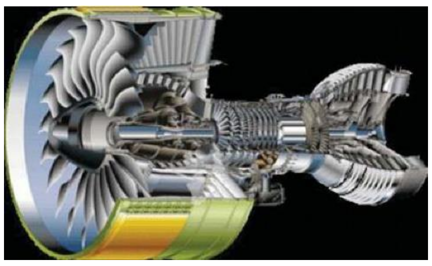
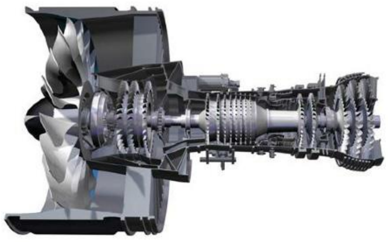
   

UNIVERSITY OF NAPLES ***FEDERICO II*** *1224 A.D.*

Propulsione Aereospaziale

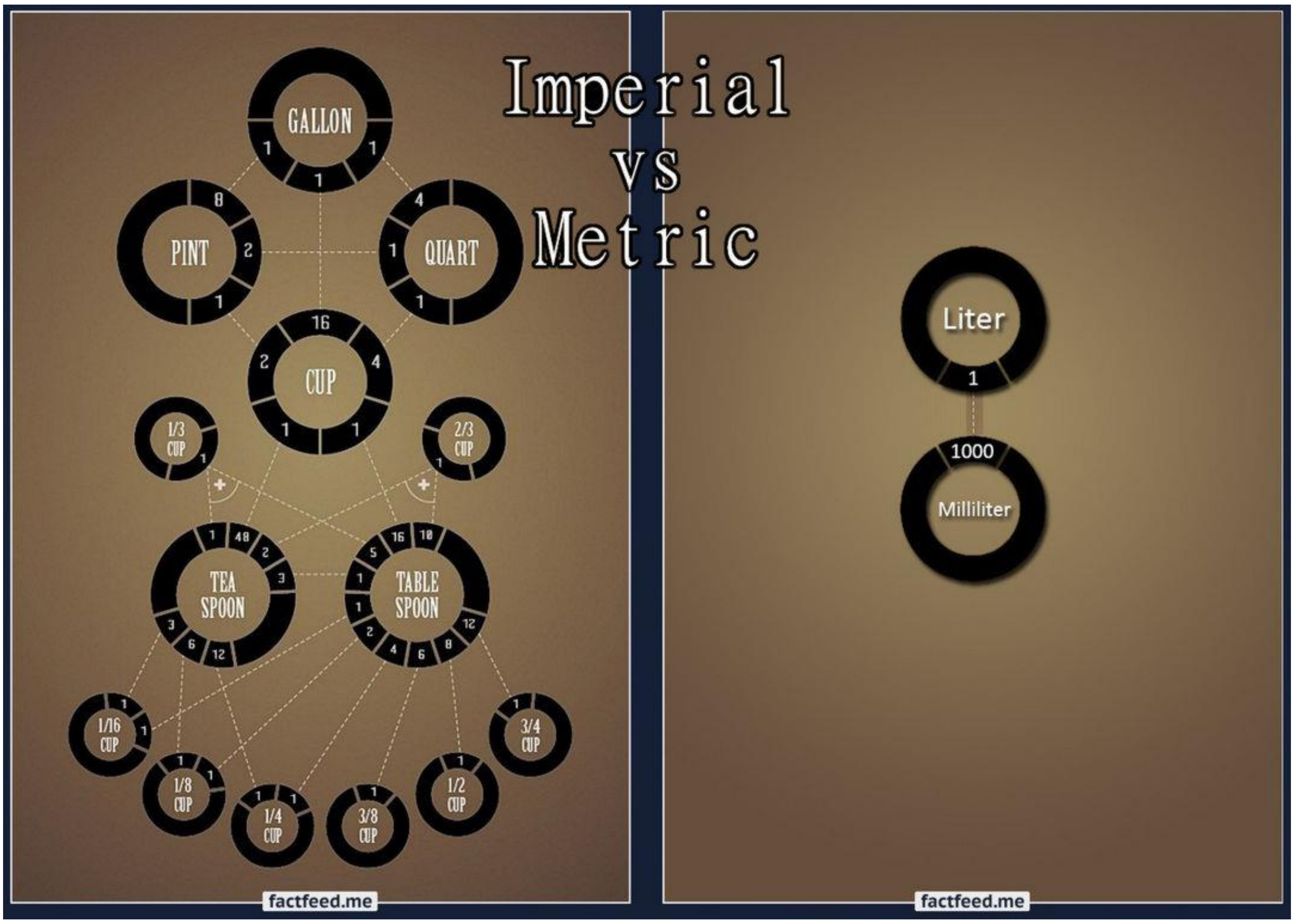
T. Astarita

astarita@unina.it

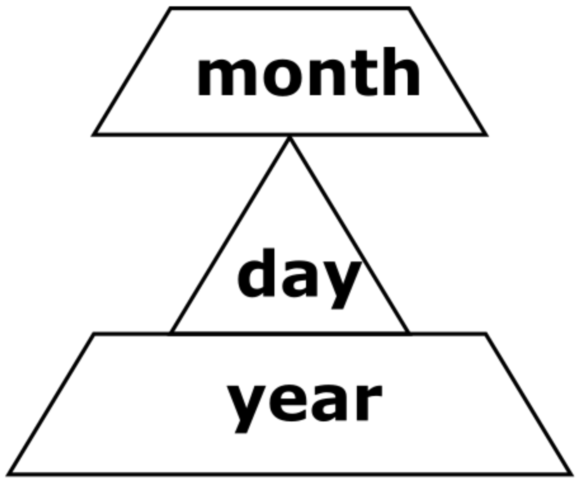
www.docenti.unina.it

Versione del 31.5.2019

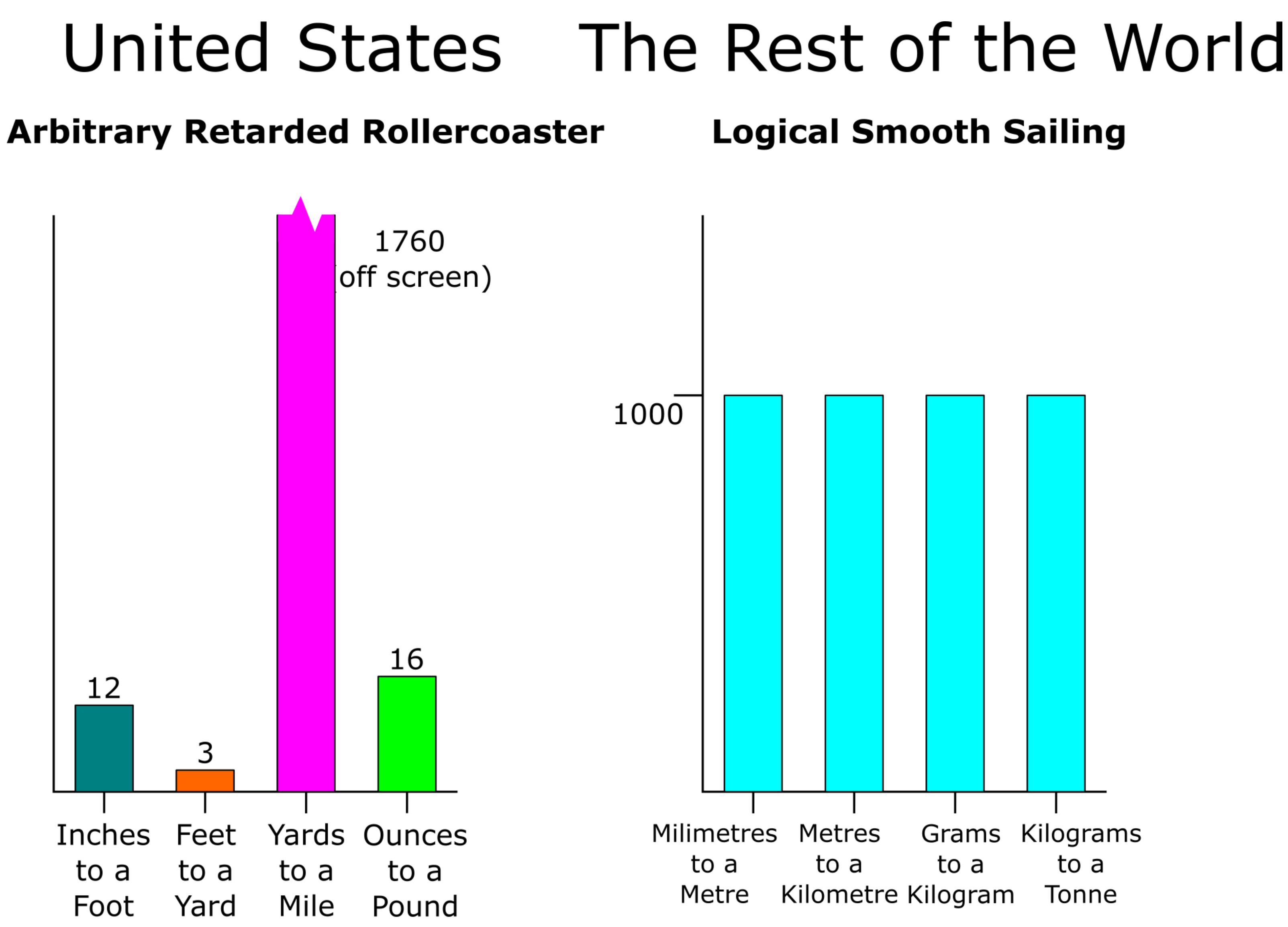
Cenni sul sistema tecnico Americano



Cenni sul sistema tecnico Americano



Lunghezze



Lunghezze

Unità	Divisioni	Equivalente SI
Relazioni esatte in **grassetto**		
Internazionale		
1 *punto*(point, p)		352,777 778 μm
1 *pica* (pc[9])	**12 p**	4,233 333 mm
1 *pollice*(inch, in)	**6 pc**	**25,4 mm**
1 *piede* (foot, ft)	**12 in**	**0,304 8 m**[10]
1 *iarda* (yard, yd)	**3 ft**	**0,914 4 m**[10]
1 *miglio*(mile, mi)	**5 280 ft** o **1 760 yd**	**1,609 344 km**
Nautica internazionale[10]		
1 braccio(fathom, ftm)	**2 yd**	**1,828 8 m**
1 cable (cb)	**120 ftm** o 1.091 fur	**219,456 m**
1 miglio nautico (NM o nmi)	8.439 cb o 1.151 mi	**1,852 km**

Volume liquido		
Unità di misura molto comuni in *corsivo* Conversione esatta in **grassetto**		
Unità	**Divisioni**	**Equivalente SI**
1 minim (min)		**61,611 519 921 875 μL**
1 dramma liquida (US fluid dram, fl dr)	**60 min**	**3,696 691 195 312 5 mL**
1 *cucchiaio da tè* (*teaspoon*, tsp)	**80 min**	**4,928 921 593 75 mL**
1 *cucchiaio da tavola* (*tablespoon*,Tbsp)	**3 tsp** o **4 fl dr**	**14,786 764 781 25 mL**
1 *oncia liquida* (*US fluid ounce*, fl oz)	**2 Tbsp**	**29,573 529 562 5 mL**
1 cicchetto(US shot, jig)	**3 Tbsp**	**44,360 294 343 75 mL**
1 US gill (gi)	**4 fl oz**	**118,294 118 25 mL**

Volume liquido		
Unità di misura molto comuni in *corsivo* Conversione esatta in **grassetto**		
Unità	**Divisioni**	**Equivalente SI**
1 *tazza* (*US cup*, cp)	**2 gi** o **8 fl oz**	**236,588 236 5 mL**
1 *pinta* (*US liquid pint*, pt)	**2 cp**	**473,176 473 mL**
1 *quarto* (*US liquid quart*, qt)	**2 pt**	**0,946 352 946 L**
1 *gallone* (*US liquidgallon*, gal)	**4 qt** o **231 in^3**	**3,785 411 784 L**
1 barile (liquid barrel, bbl[16])	**31.5 gal** o **½ hhd**	**119,240 471 196 L**
1 barile di petrolio (oil barrel, bo[16])	**42 gal** o **⅔ hhd**	**158,987 294 928 L**
1 hogshead (hhd)	**63 gal** o **8$\frac{27}{64}$ ft^3**	**238,480 942 392 L**

Volume secco

Volume secco		
Unità	**Divisioni**	**Equivalente SI**
1 pinta (pt)	33.60 in^3	550,610 5 cm³
1 quarto (qt)	**2 pt**	1,101 221 dm³
1 gallone (gal)	**4 qt**	4,404 884 dm³
1 peck (pk)	**2 gal**	8,809 768 dm³
1 staio[17] (bushel, bu)	**4 pk**	35,239 07 dm³
1 barile (*dry* barrel, bbl)	**7 056 in^3**	115,627 1 dm³

Massa

Tipo	Unità	Divisioni	Equivalenti SI
Avoirdupois	1 grano (grain gr)	**$^{1}/_{7000}$ lb**	**64,798 91 mg**
	1 dramma (dr)	**$27^{11}/_{32}$ gr**(8,859 kt)	**1,771 845 195 312 5 g**
	1 oncia (oz o oz av)	**16 dr**	**28,349 523 125 g**
	1 libbra (*pound*, lb o lb av)	**16 oz**	**453,592 37 g**
	1 quintale americano[20] (hundredweight, cwt)	**100 lb**	**45,359 237 kg**
	1 quintale inglese[20] (long hundredweight)	**112 lb**	**50,802 345 44 kg**
	1 ton (short ton)	**20 US cwt** o **2000 lb**	**907,184 74 kg**
	1 long ton	**20 long cwt** o **2240 lb**	**1 016,046 908 8 kg**

