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

Propulsione Aerospaziale

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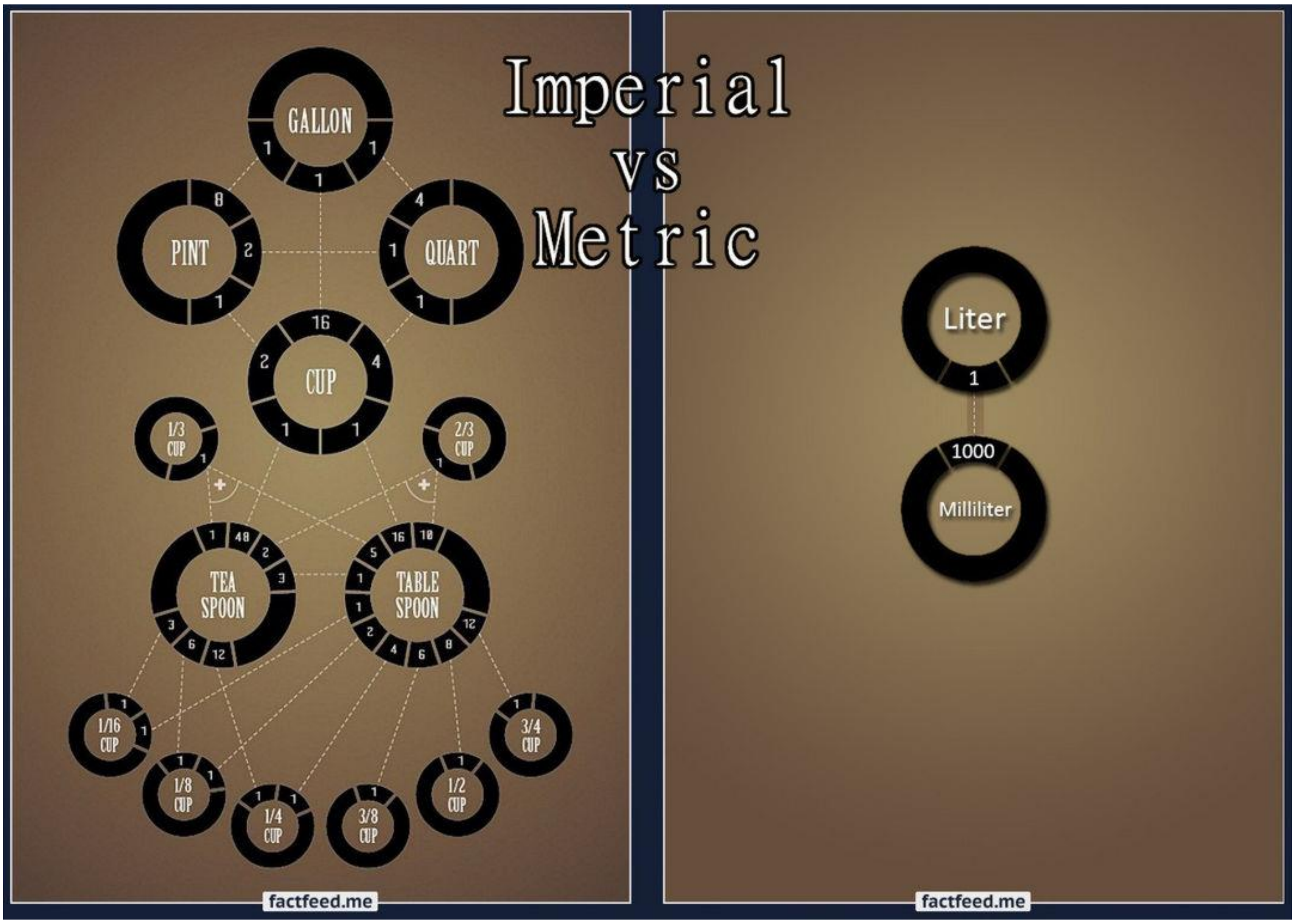
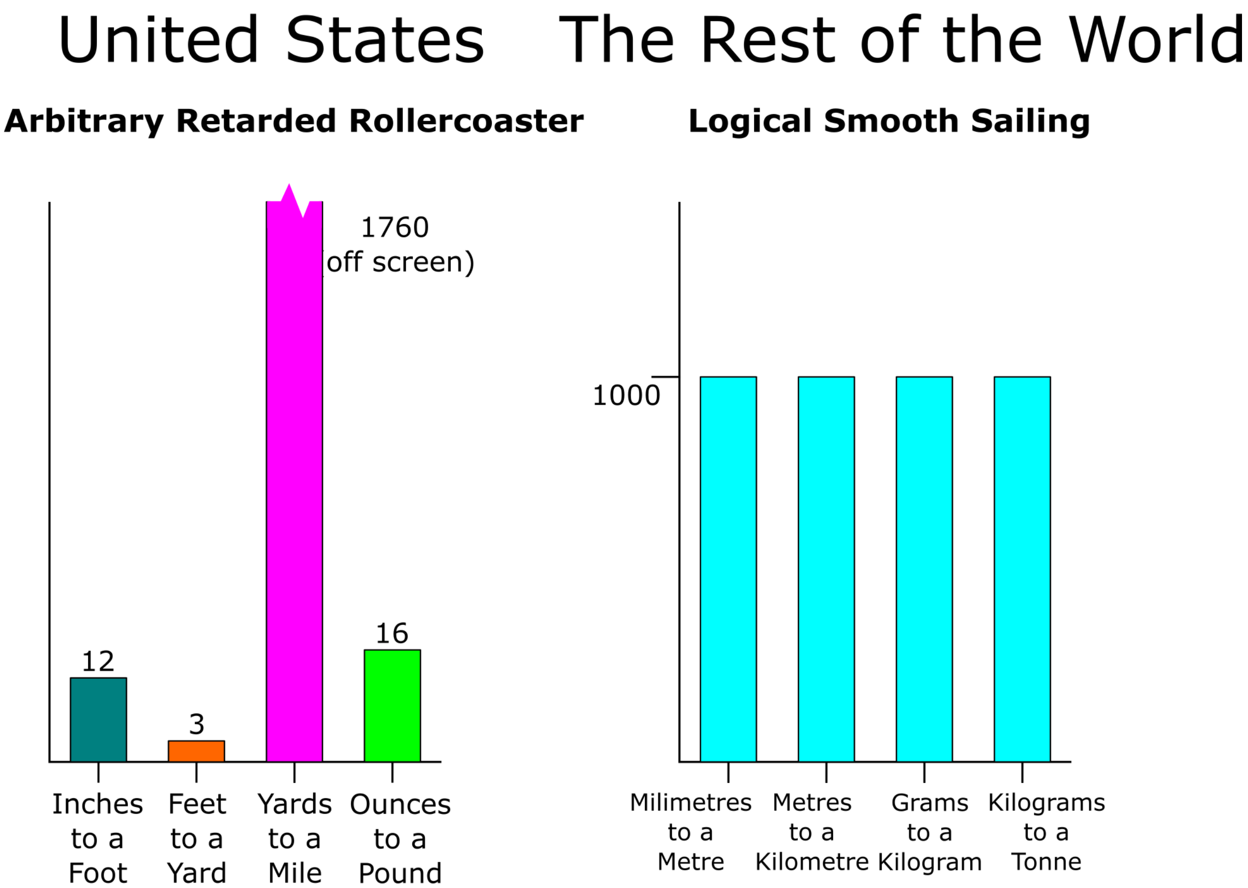
Cenni sul sistema tecnico Americano



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**

Unità	Divisioni	Equivalente SI
Volume liquido		
Unità di misura molto comuni in *corsivo* Conversione esatta in **grassetto**		
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 **$\frac{1}{2}$ hhd**	**119,240 471 196 L**
1 barile di petrolio (oil barrel, bo[16])	**42 gal** o **$\frac{2}{3}$ 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

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

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

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

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

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

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

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

$$xF = (x - 32)/1.8C = [(x - 32)/1.8 + 273.15]K$$

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 potere calorifico del combustibile (fuel heating value) sono:

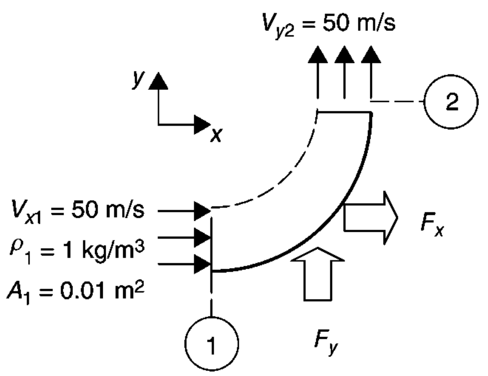
$$[Q_R] = \frac{BTU}{lbm} = \frac{1055.1J}{lbm} = \frac{1055.1kg \cdot m^2}{lbm \cdot s^2} = \frac{\frac{1055.1}{0.3048^2}ft^2}{.45359 \cdot s^2} = 25038\frac{ft^2}{s^2}$$

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

Problem Fa2.34

2.32 Consider the flow of perfect gas (air) on a vane, as shown. Apply momentum principles to the fluid to calculate the components of force F_x and F_y that act on the vane.

