Lecture 4

Design Point & Off-Design Performance ~ 2

Objective ~ Lecture 4

To show how the detailed performance of a propulsion system can be analysed.

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Design Point & Off-Design Performance

- **The Design Point of an Engine**
  - Where all of the components are matched at their ***Design Conditions.***
  - <u>*Design Conditions:*</u> where each unit achieves design pressure ratio & peak efficiency at maximum flow.
  - Also sometimes called *Synthesis Matching Point*

- The Altitude & Mach Number conditions are usually those which are critical for the Aircraft/Engine Requirements.

Design Point & Off-Design Performance

- **Subsonic Passenger Aircraft:**
  - Typically the Design Point will be at the top of Climb.
  - M = 0.8, h = 35,000 ft. T_1 = 240K.

- **Military Aircraft:**
  - The Design Point will be at the critical condition for manoeuvring in Combat.
  - M = 1.8, h = 40,000 ft. T_1 = 357K.

- **Helicopter:**
  - The Design Point will be at hovering at sea level.
  - M = 0, h = at Sea level T_1 = 288 K.

- All other conditions are ***Off-Design,*** where the pressure ratio, efficiency and flow are different from those at the design point.

Dimensional Relationships

	Group	Non-Dimensional	Quasi-dimensionless
1.	Rotational Speed	$\frac{N \cdot D}{\sqrt{R \cdot T_o}}$	$\frac{N}{\sqrt{T_o}}$
2.	Mass Flow	$\frac{\dot{m}\sqrt{R \cdot T_o}}{A \cdot P_o}$	$\frac{\dot{m}\sqrt{T_o}}{P_o}$
3.	Flow Velocity	$\frac{C}{\sqrt{\gamma R T}}\ (M)$	$\frac{C}{\sqrt{T}}$

Nomenclature: T_o ***Total Temperature, T Static Temperature***

$\dot{m}$ ***Mass flow***

C ***Flow velocity***

M ***Mach Number***

D ***a "reference" dimension***

N ***Rotational Speed***

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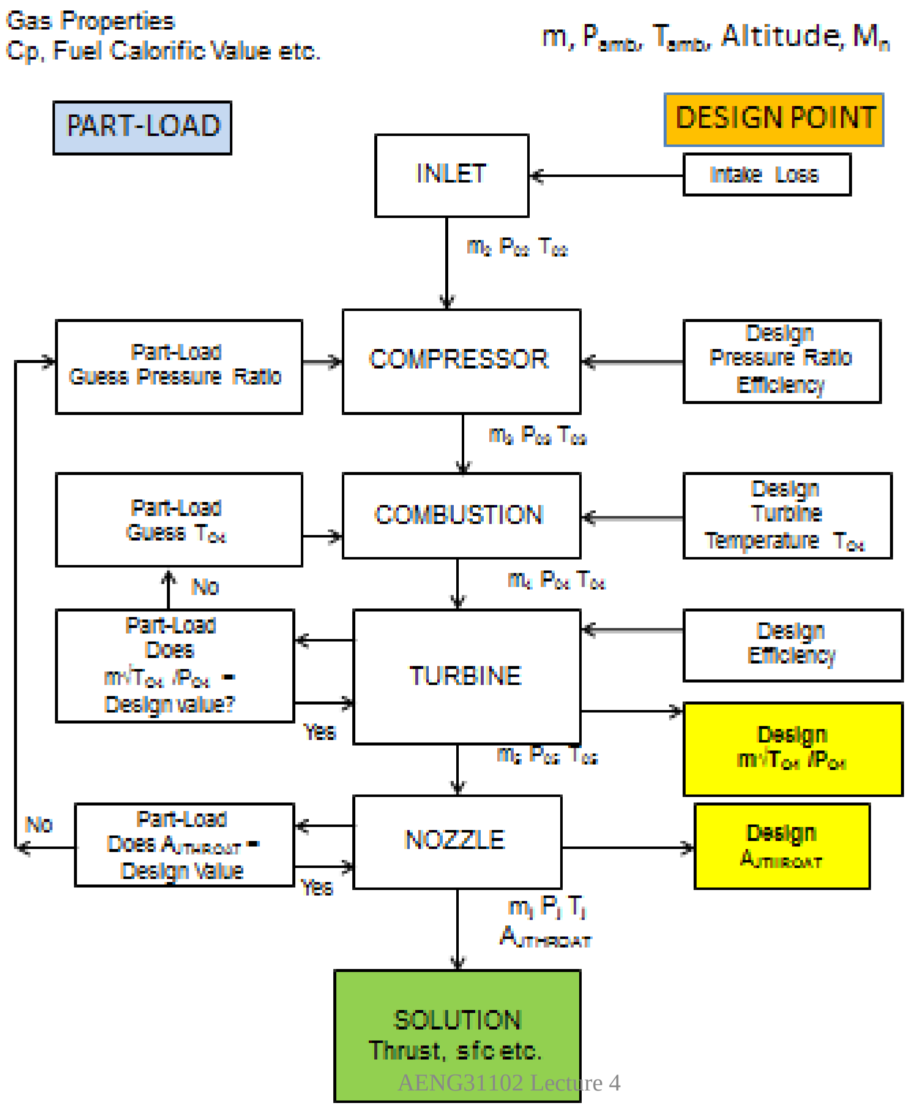
Engine Dimensional Groups

Parameter	Dimensionless Group	Quasi-dimensionless Group	Referred Parameter	Scaling parameter
Mass Flow	$\frac{m \times \sqrt{To} \times R}{Di^2 \times P1 \times \sqrt{\gamma}}$	$\frac{m \times \sqrt{To}}{P1}$	$\frac{m \times \sqrt{\theta}}{\delta}$	$\frac{m \times \theta}{Di^2 \times \delta}$
Gross Thrust	$\frac{FG}{\gamma \times Di^2 \times Po}$	$\frac{FG}{Po}$	$\frac{FG}{\delta}$	$\frac{FG}{\delta}$
Specific Thrust SFG	$\frac{SFG}{\sqrt{\gamma \times R \times To}}$	$\frac{SFG}{\sqrt{To}}$	$\frac{SFG}{\sqrt{(\theta)}}$	$\frac{SFG}{\sqrt{(\theta)}}$
Fuel Flow	$\frac{mf \times FHV \times \sqrt{(R)} \times ETA31}{Cp \times Di^2 \times Po \times \sqrt{To} \times \gamma}$	$\frac{mf \times FHV \times ETA31}{Po \times \sqrt{To}}$	$\frac{mf \times FHV \times ETA31}{\delta \times \sqrt{(\theta)}}$	$\frac{mf \times FHV \times ETA31}{\delta \times \sqrt{(\theta)}}$
SFC	$\frac{SFC \times FHV \times \sqrt{(R \times \gamma)} \times ETA31}{Cp \times \sqrt{To}}$	$\frac{SFC \times FHV \times ETA31}{\sqrt{To}}$	$\frac{SFC \times FHV \times ETA31}{\sqrt{(\theta)}}$	$\frac{SFC \times FHV \times ETA31}{\sqrt{(\theta)}}$

FG Gross Thrust
m Mass Flow
Specific Thrust SFG
mf Fuel Flow
FHV Fuel Heating Value
ETA31 Efficiency
θ Relative Temperature
δ Relative Pressure
To Total Temperature
Po Total Pressure
R gas Constant
γ = Cp/Cv
Di Dimension

Reference: *Gas Turbine Performance* Walsh & Fletcher

Flow Chart for Simple Design Point & Part-load Performance Calculation



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Turbojet Design Point ~Pressure ratio 10:1

Design Point M = 0.8, 20000 ft

Cycle parameters:

- ***Inlet***
  - Airflow 100kg/s
  - Total pressure loss 5%
- ***Compressor***
  - Pressure Ratio 10:1
  - Efficiency (isentropic) 89%
- ***Combustor***
  - T4 1200K
  - Pressure loss 8%
- ***Turbine***
  - Efficiency (isentropic) 86%
  - *No nozzle or transmission losses*

Part-load iteration procedure

- Iteration carried out on P_{03}/P_{02} and combustor temperature T_{04} until the turbine "swallowing capacity" i.e. non-dimensional $m_4\sqrt{T_{04}}/P_{04}$ and nozzle throat area are the same as the engine design point values.