Varie

$$1slug = \frac{1lbf}{ft/s^2} = \frac{1lbm \cdot g}{ft/s^2} = 32.17lbm = 14.593kg$$

$$g = 9.807\frac{m}{s^2} = 9.807\frac{\frac{1}{0.3048}ft}{s^2} = 32.175\frac{ft}{s^2}$$

$$[\rho g] = \frac{lbf}{ft^3} = \frac{g \cdot 0.45359kg}{(.3048)^3 m^3} = g \cdot 16.02\frac{kg}{m^3}$$

$$1psi = 1\frac{lbf}{in^2} = 1\frac{lbm \cdot g}{ft^2/12^2} = 4632\frac{lbm}{ft \cdot s^2} = 4632\frac{.45359kg}{.3048m \cdot s^2} = 6895N$$

$$1BTU = 1kCal\ \frac{lbm \cdot R}{kg \cdot K} = 1kCal\ \frac{.45359}{1.8} = 4186.8 \cdot \frac{.45359}{1.8}J = 1055.1J$$

$$1hp = 550ft \cdot \frac{lbf}{s} = 550 \cdot 0.3048m \cdot 0.45359 \cdot \frac{kg}{s} \cdot g = 745.6W$$

$$xF = (x + 459.67)R = (x + 459.67)/1.8K = (x/1.8 + 255.37)K$$

$$\mathrm{x}F = (x - 32)/1.8C = [(x - 32)/1.8 + 273.15]K$$

TSFC

Le unità di misura del **consumo specifico** (**TSFC** Thrust Specific Fuel Consumption) sono normalmente:

$$\left[TSFC=\frac{\dot{m}_f}{F}\right]=\frac{lbm}{hr\cdot lbf}=\frac{lbm}{3600s\cdot 1lbm\cdot 32.17ft/s^2}=\frac{1}{115830}\frac{s}{ft}$$

$$=\frac{1}{115830}\frac{s}{0.3048m}=\frac{1}{34305}\frac{s}{m}=\frac{1}{34305}\frac{10^6 mg}{s\cdot kg\cdot m/s^2}=28.325\frac{mg}{s\cdot N}$$

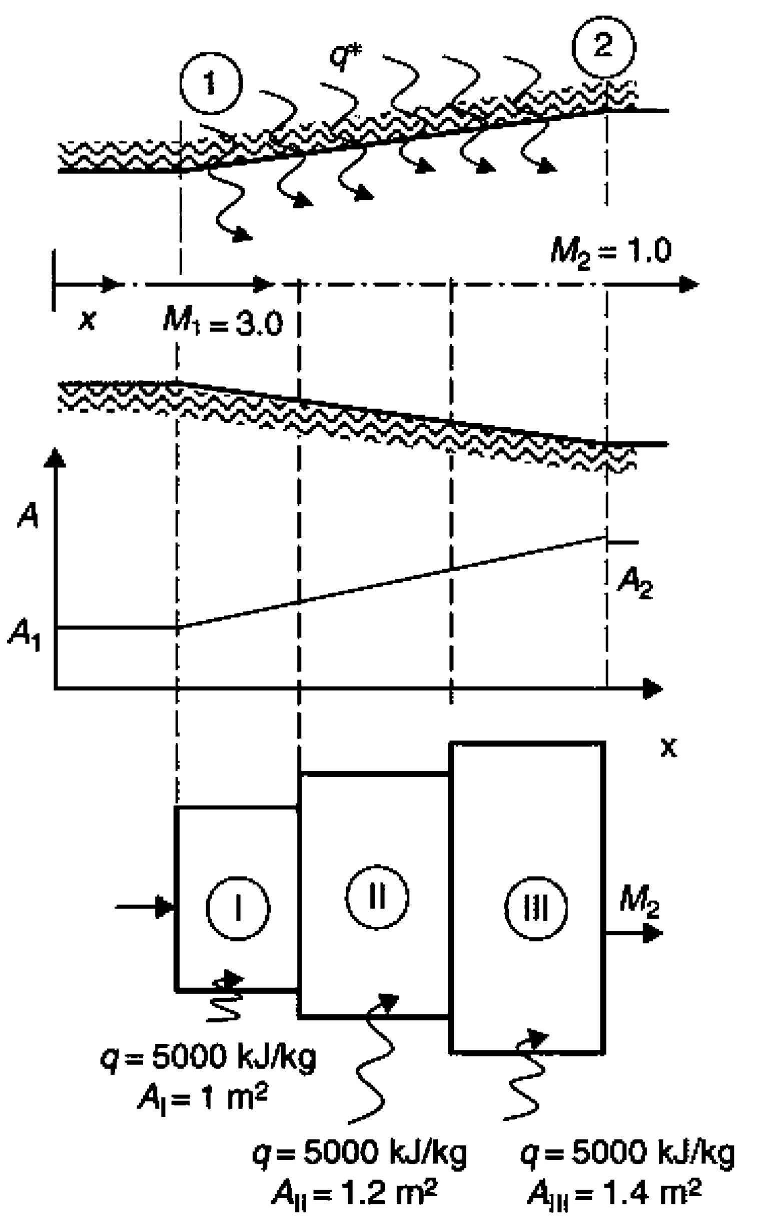
$$\left[TSFC=\frac{\dot{m}_f}{F}\right]=\frac{mg}{s\cdot N}=\frac{g}{s\cdot kN}=\frac{10^3 mg}{s\cdot kN}=\frac{10^{-6}kg}{s\cdot N}$$

Le unità di misura del calorifico del combustibile (fuel heating value) sono:

$$[Q_R]=\frac{BTU}{lbm}=\frac{1055.1J}{lbm}=\frac{1055.1kg\cdot m^2}{lbm\cdot s^2}=\frac{\frac{1055.1}{0.3048^2}ft^2}{.45359\cdot s^2}=25038\frac{ft^2}{s^2}$$

$$=25038\frac{(0.3048m)^2}{s^2}=2326.1\frac{m^2}{s^2}=2326.1\frac{J}{kg}=2.3261\frac{kJ}{kg}$$

Problem Fa2.34



2.34 A scramjet combustor has a supersonic inlet condition and a choked exit. The combustor flow area increases linearly in the flow direction, as shown.

The inlet and exit flow conditions are

$$M_1 = 3.0$$
$$p_1 = 1 \text{ bar}$$
$$T_1 = 1000 \text{ K}$$
$$A_1 = 1 \text{ m}^2$$
$$M_2 = 1.0$$
$$A_2 = 1.4 \text{ m}^2$$
$$\gamma = 1.4, R = 287 \text{ J/kg}\cdot\text{K}$$

The total heat release due to combustion, per unit flow rate in the duct, is initially *assumed* to be 15 MJ/kg. If we divide the combustor into three constant-area sections, with stepwise jumps in the duct area, we may apply Rayleigh flow principles to each segment, as shown. The heat release per segment is then 1/3 of the total heat release in the duct, i.e., 5,000 kJ/kg. As the exit condition of a segment needs to be matched to the inlet condition of the following segment, we propose to satisfy continuity equation at the boundary through an isentropic step area expansion, i.e., p_t, T_t remain the same and only the Mach number jumps isentropically through area expansion.

If we march from the inlet condition toward the exit with the assumed heat release rates, we calculate the exit Mach number M_2. Since the exit flow is specified to be choked, then we need to adjust the total heat release in order to get a choked exit. Calculate the critical heat release in the above duct that leads to thermal choking of the flow.

Problem Fa2.34

Sezione	1	6		γ	1.4									
M	3	1.2		R	287	J/kgK		cp	1004.5	J/kgK				
T	1000		K											
p	1		bar											
A	1	1.4	m/s											
T1/Tt1	0.357		Tt1	2800	K									
Si usano 3 sezioni quindi			1-2	3-4	5-6									
		A	1	1.2	1.4									
Q(kJ/kg)	500	Ra	Iso	Ra	Iso	Ra		Q	600	Ra	Iso	Ra	Iso	Ra
Sezione	1	2	3	4	5	6		Sezione	1	2	3	4	5	6
A	1	1	1.2	1.2	1.4	1.4		A	1	1	1.2	1.2	1.4	1.4
M	3	2.121	2.325	1.755	1.954	1.505		M	3	1.999	2.211	1.589	1.809	1.295
T0/T0*	0.654	0.770	0.736	0.847	0.803	0.908		T0/T0*	0.654	0.793	0.754	0.887	0.834	0.959
Tt	2800	3298	3298	3796	3796	4293		Tt	2800	3397	3397	3995	3995	4592
A/A*	4.23	1.870	2.24	1.392	1.624	1.179		A/A*	4.23	1.687	2.02	1.241	1.448	1.064
Err	0.305							Err	0.095	Q	645.43			
Q	645.43	Ra	Iso	Ra	Iso	Ra		Q	639.21	Ra	Iso	Ra	Iso	Ra
Sezione	1	2	3	4	5	6		Sezione	1	2	3	4	5	6
A	1	1	1.2	1.2	1.4	1.4		A	1	1	1.2	1.2	1.4	1.4
M	3	1.948	2.1624	1.517	1.7491	1.185		M	3	1.955	2.1689	1.527	1.7571	1.202
T0/T0*	0.654	0.804	0.763	0.905	0.848	0.981		T0/T0*	0.654	0.803	0.762	0.903	0.846	0.978
Tt	2800	3443	3443	4085	4085	4728		Tt	2800	3436	3436	4073	4073	4709
A/A*	4.23	1.616	1.9396	1.188	1.3856	1.026		A/A*	4.23	1.626	1.9507	1.195	1.3937	1.031
Err	-0.015	Q	639.21											

Problem MA3.11

At launch, the space shuttle main engine (SSME) has 1030 lbm/s of gas leaving the combustion chamber at $P_t = 3000$ psia and $T_t = 7350°$R. The exit area of the SSME nozzle is 77 times the throat area. If the flow through the nozzle is considered to be reversible and adiabatic (isentropic) with $Rg_c = 3800\ \text{ft}^2/(\text{s}^2 \cdot °\text{R})$ and $\gamma = 1.25$, find the area of the nozzle throat (in.^{2}) and the exit Mach number. *Hint*: Use the mass flow parameter to get the throat area and Eq. (3.14) to get the exit Mach number.

$$\Psi(\gamma, M) = \frac{A^*}{A}\Psi^* = \gamma M \left(1 + \frac{\gamma - 1}{2}M^2\right)^{-\frac{(\gamma+1)}{2(\gamma-1)}} = \gamma M \psi^{-K}$$

con:

$$\psi(\gamma, M) = 1 + \frac{\gamma - 1}{2}M^2 \qquad K(\gamma) = \frac{(\gamma + 1)}{2(\gamma - 1)} \qquad \Psi^* = \gamma\left(\frac{\gamma + 1}{2}\right)^{-K}$$

Model no.	Manufacturer	Takeoff Thrust, lbf	Takeoff BR[a]	Takeoff OPR[b]	Takeoff Airflow, lbm/s	Cruise Alt, kft	Cruise Mach	Cruise Thrust, lbf	Cruise TSFC[c]	Application
CF 34-8	General Electric	14,500	5	28	——	——	——	——	0.68	Bombardier CRJ700 Embraer 170/175
CF6-50-C2	General Electric	52,500	4.31	30.4	1,476	35	0.80	11,555	0.630	DC10-10, A300B, 747-200
CF6-80-C2	General Electric	52,500	5.31	27.4	1,650	35	0.80	12,000	0.576	767-200, -300, -200ER
GE90-B4	General Electric	87,400	8.40	39.3	3,037	35	0.80	17,500	——	777
GEnx	General Electric	53,000–75,000	10	42	——	——	0.85	——	——	787, 747-8
JT8D-15A	Pratt & Whitney	15,500	1.04	16.6	327	30	0.80	4,920	0.779	727, 737, DC9
JT9D-59A	Pratt & Whitney	53,000	4.90	24.5	1,639	35	0.85	11,950	0.646	DC1O-40, A300B, 747-200
PW2037	Pratt & Whitney	38,250	6.00	27.6	1,210	35	0.85	6,500	0.582	757-200
PW4052	Pratt & Whitney	52,000	5.00	27.5	1,700	——	——	——	——	767, A310-300
PW4084	Pratt & Whitney	87,900	6.41	34.4	2,550	35	0.83	——	——	777
CFM56-3C	CFM International	23,500	6.00	22.6	655	35	0.80	5,540	0.648	737-300, -400, -500
CFM56-5C	CFM International	31,200	6.60	31.5	1,027	35	0.80	6,600	0.545	A340
AE 3007	Rolls-Royce	8,600	4.8	20	——	——	——	——	——	Embraer 37, Global Hawk UAV
RB211-524B	Rolls-Royce	50,000	4.50	28.4	1,513	35	0.85	11,000	0.643	L1011-200, 747-200
RB211-535E	Rolls-Royce	40,100	4.30	25.8	1,151	35	0.80	8,495	0.607	757-200
RB211-882	Rolls-Royce	84,700	6.01	39.0	2,640	35	0.83	16,200	0.557	777
Trent 900	Rolls-Royce	70,000–76,500	8.7–8.5	——	2,655–2,745	——	——	——	——	A380
Trent 1000	Rolls-Royce	53,000–75,000	10–11	——	2,400–2,670	——	——	——	——	787
V2528-D5	International Aero Engines	28,000	4.70	30.5	825	35	0.80	5,773	0.574	MD-90
ALF502R-5	Honeywell	6,970	5.70	12.2	——	25	0.70	2,250	0.720	BAe 146-200, −200
TFE731-5	Honeywell	4,500	3.34	14.4	140	40	0.80	986	0.771	BAe 125-800
PW300	Pratt & Whitney Canada	4,750	4.50	23.0	180	40	0.80	1,113	0.675	BAe 1000
FJ44	Williams Rolls	1,900	3.28	12.8	63.3	30	0.70	600	0.750	——
Olympus 593	Rolls-Royce/SNECMA	38,000	0	11.3[d]	410	53	2.00	10,030	1.190	Concorde
GP7270	Engine Alliance	70,000	8.7[d]	45.6[e]	——	35	0.85	12,633	——	A380

[a]BR = bypass ratio. [b]OPR = overall pressure ratio. [c]TSFC = thrust specific fuel consumption. [d]At cruise. [e]max climb.
(Sources: Manufacturers' literature).