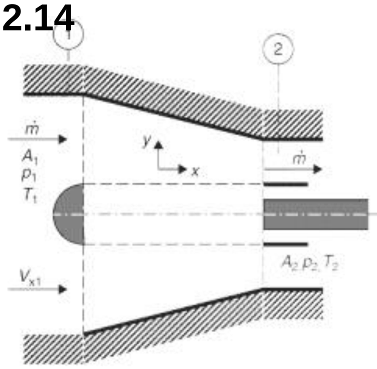
Assume the flow is uniform, and due to low speeds, $p_1 = p_2 = p_0$, where p_0 is the ambient pressure.



Farokhi Example 2.14

$$\pi_c = 20, \gamma = 1.4,$$

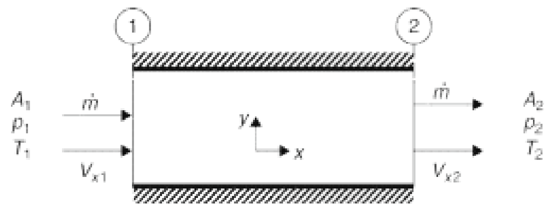
$$V_1 = V_2, \frac{F_x}{p_1 A_1} = ?$$



Farokhi Example 2.15

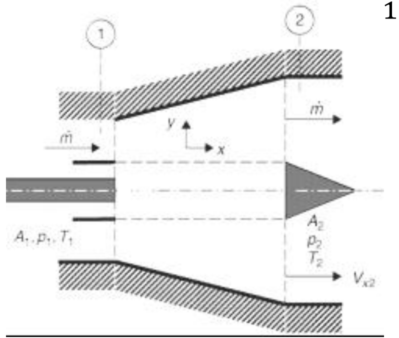
$$\tau_b = 1.8, \gamma = 1.4,$$

$$p_1 = p_2, \frac{F_x}{\dot{m} V_1} = ?$$

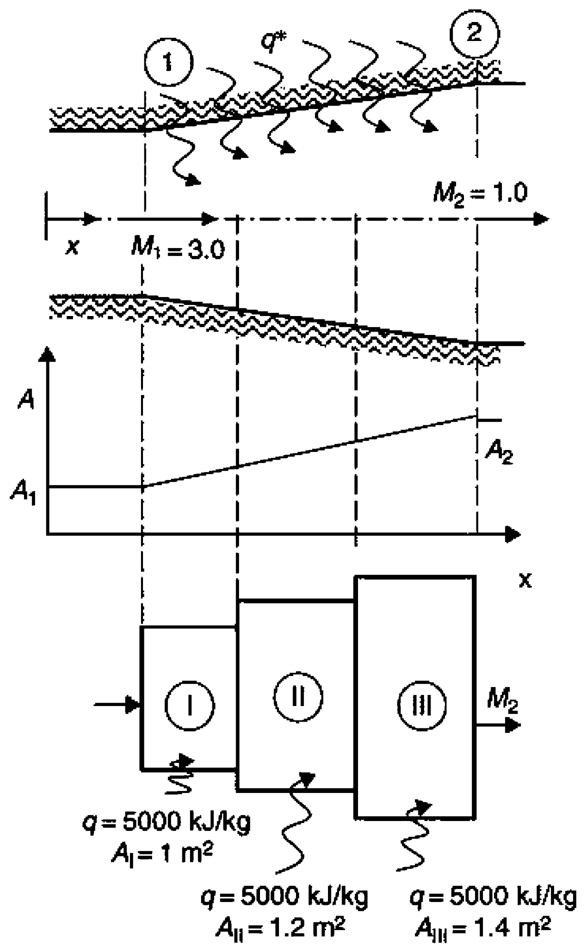


Farokhi Example 2.16

$$\tau_t = 0.79, \gamma = 1.4, V_1 = V_2, \quad \frac{F_x}{p_1 A_1} = ?$$



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:

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

Problem MA 4.1

Find the thrust of the SSME for $p_0 = 14.7 psi$ at 44,600ft and in a vacuum.

		Takeoff				Cruise				
Model no.	Manufacturer	Thrust, lbf	BR[a]	OPR[b]	Airflow, lbm/s	Alt, kft	Mach	Thrust, lbf	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).

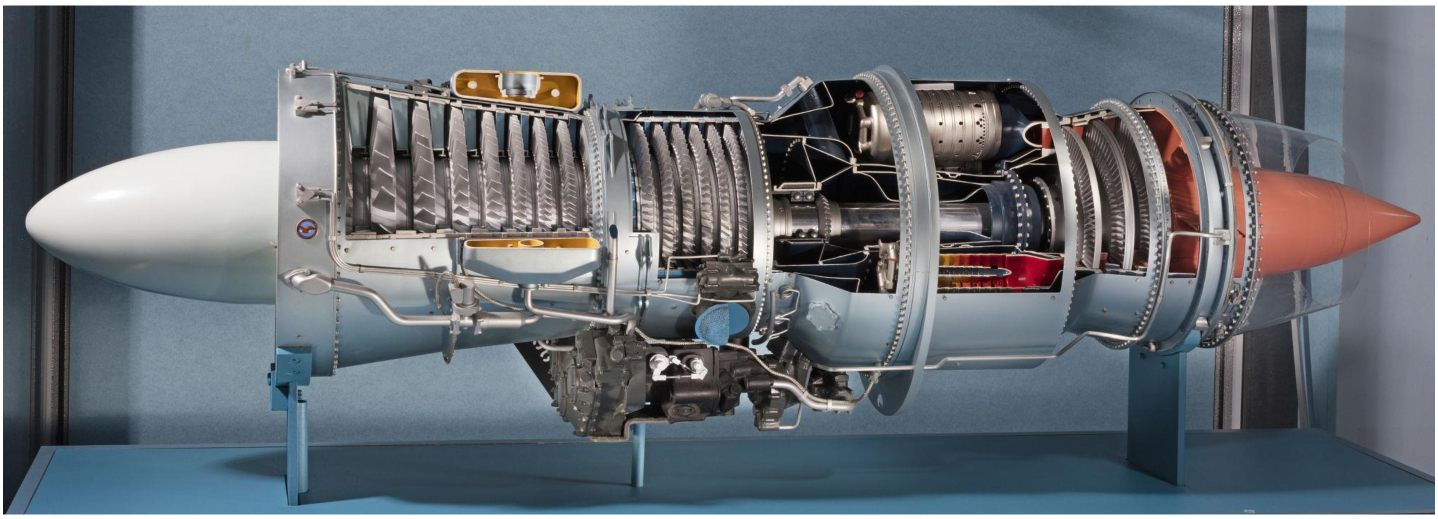
Some data

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 exhaus
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

