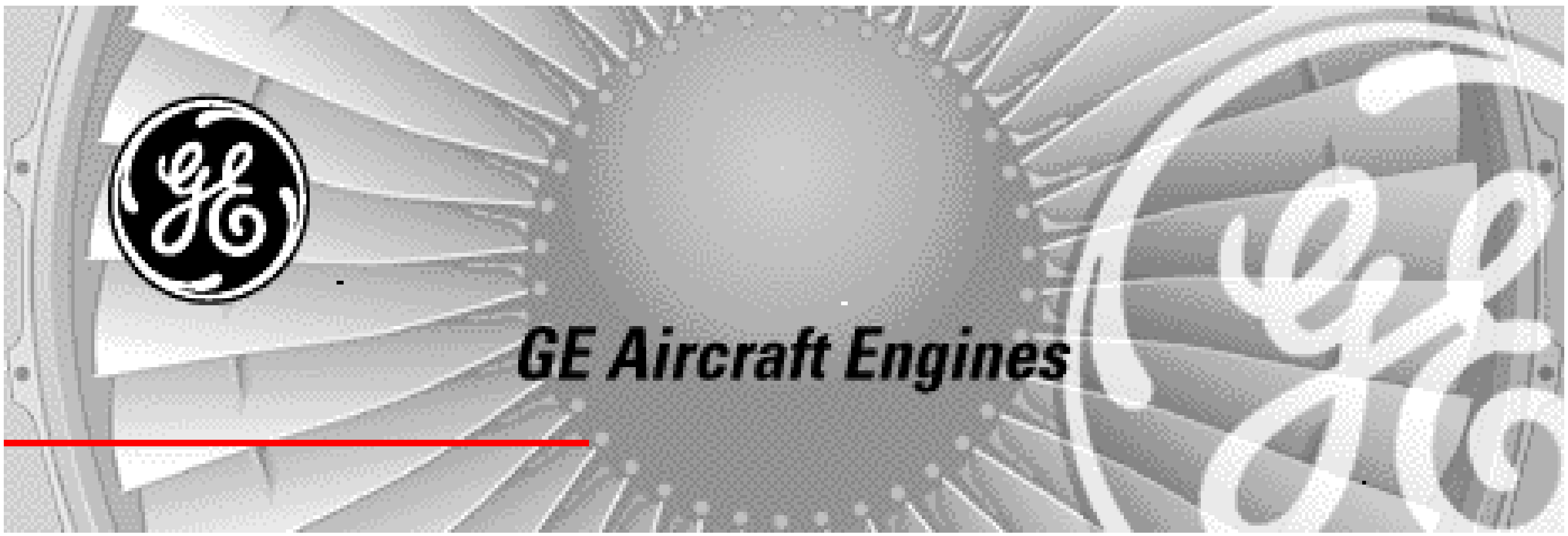
The Aircraft Engine Design Project
Fundamentals of Engine Cycles



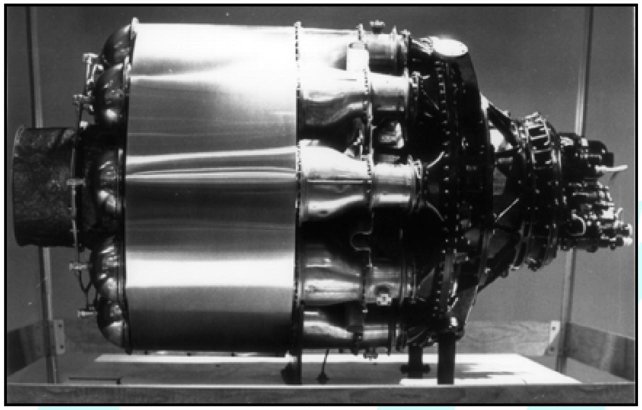
Spring 2009

Ken Gould
Phil Weed

g

GE Aviation Technical History

GE Aircraft Engines

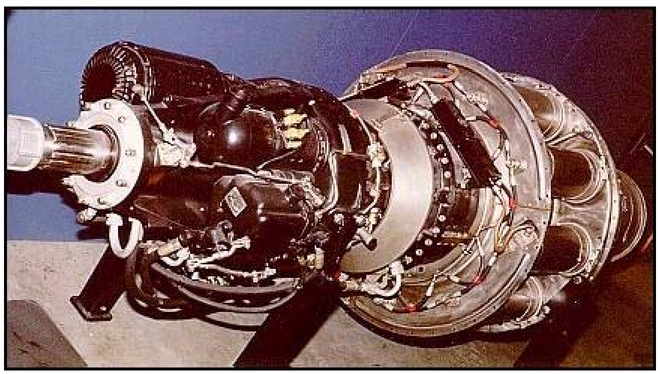


I-A - First U.S. jet engine
(Developed in Lynn, MA, 1941)

First

- U.S. jet engine
- U.S. turboprop engine
- Variable stator engine
- Mach 2 fighter engine
- Mach 3 bomber engine
- High bypass engine
- Variable cycle turbofan engine
- Unducted fan engine
- 30:1 pressure ratio engine
- Demonstration of 100k+ engine thrust
- Certified double annular combustor engine

GE90 on test

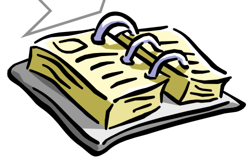


First U.S. turboprop powered aircraft, Dec. 1945

Flowdown of Requirements

The Customer: Overall system requirements

MTOW, Range, Cost per seat per mile



The Airframer: Sub-system requirements

Technical: Wing (lift/drag),**Engines**(Thrust/SFC)