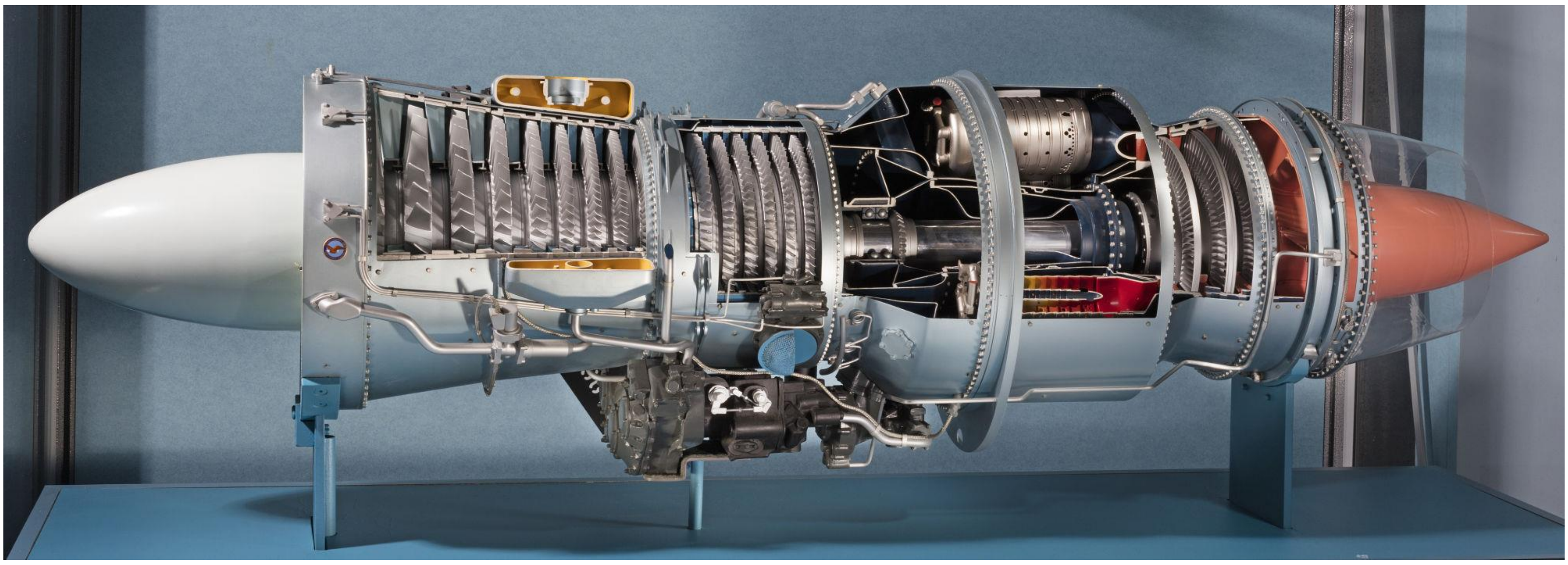
Some data

Table B.4 Temperature/pressure data for some engines

Temperature and pressure	Pegasus turbofan, separate exhaust	J57 turbojet w/AB exhaust	JT3D turbofan, separate exhaust	JT8D turbofan, mixed exhaust	JT9D turbofan, separate exhaust	F100-PW-100 turbofan, mixed w/AB exhau
P_{t2}, psia	14.7	14.7	14.7	14.7	14.7	13.1
T_{t2}, °F	59	59	59	59	59	59
$P_{t2.5}$, psia	36.1	54	63	60	32.1	39.3
$T_{t2.5}$, °F	242	330	360	355	210	297
P_{t13}, psia	36.5	——	26	28	22.6	39.3
T_{t13}, °F	257	——	170	190	130	297
P_{t3}, psia	216.9	167	200	233	316	316
T_{t3}, °F	708	660	715	800	880	1,014
P_{t4}, psia	——	158	190	220	302	304
T_{t4}, °F	1,028	1,570	1,600	1,720	1,970	2,566
P_{t5} or P_{t6}, psia	29.3	36	——	2	20.9	38.0
T_{t5} or T_{t6}, °F	510	1,013	——	——	850	1,368
P_{t16}, psia	——	——	——	——	——	36.8
T_{t16}, °F	——	——	——	——	——	303
P_{t6A}, psia	——	——	——	29	——	37.5
T_{t6A}, °F	——	——	——	890	——	960
P_{t7}, psia	——	31.9	28	29	20.9	33.8
T_{t7}, °F	——	2,540	890	890	850	3,204
P_{t17}, psia	36.5	——	26	——	22.4	——
T_{t17}, °F	257	——	170	——	130	——
Bypass ratio α	1.4	0	1.36	1.1	5.0	0.69
Thrust, lbf	21,500	16,000	18,000	14,000	43,500	23,700
Airflow, lbm/s	444	167	460	315	1,495	224

Problem 3.1

Il J57-JT3C è stato usato su: B-52, F-100, F-4 Phantom.



Problem 3.1

3.1 The total pressures and temperatures of the gas in an afterburning turbojet engine are shown (J57 "B" from Pratt & Whitney, 1988). The mass flow rates for the air and fuel are also indicated at two engine settings, the Maximum Power and the Military Power. Use the numbers specified in this engine to calculate

(a) the fuel-to-air ratio f in the primary burner and the afterburner, at both power settings

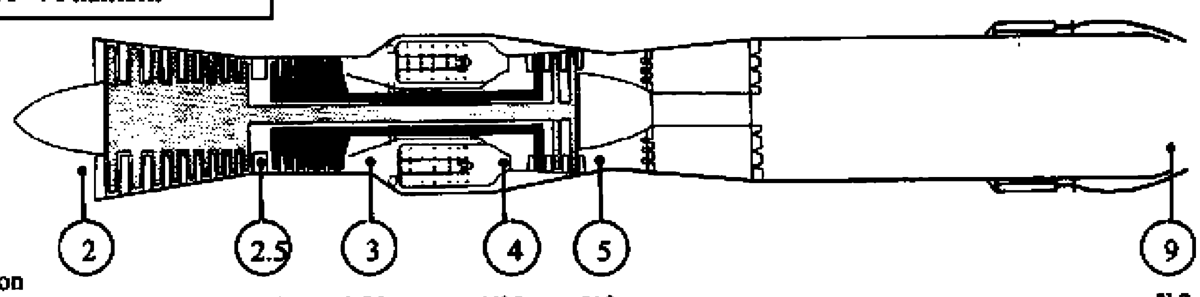
(b) the low- and high-pressure spool compressor pressure ratios and the turbine pressure ratio (note that these remain constant with the two power settings)

(c) the exhaust velocity V_9 for both power settings by assuming the specified thrust is based on the nozzle gross thrust (because of sea level static) and *neglecting any pressure thrust* at the nozzle exit

(d) the thermal efficiency of this engine for both power settings (at the sea level static operation),

J57 "B" Series afterburning military turbojet sea level static internal pressures and temperatures (typical)

J57 powered many military aircraft including B-52, F-100, and F-4 Phantom



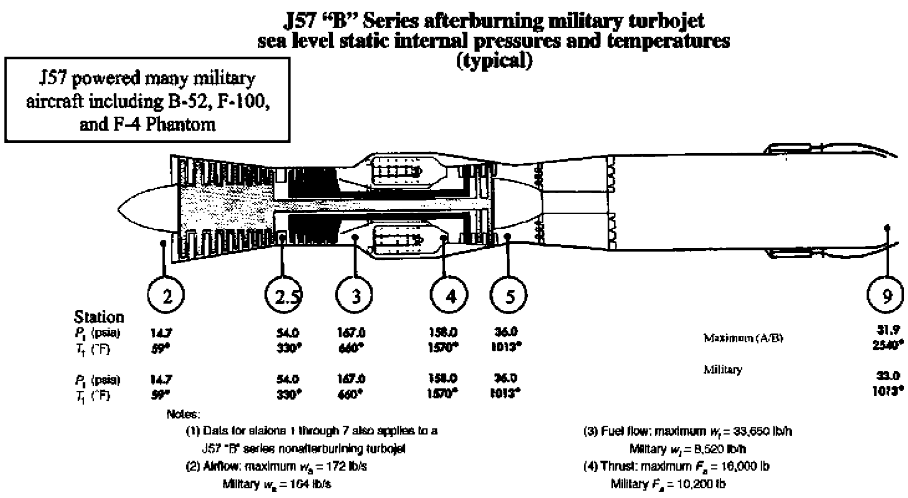
Station		2	2.5	3	4	5		9
P_t (psia)		14.7	54.0	167.0	158.0	36.0	Maximum (A/B)	31.9
T_t (°F)		59°	330°	660°	1570°	1013°		2540°
P_t (psia)		14.7	54.0	167.0	158.0	36.0	Military	33.0
T_t (°F)		59°	330°	660°	1570°	1013°		1013°

Notes:
(1) Data for stations 1 through 7 also applies to a J57 "B" series nonafterburning turbojet
(2) Airflow: maximum w_a = 172 lb/s
Military w_a = 164 lb/s
(3) Fuel flow: maximum w_f = 33,650 lb/h
Military w_f = 8,520 lb/h
(4) Thrust: maximum F_g = 16,000 lb
Military F_g = 10,200 lb

Problem 3.1

3.1 The total pressures and temperatures of the gas in an afterburning turbojet engine are shown (J57 "B" from Pratt & Whitney, 1988). The mass flow rates for the air and fuel are also indicated at two engine settings, the Maximum Power and the Military Power. Use the numbers specified in this engine to calculate

(a) the fuel-to-air ratio *f* in the primary burner and the afterburner, at both power settings



(b) the low- and high-pressure spool compressor pressure ratios and the turbine pressure ratio (note that these remain constant with the two power settings)

(c) the exhaust velocity V_9 for both power settings by assuming the specified thrust is based on the nozzle gross thrust (because of sea level static) and *neglecting any pressure thrust* at the nozzle exit

(d) the thermal efficiency of this engine for both power settings (at the sea level static operation), assuming the fuel heating value is 18,600 BTU/lbm and c_p = 0.24 BTU/lbm · °R. Explain the lower thermal efficiency of the Maximum power setting

(e) the thrust specific fuel consumption in lbm/h/lbf in both power settings

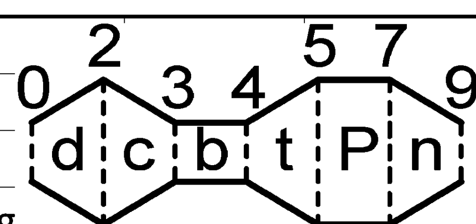
(f) the Carnot efficiency of a corresponding engine, i.e., operating at the same temperature limits, in both power settings

(g) the comparison of percent thrust increase to percent fuel flow rate increase when we turn the afterburner on

(h) why don't we get proportional thrust increase with fuel flow increase (when it is introduced in the afterburner), i.e., doubling the fuel flow in the engine (through afterburner use) does not double the thrust