- From first guess, the combustor temperature T_{04} is varied at constant compressor delivery pressure ratio until the design value of $\frac{\dot{m}_4\sqrt{T_{o4}}}{P_{o4}}$ is obtained.

- This is repeated for another value of compression ratio and so on until the final solution is reached.

Note: *The first guess would be valid if the engine could be designed with a variable turbine throat area and a variable nozzle throat area and provided that there is sufficient compressor surge margin.*

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Design Point & Part-load Performance Calculation

Sheet 1

Single Spool Turbojet M = 0.8 20,000 ft

			Design **Point**	First **Guess**	Final **Iteration**
Inlet Airflow	$\dot{m}$	kg/s	100	70	70
Ambient Temperature	T_a	kg/s	248.5	248.5	248.5
Ambient Pressure	p_a	kPa	46.6	46.6	46.6
Speed of Sound		m/s	316.2	316.2	316.2
Compressor Pressure Ratio			10:01	7:1	5.86:1
Combustion Temperature	T_{04}	K	1200	1000	855
INLET & AIR INTAKE	T_{01}	K	280.3	280.3	280.3
	P_{01}	kPa	71	71	71
	C_0	m/s	252.6	252.6	252.6
	T_{02}	K	280.3	280.3	280.3
	P_{02}	kPa	67.4	67.4	67.4
COMPRESSOR	$\frac{\dot{m}_2\sqrt{T_{o2}}}{P_{o2}}$		24.83	17.38	17.38
	P_{03}	kPa	674.2	472	399.1
	T_{03}	K	573.4	514.6	487.4
	Power	kW	29457	16479	14565
COMBUSTION	T_{04}	K	1200	1000	855
	Fuel/Air Ratio		0.0173	0.0128	0.0093
	Fuel Flow	kg/s	1.73	0.896	0.651
	P_{04}	kPa	620.3	434.2	365.5

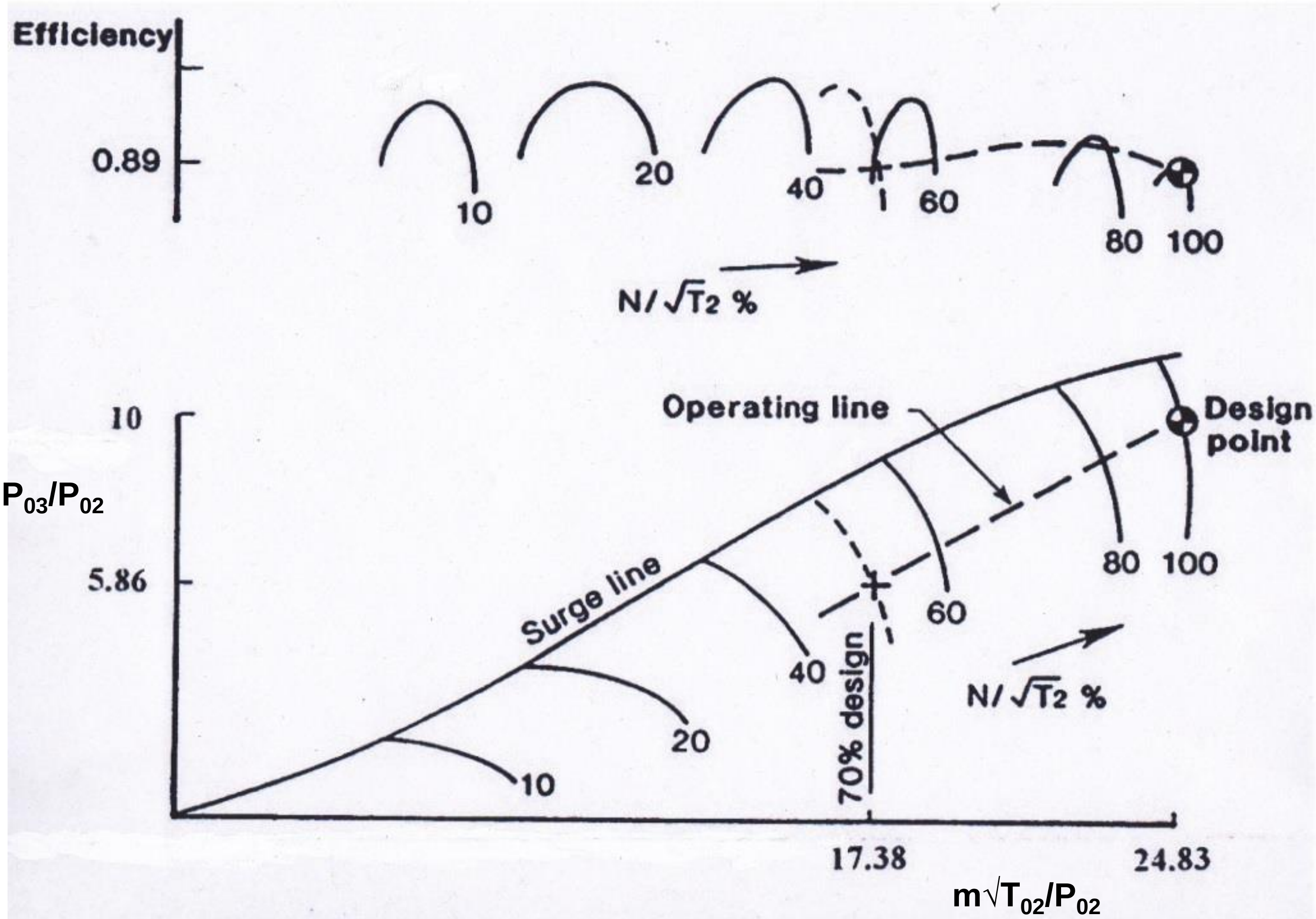
Design Point & Part-load Performance Calculation

Sheet 2

Single Spool Turbojet @ M = 0.8, 20,000ft

			Design Point	First Guess	Final Iteration
TURBINE					
Design Value	$\frac{\dot{m}_4\sqrt{T_{o4}}}{P_{o4}}$		5.681	5.163	5.683
	T_{05}	K	947.5	797.5	675.3
	P_{05}	kPa	202.2	148.2	118.5
CONVERGENT NOZZLE	P_{05}/P^*		4.336	3.184	2.544
	P_{05}/P_{oN^*}		1.853	1.853	1.853
	P_{N^*}	kPa	109	80.1	64
	T_{N^*}	K	812.3	683.6	578.9
	C_N	m/s	557.5	511.5	470.7
Design Value	$A_{JTHROAT}$	m^2	0.39	0.34	0.39
PERFORMANCE	$\dot{m}C_N$	kN	56.7	-	33.2
(Convergent Nozzle)	$A_J (P_N{}^* - P_a)$	kN	24.4	-	6.8
	Gross Thrust	kN	81.1	-	40
	Mom. Drag	kN	25.3	-	17.7
	Nett Thrust	kN	55.8	-	22.3
	SFC	kg/hr/N	0.112	-	0.105
CON-DI NOZZLE	P_{05}/P^*		4.336	-	2.544
Fully Expanded Temperature	T_{NFE}	K	656.6	-	534.7
Fully Expanded Velocity	C_{JFE}	m/s	817.3	-	568.2
√ **Fully Expanded Area**	A_{JFE}	m^2	0.504	-	0.41
Ideal Gross Thrust		Kn	83.1	-	40.1

