energy can be converted from one to others.
- If flow stagnates (U= 0) as isentropic process, the kinetic energy is converted totally to enthalpy. It is called stagnation enthalpy, or total enthalpy:

$$h^* = c_p T + \frac{U^2}{2} \qquad U \uparrow \Rightarrow h \downarrow \quad \text{and} \quad p \downarrow$$

- U: ordered movement
- T: disordered movement

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

- Corresponding stagnation temperature or total temperature:

$$T^* = T + \frac{U^2}{2c_p}$$

- Since,

$$c_p = \frac{\gamma}{\gamma - 1} R \qquad\qquad c = \sqrt{\gamma R T}$$

$$\frac{T^*}{T} = 1 + \frac{\gamma - 1}{2} M^2$$

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

- With the stagnation process under isentropic assumption:

$$\frac{P^*}{p} = \left(\frac{v}{v^*}\right)^{\gamma} = \left(\frac{T^*}{T}\right)^{\frac{\gamma}{\gamma-1}}$$

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

$$\frac{\rho^*}{\rho} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{1}{\gamma-1}}$$

2. Aerothermodynamics in Gas Turbine Engine

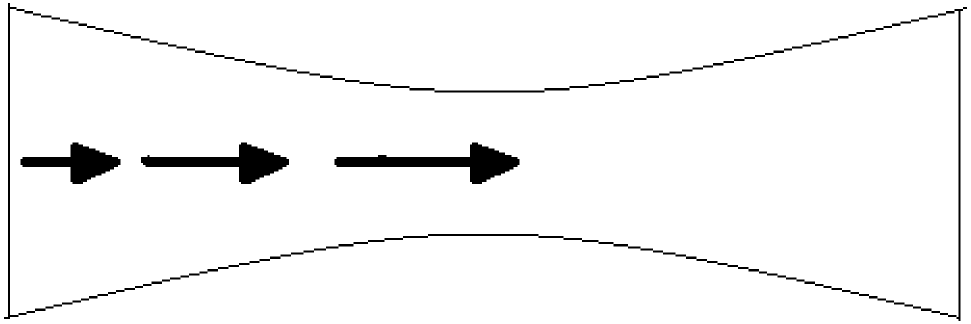
The Stagnation Process

- According to the 3 equations above, for a given gas flow, the ratios of the total parameters and steady parameters are function of Mach number.
- When air flows isentropically in a tube without energy added, the total parameters (Enthalpy, temperature, pressure and density) remain unchanged.

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

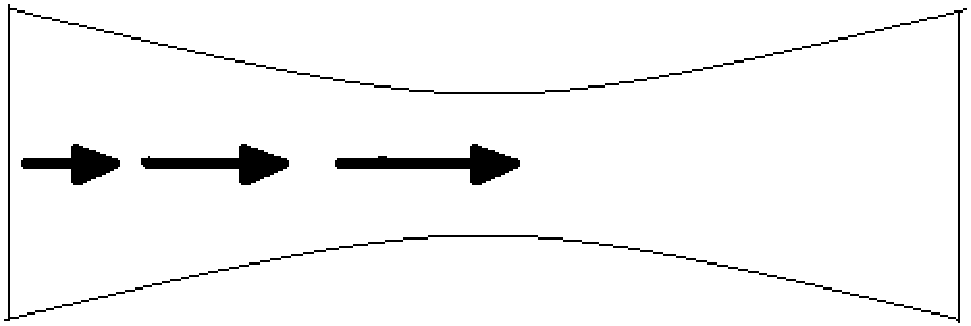
- Critical sound speed c_{cr} (in tunnel)



- U increases along the tunnel (ex. Laval nozzle)
- When U increases, T decreases. Sound speed c is function of T, it goes down.

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process



- When $U = c$, ie M= 1, this c has special meaning, called **critical sound speed c_{cr}**. This section is called critical section and it is the smallest section in the tunnel, also called **throat**.
- c_{cr} does not change, even when c does.

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

- Definition of **speed coefficient** in a section

$$\lambda = \frac{U}{c_{cr}}$$

- Writing the stagnation state relation $\frac{T^*}{T} = 1 + \frac{\gamma - 1}{2} M^2$ in terms of the critical condition, we obtain

$$T_{cr} = \frac{2}{\gamma + 1} T^*$$

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

➢ Thus

$$T_{cr} = \frac{2}{\gamma + 1} T^*$$

$$c_{cr}^2 = \frac{2}{\gamma + 1} \gamma R T^*$$

$$c_{cr} = \sqrt{\frac{2}{\gamma + 1} \gamma R T^*}$$

$$\lambda^2 = \frac{\frac{\gamma + 1}{2} M^2}{1 + \frac{\gamma - 1}{2} M^2}$$

$$M^2 = \frac{\frac{2}{\gamma + 1} \lambda^2}{1 - \frac{\gamma - 1}{\gamma + 1} \lambda^2}$$

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

- Therefore, the stagnation state equations are written in terms of the speed coefficient as,

$$\tau(\lambda) = \frac{T}{T^*} = \left(1 - \frac{\gamma - 1}{\gamma + 1}\lambda^2\right)$$

$$\pi(\lambda) = \frac{p}{p^*} = \left(1 - \frac{\gamma - 1}{\gamma + 1}\lambda^2\right)^{\frac{\gamma}{\gamma - 1}}$$

$$\varepsilon(\lambda) = \frac{\rho}{\rho^*} = \left(1 - \frac{\gamma - 1}{\gamma + 1}\lambda^2\right)^{\frac{1}{\gamma - 1}}$$

2. Aerothermodynamics in Gas Turbine Engine

The Stagnation Process

- **Flow density function**

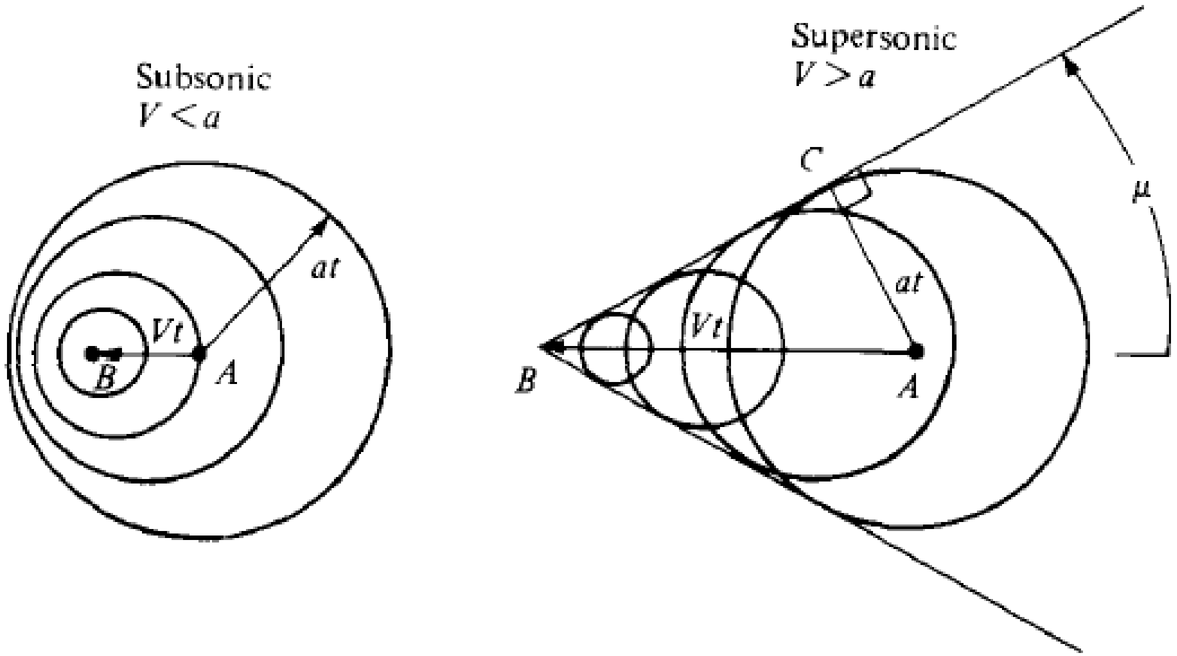
$$q_m = \rho v A$$

$$q(\lambda) = \frac{\rho v}{(\rho v)_{cr}} = \frac{A_{cr}}{A} = \left(\frac{\gamma+1}{2}\right)^{\frac{1}{\gamma-1}} \lambda \left(1 - \frac{\gamma-1}{\gamma+1}\lambda^2\right)^{\frac{1}{\gamma-1}}$$

- $q(\lambda)$ presents relative flow density in section A to the critical section even though the critical section does not exist.
- Ratio of the cross-sections.