Program: Cost and Schedule

FAA/JAA

Safety/reliability

Noise/emissions

Engines Systems: Module requirements

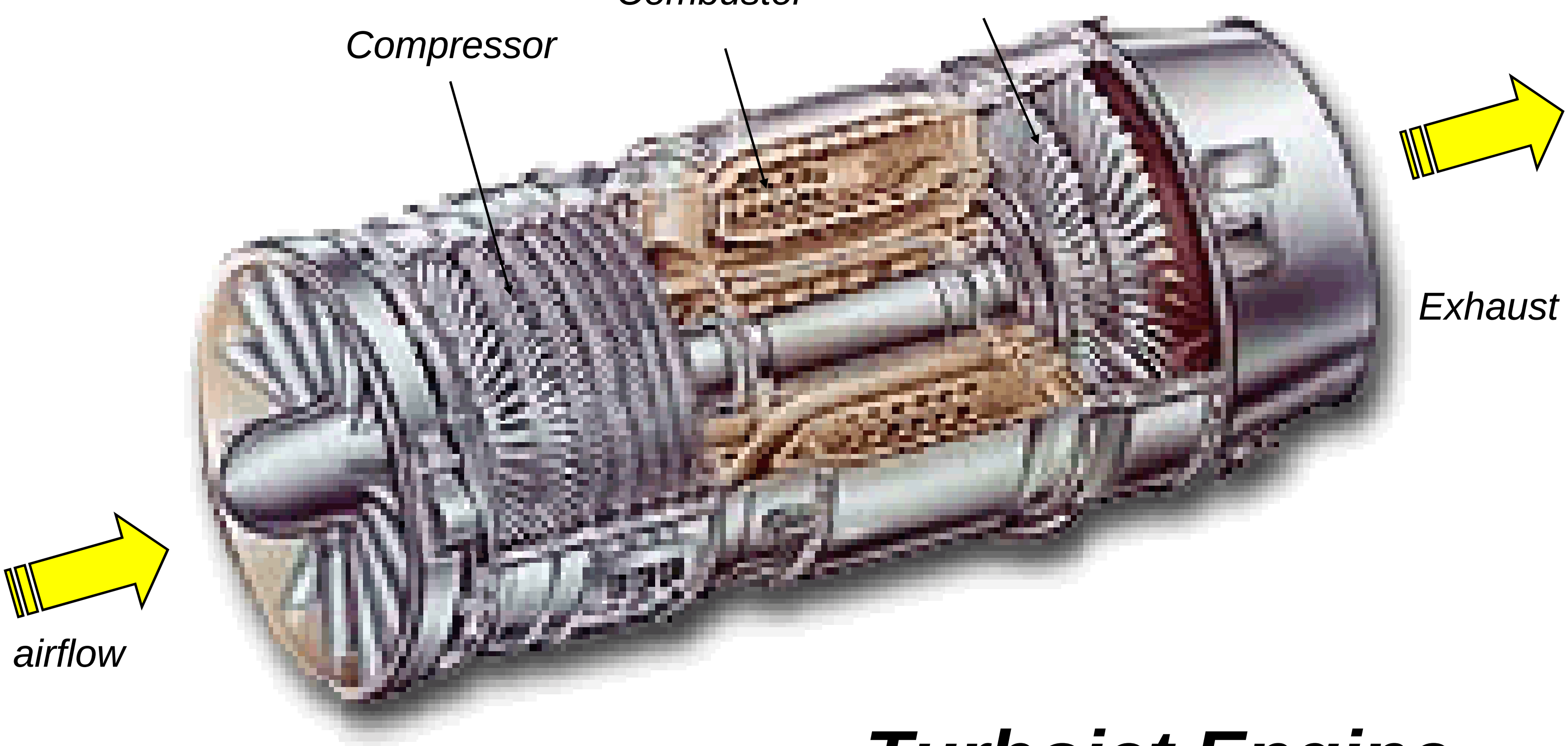
Technical: Pressure ratio, efficiency, weight, life