Typical Compressor Operating Line



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Turbojet Design Point ~Pressure ratio 20:1

Design Point: Sea level static ISA

Inlet

Airflow 100kg/s

Inlet Pressure recovery MIL-E-5007D

Compressor

Pressure Ratio 20:1

Efficiency (isentropic) 80%

Combustor

Inlet Temperature T_{03} 758K

Outlet Temperature T_{04} 1500K

Pressure loss 8.6%

Turbine

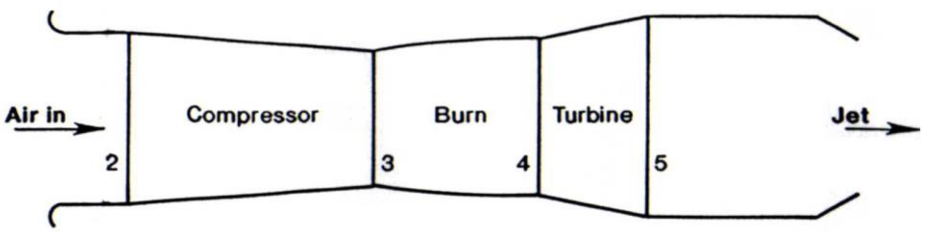
Expansion ratio P_{04}/P_{05} 6.1:1

Efficiency(isentropic) 86%

Cooling bleeds (% of compressor flow) 19%