2. Aerothermodynamics in Gas Turbine Engine

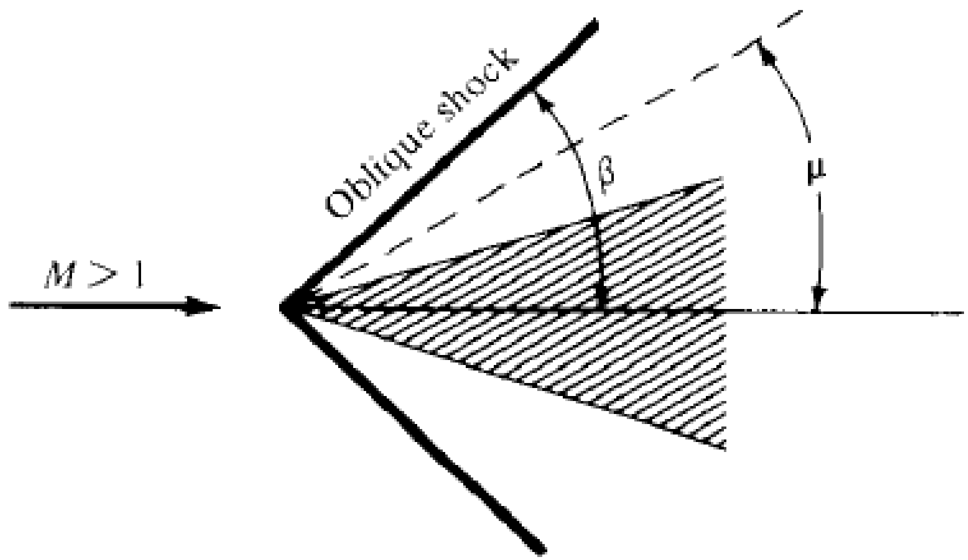
Shock Waves



- *If the disturbances are stronger than a simple sound wave, then the wave front becomes stronger than a Mach wave, creating an oblique shock wave at an angle β to the freestream, where $\beta > \mu$.*

2. Aerothermodynamics in Gas Turbine Engine

Shock Waves



- ➢ *The Mach wave is a limiting case for an oblique shock wave, i.e. it is an infinitely weak shock wave.*

2. Aerothermodynamics in Gas Turbine Engine

Shock Waves

$$M_{n,1} = M_1 \sin\beta$$

$$M_2 = \frac{M_{n,2}}{\sin(\beta - \theta)}$$

$$M_{n,2}^2 = \frac{1 + [(\gamma - 1)/2] M_{n,1}^2}{\gamma M_{n,1}^2 - (\gamma - 1)/2}$$

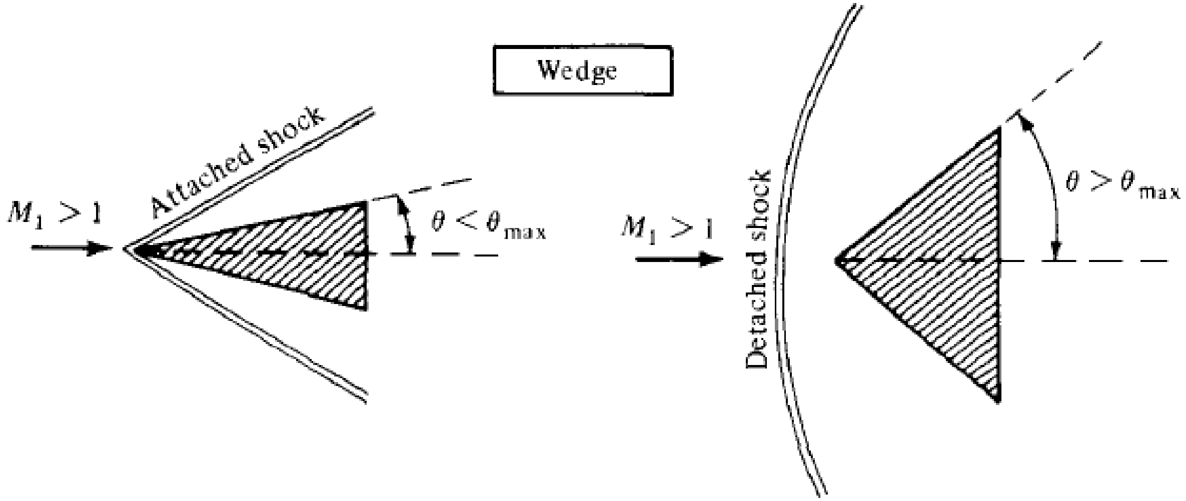
$$\frac{\rho_2}{\rho_1} = \frac{(\gamma + 1) M_{n,1}^2}{2 + (\gamma - 1) M_{n,1}^2}$$

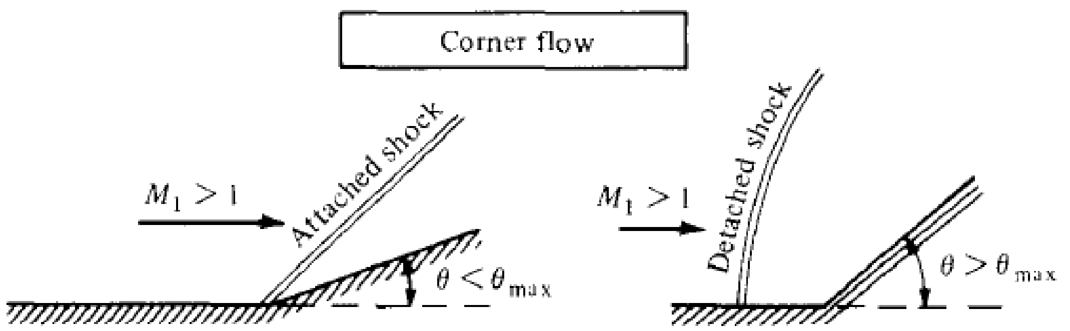
$$\frac{p_2}{p_1} = 1 + \frac{2\gamma}{\gamma + 1}(M_{n,1}^2 - 1)$$

$$\boxed{\tan\theta = 2\cot\beta \frac{M_1^2 \sin^2\beta - 1}{M_1^2(\gamma + \cos 2\beta) + 2}}$$

2. Aerothermodynamics in Gas Turbine Engine

Shock Waves





2. Aerothermodynamics in Gas Turbine Engine

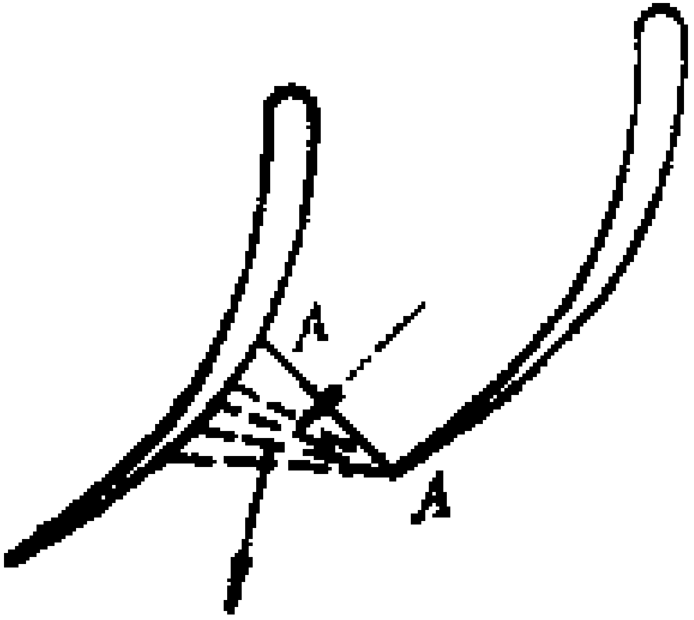
The Shock and Expansion Waves

- Supersonic flow passing through the shock wave is not isentropic process. Partial mechanical energy Irreversibly changes to heat, and total pressure decreases.
- This is shock wave loss, and usually total pressure recovery (σ) is used to present the loss. It is function of the wave strength; the stronger the wave, the greater the loss.