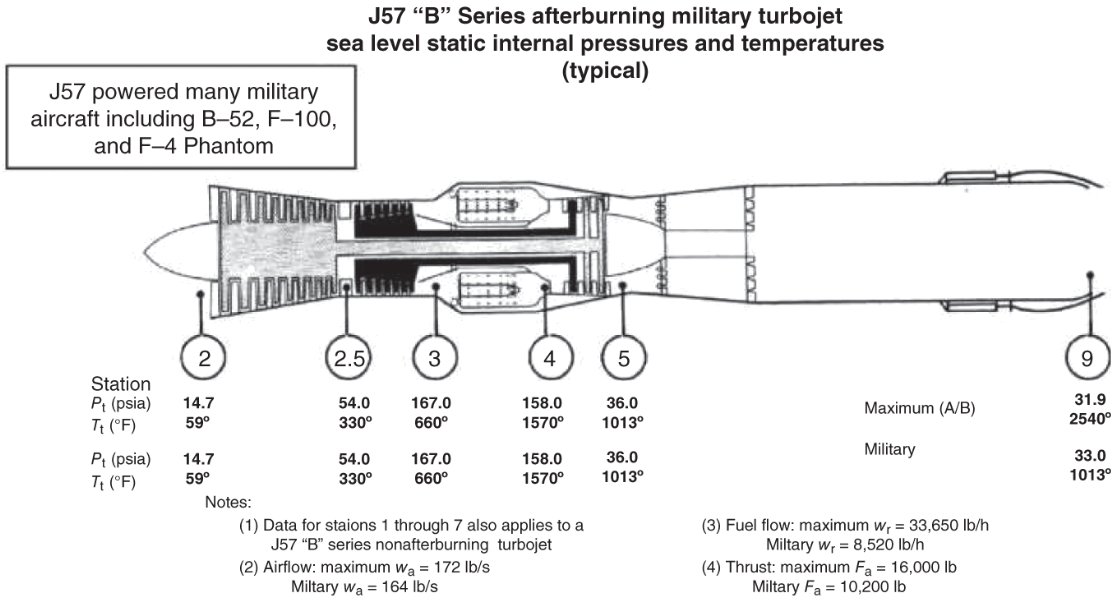
3.1 The total pressures and temperatures of the gas in an afterburning turbojet engine are shown (J57 "B" from Pratt & Whitney, 1988). The mass flow rates for the air and fuel are also indicated at two engine settings, the Maximum Power and the Military Power. Use the numbers specified in this engine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



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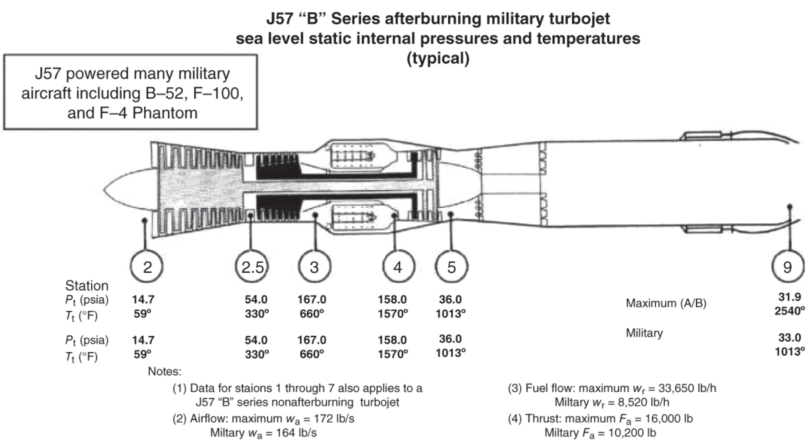
(d) the thermal efficiency of this engine for both power settings (at the sea level static operation),
assuming the fuel heating value is 18,600 BTU/lbm and c_p = 0.24 BTU/lbm · °R. Explain the lower thermal efficiency of the Maximum power setting

(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 settings

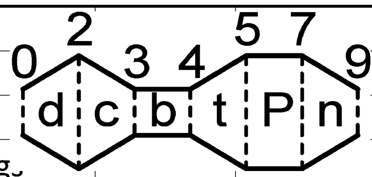
(g) the comparison of percent thrust increase to percent fuel flow rate increase when we turn the afterburner on

(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		

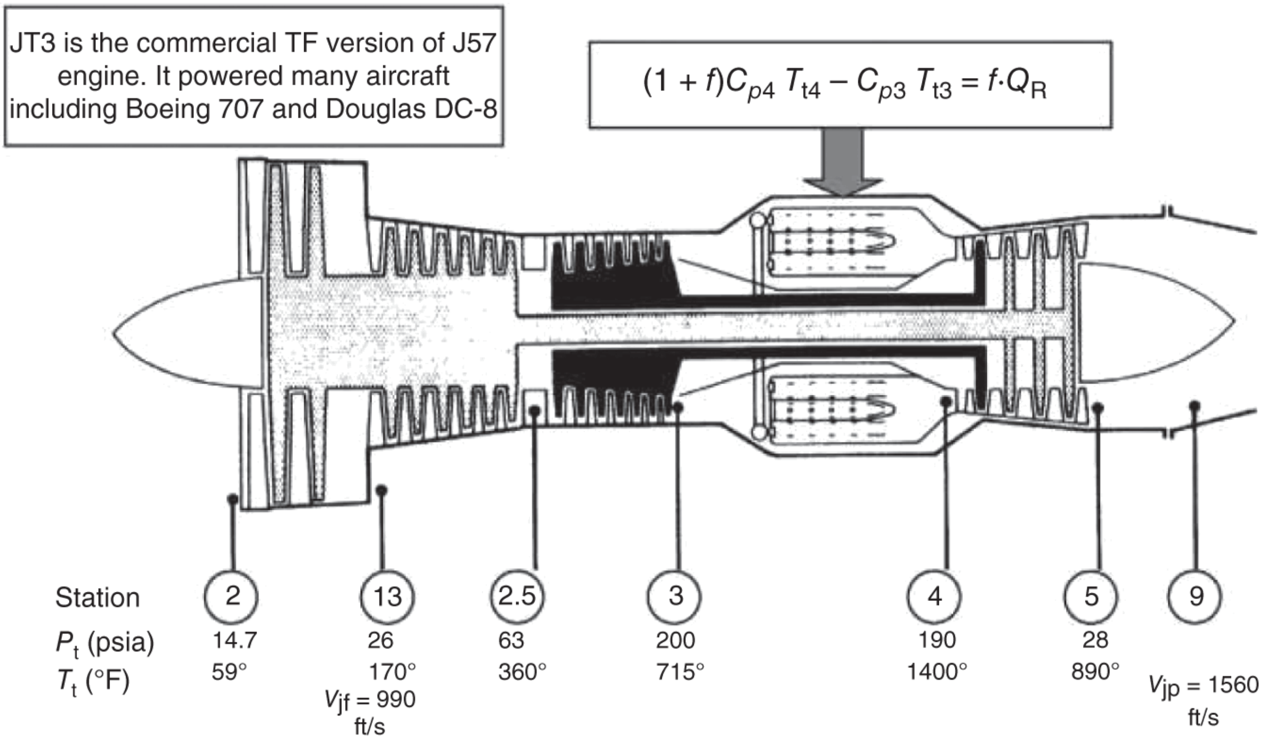


Problem 3.2

JT3D-3B è stato usato su: B707 e DC8.