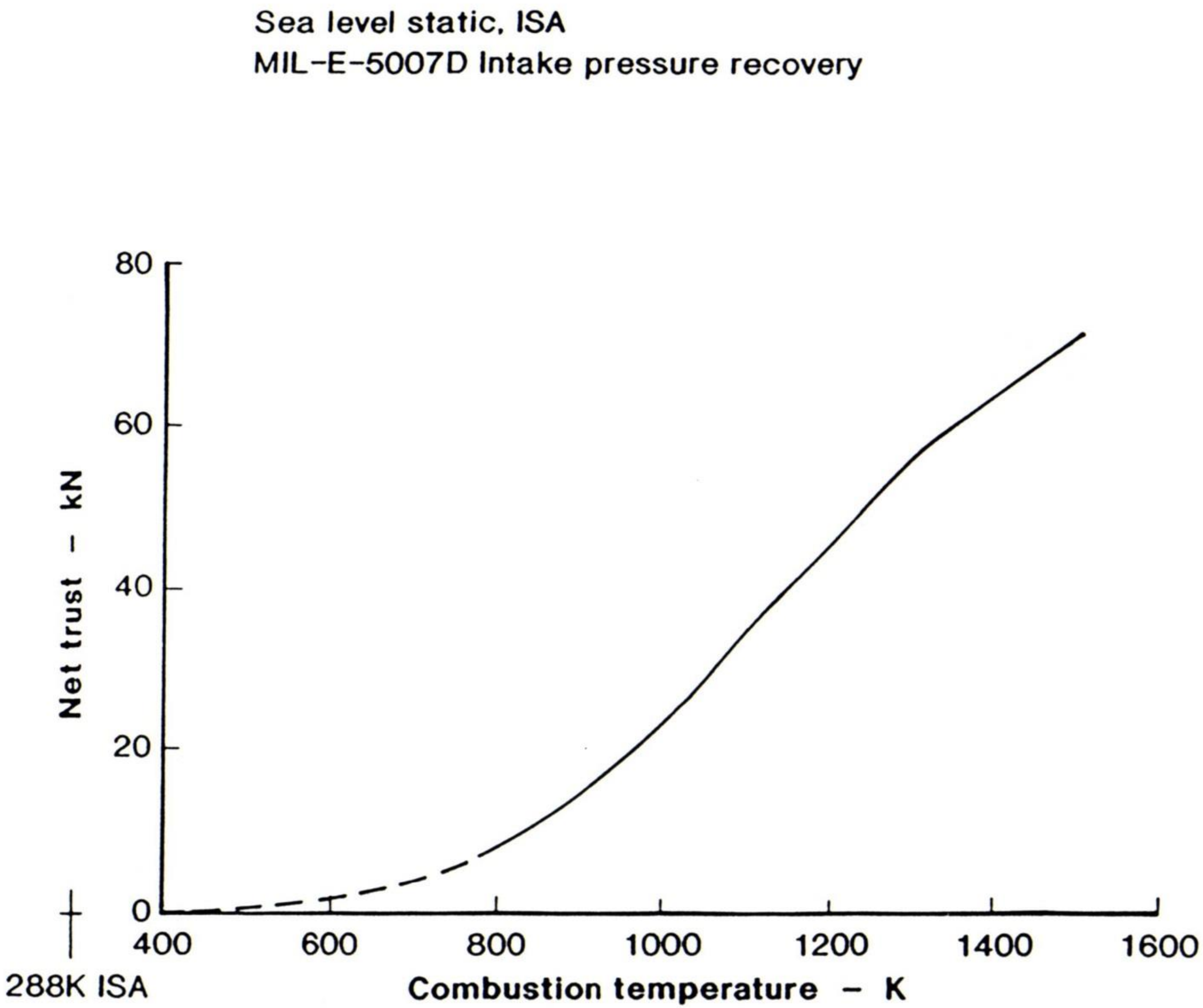
Nozzle

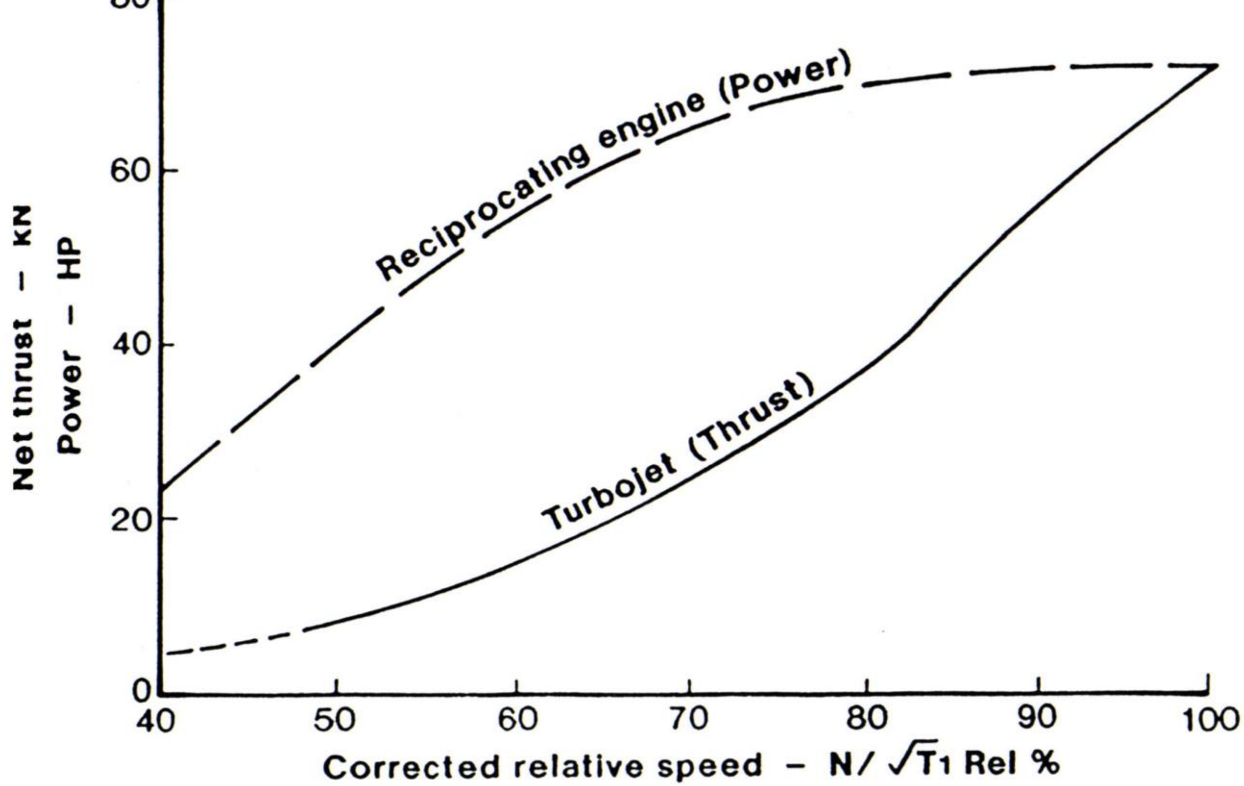
Thrust coefficient 0.96



Thrust = 71.7 kN
Fuel flow = 1.76 kg/s
SFC = 0.0245 kg/s/kN

Turbojet Thrust variation with Combustion Temperature





Sea level static, ISA
Cycle pressure ratio = 20:1
Combustion temperature = 1500K
MIL–E–5007D Intake pressure recovery

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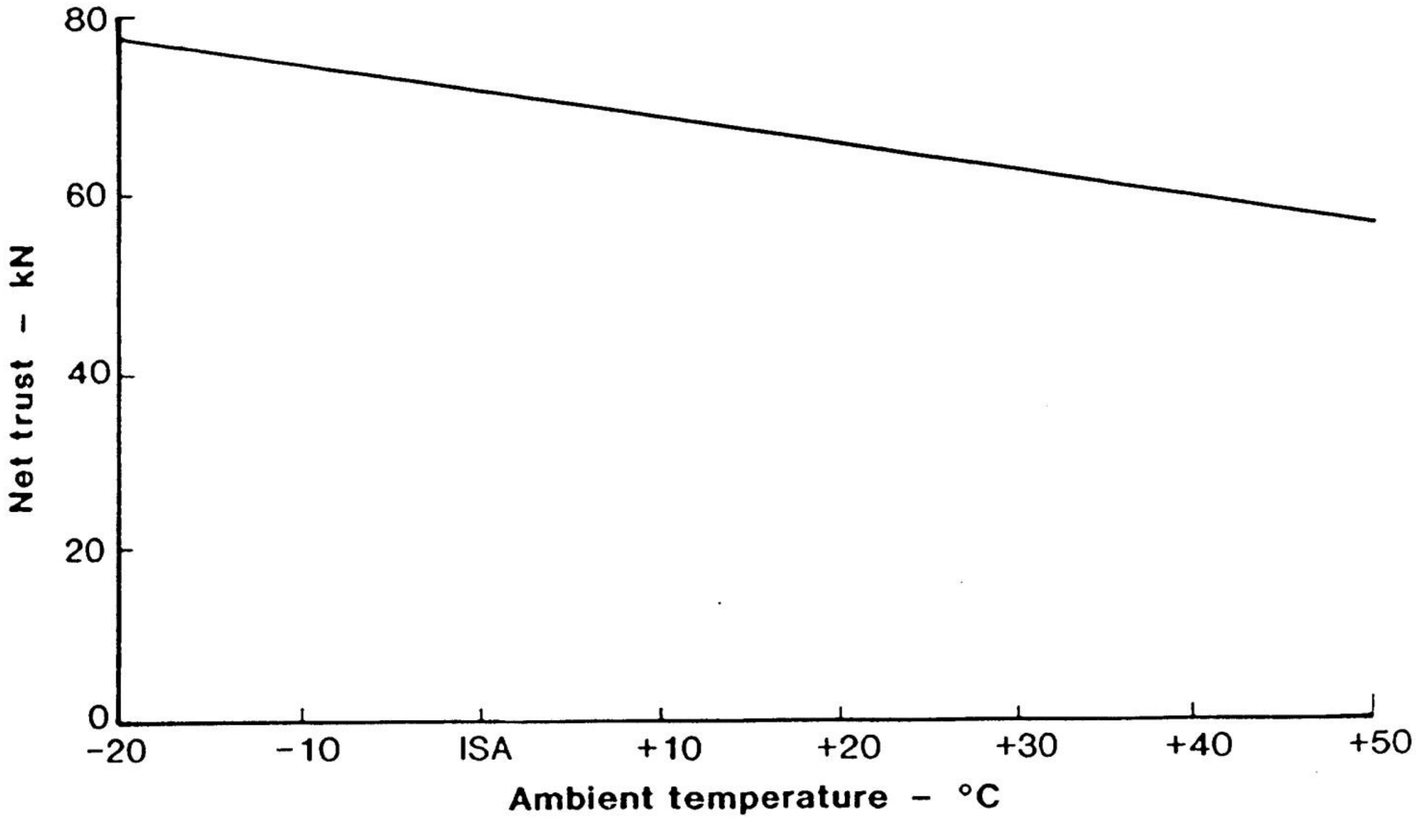
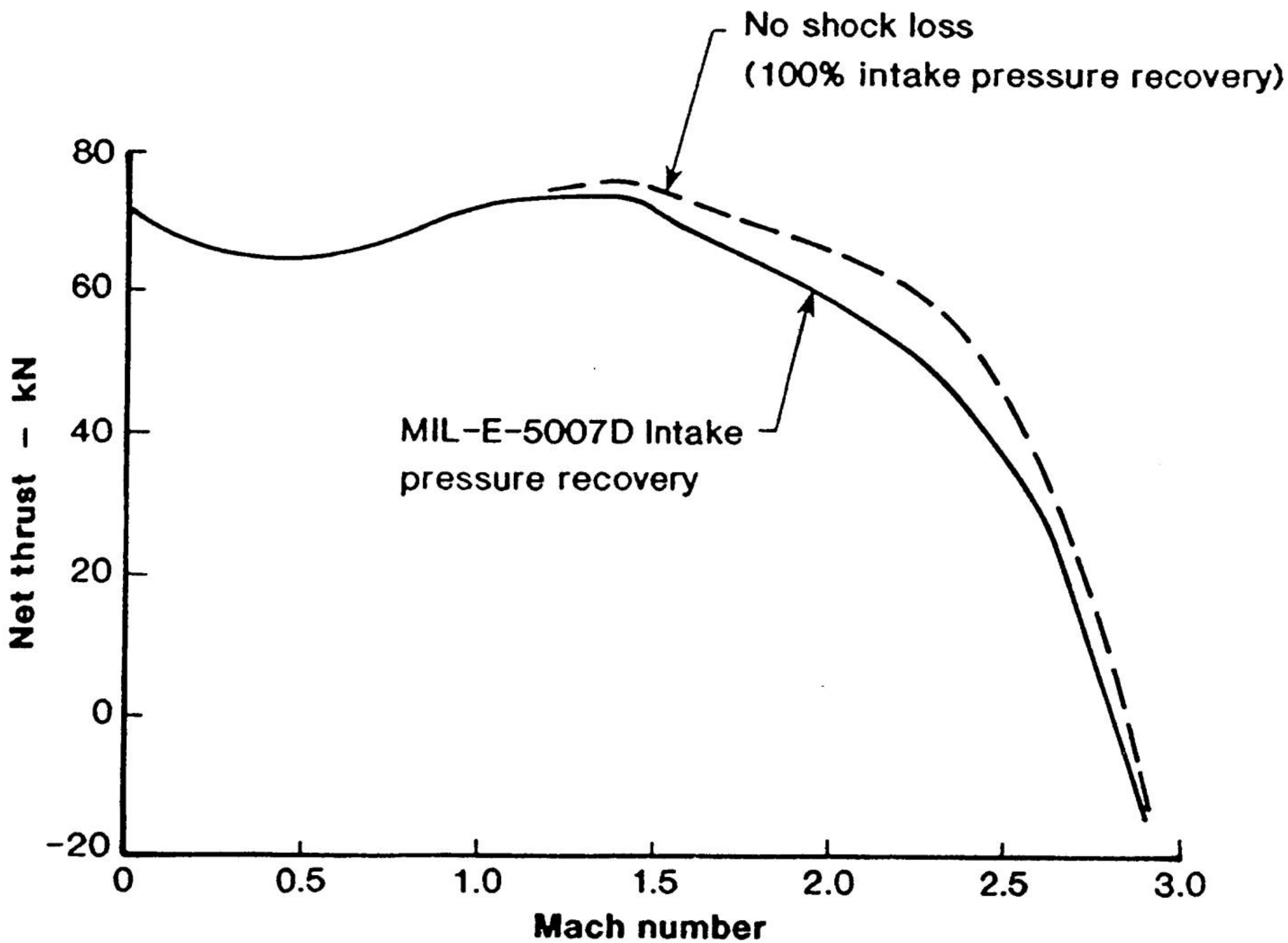
Turbojet Thrust variation with Ambient Temperature