TurboJet con e senza PB

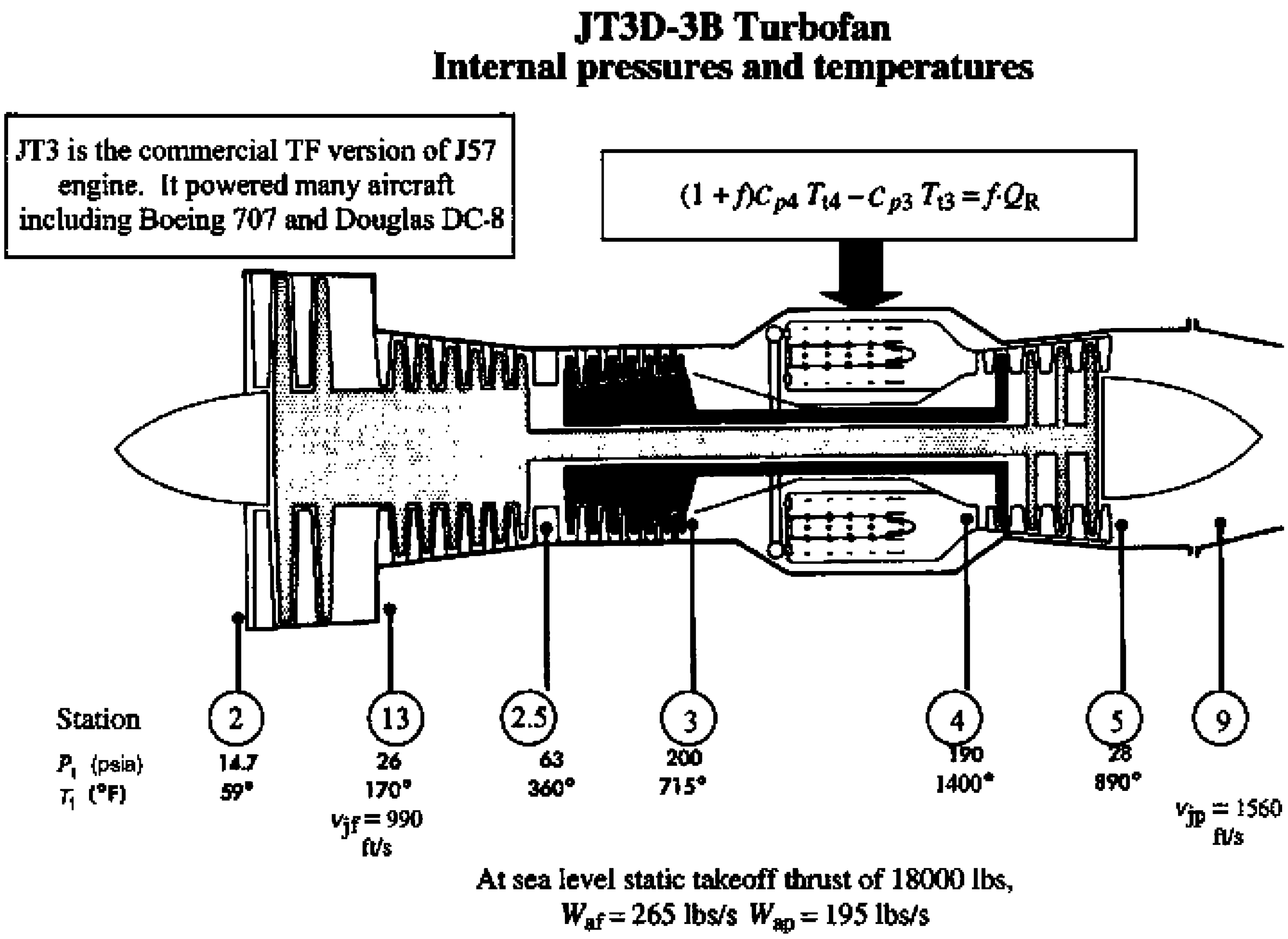
Turbo Jet By TomLevel 4							
	2	3	4	5	7	9	
	diff	comp	CC	Tur	AB	No	
c_p	1004			1152		1243	J/kg
γ	1.4			1.33		1.3	
π	0.96	10	0.95		0.98	0.97	
η, $e_{c,t}$		0.9	0.99	0.9	0.99		
Tt				1750		2250	
M0	2		po/p9	1	QR	42800	kJ/kgK
T0	250	K	p0	101,300	Pa	η_m	0.99
k	0.28571			0.24812		0.23077	
R	286.857			285.835		286.846	kJ/kgK



Problem 3.2

JT3D-3B è stato usato su: B707 e DC8.

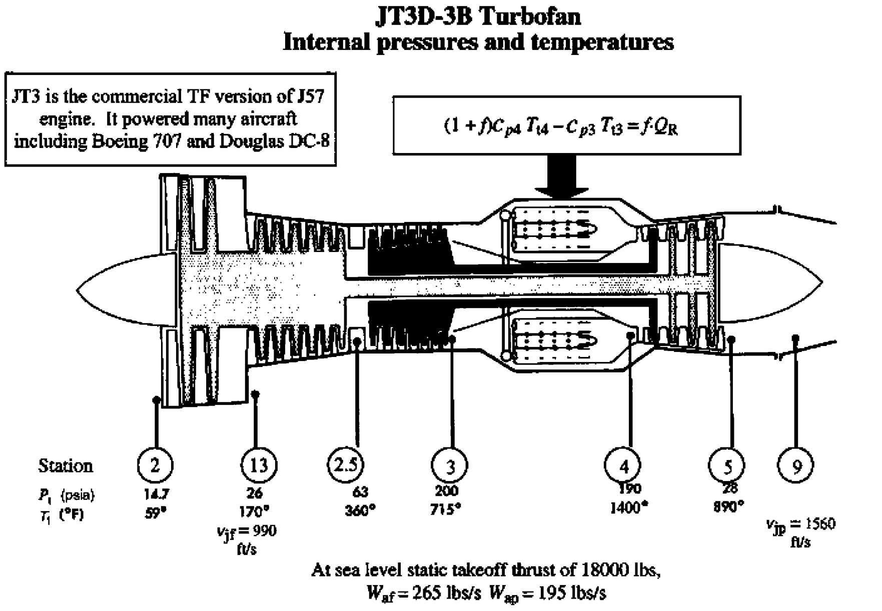
Problem 3.2



Problem 3.2

3.2 The total pressures and temperatures of the gas are specified for a turbofan engine with separate exhaust streams (JT3D-3B from Pratt & Whitney, 1974). The mass flow rates in the engine core (or primary) and the engine fan are also specified for the sea level static operation. Calculate

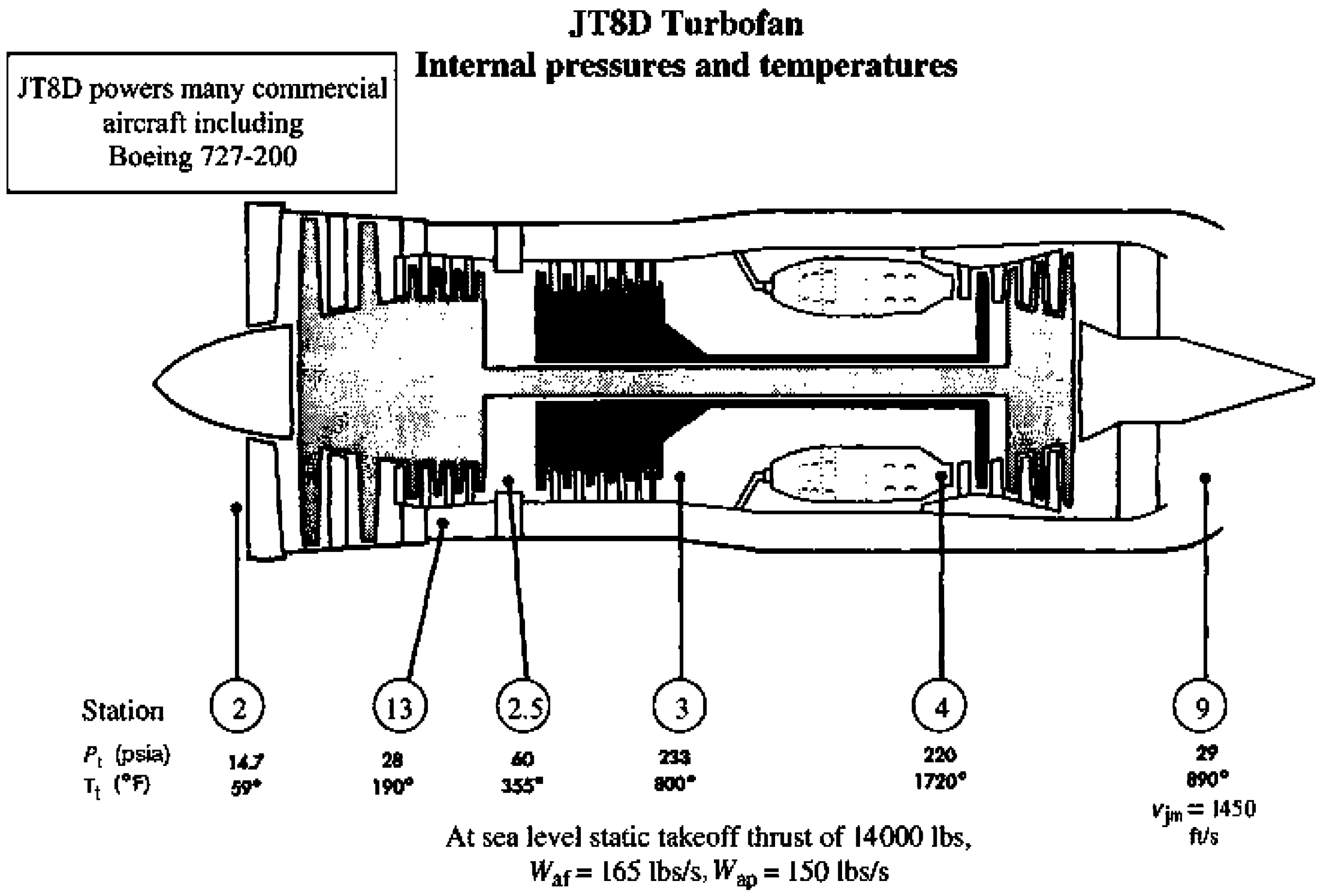
- **(a)** the engine bypass ratio α defined as the ratio of fan-to-core flow rate
- **(b)** from the total temperature rise across the burner, estimate the fuel-to-air ratio and the fuel flow rate in lbm/h, assuming the fuel heating value is $Q_R \sim 18{,}600$ BTU/ lbm and the specific heat at constant pressure is 0.24 and 0.26 BTU/lbm · °R at the entrance and exit of the burner, respectively
- **(c)** the engine static thrust based on the exhaust velocities and the mass flow rates *assuming perfectly expanded nozzles* and compare your answer to the specified thrust of 18,000 lbs
- **(d)** the engine thermal efficiency η_{th}
- **(e)** the thermal efficiency of this engine compared to the afterburning turbojet of Problem 1. Explain the major contributors to the differences in η_{th} in these two engines
- **(f)** the engine thrust specific fuel consumption in lbm/h/lbf
- **(g)** the nondimensional engine specific thrust
- **(h)** the Carnot efficiency corresponding to this engine
- **(i)** the engine overall pressure ratio p_{t3} / p_{t2}
- **(j)** fan nozzle exit Mach number [use $T_t = T + V^2/2c_p$ to calculate local static temperature at the nozzle exit, then local speed of sound $a = (\gamma RT)^{1/2}$]



Problem 3.3

JT8D è stato usato su: B727, DC9 e MD80.

Problem 3.3



Problem 3.3

3.3 A mixed exhaust turbofan engine (JT8D from Pratt and Whitney, 1974) is described by its internal pressures and temperature, as well as air mass flow rates and the mixed jet (exhaust) velocity. Let us examine a few parameters for this engine, for a ballpark approximation.

(a) Estimate the fuel flow rate from the total temperature rise across the burner assuming the fuel heating value is ~18,600 BTU/lbm and the specific heat at constant pressure is 0.24 and 0.26 BTU/lbm · °R at the entrance and exit of the burner, respectively

(b) Calculate the momentum thrust at the exhaust nozzle and compare it to the specified thrust of 14,000 lbs

(c) Estimate the thermal efficiency of this engine and compare it to Problems 3.1 and 3.2 as well as a Carnot cycle operating between the temperature extremes of this engine. Explain the differences

(d) Estimate the specific fuel consumption for this engine in lbm/h/lbf

(e) The overall pressure ratio (of the fan–compressor section) p_{t3}/p_{t2}

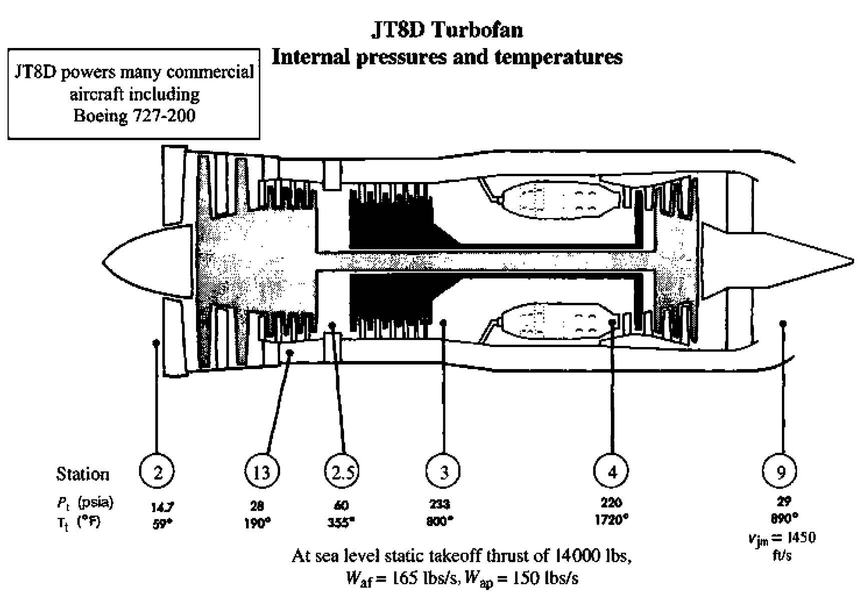
(f) What is the bypass ratio α for this engine at takeoff

(g) What is the Carnot efficiency corresponding to this engine

(h) Estimate nozzle exit Mach number [look at part (j) in Problem 3.2]