Program: NRE, part cost, schedule, validation plan

Design & Validation

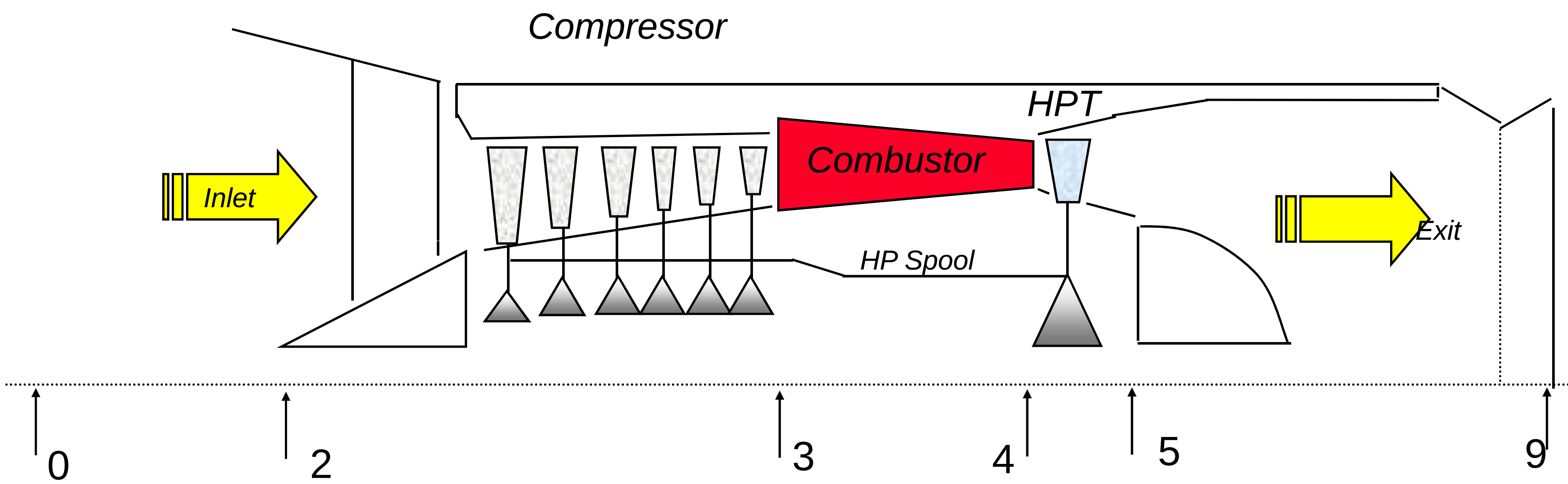
Qualified Product

The Aircraft Engine Design Project
Fundamentals of Engine Cycles



Turbojet Engine

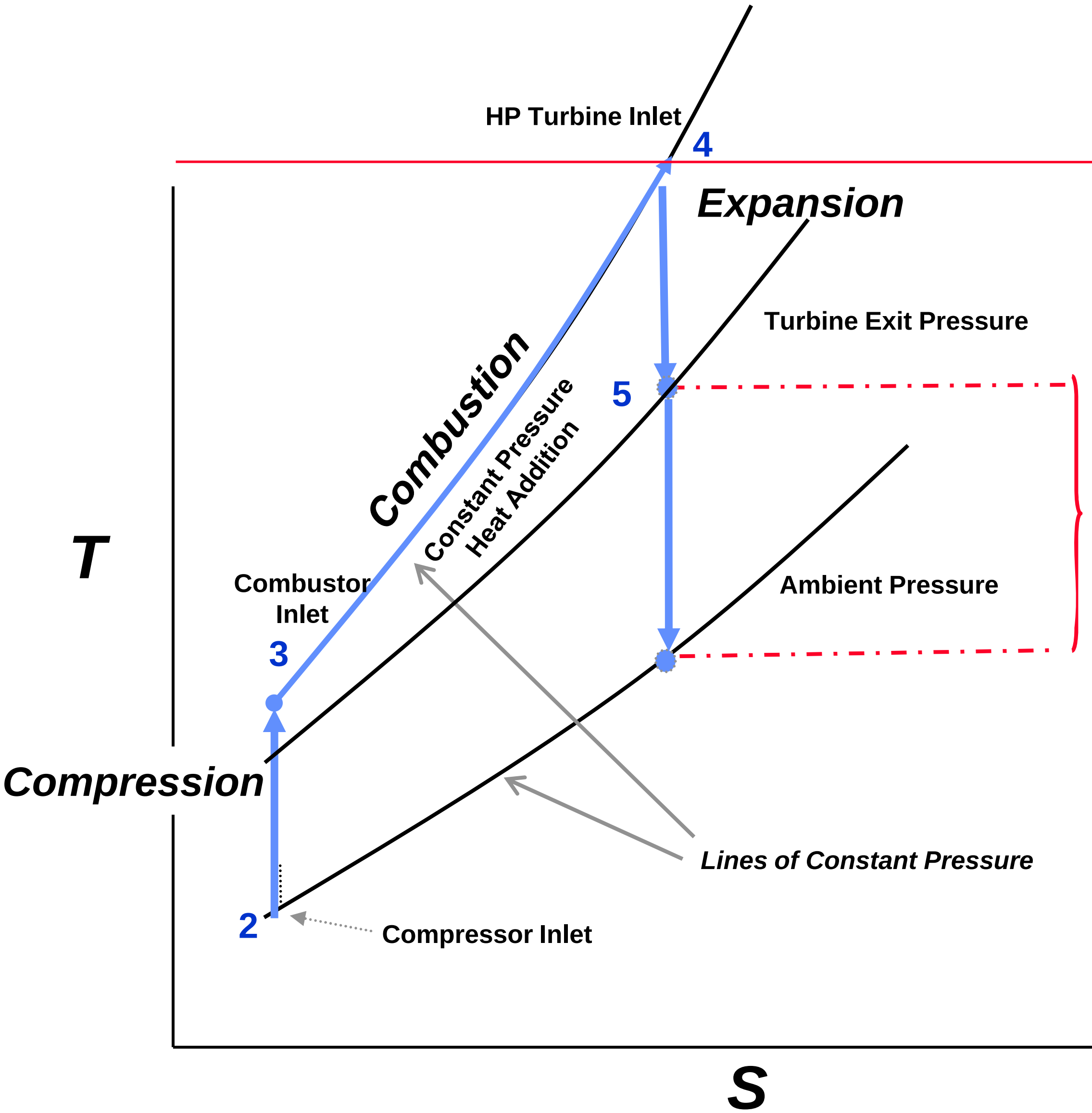
Engine Modules and Components



Turbojet Engine Cross-Section

Multi-stage compressor module powered by a single stage turbine

Ideal Brayton Cycle: T-S Representation

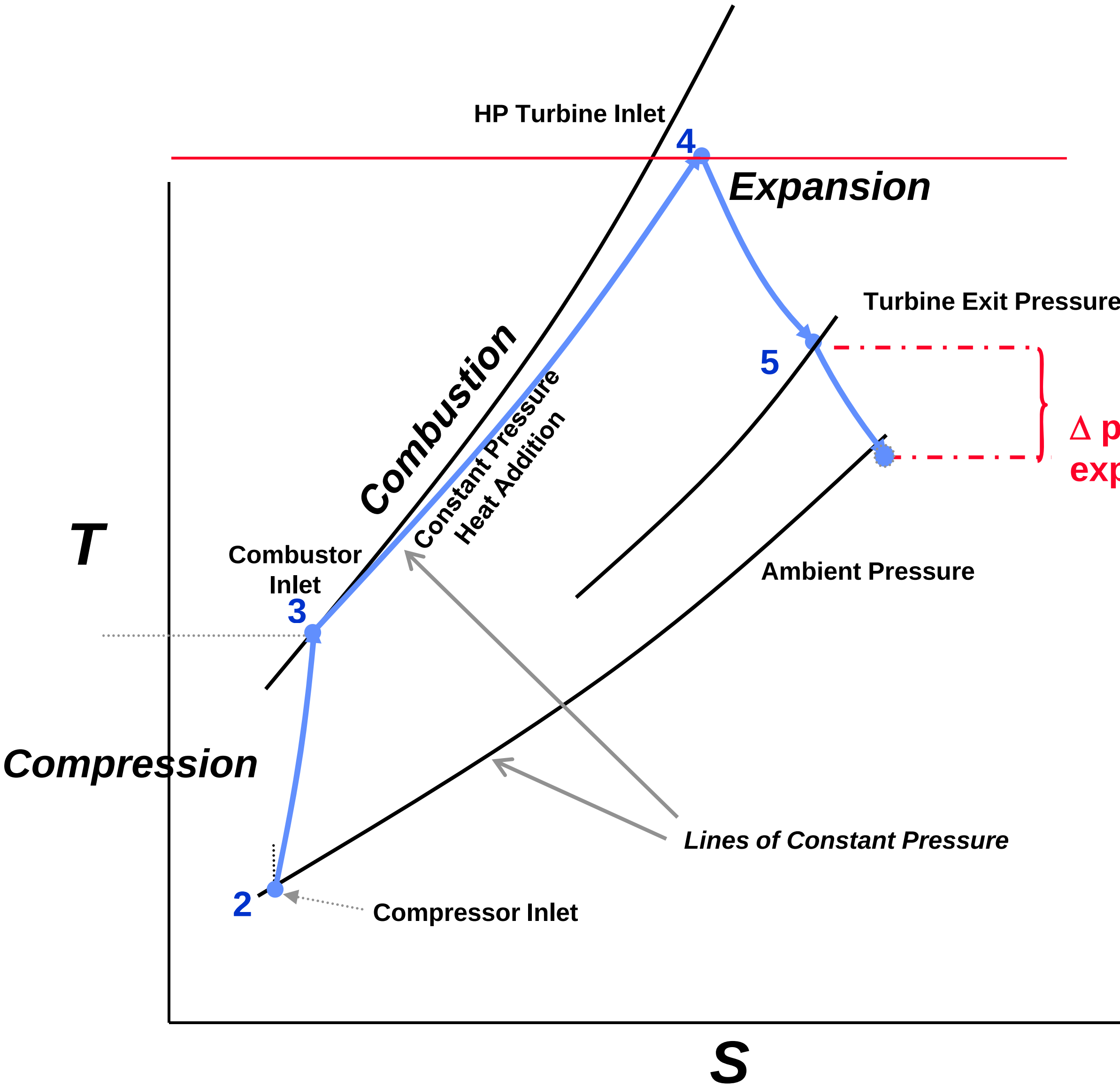


Δ pressure available for expansion across *Exhaust Nozzle*

$$\frac{P}{W} = \Delta_1^2 h_0$$

Note: 1) Flight Mach = 0
2) $P_{t2} = P_{amb}$
3) P = power
4) W = mass flow rate
5) h_0 = total enthalpy

Real Brayton Cycle: T-S Representation



Δ pressure available for expansion across *Exhaust Nozzle*

$$\frac{P}{W} = \Delta_1^2 h_0$$

Impact of Real Efficiencies:

Decreased Thrust @ if T4 is maintained

Or

Increase Temp (fuel flow) to maintain thrust!