Constant combustion temperature = 1500K
MIL-E-5007D Intake pressure recovery

Turbojet Thrust variation with Mach Number



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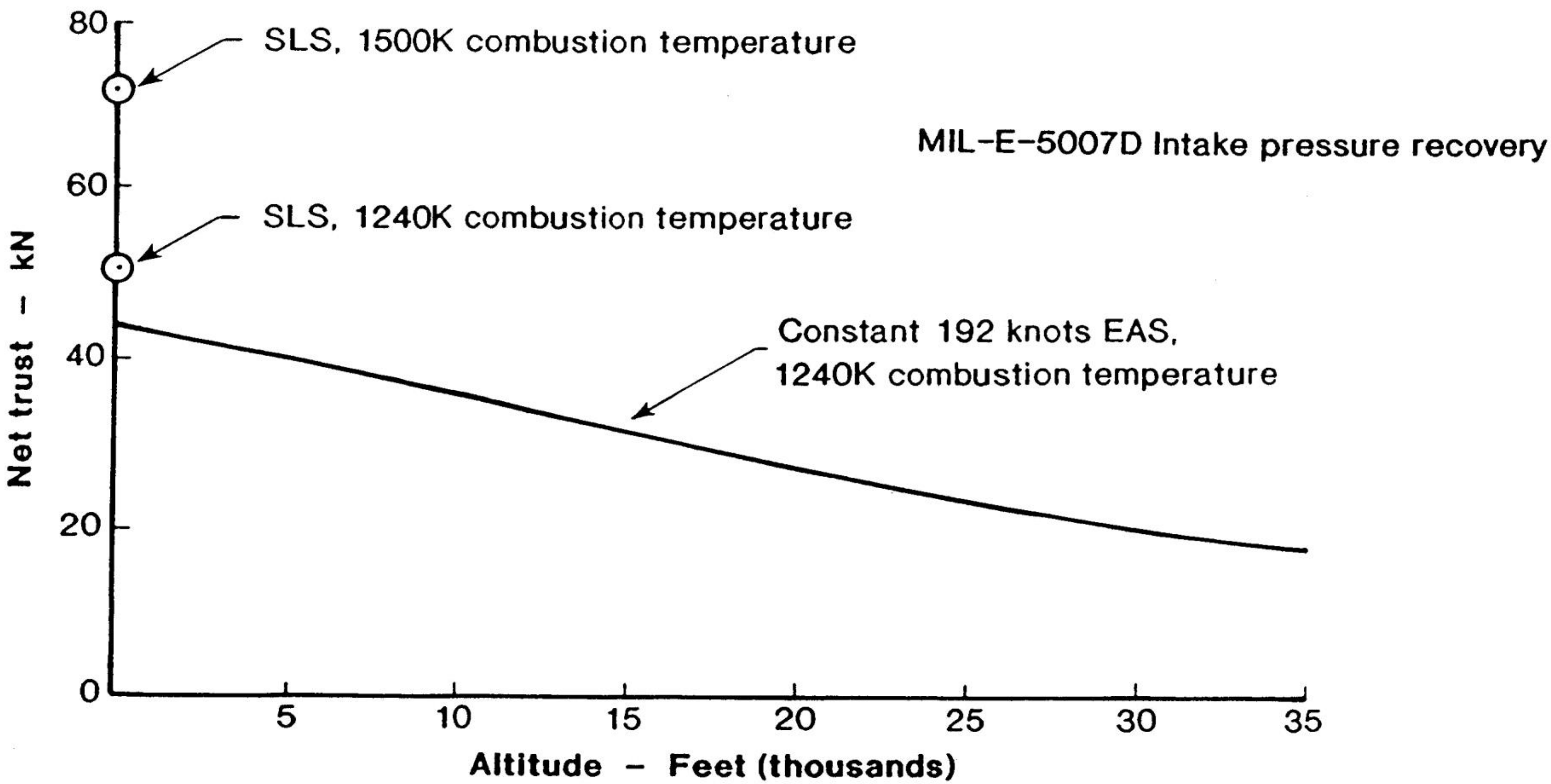
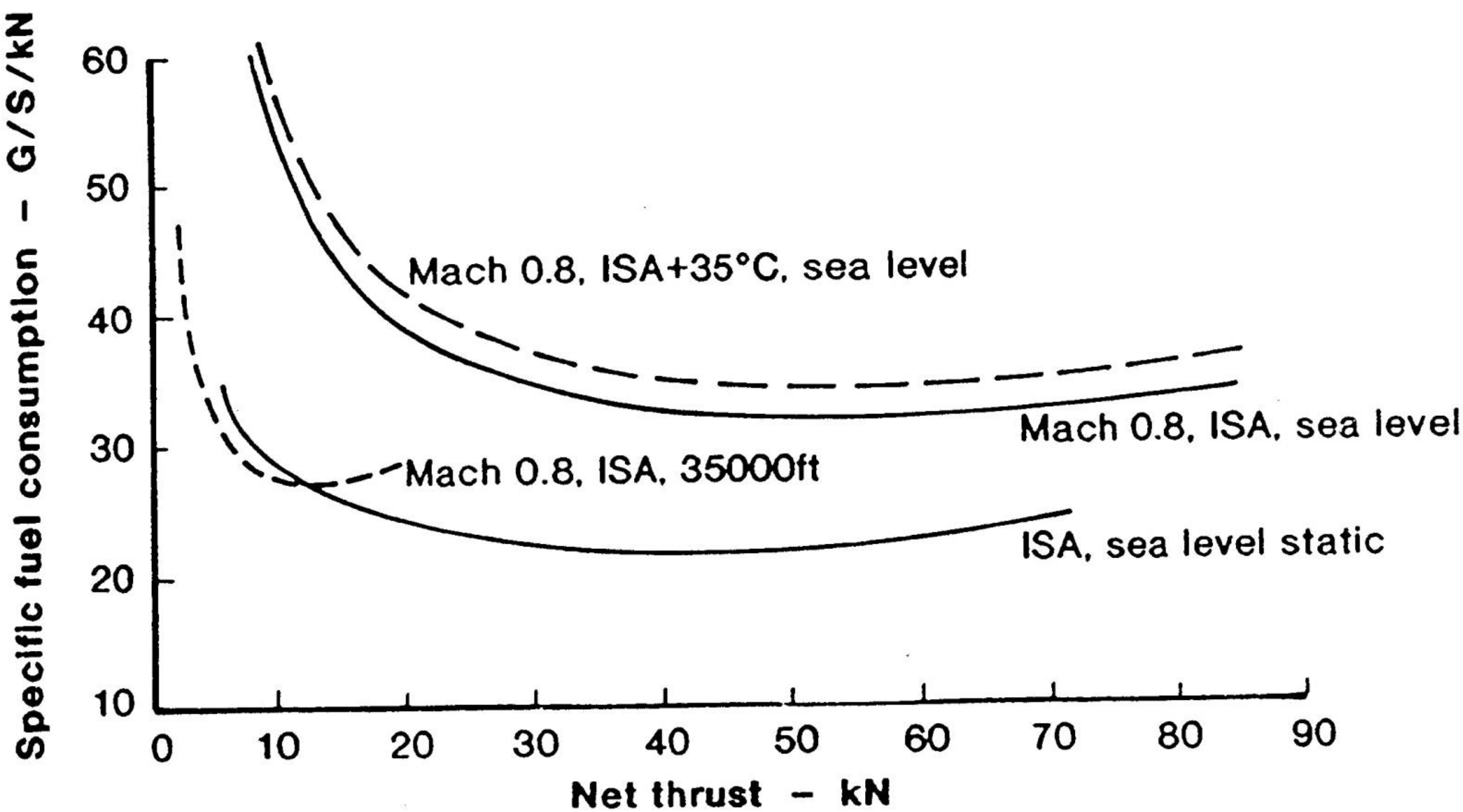
Constant combustion temperature = 1500K, sea level