Problem 3.2

JT3D-3B Turbofan
Internal pressures and temperatures



At sea level static takeoff thrust of 18000 lbs,
W_{af} = 265 lbs/s W_{ab} = 195 lbs/s

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

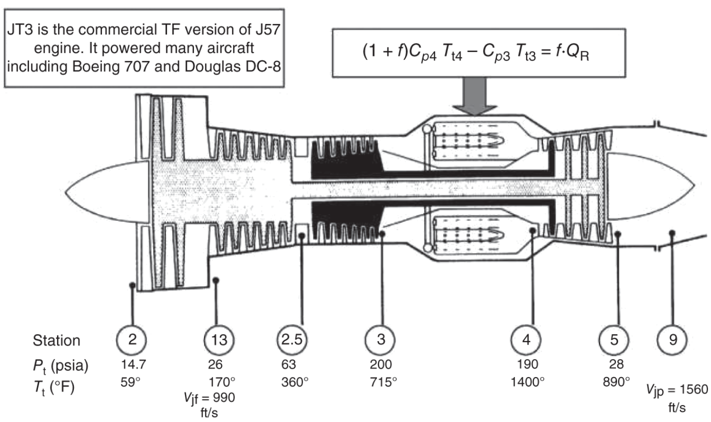
(g) the nondimensional engine specific thrust

(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 $a = (\gamma RT)^{1/2}$]

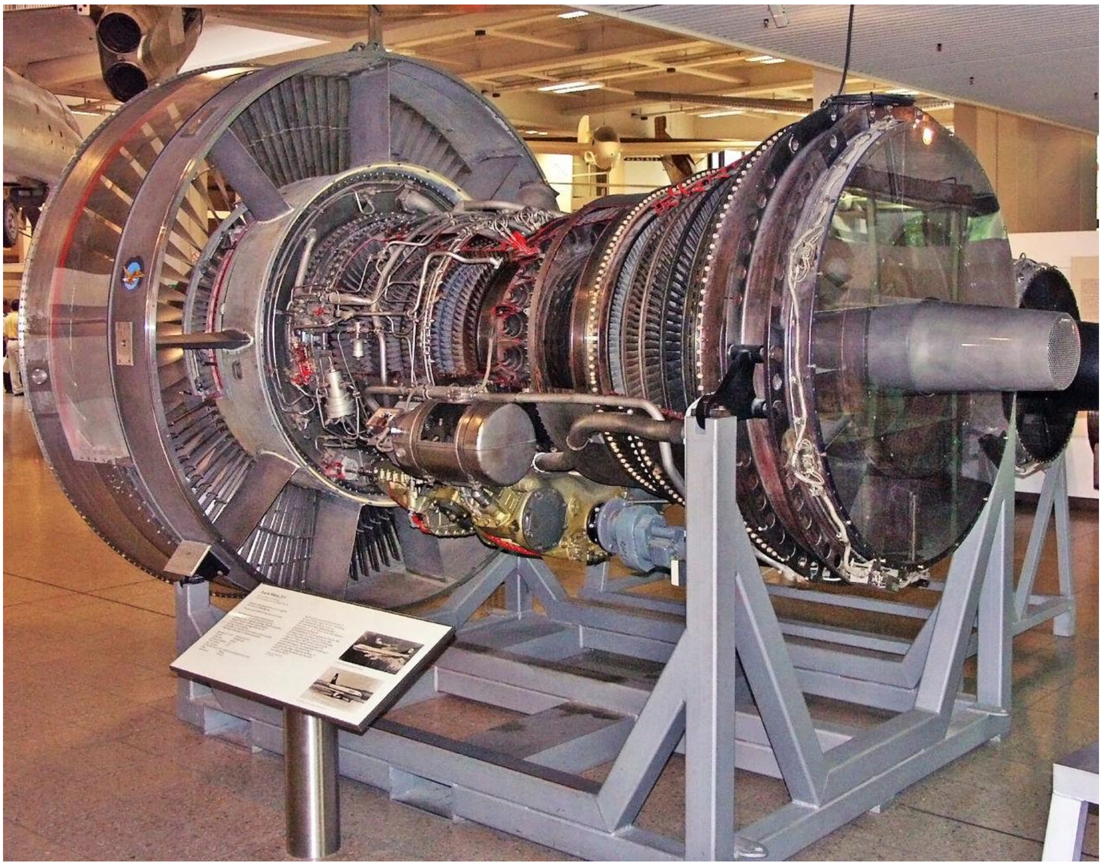
JT3D-3B Turbofan
Internal pressures and temperatures



At sea level static takeoff thrust of 18000 lbs,
W_{af} = 265 lbs/s W_{ab} = 195 lbs/s

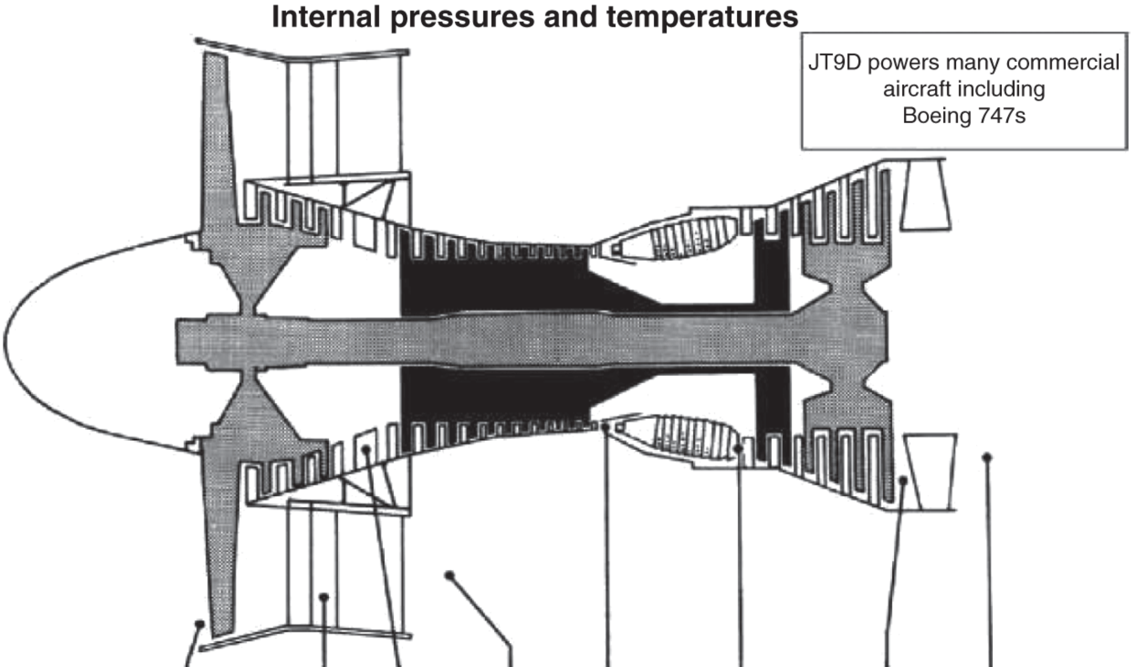
Problem 3.4

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



Problem 3.4

JT9D Turbofan
Internal pressures and temperatures



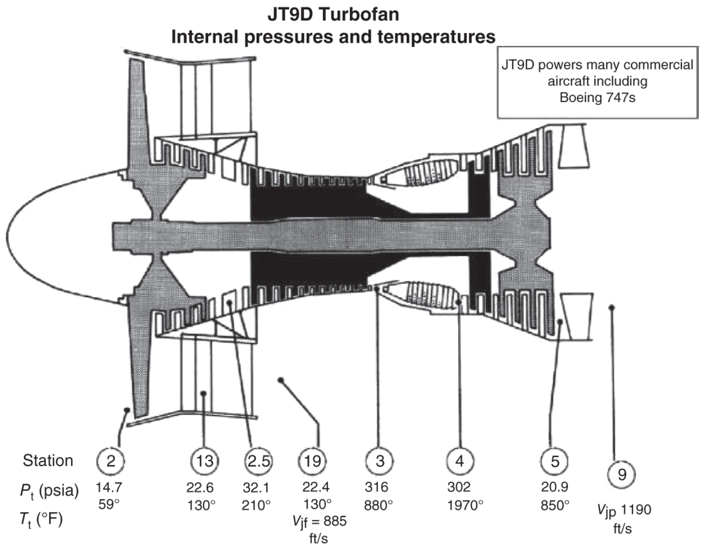
Station	2	13	2.5	19	3	4	5	9
P_t (psia)	14.7	22.6	32.1	22.4	316	302	20.9	
T_t (°F)	59°	130°	210°	130°	880°	1970°	850°	
				V_{jf} = 885 ft/s				V_{jp} 1190 ft/s

At sea level static takeoff thrust of 43500 lbs, W_{af} = 1248 lbs/s, W_{ab} = 247 lbs/s

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



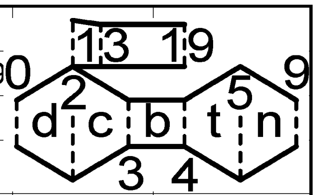
At sea level static takeoff thrust of 43500 lbs, W_{af} = 1248 lbs/s, W_{ab} = 247 lbs/s