2. Aerothermodynamics in Gas Turbine Engine

The Shock and Expansion Waves

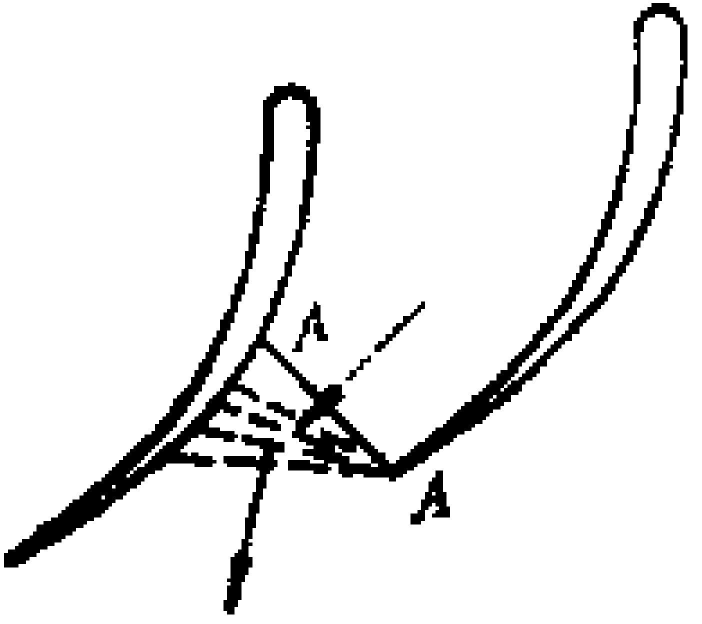
- When a supersonic air flows to a lower pressure zone, there are expansion waves due to air continuous expansion.
- In Fig, turbine cascade passage. In the throat A—A, critical section, flow becomes supersonic.
- In downstream, it is low pressure zone. The flow accelerates, it passes through a series of expansion waves, and speed increases, temperature and pressure decrease.



2. Aerothermodynamics in Gas Turbine Engine

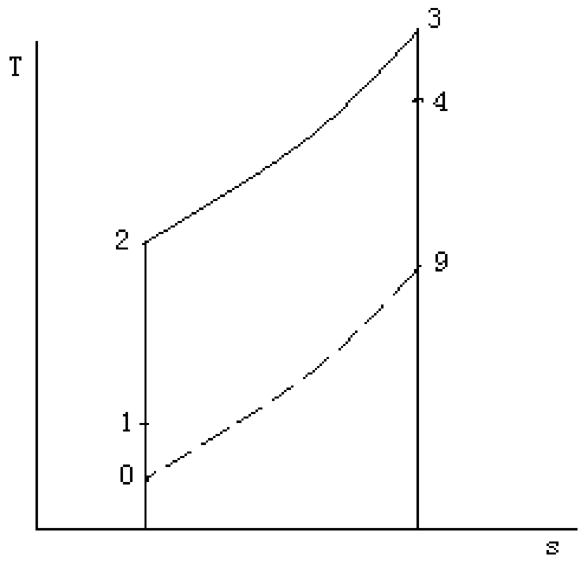
The Shock and Expansion Waves

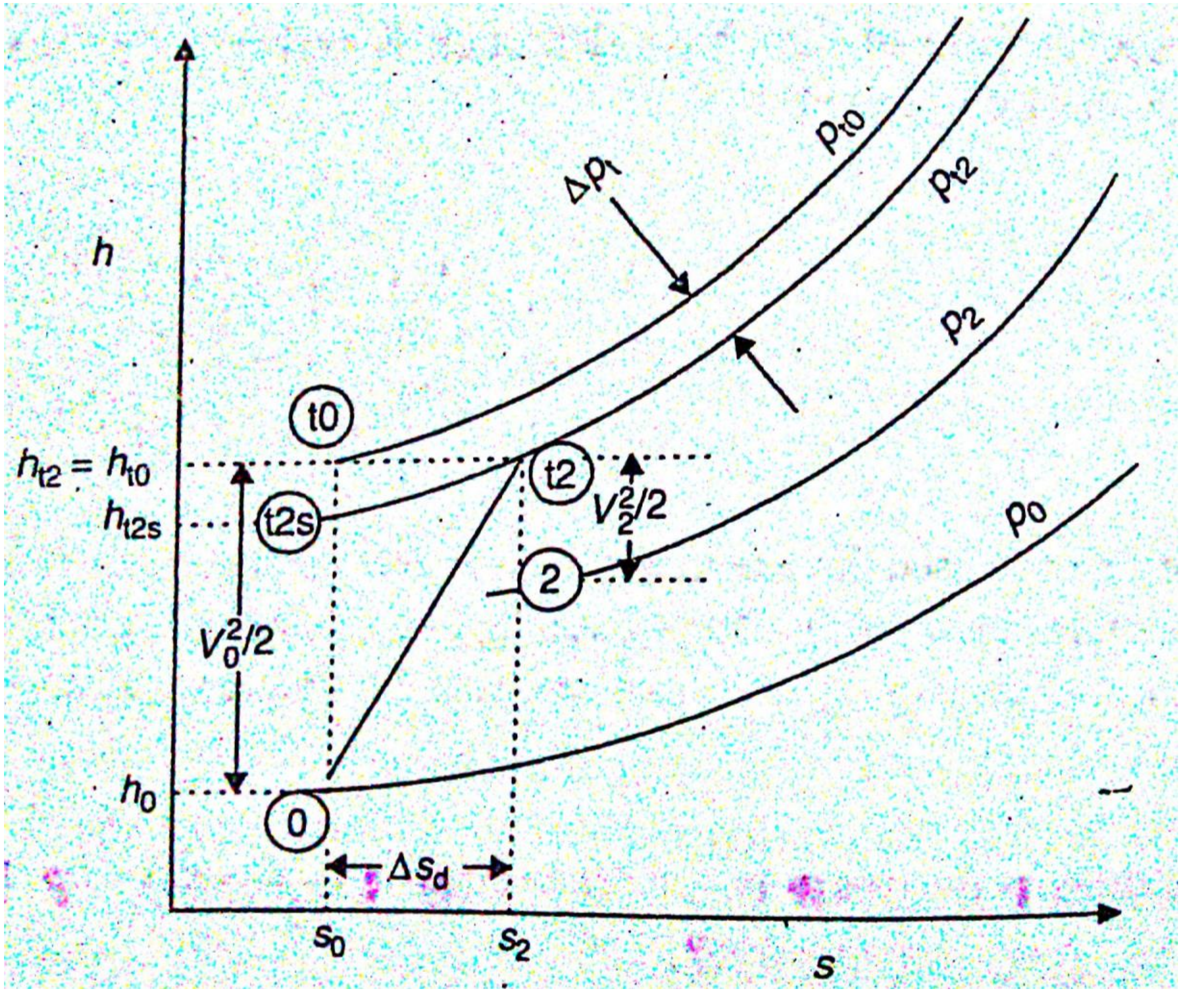
- The flow changes also the direction. The bigger the turned angle, the more expansion and flow parameters change more.
- The turned angle depends on exit pressure. The lower the pressure, the bigger the angle.
- If pressure increases, expansion waves may disappear and the flow may be subsonic.