Turbojet Climb Thrust Characteristics



Turbojet Fuel Consumption Characteristics



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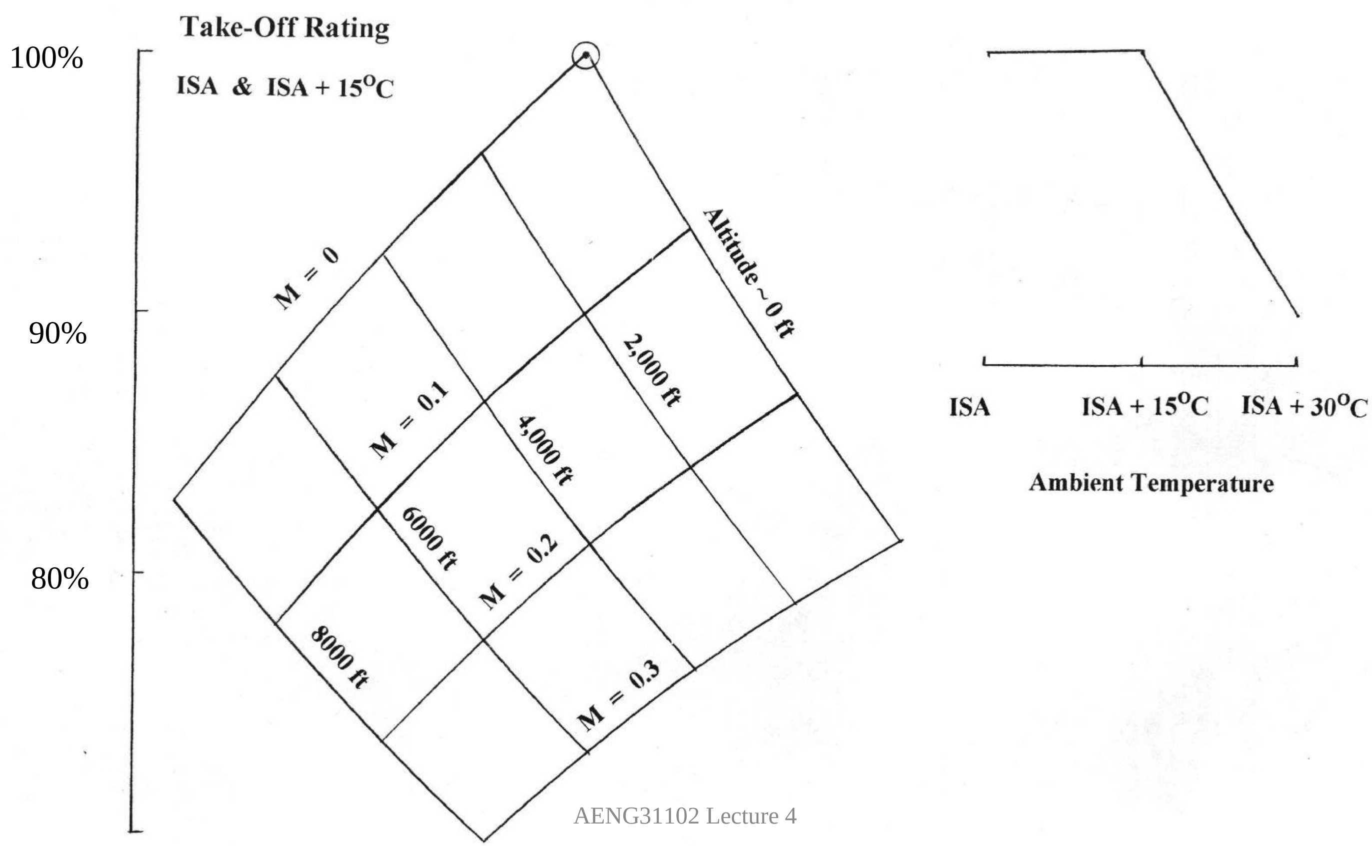
MIL–E–5007D Intake pressure recovery