Problem 3.4

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

TurboFan

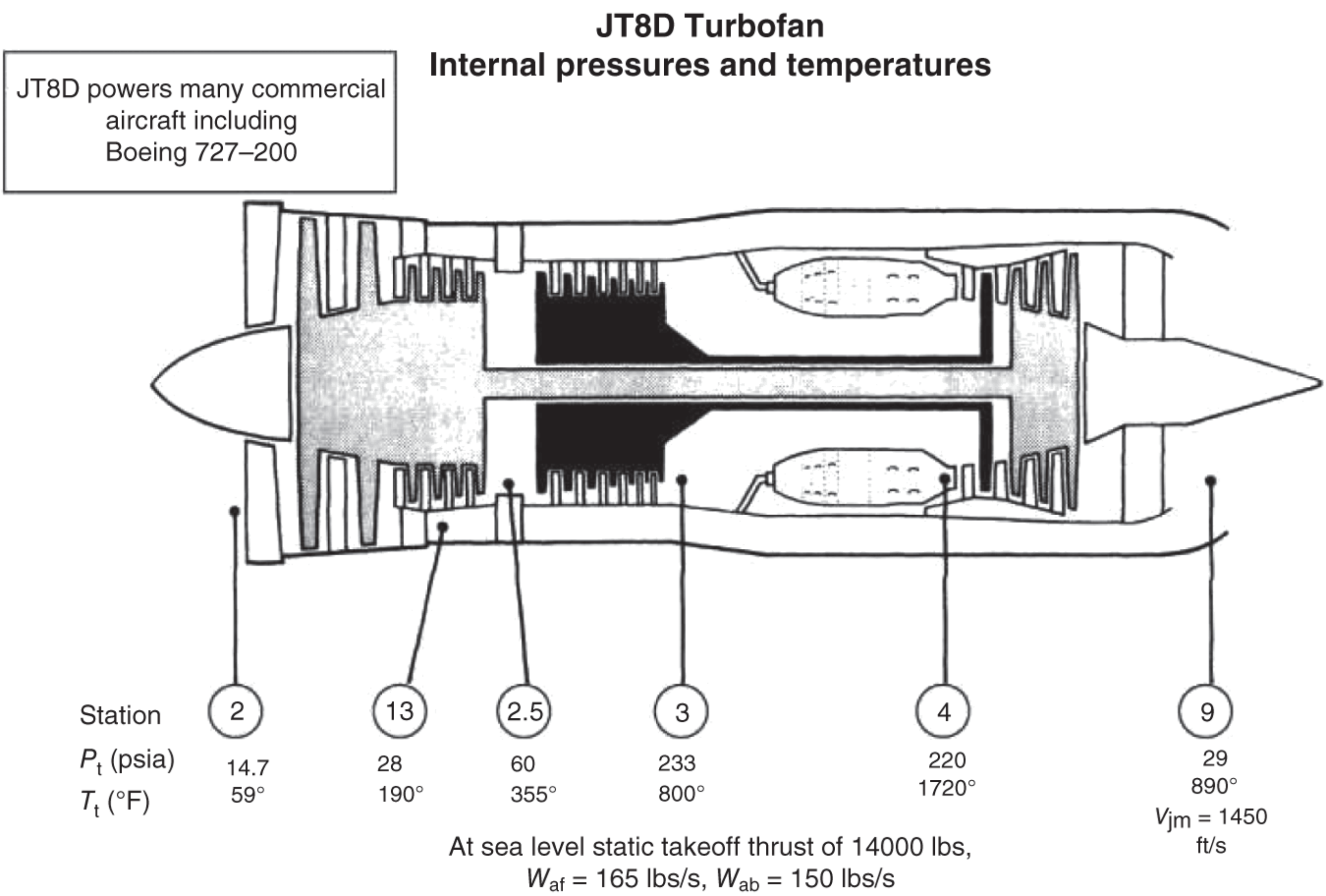
JT9d By Tom							
	2	3	4	5	9	13	19
	diff	comp	CC	Tur	No	Fan	No FAn
c_p	1004			1057			
γ	1.4			1.35			
π	1	21.5	0.955		0.98	1.53	0.99
$\eta, e_{c,t}$		0.92	0.95	0.9		0.96	
Tt				1349.8			
M0	0				QR	42800	kJ/kgK
T0	288	K	p0	101,300	Pa	η_m	0.98
alpha	5.053						
k	0.28571			0.25926			
R	286.857			274.037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



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}

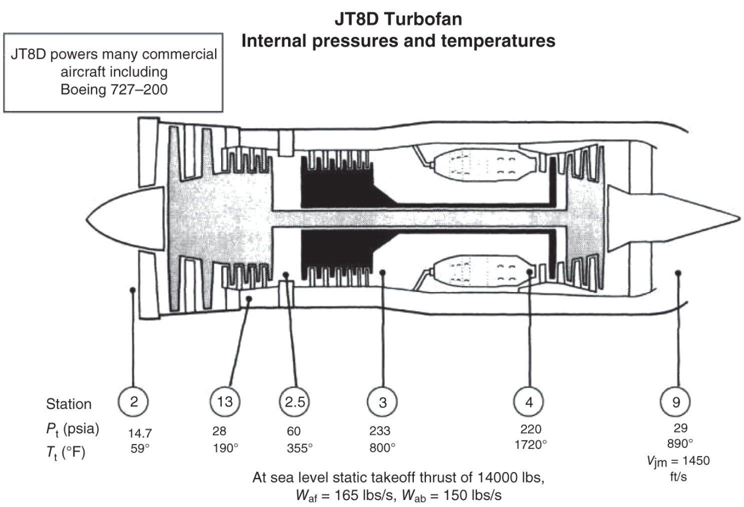
(f) What is the bypass ratio α for this engine at takeoff

(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 $p_{t3}/p_{t2.5}$

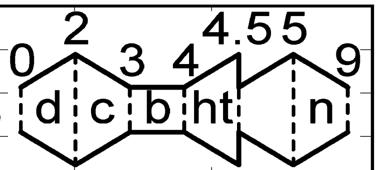


TurboFan a flussi miscelati

Example 4.16

	2	3	4	5	13	15	6	7	9
	diff	comp	CC	Tur	Fan		Mixer		No
c_p	1004			1152				1241	
γ	1.4			1.33				1.3	
π	0.9	13	0.95		1.9	0.99	0.9709	0.92	0.95
$\eta, e_{c,t}$		0.9	0.98	0.8	0.9			0.98	
Tt			1600					2000	
M0	2.0000				QR	42000	kJ/kgK		
T0	223.0	K	p0	10000.0	Pa	η_m	0.95	p9/p0	3.8
k	0.2857			0.2481				0.2308	
R	286.9			285.8	kJ/kgK			286.4	kJ/kgK
a0	299.3	m/s	V0	598.5	m/s	$\tau_r = (\psi)$	1.8		

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				

Problem MA4.3

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

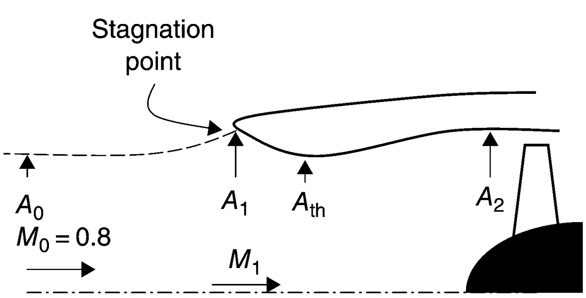
$$\frac{D_{add}}{p_0 A_1} = \gamma M_1 \left(\frac{\psi_0}{\psi_1}\right)^K \left(M_1 \sqrt{\frac{\psi_0}{\psi_1}} - M_0\right) + \left(\frac{\psi_0}{\psi_1}\right)^{\frac{\gamma}{\gamma-1}} - 1$$

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.