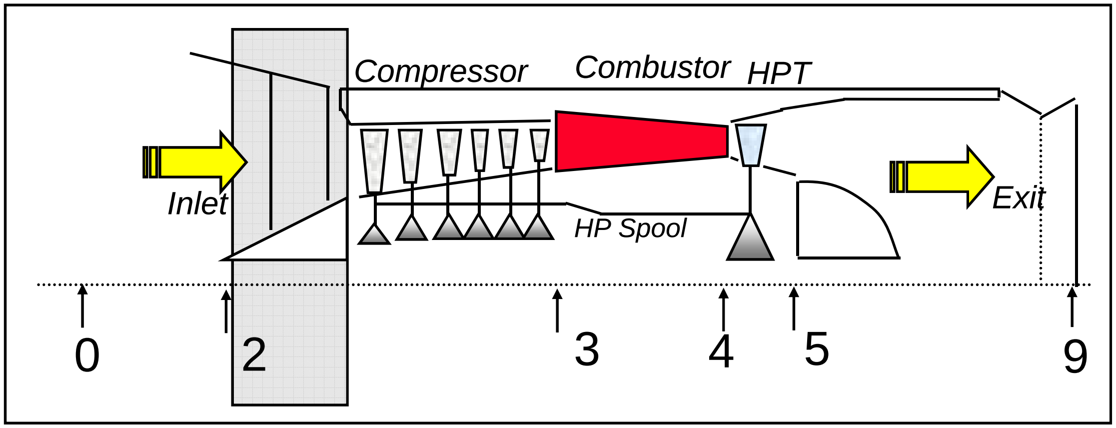
Engine Inlet

• Flow capacity (flow function relationship)
Starting with the conservation of mass and substituting the total to static relations for Pressure and Temperature, can derive:
W= Density * Area* Velocity

$$\frac{W*(\mathrm{sqrt}(Tt))}{Pt*Ae} = \frac{M*\mathrm{sqrt}(g_c*\gamma/R)}{[1+((\gamma-1)/2)*M^2]^{(\gamma+1)/[2*(\gamma-1)]}}$$

where M is Mach number
Tt is total temperature (deg R)
Pt is total pressure (psia)
W is airflow (lbm/sec)
Ae is effective area (in^2)
gc is gravitational constant
=32.17 lbm ft/(sec^2 lbf)
γ is ratio of specific heats
R is gas constant
(ft-lbf)/(lbm-deg R)

Turbojet



Compressor

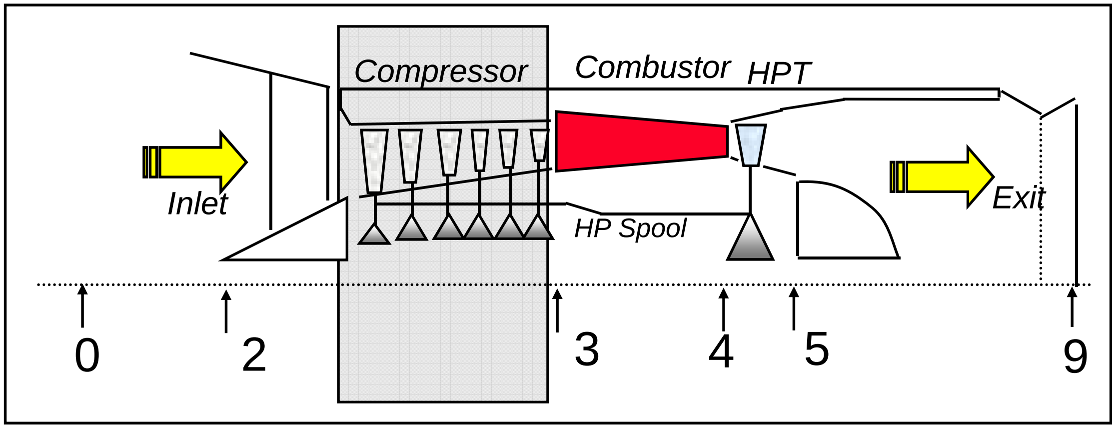
- From adiabatic efficiency relationship

$$\eta_{compressor} = \text{Ideal Work/ Actual Work} = \frac{Cp^*(Texit' - Tinlet)}{Cp^*(Texit - Tinlet)}$$

$$= \frac{(Pexit/Pinlet)^{(\gamma-1)/\gamma} - 1}{Texit/Tinlet - 1}$$

where
P_{exit} is compressor exit total pressure (psia)
P_{inlet} is compressor exit total pressure (psia)
T_{inlet} is compressor inlet total temperature (deg R)
T_{exit} is compressor exit total temperature (deg R)
T_{exit}' is ideal compressor exit temperature (deg R)

Turbojet



Combustor

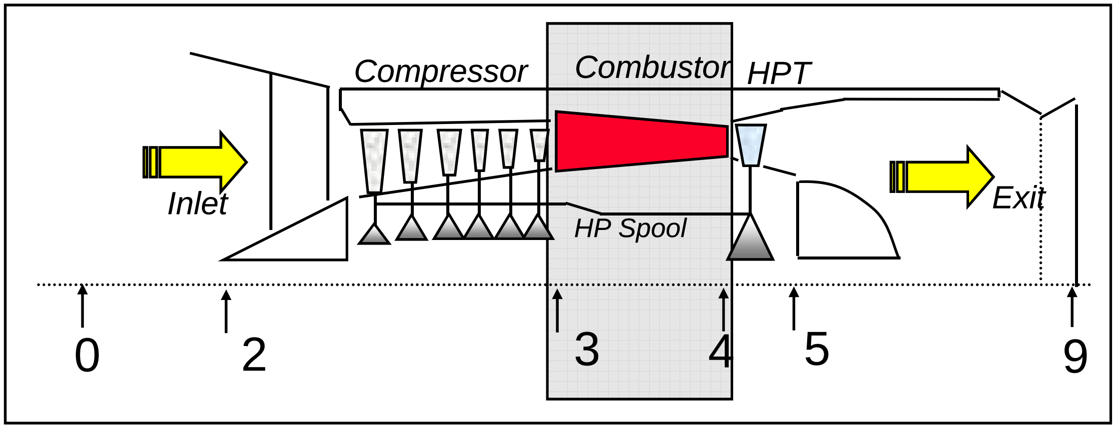
•From Energy balance/ Combustor efficiency relationship:

$$\eta_{combustor} = \text{Actual Enthalpy Rise/ Ideal Enthalpy Rise}$$

$$= \frac{(WF + W)*Cp_{combustor}Texit - W*Cp_{combustor}Tinlet}{WF * FHV}$$

where W is airflow (lbm/sec)
WF is fuel flow (lbm/sec)
FHV is fuel heating value (BTU/lbm)
T_{inlet} is combustor inlet total temperature (deg R)
T_{exit} is combustor exit total temperature (deg R)
Cp is combustor specific heat BTU/(lbm-deg R)
Can express WF/W as fuel to air ratio (FAR)