Typical Engine Data

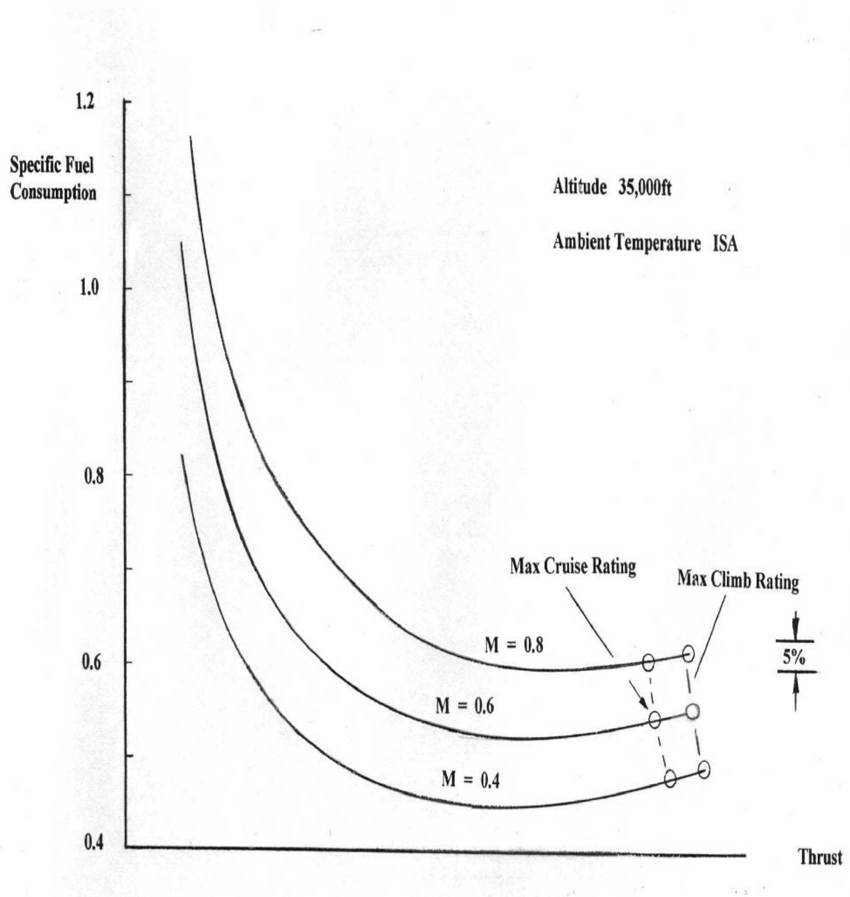
Subsonic Transport Aircraft ~ Take-off

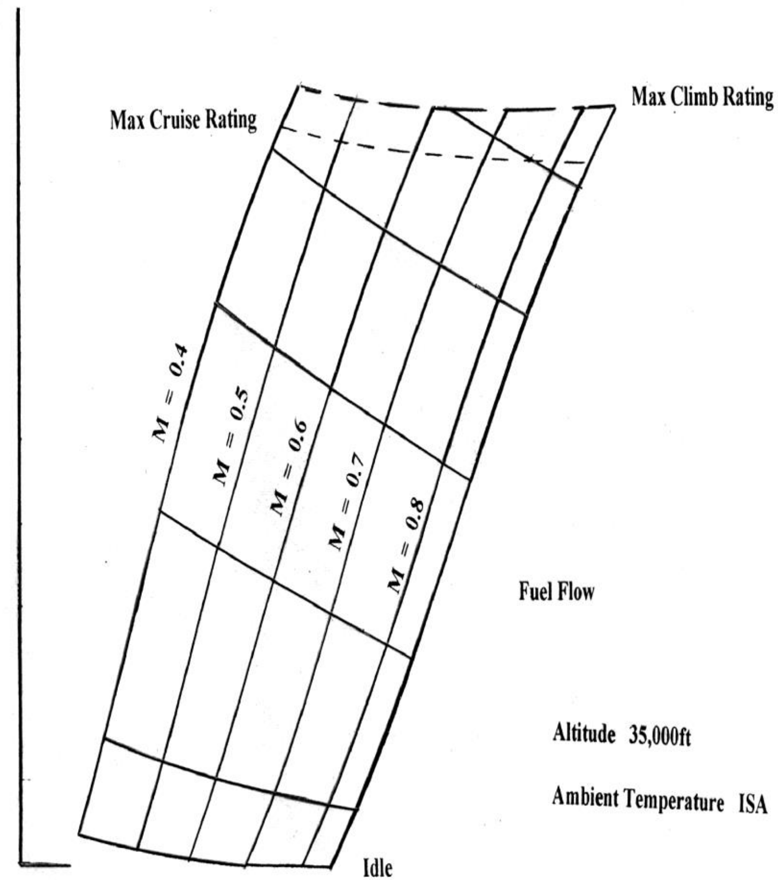
Typical High By-pass Ratio Turbofan
Take-off performance



Typical Engine Data

Subsonic Transport Aircraft ~ Cruise





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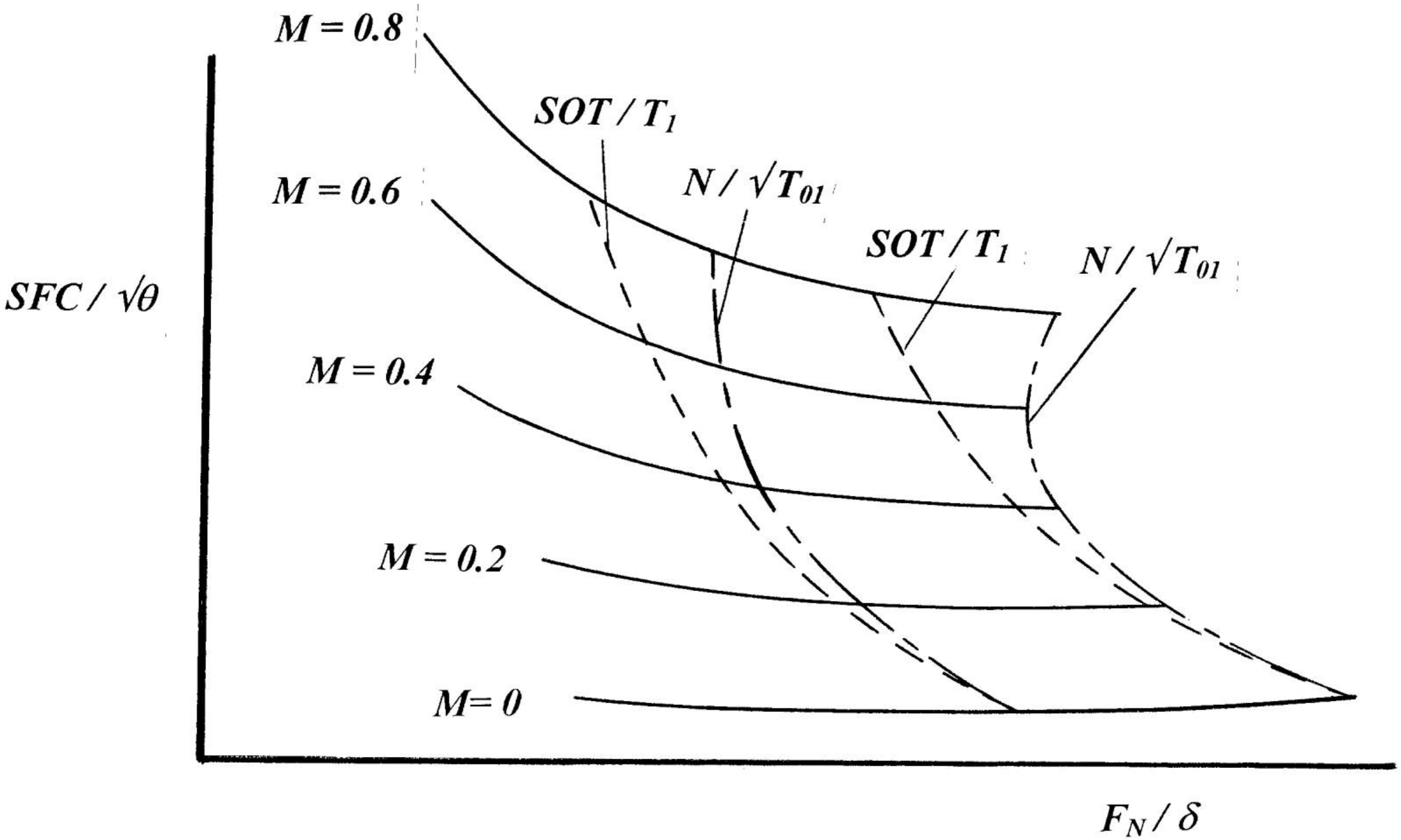
Normalised Thrust - SFC Characteristics