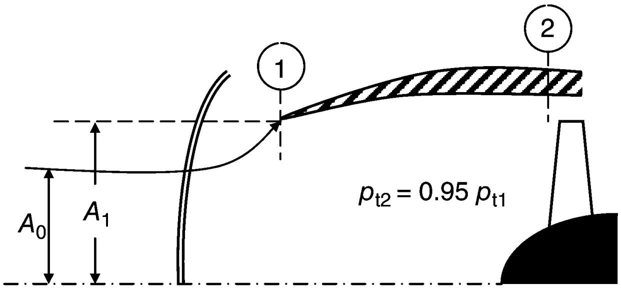
6.3 Consider a subsonic inlet at a flight cruise Mach number of 0.8. The captured streamtube undergoes a prediffusion external to the inlet lip, with an area ratio $A_0/A_1 = 0.92$, as shown. Calculate

- **(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**

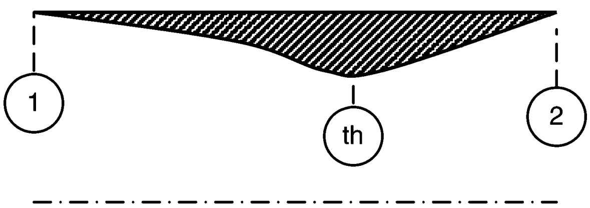
Prese d'aria ed ugelli

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.



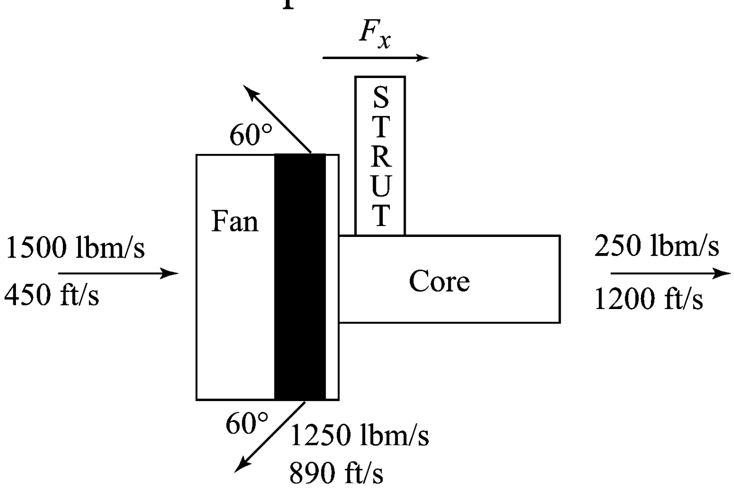
6.43 A convergent nozzle experiences π_{cn} of 0.98, the gas ratio of specific heats $\gamma = 1.30$, and the gas constant is $R = 291$ J/kg · K. First, calculate the minimum nozzle pressure ratio that will choke the expanding nozzle, i.e., NPR_{crit}. This nozzle operates, however, at a higher NPR than 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.



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

$$\bar{c}_{p\mathrm{CO}_2} = 61.9\ \mathrm{kJ/kmol\cdot K},\ \bar{c}_{p\mathrm{O}_2} = 37.8\ \mathrm{kJ/kmol\cdot K}$$
$$\bar{c}_{p\mathrm{N}_2} = 33.6\ \mathrm{kJ/kmol\cdot K},\quad \bar{c}_{p\mathrm{H_2O}} = 52.3\ \mathrm{J/kmol\cdot K}$$

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.

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
```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```
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
```

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```
%         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

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This appendix (cearun.grc.nasa.gov/ThermoBuild/) explains the format for data contained in the file thermo.inp (app. D). Equations (1) to (3) are repeated here for convenience:

$$\frac{\bar{c}_p}{\bar{R}} = a_1T^{-2} + a_2T^{-1} + a_3 + a_4T + a_5T^2 + a_6T^3 + a_7T^4$$

$$\frac{\bar{h}}{\bar{R}T} = -a_1T^{-2} + a_2\frac{\ln T}{T} + a_3 + a_4\frac{T}{2} + a_5\frac{T^2}{3} + a_6\frac{T^3}{4} + a_7\frac{T^4}{5} + b_1\frac{1}{T}$$

$$\frac{\bar{S}}{\bar{R}} = -a_1\frac{T^{-2}}{2} - a_2T^{-1} + a_3\ln T + a_4T + a_5\frac{T^2}{2} + a_6\frac{T^3}{3} + a_7\frac{T^4}{4} + b_2$$

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:

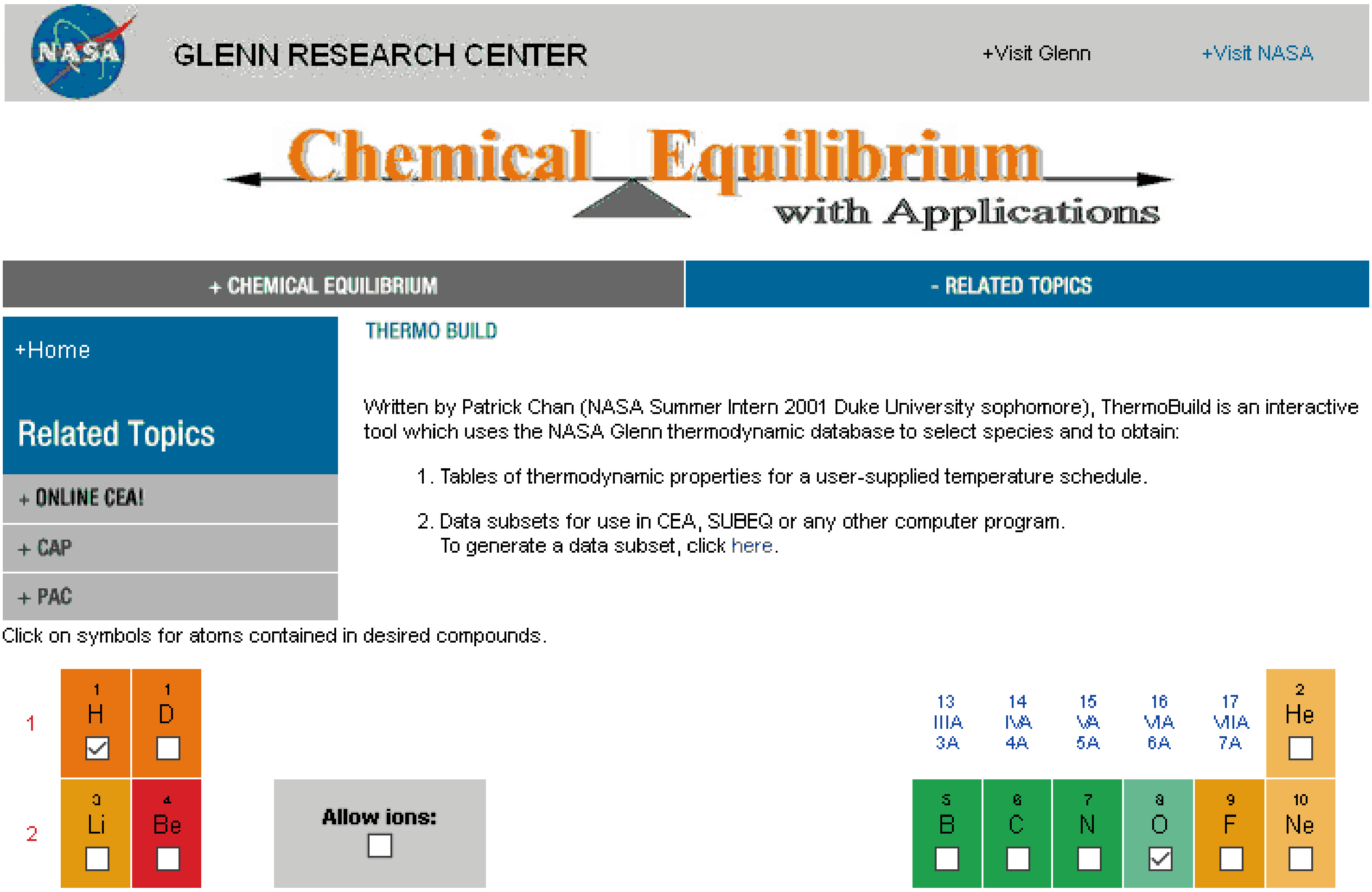
```
NInt: 2
WM: 61.87374
DH0: -337648.8
TInt: [2×2 double]
Esp: [2×7 double]
DH1: [2×2 double]
a: [2×7 double]
b: [2×2 double]
```