2. Aerothermodynamics in Gas Turbine Engine

Gas Turbine Cycle



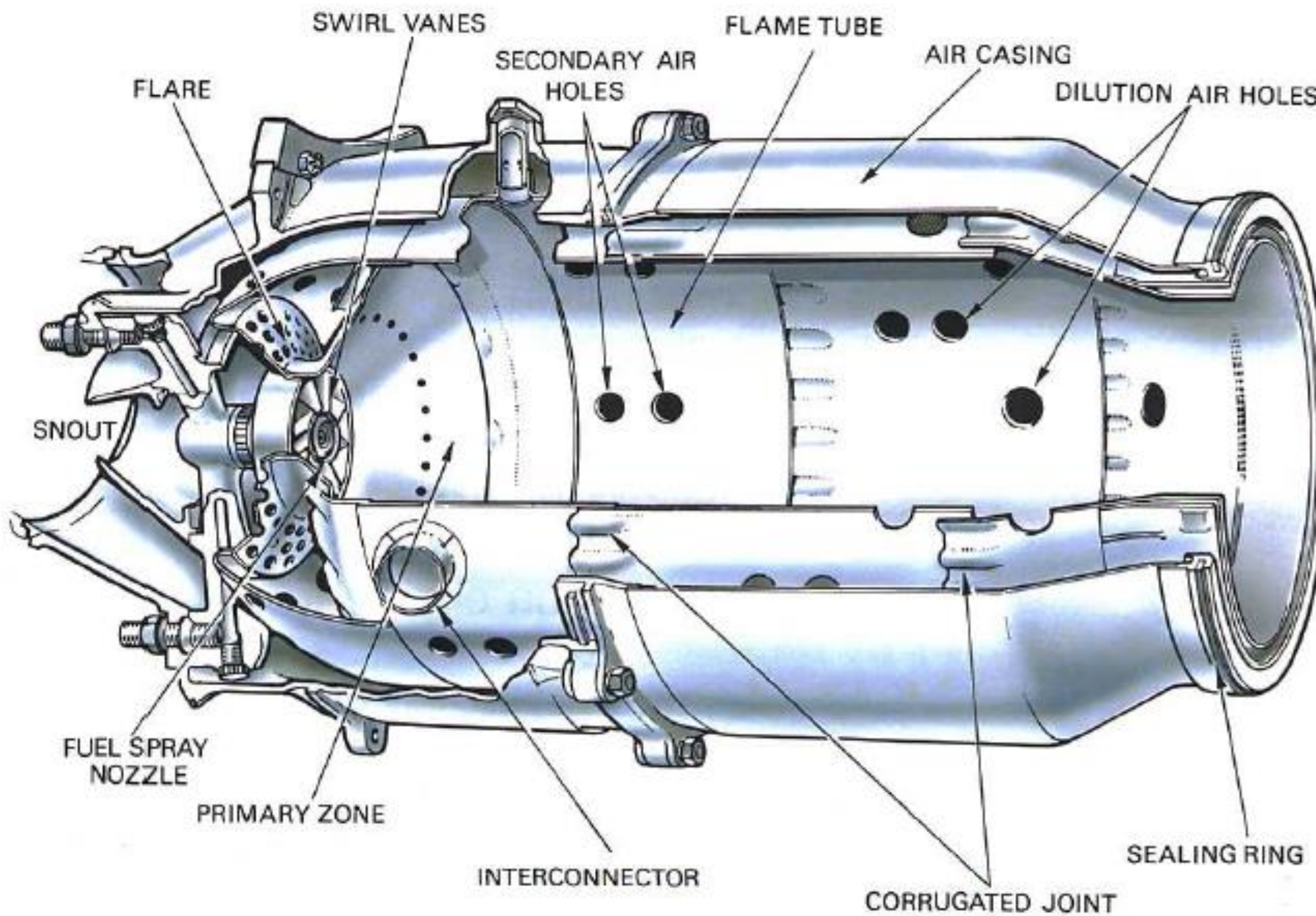


2. Aerothermodynamics in Gas Turbine Engine

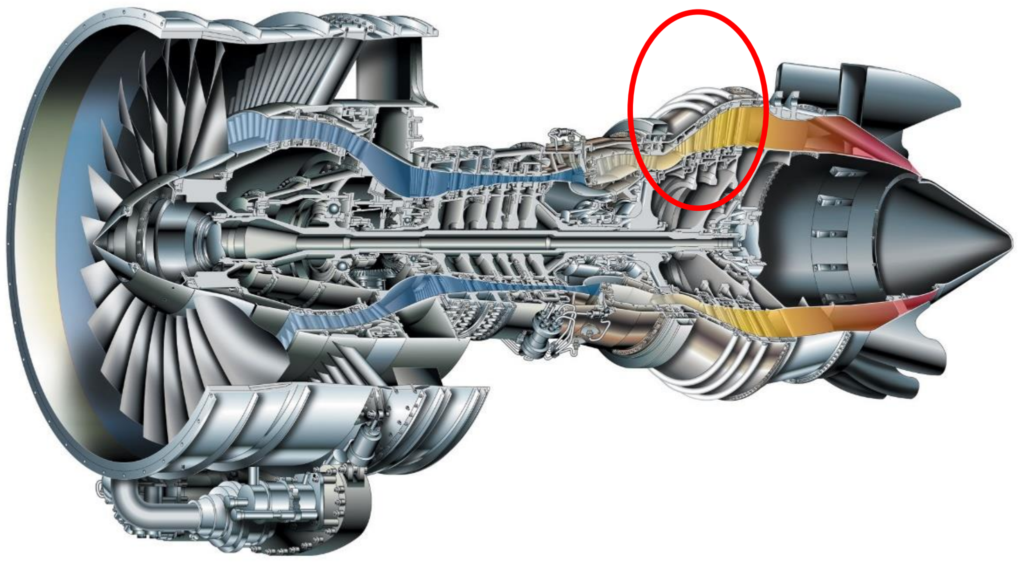
The Shock and Expansion Waves

- When supersonic flow passes through the shock wave, sharply speed decreases, pressure and temperature increase.
- After normal wave, the flow is certainly subsonic. But after oblique shock wave, it is still supersonic.
- Strength of the shock wave is described by pressure ratio of after and before. It is only function of M for normal shock wave, the greater M, the stronger the wave.
- For oblique shock wave, the greater M and q, the stronger the wave.