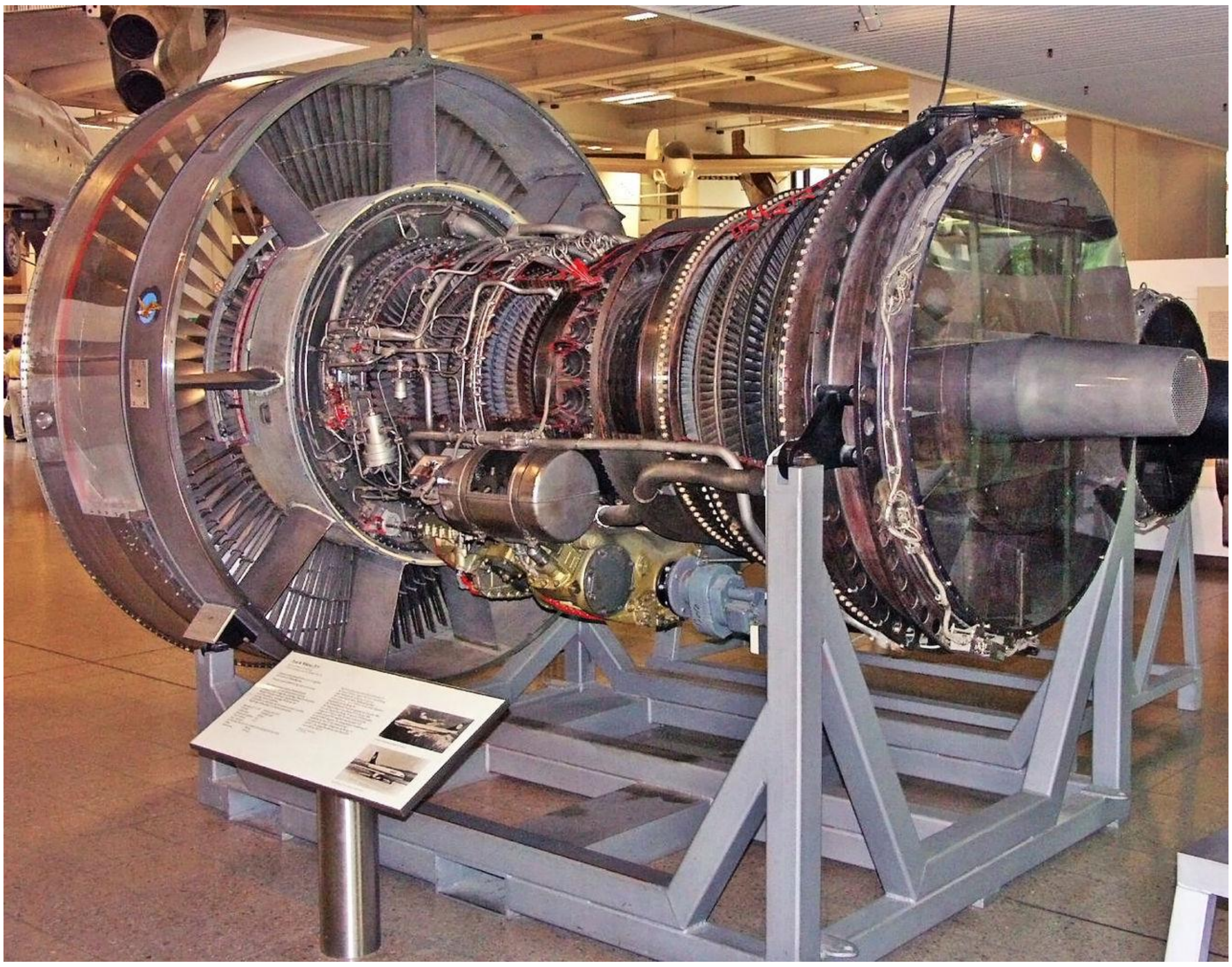
(i) What is the low-pressure compressor (LPC) pressure ratio $p_{t2.5}/p_{t2}$

(j) What is the high-pressure compressor (HPC) pressure ratio $p_{t3}/p_{t2.5}$

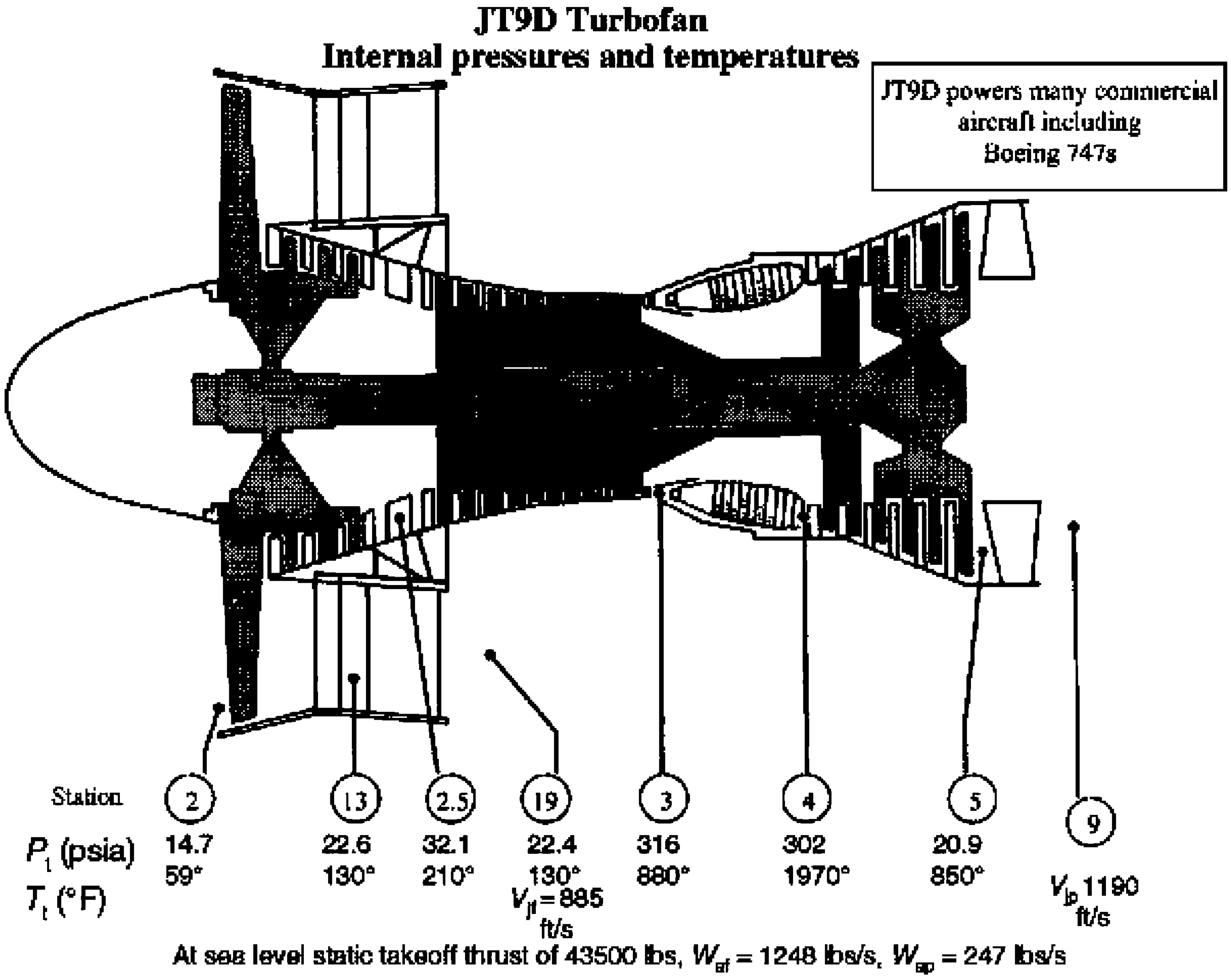


Problem 3.4

JT9D è stato usato su: B747, A310 e 767.



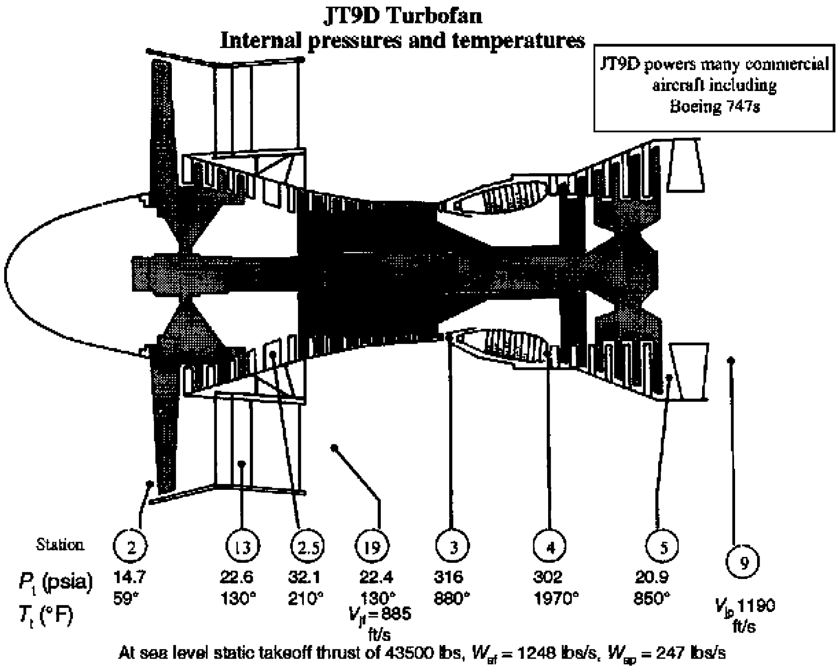
Problem 3.4



Problem 3.4

3.4 A large bypass ratio turbofan engine (JT9D engine from Pratt and Whitney, 1974) is described by its fan and core engine gas flow properties.

- **(a)** What is the overall pressure ratio (OPR) of this engine
- **(b)** Estimate the fan gross thrust $F_{g,fan}$ in lbf
- **(c)** Estimate the fuel-to-air ratio based on the energy balance across the burner, assuming the fuel heating value is ~18,600 BTU/lbm and the specific heat at constant pressure is 0.24 and 0.26 BTU/lbm · °R at the entrance and exit of the burner, respectively
- **(d)** Calculate the core gross thrust and compare the sum of the fan and the core thrusts to the specified engine thrust of 43,500 lbf
- **(e)** Calculate the engine thermal efficiency and compare it to Problems 3.1–3.3. Explain the differences
- **(f)** Estimate the thrust-specific fuel consumption (TSFC), in lbm/h/lbf
- **(g)** What is the bypass ratio of this turbofan engine
- **(h)** What is the Carnot efficiency η_{Carnot} corresponding to this engine
- **(i)** What is the LPC pressure ratio $p_{t2.5} / p_{t2}$
- **(j)** What is the HPC pressure ratio $p_{t3} / p_{t2.5}$
- **(k)** Estimate the fan nozzle exit Mach number [see part (j) in Problem 3.2]
- **(l)** Estimate the primary nozzle exit Mach number

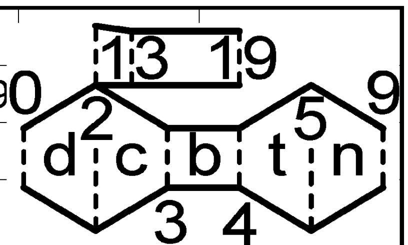


Problem 3.4

Valutare inoltre il lavoro nei vari stadi del compressore, nella turbina e il calore scambiato nella camera di combustione.

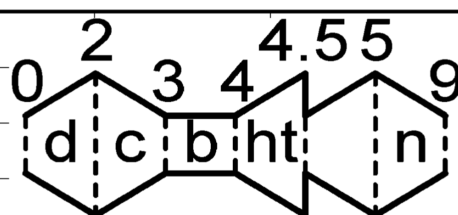
TurboFan

JT9d By Tom									
	2	3	4	5	9	13	19		
	diff	comp	CC	Tur	No	Fan	No FAn		
c_p	1004			1057					
γ	1.4			1.35					
π	1	21.5	0.955		0.98	1.53	0.99		
$\eta, e_{c,t}$		0.92	0.95	0.9		0.96			
Tt				1349.8					
M0	0				QR	42800	kJ/kgK		
T0	288	K	p0	101,300	Pa	η_m	0.98		
alpha	5.053								
k	0.28571			0.25926					
R	286.857			274.037	kJ/kgK				
a0	340.1	m/s	V0	0.0	m/s				



TurboProp

Turbo Prop FaE4.38									
	2	3	4	4.5	5	9			
	diff	comp	CC	Tur	Tur	No	Prop		
c_p	1004			1152	J/kgs				
γ	1.4			1.33					
π	0.99	35	0.96						
$\eta, e_{c,t}$		0.92	0.99	0.8	0.859	0.95	0.85		
Tt				1650					
M0	0.82				QR	42000	kJ/kgK		
T0	258	K	p0	30,000	Pa	η_m H	0.99		
alpha	0.75	m0	50	kg/s		η_m L	0.99		
k	0.28571			0.24812		η_{gb}	0.995		
R	286.857			285.835	kJ/kgK				
a0	321.9	m/s	V0	263.9	m/s				

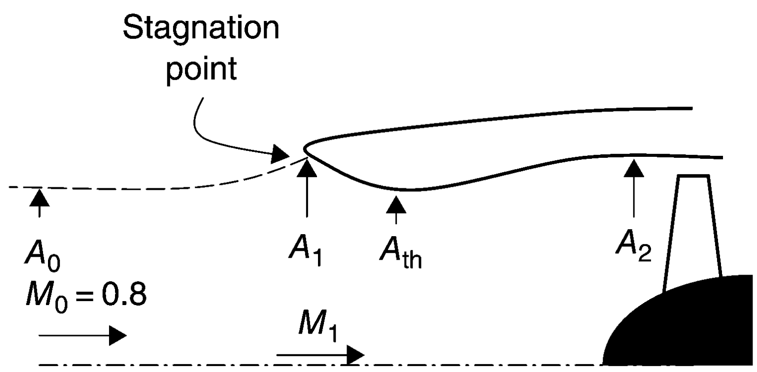


Prese d'aria ed ugelli

MA4.3 The inlet for a high-bypass-ratio turbofan engine has an area A_1 of 6.0 m^2 and is designed to have an inlet Mach number M_1 of 0.6. Determine the additive drag at the flight conditions of sea-level static test and Mach number of 0.8 at 12-km altitude.