Turbojet



Turbine

• From efficiency relationship

$$\eta_{turbine} = \text{Actual Work/Ideal Work} = \frac{Cp^*(Tinlet - Texit)}{Cp^*(Tinlet - Texit')}$$

$$= \frac{1 - (Texit/Tinlet)}{1 - (Pexit/Pinlet)^{(\gamma-1)/\gamma}}$$

• Work Balance: From conservation of energy

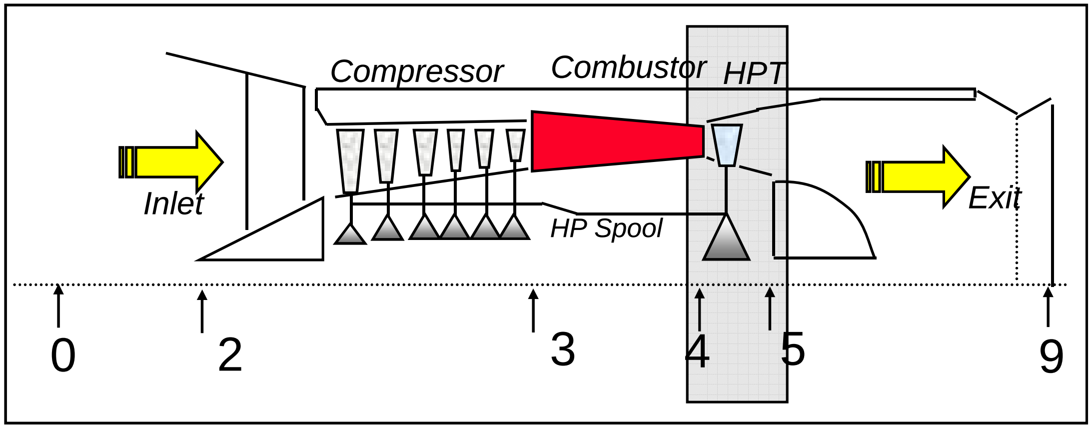
Turbine Work = Compressor Work + Losses

$$(W+ WF)^* Cp_{turb}^* (Tinlet - Texit)|_{turb} = W * Cp_{compressor}^* (Texit - Tinlet)|_{comp}$$

where

Pexit is turbine exit total pressure (psia)
Pinlet is turbine exit total pressure (psia)
Tinlet is inlet total temperature (deg R)
Texit is exit total temperature (deg R)
Texit' is ideal exit total temperature (deg R)
Cp is specific heat for turbine or compressor BTU/(lbm-deg R)

Turbojet



Nozzle

- Isentropic relationship, can determine exhaust properties

$$Tt/Ts = (Pt/Ps)^{(\gamma-1)/\gamma}$$
$$= 1 + ((\gamma - 1)/2) * M^2$$

- From Mach number relationship can determine exhaust velocity

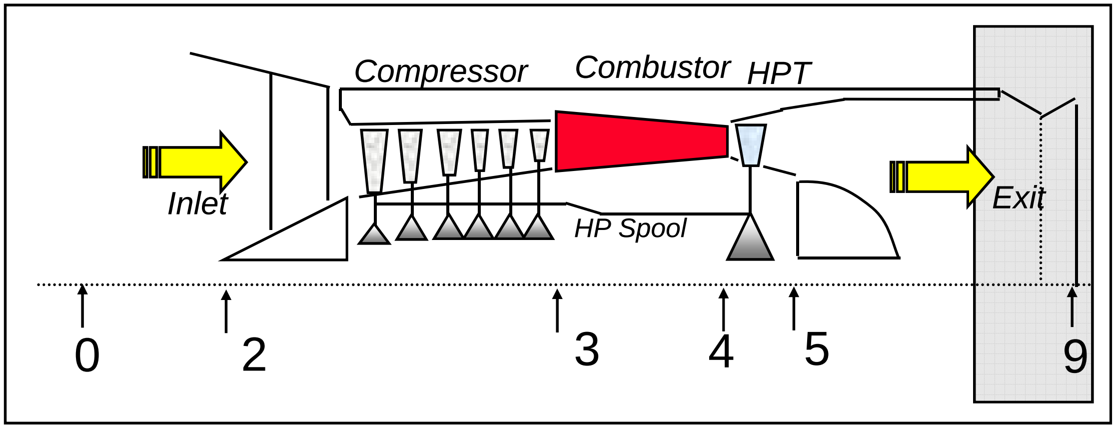
$$v = M*a$$

where a, speed of sound= $\mathrm{sqrt}(\gamma * g_c * R * Ts)$

where

Tt is total temperature (deg R)
Pt is total pressure (psia)
Ps is static pressure (psia)
Ts is static temp (deg R)
g_c is gravitational constant
=32.17 lbm ft/(sec^2 lbf)
γ is ratio of specific heats
R is gas constant (ft-lbf)/(lbm-deg R)
v is flow velocity (ft/sec)
a is speed of sound (ft/sec)
M is Mach number