Combustion chamber



Turbine



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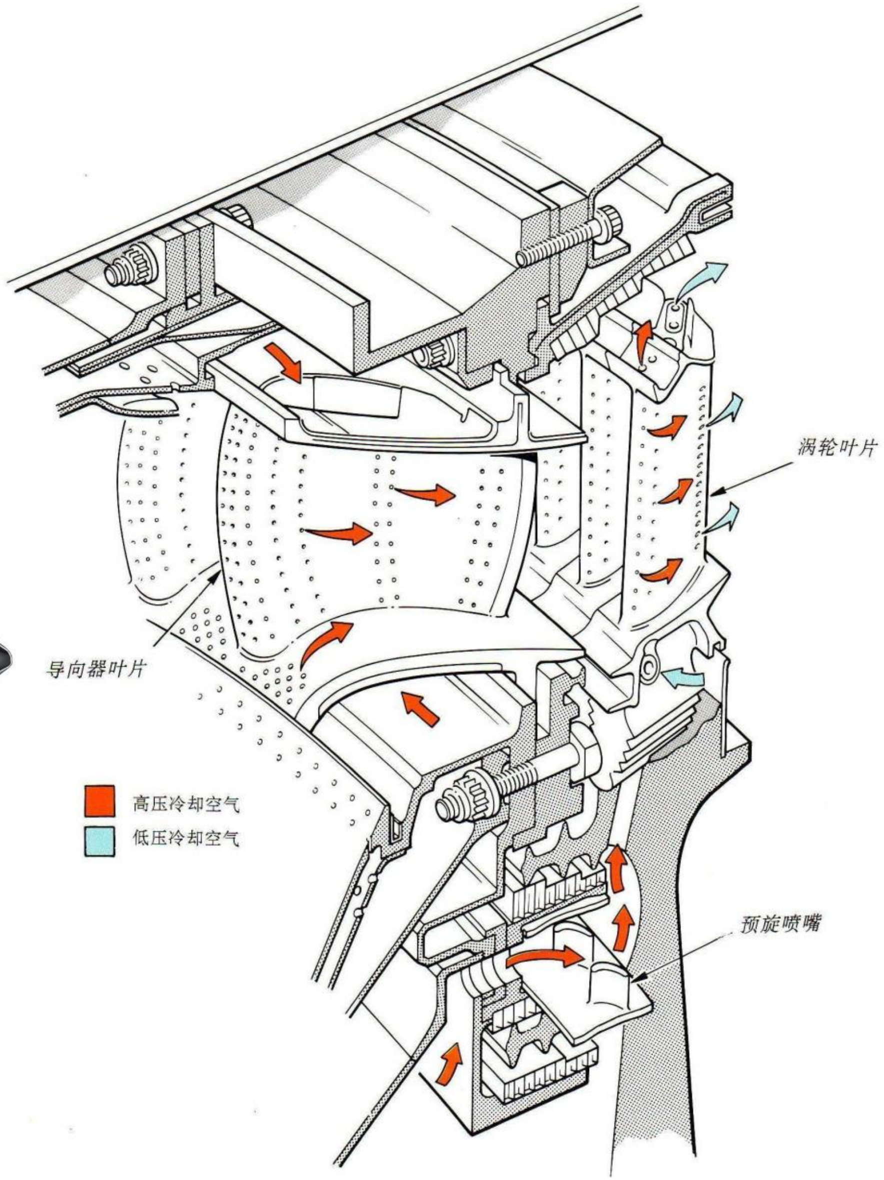
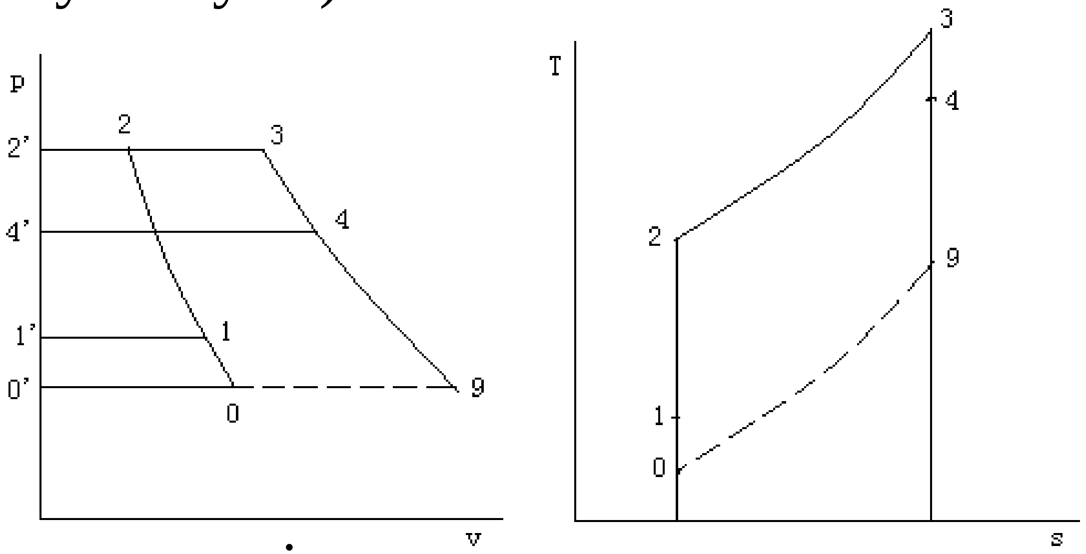


2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- 1. Ideal cycle (Brayton cycle)

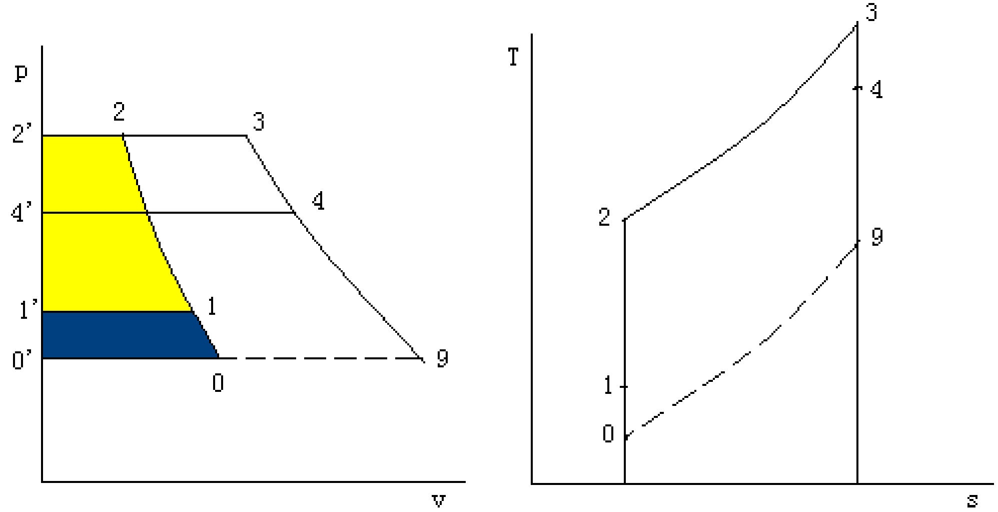


- 0-2 Isentropic compression
- 2-3 Isobar heating
- 3-9 Isentropic expansion
- 9-0 Isobar cooling

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- 0-2 Isentropic compression

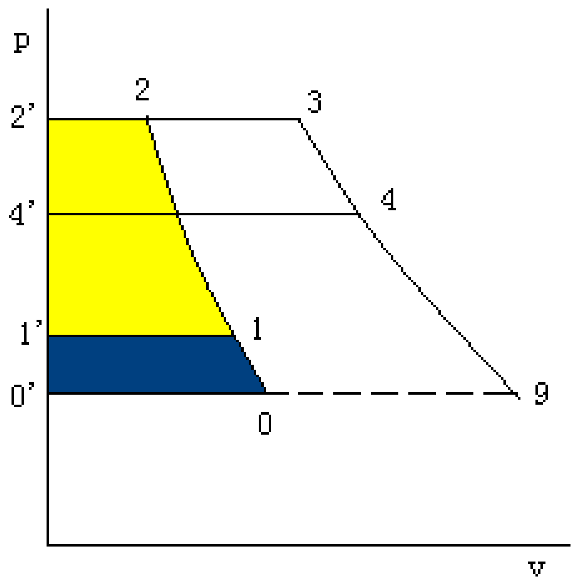


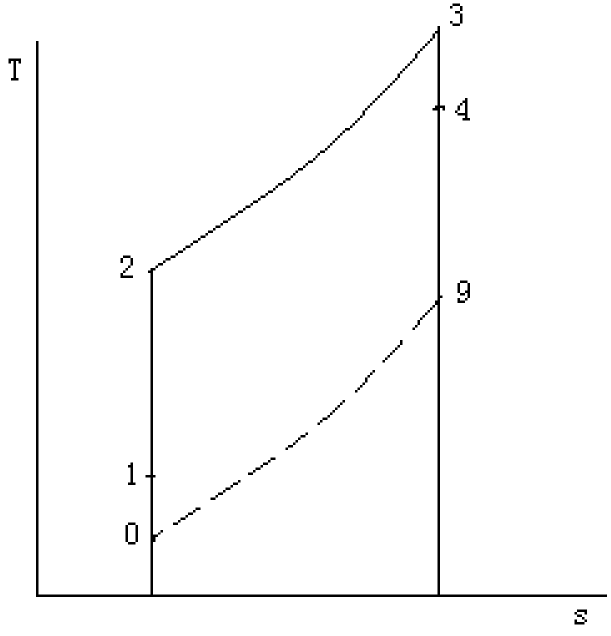
- Diffuser and compressor
- 0-1 speed pressure rise. 0 atmosphere condition. Add dynamic energy to substance and to increase pressure to 1. Area 011'0'0 presents dynamic energy difference.