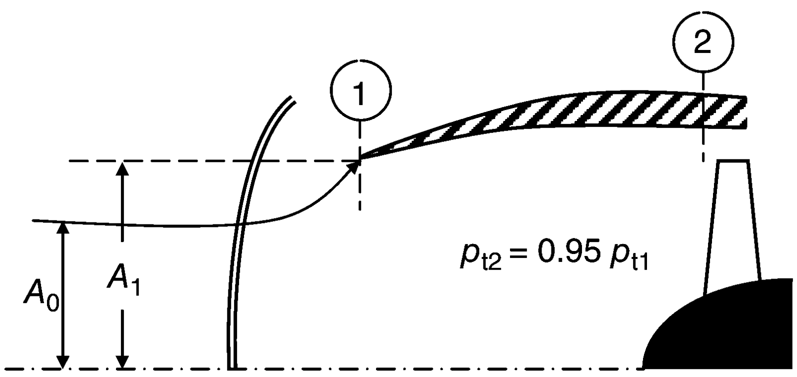
6.3 Consider a subsonic inlet at a flight cruise Mach number of 0.8. The captured streamtube undergoes a prediffusion external to the inlet lip, with an area ratio $A_0/A_1 = 0.92$, as shown. Calculate

- **(a)** C_p (i.e., the pressure coefficient) at the stagnation point
- **(b)** inlet lip Mach number M_1
- **(c)** lip contraction ratio A_1/A_{th} for a throat Mach number $M_{th} = 0.75$ (assume $p_{t,th}/p_{t1} = 1$)
- **(d)** the diffuser area ratio A_2/A_{th} if $M_2 = 0.5$ and $p_{t2}/p_{t,th} = 0.98$
- **(e)** the nondimensional inlet additive drag D_{add}/p_0A_1.



6.17 A normal-shock inlet is operating in a supercritical mode, as shown. Flight Mach number is $M_0 = 1:6$. The inlet capture area ratio $A_0/A_1 = 0:90$ and the diffuser area ratio $A_2/A_1 = 2$. Calculate

- **(a)** M_1
- **(b)** inlet total pressure recovery π_d, i.e., p_{t2}/p_{t0}



■ **FIGURE P6.17**

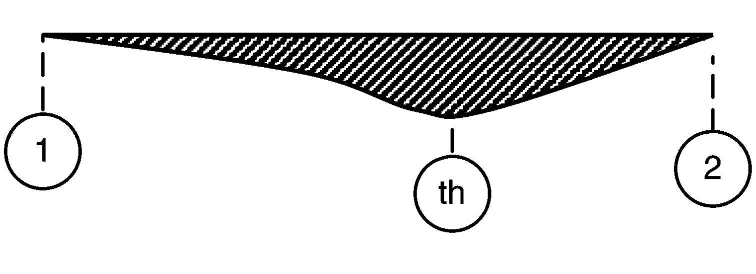
Prese d'aria ed ugelli

6.18 An isentropic convergent–divergent supersonic inlet is designed for $M_D = 1.6$. Calculate the inlet's

- **(a)** area contraction ratio A_1/A_{th}
- **(b)** subsonic Mach number where the throat first chokes
- **(c)** percent spillage at $M_0 = 0.7$
- **(d)** percent spillage at $M_0 = 1.6$ (in the unstarted mode)
- **(e)** overspeed Mach number to start this inlet, $M_{overspeed}$
- **(f)** throat Mach number after the inlet was started, with still $M_{overspeed}$ as the flight Mach number

6.23 A Kantrowitz–Donaldson inlet is designed for $M_D = 1.7$. Calculate

- **(a)** the inlet contraction ratio A_1/A_{th}
- **(b)** the throat Mach number after the inlet self started
- **(c)** the total pressure recovery with the best backpressure.



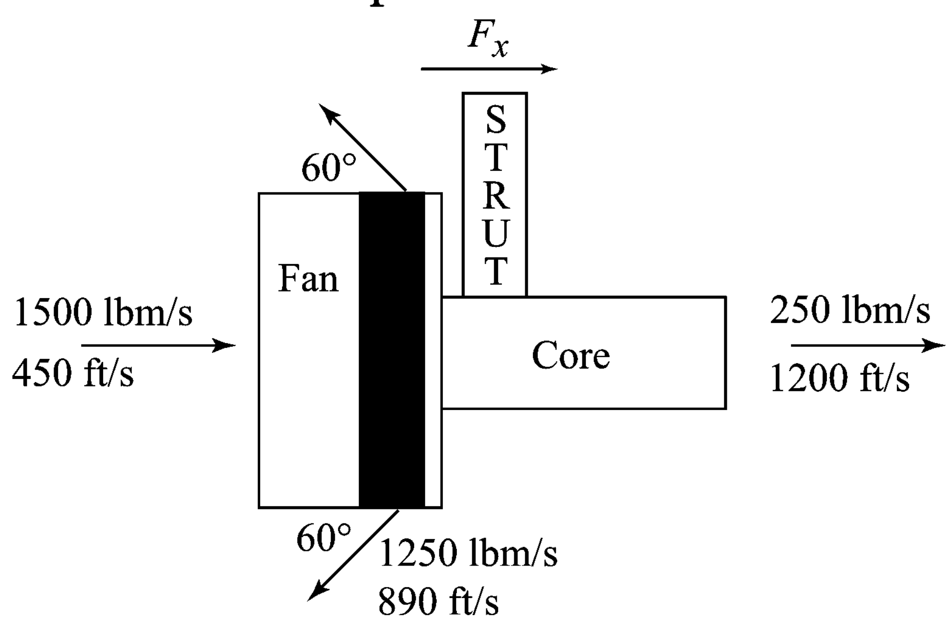
6.43 A convergent nozzle experiences π_{cn} of 0.98, the gas ratio of specific heats $\gamma = 1.30$, and the gas constant is $R = 291$ J/kg · K. First, calculate the minimum nozzle pressure ratio that will choke the expanding nozzle, i.e., NPR_{crit}. This nozzle operates, however, at a higher NPR than the critical, namely, NPR = 4. 2 and with an inlet stagnation temperature of $T_{t7} = 939$ K. Assuming this nozzle operates in $p_0 = 100$ kPa ambient static pressure, calculate

- **(a)** the exit static pressure and temperature p_9 and T_9, respectively
- **(b)** the actual exit velocity V_9 in m/s
- **(c)** nozzle adiabatic efficiency η_n
- **(d)** the ideal exit velocity V_{9s} in m/s
- **(e)** percentage gross thrust gain, had we used a convergent–divergent nozzle with perfect expansion
- **(f)** nozzle discharge coefficient C_{D8}
- **(g)** draw a qualitative wave pattern in the exhaust plume

6.44 A convergent–divergent nozzle has a conical exhaust shape with the half-cone angle of $\alpha = 25°$. Calculate the divergence loss C_A for this nozzle due to nonaxial exhaust flow. Assuming the same (half) divergence angle of 25°, but in a 2D rectangular nozzle, calculate the flow angularity loss and compare it to the conical case.

Problem MA2.6

One method of reducing an aircarft's landing distance is through the use of thrust reversers. Consider the turbofan engine in Fig. P2.5 with thrust reverser of the bypass airstream. It is given that 1500 lbm/s of air at 60°F and 14.7 psia enters the engine at a velocity of 450 ft/s and that 1250 lbm/s of bypass air leaves the engine at 60 deg to the horizontal, velocity of 890 ft/s, and pressure of 14.7 psia. The remaining 250 lbm/s leaves the engine core at a velocity of 1200 ft/s and pressure of 14.7 psia. Determine the force on the strut F_x. Assume the outside of the engine sees a pressure of 14.7 psia.



Camera di combustione

7.4 Write the chemical reaction for the complete combustion of JP-4 and air. JP-4 has the formula $CH_{1.93}$. Also, calculate the stoichiometric fuel-to-air ratio for this blended jet fuel.

7.5 Calculate the lower and higher heating values of octane, C_8H_{18}, in the stoichiometric chemical reaction *with oxygen* at a reference temperature of 298.16 K and the pressure of 1 bar.

$$C_8H_{18(g)} + 12.5O_{2(g)} \rightarrow 8CO_{2(g)} + 9H_2O$$

7.9 One mole of octane is burned with 120% theoretical air. Assuming that the octane and air enter the combustion chamber at 25°C and the excess oxygen and nitrogen in the reaction will not dissociate, calculate

(a) the fuel–air ratio
(b) the equivalence ratio ϕ
(c) the adiabatic flame temperature T_{af}

Assume

$$\bar{c}_{p_{CO_2}} = 61.9\ \text{kJ/kmol}\cdot\text{K},\ \bar{c}_{p_{O_2}} = 37.8\ \text{kJ/kmol}\cdot\text{K}$$
$$\bar{c}_{p_{N_2}} = 33.6\ \text{kJ/kmol}\cdot\text{K},\quad \bar{c}_{p_{H_2O}} = 52.3\ \text{J/kmol}\cdot\text{K}$$

7.10 One mole of oxygen, $O_{2(g)}$, is heated to 4000 K at the pressure of p_m. A fraction of the oxygen dissociates to oxygen atoms according to

$$xO_2 \rightarrow 2xO$$

Assuming a state of equilibrium is reached in the mixture, calculate

(a) mole fraction of O_2 at equilibrium when p_m is 1 atm.
(b) mole fraction of O_2 at equilibrium when p_m is 10 atm.

Assume the equilibrium constant for the reaction

$$O_2 \leftrightarrow 2O$$

is $K_p = 2.19$ atm at the temperature of 4000 K. Explain the effect of pressure on dissociation.

Camera di combustione

Consideriamo ora la reazione:

$$\mathrm{H_2} + \frac{1}{2}\mathrm{O_2} \leftrightarrow n_{\mathrm{H_2O}}\mathrm{H_2O} + n_{\mathrm{H_2}}\mathrm{H_2} + n_{\mathrm{O_2}}\mathrm{O_2} + n_{\mathrm{OH}}\mathrm{OH} + n_{\mathrm{O}}\mathrm{O} + n_{\mathrm{H}}\mathrm{H}$$

per equilibrare questa reazione si procede inizialmente ad equilibrare le singole specie ottenendo:

$$2 = 2n_{\mathrm{H_2O}} + 2n_{\mathrm{H_2}} + n_{\mathrm{OH}} + n_{\mathrm{H}} \qquad 1 = n_{\mathrm{H_2O}} + 2n_{\mathrm{O_2}} + n_{\mathrm{OH}} + n_{\mathrm{O}}$$

```
%        1     2    3     4     5    6
Spe={'H2O','H2','O2','OH','O','H'};
Bil=[   2     2     0     1     0    1  2*1/1.00794;
        1     0     2     1     1    0  1*16/15.9994];

%Bilancio stechiometrico
for i=1:Nb
    R(i)=sum(Bil(i,1:end-1).*n)-Bil(i,end);
end
```


Camera di combustione

Si devono quindi analizzare le sotto reazioni:

- $\frac{1}{2}\mathrm{O_2} \leftrightarrow \mathrm{O} \quad \rightarrow \quad K_{p1} = \frac{p_{\mathrm{O}}}{p_{\mathrm{O_2}}^{1/2}} = \frac{\chi_{\mathrm{O}}}{\chi_{\mathrm{O_2}}^{1/2}} p_m^{1/2}$
- $\frac{1}{2}\mathrm{H_2} \leftrightarrow \mathrm{H} \quad \rightarrow \quad K_{p2} = \frac{p_H}{p_{\mathrm{H_2}}^{1/2}} = \frac{\chi_H}{\chi_{\mathrm{H_2}}^{1/2}} p_m^{1/2}$
- $\frac{1}{2}\mathrm{H_2} + \frac{1}{2}\mathrm{O_2} \leftrightarrow \mathrm{OH} \quad \rightarrow \quad K_{p3} = \frac{p_{OH}}{p_{\mathrm{H_2}}^{1/2} p_{\mathrm{O_2}}^{1/2}} = \frac{\chi_{OH}}{\chi_{\mathrm{H_2}}^{1/2} \chi_{\mathrm{O_2}}^{1/2}}$
- $\mathrm{H_2} + \frac{1}{2}\mathrm{O_2} \leftrightarrow \mathrm{H_2O} \quad \rightarrow \quad K_{p4} = \frac{p_{\mathrm{H_2O}}}{p_{\mathrm{H_2}} p_{\mathrm{O_2}}^{1/2}} = \frac{\chi_{\mathrm{H_2O}}}{\chi_{\mathrm{H_2}} \chi_{\mathrm{O_2}}^{1/2}} p_m^{-1/2}$

```
%        1     2    3     4     5    6
Spe={'H2O','H2','O2','OH','O','H'};
Rea=[   0     0   -0.5    0     1   0;
        0   -0.5    0     0     0   1;
        0   -0.5  -0.5    1     0   0;
        1    -1   -0.5    0     0   0];
for i=1:Nr
    ind=Rea(i,:)~=0;
    R(Nb+i)=K(i)-prod(chi(ind).^Rea(i,ind))*p^sum(Rea(i,ind));
end
```