Turbojet



Engine Performance

- Thrust relationship: from conservation of momentum

$$Fnet = W9\ V9/\ g_c\ -\ W0\ V0/\ g_c\ +\ (Ps9-Ps0)\ A9$$

If flight Mach number is 0, v0 = 0

and if nozzle expands to ambient, PS9=Ps0 and

$$Fnet\ = W9\ V9/\ g_c$$

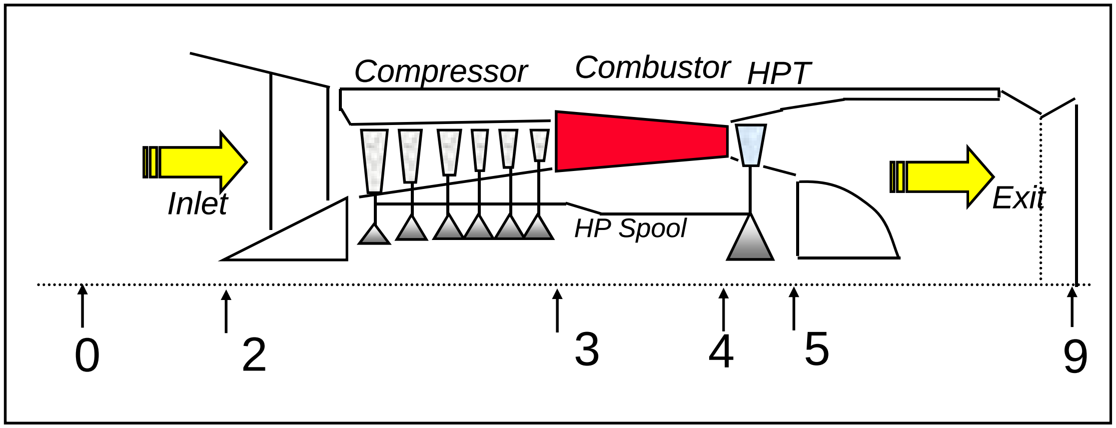
where g_c is gravitational constant

- Specific Fuel Consumption (SFC)

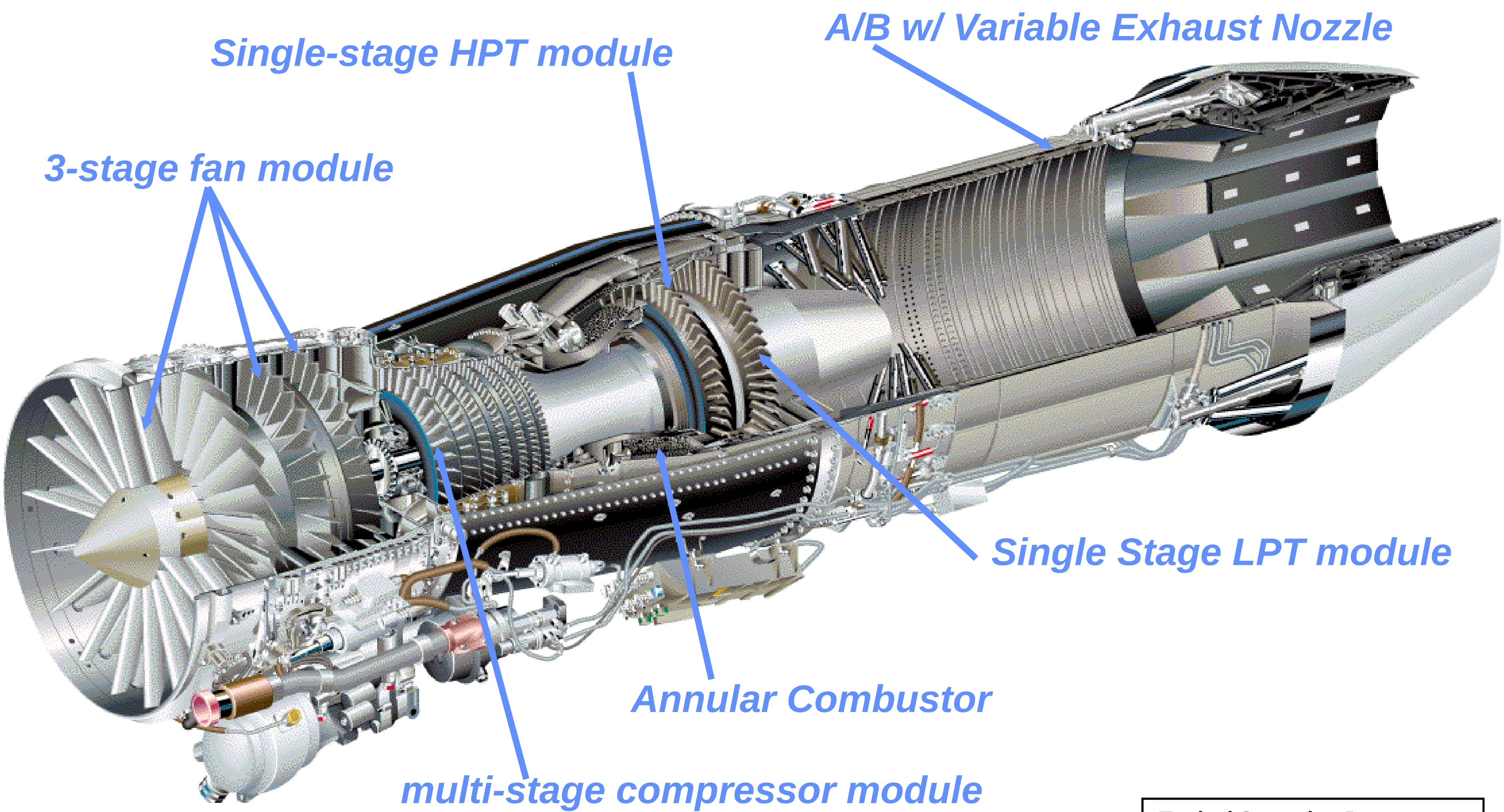
SFC = Wf/ Fnet (lbm/hr/ lbf)

(lower SFC is better)