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- 0-2 Isentropic compression



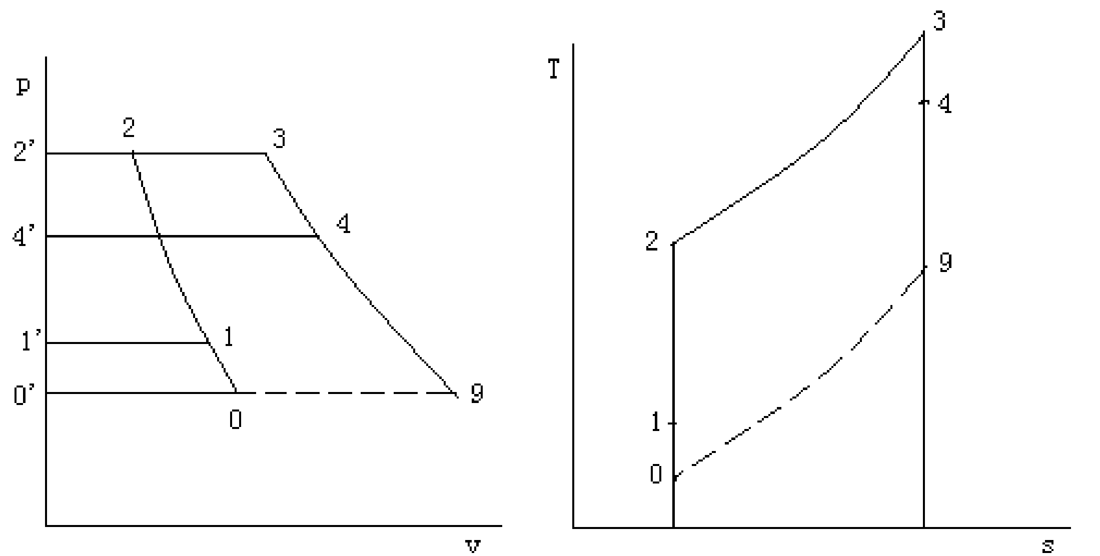


- 1-2: compressor. Pressure from 1 to 2, work added is the area 122'1'1.

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- 2-3 Isobar heating

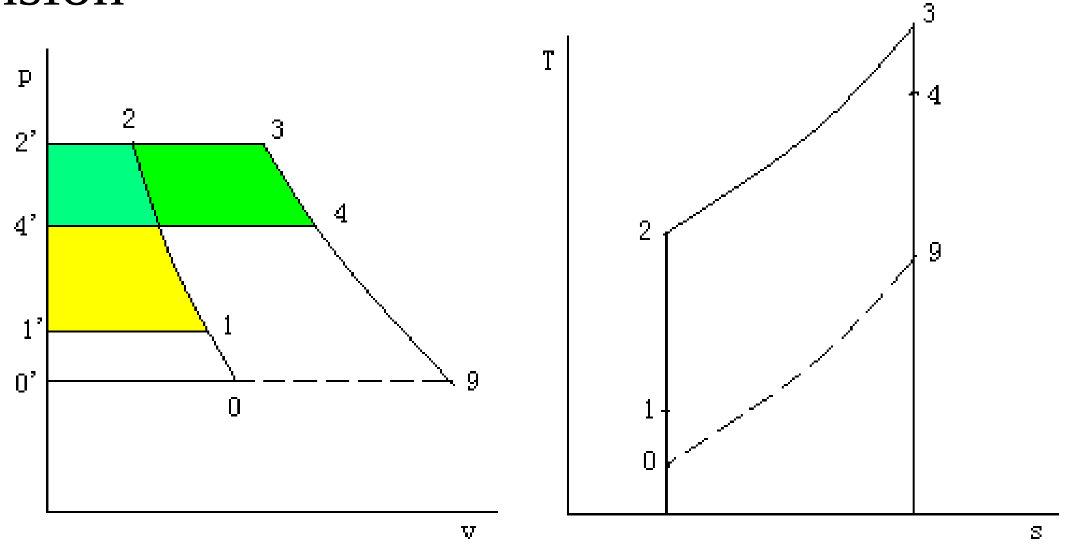


  - Combustion chamber
  - Burn ideally kerosene at constant pressure in combustion chamber and substance properties unchanged.
  - Total temperature $T_2^* \Rightarrow T_3^*$ 。

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- 3-9 Isentropic expansion

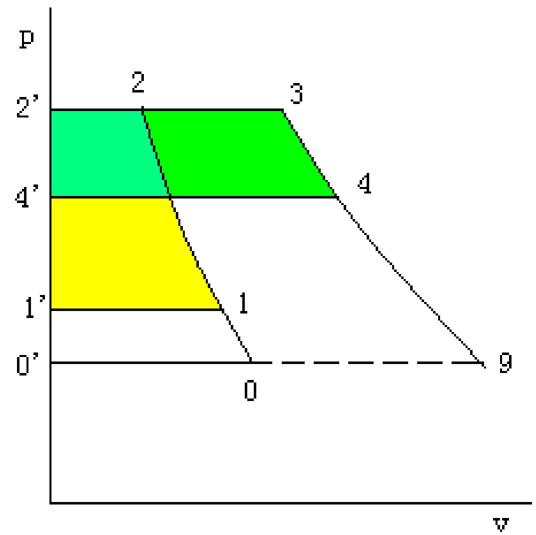


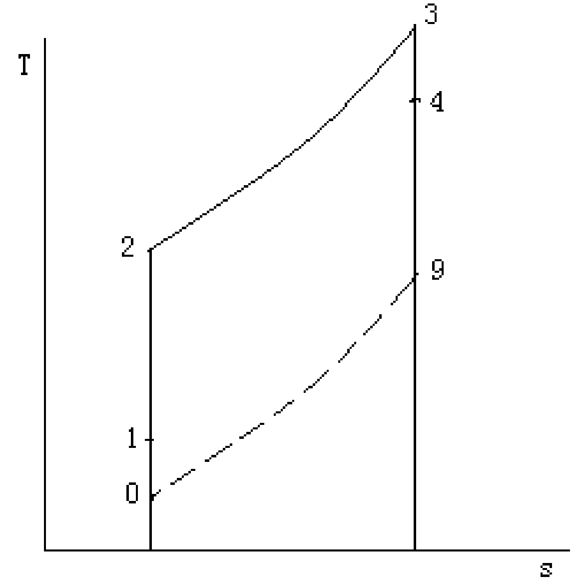
  - Turbine and nozzle
  - 3-4 presents expansion in turbine, heat ⇒ mechanical energy giving to compressor. Area 344'2'3=Area 122'1'1, total pressure $p_3^* \Rightarrow p_4^*$.

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- 3-9 Isentropic expansion



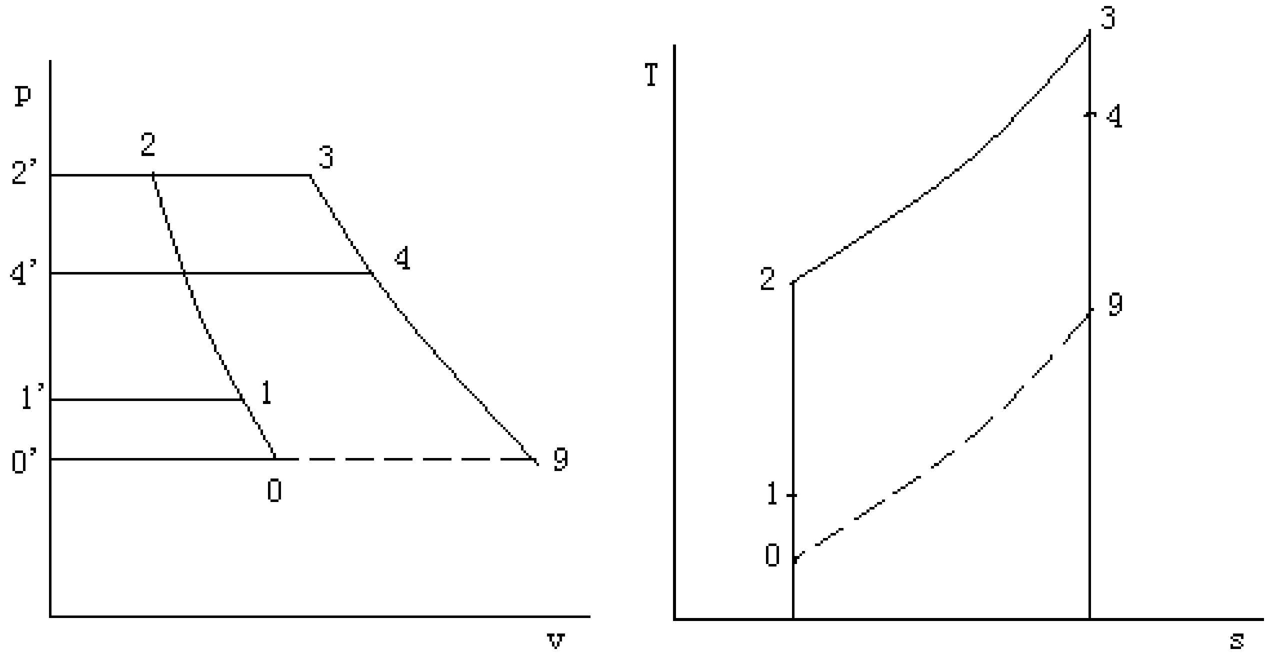


  - 4-9 complete expansion in exhaust system. Heat changes kinetic energy in substance, exits from the nozzle.
  - As diffuser, kinetic energy change can be seen as output work.