$\Delta \bar{h}^0_{fi}$

```
     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



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

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
```

$\Delta\bar{h}^0_{fi}$

```
    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
```

$\Delta\bar{h}^0_{fi}$

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

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

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);
```

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

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/

Select problem and code and click on the 'Submit' button:

Chemical Equilibrium Problem Types

	Type Code	Description
○	rocket	Rocket
○	hp	Assigned Enthalpy & Pressure
◉	tp	Assigned Temperature & Pressure
○	det	Chapman-Jouguet Detonation
○	shock	Shock Tube
○	tv	Assigned Temperature & Density
○	uv	Combustion at Assigned Density
○	sp	Assigned Entropy & Pressure
○	sv	Assigned Entropy & Density

Enter an alphanumeric code: 123

The code is optional and is used to identify your data. Use no more than *15 characters*. *Blank spaces, hyphens and underscores are allowed, but no special characters ($, #, etc).*

Submit

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

Camera di combustione

ASSIGNED TEMPERATURE AND PRESSURE PROBLEM (TP)

Problem Type | **Temperatures&Pressures** | Fuel(s) | Oxidizer(s) | Oxid/Fuel | Final

Enter Temperature and Pressure Values

Use either Low-High-Interval fields (on left side) or numbered fields (on right side)--*Not both.*

Temperature

Enter Low/High/Interval values for no more than **24** datapoints.

Low Value:

High Value:

Interval:

Clear Low/High/Int. Fields

1. 3500   2.   3.   4.   5.   6.   7.   8.   9.   10.   11.   12.   13.   14.   15.   16.   17.   18.   19.   20.   21.   22.   23.   24.

Clear Numbered Fields

Select one: (•) K ( ) C ( ) F ( ) R

*Ref: NASA RP1311 Part II (Users Manual), Section 2.4.10

Pressure

Enter Low/High/Interval values for no more than **24** datapoints.

Low Value:

High Value:

Interval:

Clear Low/High/Int. Fields

1. 20   2.   3.   4.   5.   6.   7.   8.   9.   10.   11.   12.   13.   14.   15.   16.   17.   18.   19.   20.   21.   22.   23.   24.

Clear Numbered Fields

Select one: (•) atm ( ) bar ( ) mmHg ( ) psia

*Ref: NASA RP1311 Part II (Users Manual), p. 13, Section 2.4.8

Accept Input & Continue to Next Form.   Reset

Camera di combustione

Select one of the following:

Select one of the following compounds for *simple* (1-component) fuels, or select a mixture using the periodic table:

( ) CH4 ( ) CH4(L) (•) H2 ( ) H2(L) ( ) RP-1 ( ) Use Periodic Table (mixtures)

- The species listed above are assumed to be **pure**. If your reactant is not shown here, or you need to blend one or more compounds, use the Periodic Table.
- **Please note that any fuel combinations using the Periodic Table will cancel out a current simple-fuel selection.**
- Be careful to select the appropriate compounds. Some compounds are represented in the CEA database in more than one form. For example, H2 refers to gas while H2(L) is liquid.
- To specify reactants without distinguishing between 'fuels' and 'oxidants', select **'None'** from the Oxidizer Selection page, and CEA will be instructed to skip the Oxidizer Selection Form.

Enter Reactant Temperature(K), if needed: 298.15

Please specify how to define reactant mixtures ***(both Fuels & Oxidizers)***: (•) wt% ( ) mole

Select one of the following:

Select one of the following compounds for *simple* (1-component) oxidizers, or select a mixture using the periodic table:

( ) Air ( ) CL2 ( ) CL2(L) ( ) F2 ( ) F2(L) ( ) H2O2(L) ( ) N2H4(L) ( ) N2O ( ) NH4NO3(I) (•) O2 ( ) O2(L)

Enter Oxidizer Temperature(K), if needed: 298.15

( ) Use Periodic Table (mixtures) ( ) None

- Use the Periodic Table if your oxidizer does not appear above or you need a mixture of two or more compounds.
- **Please note that selecting oxidizer(s) using the Periodic Table cancels out any simple-oxidizer selection.**
- If you do not want your analysis to distinguish between fuels and oxidizers, select **None.**
- Select your reactants carefully, since some compounds are represented in the CEA Database in more than one form. For example, H2O refers to water vapor while H2O(L) is liquid.

Camera di combustione

ASSIGNED TEMPERATURE AND PRESSURE PROBLEM (TP)

Problem Type | Temperatures&Pressures | Fuel(s) | Oxidizer(s) | Oxid/Fuel | Final

Fuel(s): Pure H2
Oxid(s): Pure O2

Enter Proportions of Oxidizer/Fuel (Optional)

How do you want to specify the Oxidizer/Fuel Ratio?

- (●) %fuel: %Fuel by Weight
- ( ) o/f: Oxidizer/Fuel Wt. ratio
- ( ) phi: Equivalence based on Fuel/Oxid. wt ratio (Eq 9.19*)
- ( ) r,eq.ratio: Equivalence based on Valence (Eq 9.18*)

If these values are not filled, CEA will assume 1:1 Oxidizer/Fuel ratio by weight.
*Reference: NASA RP1311 Part II (Users Manual), p. 13, Sect. 2.4.3

Enter Low-Value, High-Value and Interval

Low Value:
High Value:
Interval:

Clear Low/High/Int. Fields

The Low-High-Interval values must result in an array of no more than **30** datapoints.

Or, Fill in these numbered fields. *Do not use both sides.*

1.	11.11111	11.		21.	
2.		12.		22.	
3.		13.		23.	
4.		14.		24.	
5.		15.		25.	
6.		16.		26.	
7.		17.		27.	
8.		18.		28.	
9.		19.		29.	
10.		20.		30.	

Clear Numbered Fields

Camera di combustione

ASSIGNED TEMPERATURE AND PRESSURE PROBLEM (TP)

Problem Type | Temperatures&Pressures | Fuel(s) | Oxidizer(s) | Oxid/Fuel | Final

Fuel(s): Pure H2
Oxid(s): Pure O2

Enter Your Final Choices Before Running CEA.

Select Your Output File Length:

- (●) Short: Prints only error messages and final tables, excluding atom ratios and species being considered during calculation.
- ( ) Long: Prints all output tables.

CEA Options

Express **Products** as: ( ) Mass-Fractions (●) Mole-Fractions

Express heat as: (●) SI units ( ) Calories

☐ Include Transport Properties?
Ref: NASA RP1311 Part I (Analysis), Ch. 5

☐ Consider Ionized Species as possible products?

☐ Set Trace Value: 1 x 10^-5

The Trace Option prints species compositions having mole or mass fractions exceeding the Trace Value. The criteria for equilibrium composition convergence will be tighter to ensure accuracy. Mass- and mole-fractions will be expressed in E-format (eg, '2.0089-4' = '0.0002').

Other Parameters

Name	Definition	Qty.	Value(s)
P	Pressure (atm)	1	20
T	Temperature (K)	1	3500
OFP	%fuel	1	11.11111

What do you want to do upon clicking 'Submit'?

- ( ) Perform CEA Analysis.
- ( ) Change Problem Type
- ( ) Tabulate results for insertion into a spreadsheet.
- (●) Select species for 'Omit/Only/Insert' options.

Submit input & Perform CEA Analysis. | Reset

Camera di combustione

Select desired action for each species. Click 'Accept Checked Input...' button when done:

Species	Action	Species	Action	Species	Action
H	● default ○ only ○ omit ○ insert	H2O(cr)	○ default ○ only ● omit ○ insert	O2	● default ○ only ○ omit ○ insert
H2	● default ○ only ○ omit ○ insert	H2O2	○ default ○ only ● omit ○ insert	O2(L)	○ default ○ only ● omit ○ insert
H2(L)	○ default ○ only ● omit ○ insert	H2O2(L)	○ default ○ only ● omit ○ insert	O3	○ default ○ only ● omit ○ insert
H2O	● default ○ only ○ omit ○ insert	HO2	○ default ○ only ● omit ○ insert	O3(L)	○ default ○ only ● omit ○ insert
H2O(L)	○ default ○ only ● omit ○ insert	O	● default ○ only ○ omit ○ insert	OH	● default ○ only ○ omit ○ insert

[Deselect All] [Omit All] [Accept Checked Input and Perform CEA Analysis.]

Camera di combustione

ASSIGNED TEMPERATURE AND PRESSURE PROBLEM (TP)

Problem Type | Temperatures&Pressures | Fuel(s) | Oxidizer(s) | Oxid/Fuel | **Final**

Fuel(s): Pure H2

Oxid(s): Pure O2

Enter Your Final Choices Before Running CEA.

Select Your Output File Length:

● Short: Prints only error messages and final tables, excluding atom ratios and species being considered during calculation.

○ Long: Prints all output tables.

CEA Options

Express **Products** as: ○ Mass-Fractions ● Mole-Fractions

Express heat as: ● SI units ○ Calories

☐ Include Transport Properties?
Ref: NASA RP1311 Part I (Analysis), Ch. 5

☐ Consider Ionized Species as possible products?

☐ Set Trace Value: [1] x 10^-[5]
The Trace Option prints species compositions having mole or mass fractions exceeding the Trace Value. The criteria for equilibrium composition convergence will be tighter to ensure accuracy. Mass- and mole-fractions will be expressed in E-format (eg, '2.0089-4' = '0.0002').

Other Parameters

Name	Definition	Qty.	Value(s)
P	Pressure (atm)	1	20
T	Temperature (K)	1	3500
OFP	%fuel	1	11.11111
omits		9	H2(L) H2O(L) H2O(cr) H2O2 H2O2(L) HO2 O2(L) O3 O3(L)

What do you want to do upon clicking 'Submit'?

● Perform CEA Analysis.

○ Change Problem Type

○ Tabulate results for insertion into a spreadsheet.

○ Select species for 'Omit/Only/Insert' options.

[Submit input & Perform CEA Analysis.] [Reset]

Camera di combustione