Camera di combustione

```
function R=Reazione (n,p,K,Rea,Bil)
R=0*n;
n=abs(n);

%Bilancio stechiometrico
Nb=size(Bil,1);
for i=1:Nb
    R(i)=sum(Bil(i,1:end-1).*n)-Bil(i,end);
end

% costanti
chi=n/sum(n);
Nr=size(Rea,1);
for i=1:Nr
    ind=Rea(i,:)~=0;
    R(Nb+i)=K(i)-prod(chi(ind).^Rea(i,ind))*p^sum(Rea(i,ind));
end
```


Camera di combustione

```
%       H2O2
%       1    2    3     4     5    6
Spe={'H2O','H2','O2','OH','O','H'};
Rea=[   0    0   -0.5   0     1   0;
        0  -0.5    0    0     0   1;
        0  -0.5  -0.5   1     0   0;
        1   -1   -0.5   0     0   0];
Bil=[   2    2     0    1     0    1   2*1/1.00794;
        1    0     2    1     1    0   1*16/15.9994];
n=  [   4   .1   0.1    .1   .1  .1];
p=20*1.013;
K=[0.489778819 0.587489353 1.445439771 5.152286446];
options = optimset('Display','off','TolX',1e-18,'TolFun',1e-18);
n2=fsolve(@Reazione, n,options ,p,K,Rea,Bil);
fprintf('csi= ');fprintf('%.3g ',n2/sum(n2));
```

T(K)	$\frac{1}{2}O_2 \leftrightarrow O$	$\frac{1}{2}H_2 \leftrightarrow H$	$\frac{1}{2}H_2 + \frac{1}{2}O_2 \leftrightarrow OH$	$H_2 + \frac{1}{2}O_2 \leftrightarrow H_2O$
3500	-0.310	-0.231	0.160	0.712
4000	0.170	0.201	0.233	0.238

Logaritmo in base 10 delle cosanti d'equilibrio Kp

Camera di combustione

This appendix (ceaThermoBuild) explains the format for data contained in the file thermo.inp (app. D). Equations (1) to (3) are repeated here for convenience:

$$\frac{\bar{c}_p}{\bar{R}} = a_1 T^{-2} + a_2 T^{-1} + a_3 + a_4 T + a_5 T^2 + a_6 T^3 + a_7 T^4$$

$$\frac{\bar{h}}{\bar{R}T} = -a_1 T^{-2} + a_2 \frac{\ln T}{T} + a_3 + a_4 \frac{T}{2} + a_5 \frac{T^2}{3} + a_6 \frac{T^3}{4} + a_7 \frac{T^4}{5} + b_1 \frac{1}{T}$$

$$\frac{\bar{S}}{\bar{R}} = -a_1 \frac{T^{-2}}{2} - a_2 T^{-1} + a_3 \ln T + a_4 T + a_5 \frac{T^2}{2} + a_6 \frac{T^3}{3} + a_7 \frac{T^4}{4} + b_2$$

Camera di combustione

Si dimostra che per una generica reazione chimica:

$$\ln K_p = \sum \left(\frac{\bar{h}}{\bar{R}T} - \frac{\bar{S}}{\bar{R}} \right)^{-r}$$

dove *r* è il coefficiente stechiometrico della reazione.

Per esempio:

- $\frac{1}{2}\mathrm{O}_2 \leftrightarrow \mathrm{O} \qquad \rightarrow \qquad K_{p1} = \frac{p_\mathrm{O}}{p_{\mathrm{O}_2}^{1/2}} = \frac{\chi_\mathrm{O}}{\chi_{\mathrm{O}_2}^{1/2}} p_m^{1/2}$

$$\ln K_{p1} = \left(\frac{\bar{h}}{\bar{R}T} - \frac{\bar{S}}{\bar{R}} \right)_{O_2}^{\frac{1}{2}} + \left(\frac{\bar{h}}{\bar{R}T} - \frac{\bar{S}}{\bar{R}} \right)_{O}^{-1}$$

Camera di combustione

TABLE C1.—FORTRAN FORMAT USED FOR DATA IN APPENDIX D

Record	Contents	FORTRAN format	Columns
1	Species name or formula Comments and data sources	A16 A62	1 to 16 19 to 80
2	Number of T intervals Reference date code Chemical formula—symbols (all capitals) and numbers Zero for gas; nonzero for condensed[a] Molecular weight Heat of formation at 298.15 K, J/mol	I2 A6 5(A2, F6.2) I2 F13.7 F15.5	1 to 2 4 to 9 11 to 50 51 to 52 53 to 65 66 to 80
3	Temperature range Number of coefficients for $C_p^o(T)/R$ (always seven) T exponents in empirical equation for $C_p^o(T)/R$ [always –2, –1, 0, 1, 2, 3, 4; see eq. (1)] $H^o(298.15) - H^o(0)$ J/mol, if available	2F11.3 I1 8F5.1 F15.3	1 to 22 23 24 to 63 66 to 80
4	First five coefficients for $C_p^o(T)/R$, eq. (1)	5D16.9	1 to 80
5	Last two coefficients for $C_p^o(T)/R$, eq. (1) Integration constants b_1 and b_2, eqs. (2) and (3)	2D16.9 2D16.9	1 to 32 49 to 80
-	Repeat 3, 4, and 5 for each interval	---------	----------

Camera di combustione

La funzione (LeggiFileCea.m) risolve il problema della lettura dei file di testo restituendo una struttura con i seguenti campi:

```
NInt: 2
WM: 18.0153
DH0: -241826
TInt: [2×2 double]
Esp: [2×7 double]
DH1: [2×2 double]
a: [2×7 double]
b: [2×2 double]
```

```
12345678901234567890123456789012345678901234567890123456789012345678901234567890
        10        20        30        40        50        60        70        80

TiN(cr)           Chase,1998 pp1612-4.
 2 j 6/68 TI  1.00N   1.00    0.00    0.00    0.00 1     61.87374    -337648.800
    200.000    800.0007 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0           5487.000
-5.479117220D+05 9.328691110D+03-6.386263890D+01 2.429925456D-01-4.304234520D-04
 3.792645100D-07-1.317412256D-10                -8.424256140D+04 3.392988560D+02
    800.000   3220.0007 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0           5487.000
-3.656247060D+05 1.265730431D+03 3.831711190D+00 1.632900455D-03-1.062786626D-07
 1.310931390D-11-5.770548410D-16                -5.027654400D+04-1.652632899D+01
TiN(L)            Chase,1998 pp1612-4.
 1 j 6/68 TI  1.00N   1.00    0.00    0.00    0.00 2     61.87374    -337648.800
   3220.000   6000.0007 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0           5487.000
 0.000000000D+00 0.000000000D+00 7.548249987D+00 0.000000000D+00 0.000000000D+00
 0.000000000D+00 0.000000000D+00                -3.626039860D+04-3.958296649D+01
```

Camera di combustione

GLENN RESEARCH CENTER +Visit Glenn +Visit NASA

Chemical Equilibrium with Applications

+ CHEMICAL EQUILIBRIUM | - RELATED TOPICS

+Home

Related Topics

+ ONLINE CEA!

+ CAP

+ PAC

THERMO BUILD

Written by Patrick Chan (NASA Summer Intern 2001 Duke University sophomore), ThermoBuild is an interactive tool which uses the NASA Glenn thermodynamic database to select species and to obtain:

1. Tables of thermodynamic properties for a user-supplied temperature schedule.
2. Data subsets for use in CEA, SUBEQ or any other computer program.
   To generate a data subset, click here.

Click on symbols for atoms contained in desired compounds.

Camera di combustione

ThermoBuild (page 2)

Jump to bottom of page

Instructions: To generate the desired thermodynamic data, please choose from the following species (Air is provided as a convenience):

Choose	Species Name	Beginning Temperature (K)	End Temperature (K)
☐	H	200.000	20000.000
☐	HO2	200.000	6000.000
☐	H2	200.000	20000.000
☑	H2O	200.000	6000.000
☐	H2O2	200.000	6000.000
☐	O	200.000	20000.000
☐	OH	200.000	20000.000
☐	O2	200.000	20000.000
☐	O3	200.000	6000.000
☐	H2O(cr)	200.000	273.150
☐	H2O(L)	273.150	600.000
☐	Air	200.000	6000.000

Camera di combustione

ThermoBuild

(page 3)

Energy Units:

Joules ☑ Calories ☐

Temperature Schedule (Kelvin)
Each row used must be complete. Intervals may be zero (0.0).

Ranges:

Begin Temp	Intervals	End Temp
298.15	500	4000

Camera di combustione

```
TEMPERATURE SCHEDULE

   298.150     798.150    1298.150    1798.150    2298.150    2798.150
  3298.150    3798.150    4000.000


 NOTE:  Thermodynamic properties calculated for temperatures outside the range of
      the fitted data may have large errors.  This program allows calculations only
      for temperatures within 20% above or below the fitted temperature range.



                     COEFFICIENTS FOR FITTED THERMODYNAMIC FUNCTIONS


    H2O                 Hf:Cox,1989. Woolley,1987. TRC(10/88) tuv25.
     2 g 8/89 H    2.000    1.00     0.00     0.00     0.00 0     18.01528    -241826.000
        200.000   1000.000 7 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0           9904.092
    -3.947960830E+04 5.755731020E+02 9.317826530E-01 7.222712860E-03-7.342557370E-06
     4.955043490E-09-1.336933246E-12 0.000000000E+00-3.303974310E+04 1.724205775E+01
       1000.000   6000.000 7 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0           9904.092
     1.034972096E+06-2.412698562E+03 4.646110780E+00 2.291998307E-03-6.836830480E-07
     9.426468930E-11-4.822380530E-15 0.000000000E+00-1.384286509E+04-7.978148510E+00
```

Camera di combustione

```
   -3.947960830E+04 5.755731020E+02 9.317826530E-01 7.222712860E-03-7.342557370E-06
    4.955043490E-09-1.336933246E-12 0.000000000E+00-3.303974310E+04 1.724205775E+01
      1000.000   6000.000 7 -2.0 -1.0   0.0   1.0   2.0   3.0   4.0   0.0         9904.092
    1.034972096E+06-2.412698562E+03 4.646110780E+00 2.291998307E-03-6.836830480E-07
    9.426468930E-11-4.822380530E-15 0.000000000E+00-1.384286509E+04-7.978148510E+00
```

THERMODYNAMIC FUNCTIONS CALCULATED FROM COEFFICIENTS FOR H2O

T deg-K	Cp J/mol-K	H-H298 kJ/mol	S J/mol-K	-(G-H298)/T J/mol-K	H kJ/mol	delta Hf kJ/mol	log K
0	0.	-9.904	0.	INFINITE	-251.730	-238.922	INFINITE
298.15	33.588	0.000	188.829	188.829	-241.826	-241.826	40.0453
798.15	38.705	17.931	223.732	201.265	-223.895	-246.429	13.3235
1298.15	45.043	38.880	243.985	214.035	-202.946	-249.447	7.0765
1798.15	50.160	62.753	259.496	224.598	-179.073	-250.990	4.2768
2298.15	53.698	88.775	272.247	233.618	-153.051	-251.655	2.6883
2798.15	56.090	116.261	283.061	241.512	-125.565	-251.903	1.6657
3298.15	57.731	144.740	292.421	248.536	-97.086	-252.031	0.9526
3798.15	58.921	173.917	300.656	254.866	-67.909	-252.212	0.4270
4000	59.325	185.852	303.718	257.255	-55.974	-252.323	0.2519

Camera di combustione

```
function [cp_R H_RT S_R]=TermoCea(T,S)
% Con R =8314.5 J/kmole K
% cp=cp_R*R [J/kmole K]
% H*H_RT*R*T [J/kmole]
% S=S_R*R [J/kmole K]
% G=S_R*R*T [J/kmole]

%Verifica intervallo di temperature
if T<S.TInt(1,1) |T>S.TInt(:,:)
   error ('Temperture outside  limits')
end
%Trova l'indice dell'intervallo di temperature
for i=1:S.NInt
   if T<S.TInt(i,2)
      ic=i;
      break;
   end
end
```

Camera di combustione

```
function [cp_R H_RT S_R]=TermoCea(T,S)
...
```

- $\frac{\bar{c}_p}{\bar{R}} = a_1 T^{-2} + a_2 T^{-1} + a_3 + a_4 T + a_5 T^2 + a_6 T^3 + a_7 T^4$

```
cp_R=sum(S.a(ic,:).*T.^S.Esp(ic,:));
```

- $\frac{\bar{h}}{\bar{R}T} = -a_1 T^{-2} + a_2 \frac{\ln T}{T} + a_3 + a_4 \frac{T}{2} + a_5 \frac{T^2}{3} + a_6 \frac{T^3}{4} + a_7 \frac{T^4}{5} + b_1 \frac{1}{T}$

```
dum=S.Esp(ic,:)~=-1;
dum1=S.a(ic,dum)./(S.Esp(ic,dum)+1).*T.^S.Esp(ic,dum);
dum2=S.a(ic,~dum)*log(T)/T;
H_RT=(sum(dum1)+sum(dum2)+S.b(ic,1)/T);
```