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- 9-0 Isobar heat release



- Dash line, accomplished in atmosphere. This process is unavoidable, The cycle is closed.

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- Specific heat added in the cycle

$$q_1 = c_p (T_3^* - T_2^*)$$

- Specific heat lost to the surrounding

$$q_2 = c_p (T_9 - T_0)$$

- Specific work in the cycle

$$W = q_1 - q_2$$

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- Thermal efficiency in the cycle

$$\eta_t = \frac{W}{q_1} = \frac{q_1 - q_2}{q_1} = 1 - \frac{1}{\pi^{\frac{\gamma-1}{\gamma}}}$$

- Where pressure ratio

$$\pi = p_2^* / p_0$$

2. Aerothermodynamics in Gas Turbine Engine

Thermodynamic Cycle

- Cycle work as mechanical energy

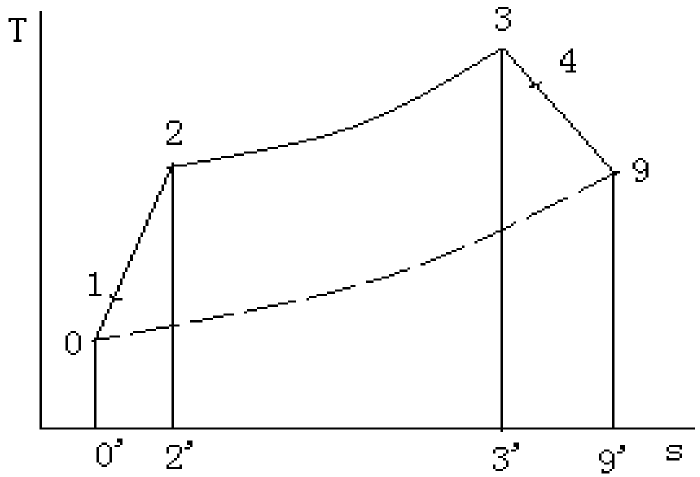
$$W = W_T + \frac{v_9^2}{2} - W_C - \frac{v_0^2}{2}$$

- if $W_T = W_C$

$$W = \frac{v_9^2 - v_0^2}{2}$$

2. Aerothermodynamics in Gas Turbine Engine

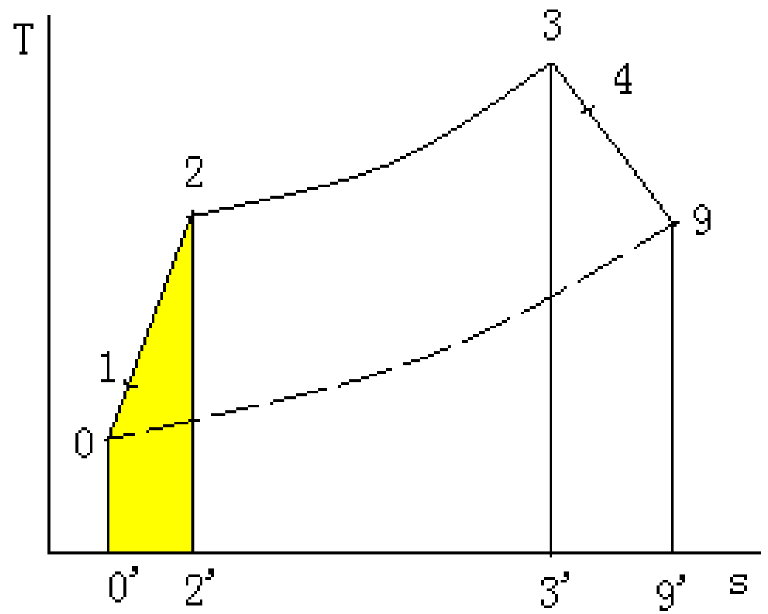
Actual Thermodynamic Cycle



- 0-2 Compression (non isentropic)
- 2-3 Heating (non isobar)
- 3-9 Expansion (non isentropic)
- 9-0 Isobar heat release

2. Aerothermodynamics in Gas Turbine Engine

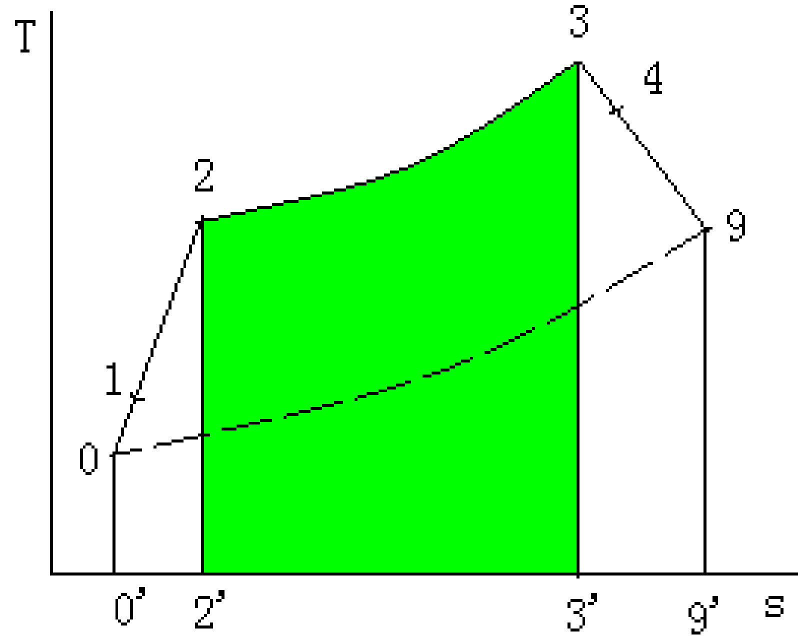
Actual Thermodynamic Cycle



- 0-2 Compression
  - Stagnation in diffuser and compression in compressor suffer from many types of losses.
  - Non isentropic

2. Aerothermodynamics in Gas Turbine Engine

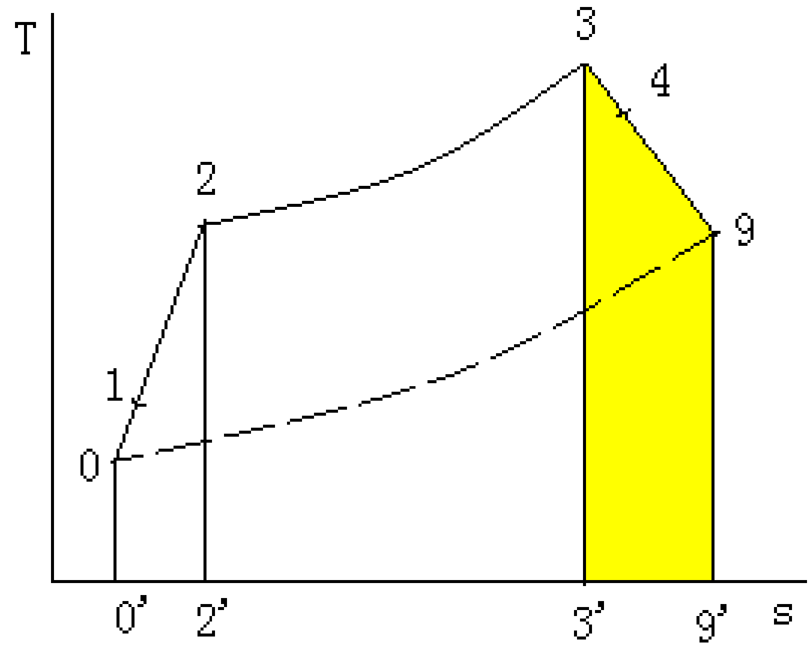
Actual Thermodynamic Cycle



- 2-3 Non isobar combustion
  - Existing flow losses and thermal resistance losses lowers the pressure in combustion chamber.
  - Composition of substance changes.