```
### CEA analysis performed on Thu 29-Apr-2021 03:19:29
# Problem Type: "Assigned Temperature and Pressure"
prob case=123____________6960 tp
# Pressure (1 value):
p,atm= 20
# Temperature (1 value):
t,k= 3500
# %Fuel by Weight (1 value):
%fuel= 11.11111

# You selected the following fuels and oxidizers:
 reac
 fuel H2                  wt%=100.0000  t,k= 298.150
 oxid O2                  wt%=100.0000  t,k= 298.150
# OMIT the selected species as possible products. (9 selected)
 omit H2(L) H2O(L) H2O(cr) H2O2 H2O2(L) HO2 O2(L) O3 O3(L)

# You selected these options for output:
# short version of output
output short
# Proportions of any products will be expressed as Mole Fractions.
# Heat will be expressed as siunits
output siunits
```


Camera di combustione

Reazione a *T=3500K*

```
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=8.00000  %FUEL=11.111110 R,EQ.RATIO= 0.992085 PHI,EQ.RATIO= 0.992085


THERMODYNAMIC PROPERTIES              Cp, KJ/(KG)(K)    12.9462
                                      GAMMAs             1.1254
 P, BAR             20.265            SON VEL,M/SEC     1453.1
 T, K              3500.00
 RHO, KG/CU M     1.0801 0            MOLE FRACTIONS
 H, KJ/KG           109.70
 U, KJ/KG         -1766.52            *H                0.04777
 G, KJ/KG         -57866.6            *H2               0.13223
 S, KJ/(KG)(K)     16.5647            H2O               0.64164
                                      *O                0.02236
 M, (1/n)           15.510            *OH               0.11428
 (dLV/dLP)t       -1.05824            *O2               0.04172
 (dLV/dLT)p         2.0240
```

Camera di combustione

Chemical Equilibrium Problem Types

	Type Code	Description
○	**rocket**	Rocket
◉	**hp**	Assigned Enthalpy & Pressure
○	**tp**	Assigned Temperature & Pressure
○	**det**	Chapman-Jouguet Detonation
○	**shock**	Shock Tube
○	**tv**	Assigned Temperature & Density
○	**uv**	Combustion at Assigned Density
○	**sp**	Assigned Entropy & Pressure
○	**sv**	Assigned Entropy & Density

Change Problem Type & Continue to Next Form.

What do you want to do upon clicking 'Submit'?

◉ Perform CEA Analysis.
○ Change Problem Type
○ Tabulate results for insertion into a spreadsheet.
○ Select species for 'Omit/Only/Insert' options.

Camera di combustione

```
### CEA analysis performed on Thu 29-Apr-2021 03:35:06
# Problem Type: "Assigned Enthalpy and Pressure"
prob case=123____________6960 hp
# Pressure (1 value):
p,atm= 20
# %Fuel by Weight (1 value):
%fuel= 11.11111

# You selected the following fuels and oxidizers:
 reac
 fuel H2                  wt%=100.0000  t,k= 298.150
 oxid O2                  wt%=100.0000  t,k= 298.150
# OMIT the selected species as possible products. (9 selected)
 omit H2(L) H2O(L) H2O(cr) H2O2 H2O2(L) HO2 O2(L) O3 O3(L)

# You selected these options for output:
# short version of output
output short
# Proportions of any products will be expressed as Mole Fractions.
# Heat will be expressed as siunits
output siunits
```

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=8.00000  %FUEL=11.111110 R,EQ.RATIO= 0.992085 PHI,EQ.RATIO= 0.992085


THERMODYNAMIC PROPERTIES                  Cp, KJ/(KG)(K)    12.8184
                                          GAMMAs             1.1252
 P, BAR               20.265              SON VEL,M/SEC      1449.4
 T, K                3491.48              MOLE FRACTIONS
 RHO, KG/CU M      1.0854 0
 H, KJ/KG            0.00000              *H                0.04662
 U, KJ/KG           -1867.02              *H2               0.13092
 G, KJ/KG           -57725.7              H2O               0.64631
 S, KJ/(KG)(K)       16.5333              *O                0.02180
                                          *OH               0.11296
 M, (1/n)             15.549              *O2               0.04140
 (dLV/dLP)t         -1.05729
 (dLV/dLT)p           2.0103
```


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.

Razzi

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 gas. 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:

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

Off design - Fa 11.10

A turbojet engine has the following design parameters (which is at takeoff):

$M_0 = 0, \quad p_0 = 101.3\,kPa, \quad T_0 = 15C, \; \gamma_c = 1.4,$
$c_{pc} = 1004\,J/kg\,K, \quad \pi_d = 0.95, \quad \pi_c = 30, \quad e_c = 0.90,$
$Q_R = 42{,}600\,kJ/kg, \quad \pi_b = 0.95, \quad \eta_b = 0.98,$
$T_{t4} = 1700\,C, \quad \eta_m = 0.98, \quad e_t = 0.85, \quad \gamma_t = 1.33,$
$c_{pt} = 1156\,J/kgK, \quad \pi_n = 0.90, \quad p_9 = p_0$

This engine powers an aircraft that cruises at $M_0 = 0.80$ at an altitude where $T_0 = -35C$, $p_0 = 20\,kPa$. The turbine entry temperature at cruise is $T_{t4} = 1500C$. Assume that the engine has the same component efficiencies at cruise and takeoff, and the nozzle is perfectly expanded at cruise, as well. Calculate:

Off design - Fa 11.10

This engine powers an aircraft that cruises at $M_0 = 0.80$ at an altitude where $T_0 = -35C$, $p_0 = 20\,kPa$. The turbine entry temperature at cruise is $T_{t4} = 1500C$. Assume that the engine has the same component efficiencies at cruise and takeoff, and the nozzle is perfectly expanded at cruise, as well. Calculate

(a) the exhaust velocity V_9 (in m/s) at the design point, i.e., at takeoff

(b) the thermal efficiency η_{th} at the design point

(c) thrust-specific fuel consumption at the design point

(d) the compressor pressure ratio at cruise

(e) the exhaust velocity V_9 (in m/s) at cruise