- $\frac{\bar{s}}{\bar{R}} = -a_1 \frac{T^{-2}}{2} - a_2 T^{-1} + a_3 \ln T + a_4 T + a_5 \frac{T^2}{2} + a_6 \frac{T^3}{3} + a_7 \frac{T^4}{4} + b_2$

```
dum=S.Esp(ic,:)~=0;
dum1=S.a(ic,dum)./(S.Esp(ic,dum)).*T.^S.Esp(ic,dum);
dum2=S.a(ic,~dum)*log(T);
S_R=(sum(dum1)+sum(dum2)+S.b(ic,2));
```


Camera di combustione

```
function [T2n n2]=ReazioneTP(T2,T1,n,n1,p,Rea,Bil, S)
Tf=298.15;R=8314.5;
NSpe=numel(S);
for i=NSpe:-1:1
   [cp_R(i) H_RT(i) S_R(i)]=TermoCea(T1,S(i));
end
H1=H_RT*R*T1;
for i=NSpe:-1:1
   [cp_R(i) H_RT(i) S_R(i)]=TermoCea(T2,S(i));
end
H2=H_RT*R*T2;
cpm2=(H2-[S(:).DH0]*1000)/(T2-Tf);
K=exp(sum((H_RT-S_R).*-Rea,2));

options = optimset('Display','off','TolX',1e-18,'TolFun',1e-18);
n2=fsolve(@Reazione, n,options ,p ,K ,Rea, Bil);
chi2=n2/sum(n2);
ntot2=sum(n2);
cpmm2=sum(chi2.*cpm2);
T2n=Tf-(sum(n2.*[S(:).DH0]*1000)-sum(n1.*H1))/(ntot2*cpmm2);
```

Camera di combustione

```
% H2O2Cea.m
clear
%        1     2    3    4     5    6
Spe={'H2O','H2','O2','OH','O','H'};
Rea=[   0     0   -0.5   0     1   0;
        0   -0.5    0    0     0   1;
        0   -0.5  -0.5   1     0   0;
        1    -1   -0.5   0     0   0];
Bil=[   2     2     0    1     0    1   2*1/1.00794;
        1     0     2    1     1    0   1*16/15.9994];
n=  [   4    .1   0.1    .1   .1 .1];
n1 =[   0     1    0.5   0    0   0];

T1=298.15;
%T1=350;
T2 =3000;
p=20*1.013;
```


Camera di combustione

```
% H2O2Cea.m
...

NSpe=numel (Spe);

for i=NSpe:-1:1
   S(i)=LeggiFileCea(strcat(Spe{i},'.txt'));
end
% T e p assegnati
[~, n2 ]=ReazioneTP(T2,T1,n,n1,p,Rea,Bil, S);
fprintf('csi= ');fprintf('%.4g ',n2/sum(n2));

% adiabatica p assegnato
FunErr=@(T2) ReazioneTP(T2,T1,n,n1,p,Rea,Bil, S)-T2;
options = optimset('Display','off','TolX',1e-18,'TolFun',1e-18);
T2=fsolve(FunErr,T2,options );
%T2=fzero(FunErr,T2,options );
[T2 n2 ]=ReazioneTP(T2,T1,n,n1,p,Rea,Bil, S);
fprintf('csi= ');fprintf('%.3g ',n2/sum(n2));
fprintf('\nT2=%g \n', T2);
```

Camera di combustione

https://cearun.grc.nasa.gov/

Enter a 4-character alphanumeric code of your own choosing: 5896

Problem type:

○ hp ○ rocket ◉ tp ○ det ○ sp ○ tv ○ uv ○ sv ○ shock

Output:

○ long ○ debug ◉ short

Output:

○ kcal/mole ◉ kJ/mole

Output species:

◉ mole fraction ○ mass fraction

Ions:

◉ no_ions ○ ions

Transport properties:

○ Y ◉ N

Submit Reset

Dove hp -> adiabatica e p assegnata. tp -> T e P assegnate

Camera di combustione

You have chosen problem type = tp

Please type in up to 8 temperatures and 8 pressures:(blank boxes are OK)

3500

◉ K ○ C ○ F ○ R

20

○ MPa ○ kPa ◉ atm ○ bar ○ psia

This is the reactants selection page

Choose your reactants here

Reactant:

○ Air ○ RP-1 ○ Jet-A(L) ○ CH4 ◉ Use Periodic Chart

Submit Reset

Reactant selection page

Click on symbols for atoms contained in desired compounds.

1 H ☑	1 D ☐	13 IIIA 3A	14 IVA 4A	15 VA 5A	16 VIA 6A	17 VIIA 7A	2 He ☐

Choose your reactant(s) (ASCII-betical vertically):

☐H ☑H2 ☐H2(L)

More reactants?

◉Y ○N

Reactant selection page

lick on symbols for atoms contained in desired compounds.

1 D ☐	13 IIIA 3A	14 IVA 4A	15 VA 5A	16 VIA 6A	17 VIIA 7A	2 He ☐
4 Be ☐	5 B ☐	6 C ☐	7 N ☐	8 O ☑	9 F ☐	10 Ne ☐

You have designated 2 molecules in your reaction mix.

Please indicate relative amounts (mass):

O2 16

H2 2

Invia richiesta Reimposta

Check if you wish to use special CEA options:

(All boxes empty for none):

☐omit ☑only ☐insert

Your choices for 'ONLY' are:

☑H	☑H2O	☐H2O2	☑O	☐O3
☑H2	☐H2O(L)	☐H2O2(L)	☑O2	☐O3(L)
☐H2(L)	☐H2O(cr)	☐HO2	☐O2(L)	☑OH

Submit Reset

Camera di combustione

Input data

```
prob case=58965686 tp
   p(atm) = 20
   t,k=  3500
 reac
   name O2               wt%= 88.89
   name H2               wt%= 11.11
 output trace=1e-5
 only H  H2O  O  H2  O2  OH
 end
```


Camera di combustione

Reazione a *T*=3500*K*

```
REACTANT   WT.FRAC    (ENERGY/R),K    TEMP,K  DENSITY      EXPLODED FORMULA
 N: O2     0.888900    0.000000E+00      0.00   0.0000       O   2.00000
 N: H2     0.111100    0.000000E+00      0.00   0.0000       H   2.00000
```

```
THERMODYNAMIC PROPERTIES                  Cp, KJ/(KG)(K)   12.9454
                                          GAMMAs            1.1254
 P, BAR               20.265              SON VEL,M/SEC     1453.0
 T, K                3500.00
 RHO, KG/CU M     1.0802 0                MOLE FRACTIONS
 H, KJ/KG             109.77
 U, KJ/KG           -1766.33              *H               4.7764-2
 G, KJ/KG           -57863.7              *H2              1.3220-1
 S, KJ/(KG)(K)       16.5638              H2O              6.4164-1
                                          *O               2.2364-2
 M, (1/n)             15.511              *OH              1.1429-1
 (dLV/dLP)t         -1.05824              *O2              4.1743-2
 (dLV/dLT)p           2.0240
```

Camera di combustione

Input data

```
prob case=44623075  hp p(atm)=20
 phi=1
 reac
  fuel  H2              wt%= 100.0 t,k= 298.15
  oxid  O2              wt%= 100.0 t,k=  298.15
   output short
 output trace= 1e-5
 only H  H2O  O  H2  O2  OH
 end
```


Camera di combustione

Reazione adiabatica

```
 REACTANT                  WT FRACTION      ENERGY      TEMP
                           (SEE NOTE)     KJ/KG-MOL      K
 FUEL         H2            1.0000000         0.000    298.150
 OXIDANT      O2            1.0000000         0.000    298.150
O/F=     7.93668   %FUEL= 11.189834   R,EQ.RATIO= 1.000000   PHI,EQ.RATIO=
1.000000


THERMODYNAMIC PROPERTIES                     Cp, KJ/(KG)(K)    12.8800
                                             GAMMAs             1.1253
 P, BAR              20.265                  SON VEL,M/SEC     1452.8
 T, K               3491.86
 RHO, KG/CU M     1.0804 0                   MOLE FRACTIONS
 H, KJ/KG           0.00000
 U, KJ/KG         -1875.66                   *H               4.7123-2
 G, KJ/KG         -57941.0                   *H2              1.3352-1
 S, KJ/(KG)(K)     16.5932                   H2O              6.4615-1
                                             *O               2.1406-2
 M, (1/n)           15.479                   *OH              1.1194-1
 (dLV/dLP)t       -1.05735                   *O2              3.9854-2
 (dLV/dLT)p         2.0110
```

Compressori MA9.12

Air enters a compressor stage that has the following properties:

$$\dot{m} = 50\ \text{kg/s}, \quad \omega = 800\ \text{rad/s}, \quad r = 0.5\ \text{m}$$

$$M_1 = M_3 = 0.5, \quad \alpha_1 = \alpha_3 = 40\ \text{deg}, \quad T_{t1} = 290\ \text{K}$$

$$P_{t1} = 101.3\ \text{kPa}, \quad u_2/u_1 = 1.0, \quad T_{t3} - T_{t1} = 45\ \text{K}$$

$$\phi_{cr} = 0.10, \quad \phi_{cs} = 0.03, \quad \sigma = 1$$

Note: For air, use $\gamma = 1.4$ and $R = 0.286\ \text{kJ/(kg}\cdot\text{K)}$. Make and fill out a table of flow properties like Table 9.3, and determine the diffusion factors, degree of reaction, stage efficiency, polytropic efficiency, and flow areas and associated hub and tip radii at stations 1, 2, and 3.

Turbine MA9.35

Products of combustion enter a turbine stage with the following properties:

$$\dot{m} = 40\,\text{kg/s}, \quad T_{t1} = 1780\,\text{K}, \quad P_{t1} = 1.40\ \text{MPa}, \quad M_1 = 0.3$$

$$M_2 = 1.15, \quad \omega r = 400\ \text{m/s}, \quad T_{t3} = 1550\,\text{K}, \quad \alpha_1 = \alpha_3 = 0$$

$$r_m = 0.4\ \text{m}, \quad u_3/u_2 = 1.0, \quad \phi_{t\,\text{stator}} = 0.04, \quad \phi_{t\,\text{rotor}} = 0.08$$

Note: For the gas, use $\gamma = 1.3$ and $R = 0.287\ \text{kJ/(kg}\cdot\text{K)}$. Make and fill out a table of flow properties like Table 9.12 for the mean line, and determine the degree of reaction, total temperature change, stage efficiency, polytropic efficiency, and flow areas and associated hub and tip radii at stations 1, 2, and 3.

MA Example 3.3

Consider both a two-stage vehicle and a three-stage vehicle for the launch of the 900-kg (2000-lbm) payload. Each stage uses a liquid H_2 -O_2 chemical rocket (C ¼ 4115 m/s, 13,500 ft/s), and the DV total of 14,300 m/s (46,900 ft/s) is split evenly between the stages. Suppose $\delta = m_{str}/m_0 = 0.03$.

Mattingly example 3.7

The space shuttle main engine (SSME) operates for up to 520 s in one mission at altitudes over 100 miles. The nozzle expansion ratio 1 is 77:1, and the inside exit diameter is 2.30m. Assume a calorically perfect. We want to calculate the following:

- Characteristic velocity c^*
- Mass flow rate of gases through the nozzle.
- Pressure at which the nozzle is “on-design.”
- Pressure at which the nozzle is just separated (assume separation occurs when p_a>3.5 p_{r3}).
- Thrust coefficient C_F, specific impulse and thrust for p_0=0, p_s.

Ae/Ag	77	
De	2.298	m
g	1.25	
pt	2.068E+07	Pa
Tt	4083	K
R	602.6	J/kgK
M2	5.092	

Farokhi problem 12.2

A solid rocket motor has a design chamber pressure of 10 MPa, an end-burning grain with n = 0.4 and $\dot{r}$ = 3cm/s at the design chamber pressure and design grain temperature of 15C. The temperature sensitivity of the burning rate is σ_p= 0.002⁄C, and chamber pressure sensitivity to initial grain temperature is π_K=0.005⁄C. The nominal effective burn time for the rocket is 120 s, i.e., at design conditions. Calculate:

- the new chamber pressure and burning rate when the initial grain temperature is 75C;
- the corresponding reduction in burn time Δt_b in seconds.

Farokhi problem 12.20

Consider a scramjet in a Mach 6 flight. The fuel for this engine is hydrogen with Q_R=120,000kJ⁄kg. The inlet uses multiple oblique shocks with a total pressure recovery following MIL-E-5008B standards for M0 > 5, i.e.,

$$\pi_d = \frac{800}{M_0^4 + 935}$$

The combustor entrance Mach number is M2=2.6. Use frictionless, constant-pressure heating, i.e., p4= p2, to simulate the combustor with combustor exit Mach number M4= 1.0. All component parameters and gas constants are shown in the schematic drawing below. Calculate:

Razzi

Farokhi problem 12.20

(a) Inlet static temperature ratio T2/T0

(b) combustor exit temperature T4 in K

(c) fuel-to-air ratio f

(d) nozzle exit Mach number M10

(e) nondimensional ram drag Dram/p0A1 (note that A0= A1)

(f) nondimensional gross thrust Fg⁄p0A1

(g) fuel-specific impulse Is in seconds

(h) combustor area ratio A4/A2

(i) nozzle area ratio A10/A4

(j) thermal efficiency

(k) propulsive efficiency