2. Aerothermodynamics in Gas Turbine Engine

Actual Thermodynamic Cycle



- 3-9 Expansion
  - There are always losses in turbine and nozzle.
  - Non isentropic

2. Aerothermodynamics in Gas Turbine Engine

Actual Thermodynamic Cycle

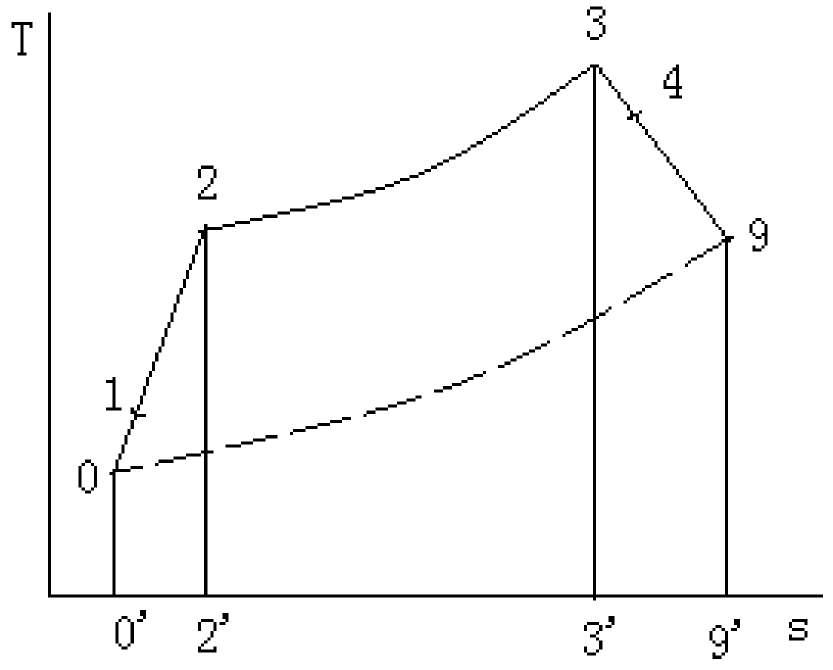
- Heat added:
  - Area 2’233’2’

$$q_1 = c_p (T_3^* - T_2^*)$$

- Heat released
  - Area 0’099’0’

$$q_2 = c_p^{'} (T_9 - T_0)$$

- c_p’ gas specific heat



2. Aerothermodynamics in Gas Turbine Engine

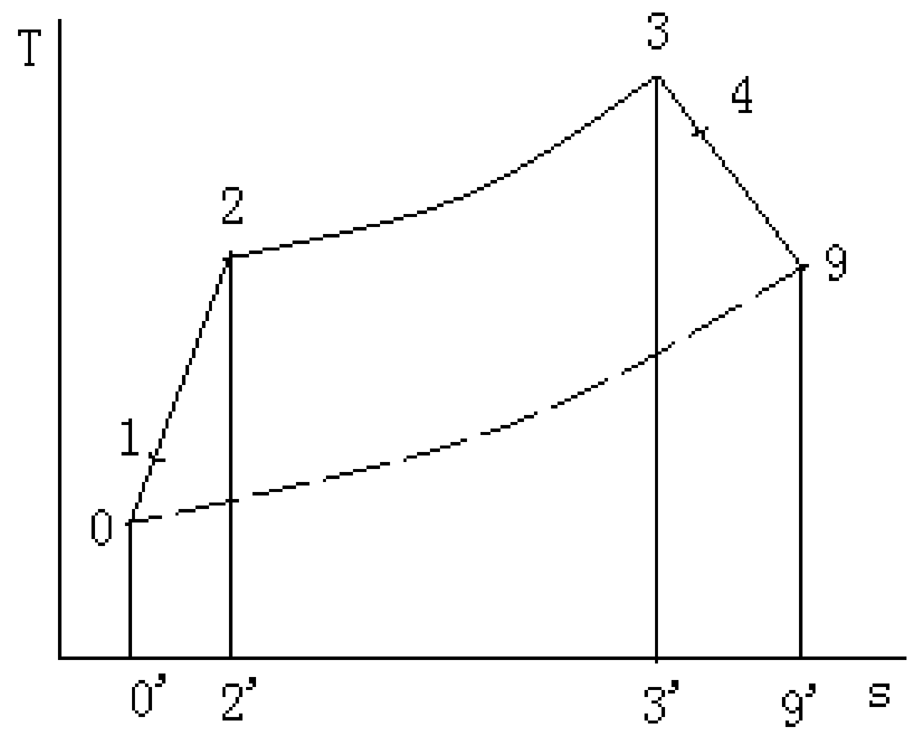
Actual Thermodynamic Cycle

- Efficiency

$$\eta_t = \frac{q_1 - q_2}{q_1}$$

- Work

$$W = q_1 - q_2$$

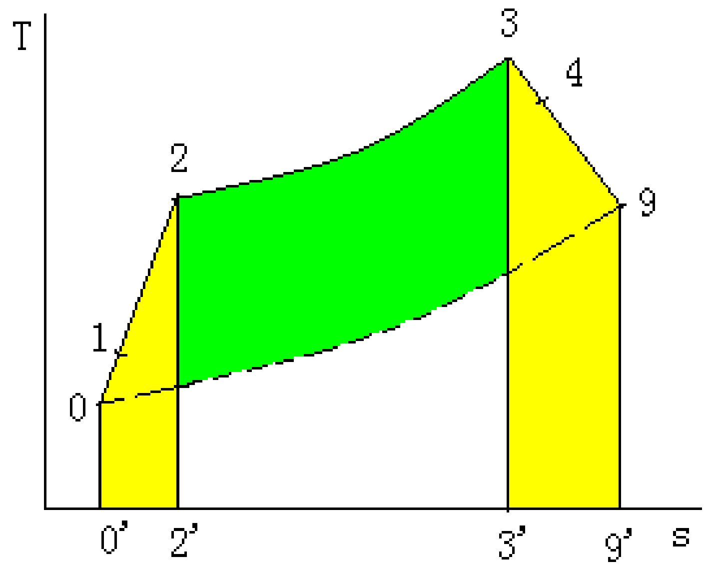


2. Aerothermodynamics in Gas Turbine Engine

Actual Thermodynamic Cycle

- If T_3* lower, q_1=q_2, then η_t=0, no output work.
- Work can be presented by mechanical energy, same as ideal cycle:

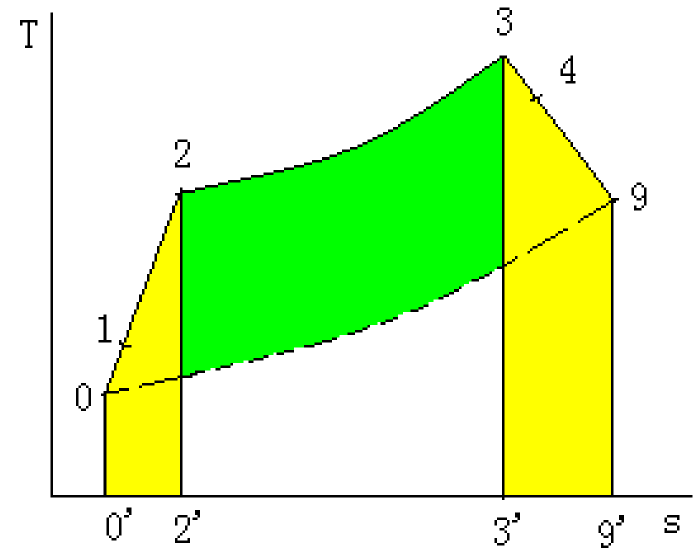
$$W=\frac{v_9^2-v_0^2}{2}$$



2. Aerothermodynamics in Gas Turbine Engine

Actual Thermodynamic Cycle

- Under the same pressure ratio *p* and the same T_3*, work is smaller in real cycle than ideal cycle.
- Note that area in diagram T-s is heat, not work.



2. Aerothermodynamics in Gas Turbine Engine

Actual Thermodynamic Cycle

- If not take into account the change in and increase in mass flow, differences between real cycle and ideal cycle are:
  - Friction and flow losses
  - Total pressure loss
  - Heating resistance
- If nozzle gas kinetic energy is smaller, velocity of air jet is smaller.
- To improve the engine’s efficiency, components with high efficiency and performance should be used.