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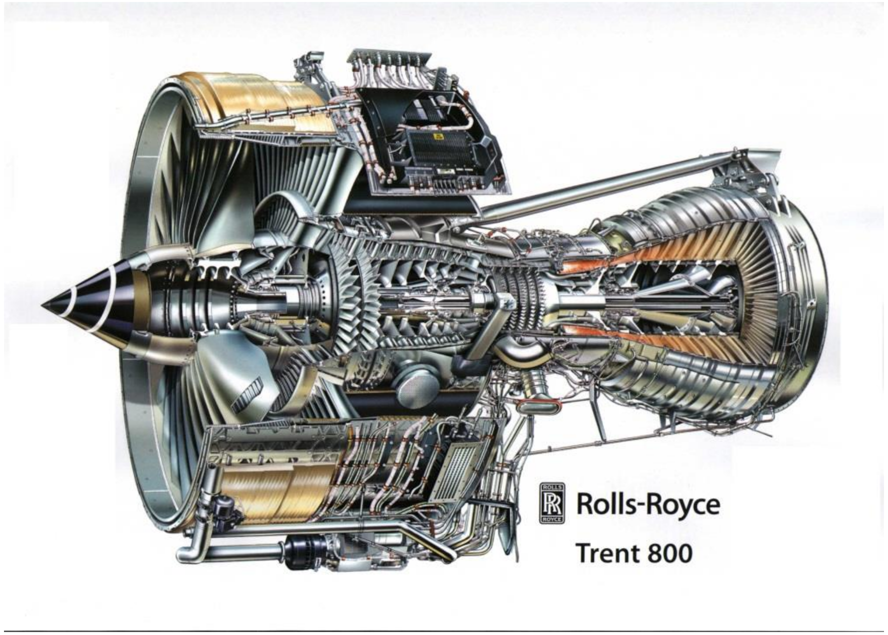
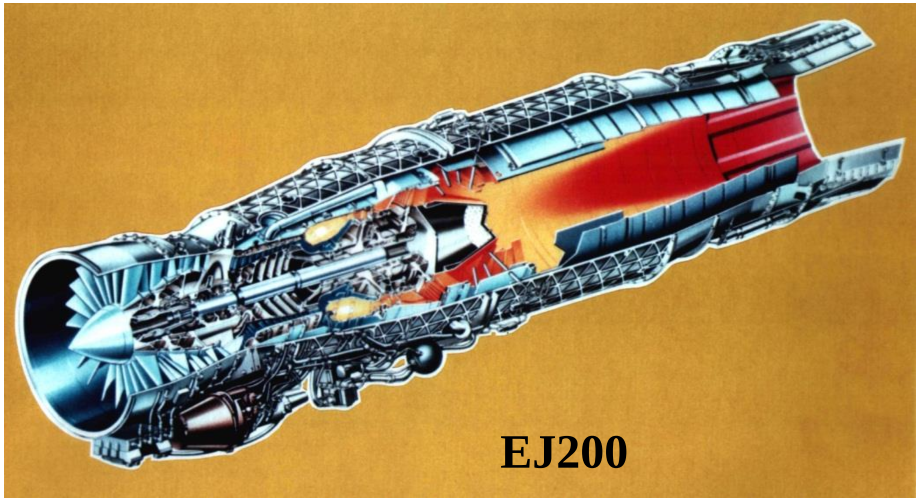
Propulsion Systems for Transport & Combat Aircraft





High by-pass ratio Turbofan

Thrust ~2000 to 100,000 lb

By-pass ratio 4 – 10

OPR ~ 40

Fan PR ~ 1.9

Specific Thrust ~ 25 – 35 lb/lb/sec

Low by-pass ratio Reheated Turbofan

Thrust ~10,000 to 40,000 lb (inc R/H)

By-pass ratio 0.3 – 1

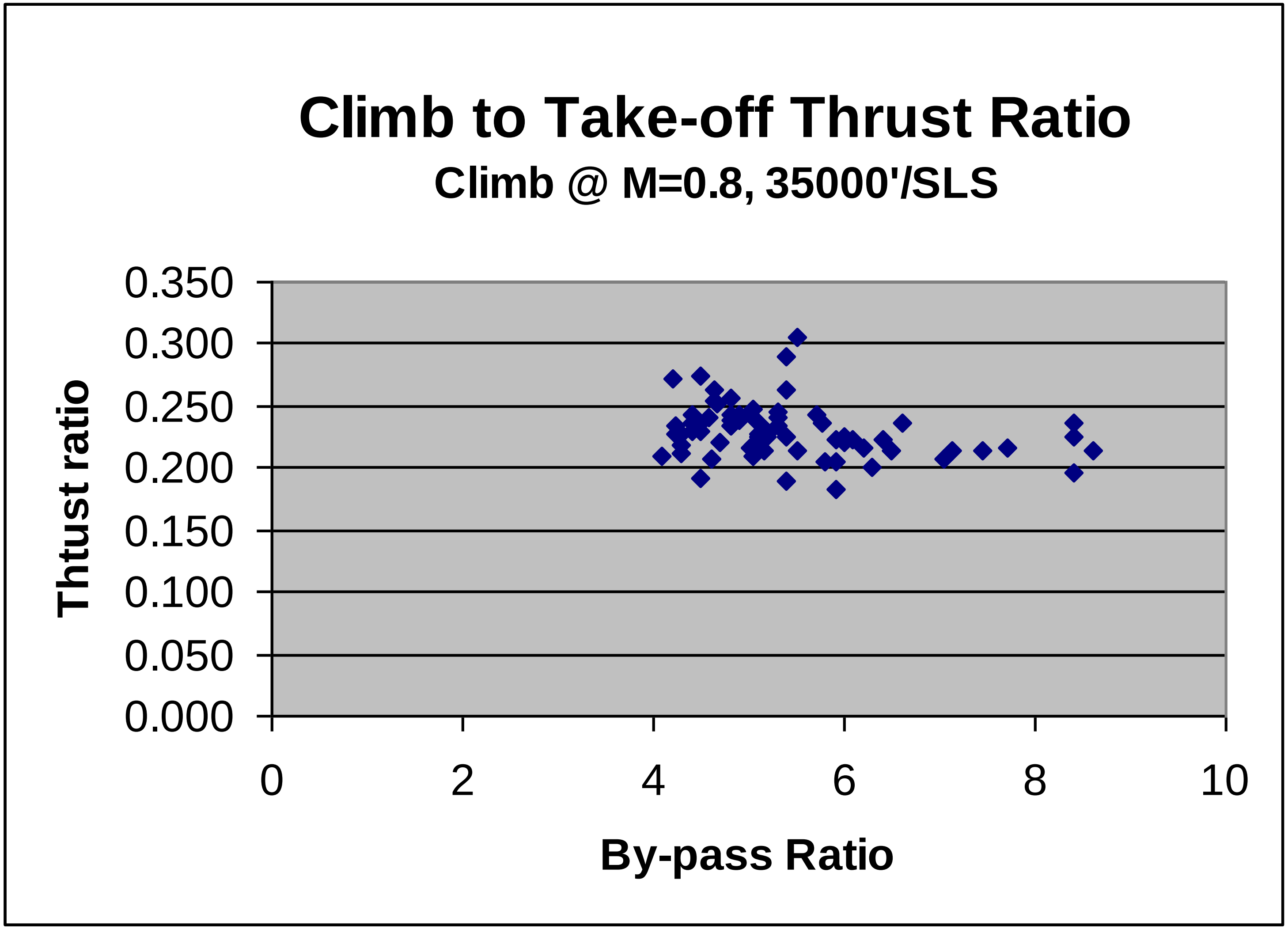
OPR ~ 25 – 30

Fan PR ~ 3 – 5

Specific Thrust ~ 120 lb/lb/sec (inc R/H)

Typical Engine Characteristics

High by-pass ratio Engines



Overall Performance Characteristics

The major points of interest are:

- As combustion temperature and spool speed increase at fixed atmospheric conditions, the thrust increase is non-linear – the thrust increases rapidly at the higher values of combustion temperature & spool speed. The gas turbine has a different characteristic to that of the reciprocating engine.

- At a fixed combustion temperature, thrust falls almost linearly with atmospheric conditions due to falling density and quantity of inlet airflow.

- Initially thrust decreases with forward speed (at fixed atmospheric conditions & combustor temperature). This is due to increasing momentum drag. Subsequently thrust rises as inlet pressure rises giving higher values of jet pipe pressure & jet velocity. Finally thrust falls as compressor delivery temperature rises towards the combustor temperature.

- $SFC/\sqrt{\theta}$ and F_N/δ may be used to normalise engine characteristics. In general SFC increases with Mach Number.

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Key Points from Lecture 4

- Design Points for different types of platforms
- Fundamental Dimensionless relationships
- The calculation of Off-design Performance
- How thrust & fuel consumption varies with inlet conditions i.e. Altitude, Mach Number & throttle setting

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Lecture 5

Engine – Airframe Integration

Objective ~ Lecture 5

To detail the issues arising from the installation of a propulsion system into a vehicle.