Turbojet



Modern Afterburning Turbofan Engine

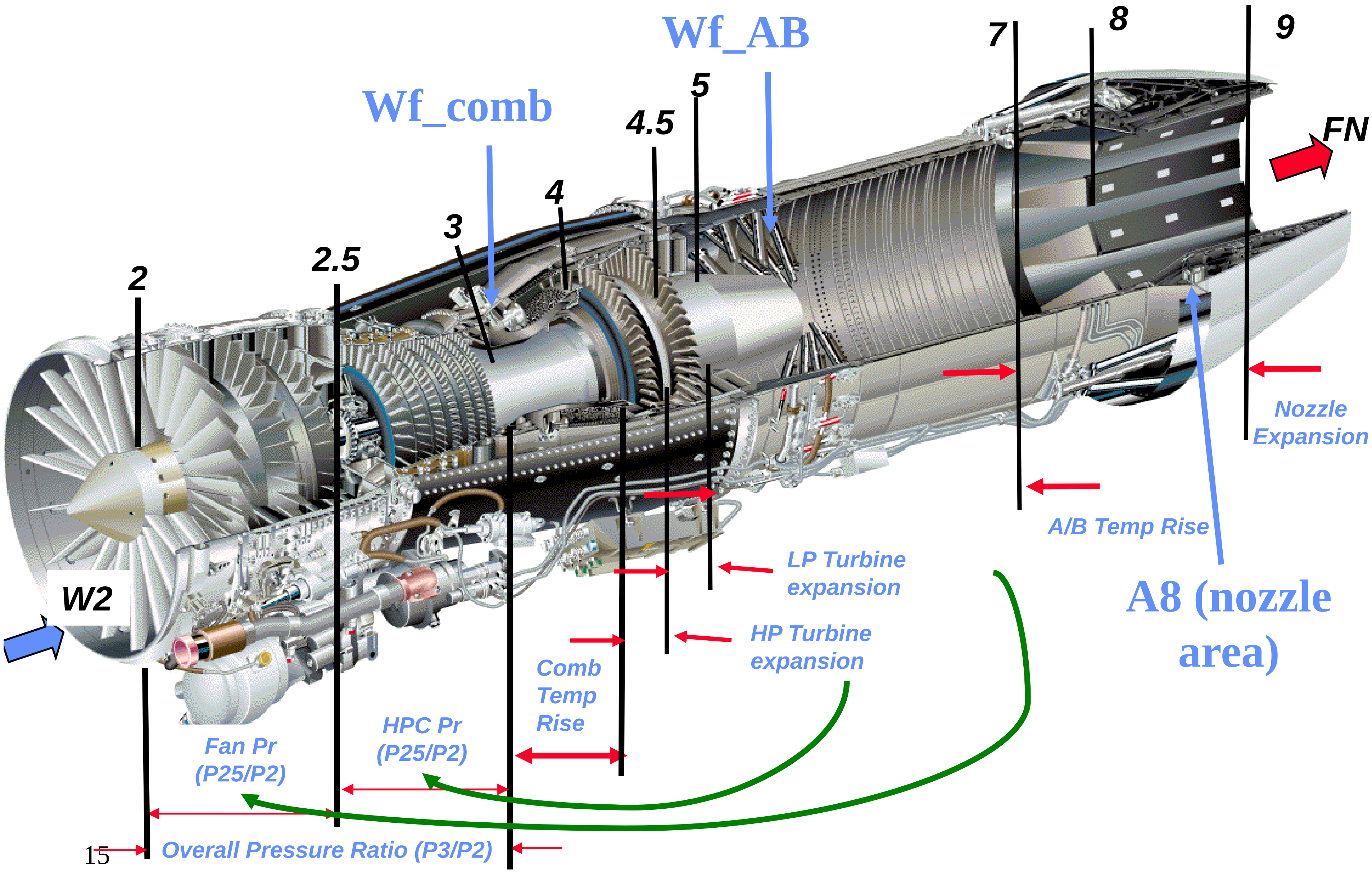


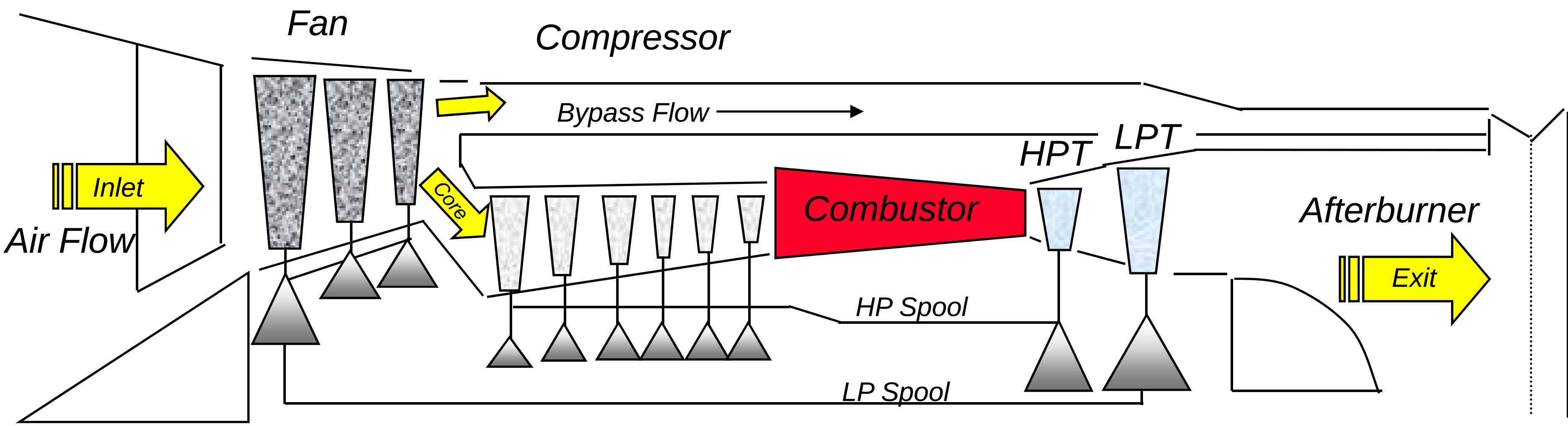
Terms:	
blade	***rotating airfoil***
vane	***static airfoil***
stage	***rotor/stator pair***
PLA	***pilot's throttle***

Typical Operating Parameters:	
OPR	***25:1***
BPR	***0.34***
ITT	***2520°F***
Airflow	***142 lbm/sec***
Thrust Class	***16K-22K lbf***

Thermodynamic Station Representation

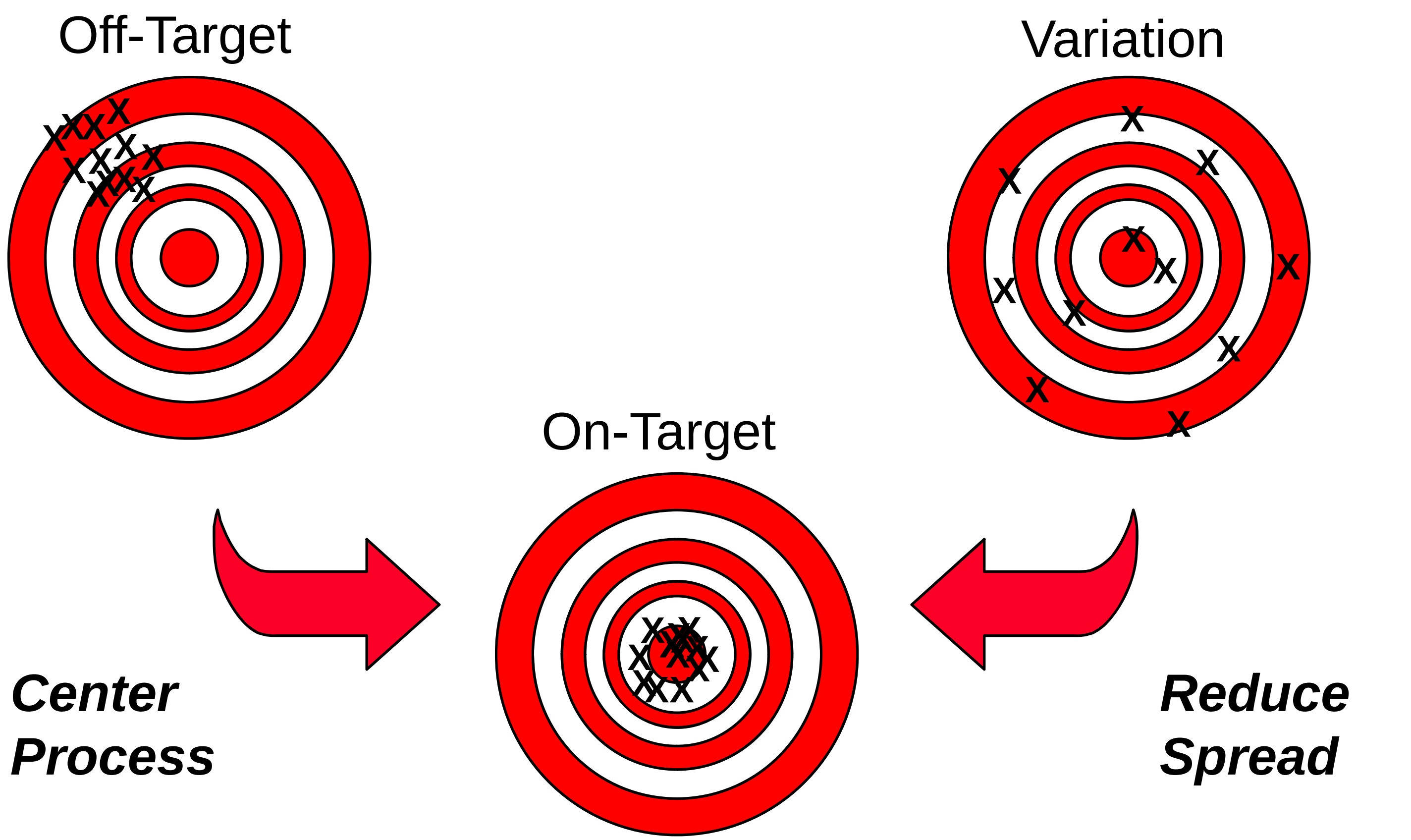
GE Aircraft Engines





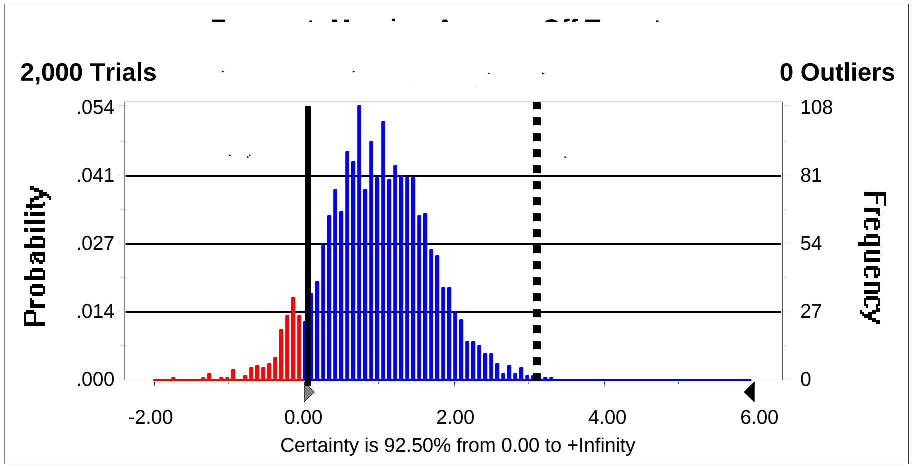
Augmented Turbofan Engine Cross-Section

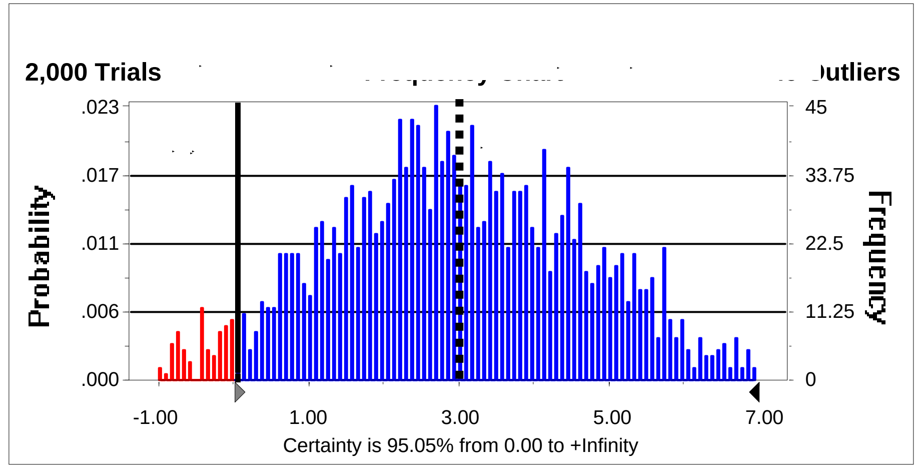
Design Considerations- Process Centering and Variation



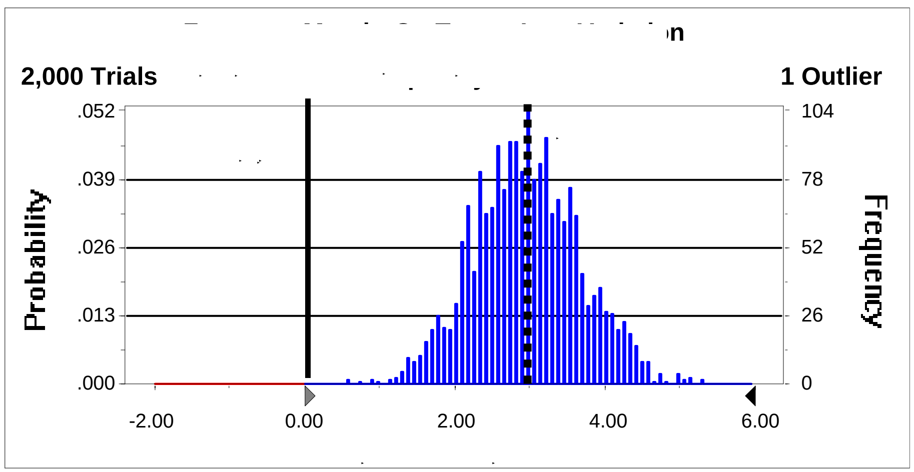
Six Sigma Methodology Applies Statistical Analyses to Center Processes and Minimize Variation

Probabilistic Design Techniques Account for Process Variation





Center Process



Reduce Spread

Understanding and Accounting for Process Variation Assures Compliance with Design Limits