Jet Aircraft Propulsion

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Lecture 41

Lecture with Numerical Examples of Ramjet, Pulsejet and Scramjet

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Problem-1 Ramjet

A ramjet is flying at Mach 1.818 at an altitude 16.750 km altitude (Pa = 9.122 kPa, Ta= -56.5^0 C = 216.5 K., sonic speed, a = 295 m/s). The flow is assumed to enter the intake of the ramjet through a normal shock standing at the intake face. No pre-entry loss or friction loss inside the engine is assumed to exist. Combustion delivery temperature is 1280 K. and the fuel –air ratio is 1:40. The area at the intake face is A_1 = 0.0929 m^2 and at the Combustion chamber , A_3 = 0.1858 m^2

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Calculate :

i) Mass flow rate through the engine

ii) Throat area in the nozzle, A_5

iii) Combustion related pressure drop in the combustion chamber

iv) If the nozzle expands to ambient pressure – find the thrust produced

v) If the nozzle expands only in a convergent nozzle – find the thrust produced

vi) Calculate the propulsive efficiencies for (iv) and (v)

vii) Calculate TSFC in both the cases

viii) Complete and draw the cycles for the cases – (a) with C-D nozzle and (b) Convergent nozzle

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Solution :

Flight velocity=Intake velocity of air=$M_1.a$=536 m/s

From isentropic relations,

Total temperature at entry, T_{0a} = 360 K

Total Pressure at entry, P_{0a} = 53.85 kPa

Ambient air density, $\rho_1 = P_a/R.T_a$ = 0.147 kg/m^3

Mass flow through the Intake = $\rho_1.V_a.A_1$ =7.3 kg/s

Normal shock : From shock tables :

At intake face, for M_1 =1.82, M_2=0.612, T_2= 334.8K

$P_{02} = 0.803.P_{01}$= 43.25 kPa, $T_{02}=T_{01}$=360 K

Since the duct losses due to friction etc are zero,

$P_{03}=P_{02}$= 43.25 kPa, $T_{03}=T_{02}$= 360 K

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Using isentropic tables, at stn 2 behind the shock, $M_2 = 0.6121$, the area may be computed from $A_2/A_l = 1.16565$;

Then , $A_{cc}/A_l = (A_{cc}/A_2).(A_2/A_l) = 2 \times 1.16565 = 2.33$

Inside the combustion chamber, the Mach number may be computed from isentropic tables or relations as : $M_3 = 0.26$

Combustion chamber calculations :

Due to accompanying heat addition, Rayleigh flow tables or relations need to be utilized for calculation of parametric variations.

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Flow in the C-D nozzle may be assessed as :
Starting from Mach 0.83 (CC delivery) assuming isentropic flow, $A_4/A_t=1.027$ in the convergent duct
The exit area may be calculated from
$A_t=A_4/(A_4/A_t)=0.1858/1.02696=0.181\ m^2$
As no duct loss is prescribed, $P_{0t}=P_{04}$, $P_t=0.25 \times P_{04}$
At the exit, after flow through the divergent duct,
$M_e=1.55$; $A_t/A_e=1.211$, $T_e/T_{0e}=0.675$
$A_e=1.211 \times 0.181=0.22\ m^2$, $P_e = P_a$ as prescribed
From $T_{0t}=T_{0e}=T_{04}$, as no heat / work is transacted
Since, $T_{0t}=T_{0e}=T_{04}=1280$ K, then $\overline{T_e = }\ 864.5$ K
Jet Vel is now calculated $V_e = M_e\sqrt{\gamma . R . T_e}=916.5$ m/s
For fuel-air ratio of 1:40 thrust is calculated as :

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At the exit, $M_e=1.55$; $A_t/A_e=1.211$, $T_e/T_{0e}=0.675$
$A_e=1.211 \times 0.181=0.22\ m^2$
From $T_{0t}=T_{04}$, as no heat / work is transacted
Since, $T_{0t}=T_{04}=1280$ K, then $T_e = 864.5$ K
Jet Vel is now calculated $V_e = M_e\sqrt{\gamma . R . T_e}=916.5$ m/s
For a fuel-air ratio of 1:40 the thrust may now be calculated as :

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Thrust is given by : $F = \dot{m}[(1+f)V_e - V_1] + A_{e.}(P_e - P_a)$

C-D nozzle Thrust is, $F = 4090$ N

Propulsive Efficiency,

$$\eta_p = F.V_a / F.V_a + ½.\dot{m}.[(1+f)V_e^2 - V_a^2]$$
$$= 51.2\ \%$$

For fuel-air ratio prescribed as $f = 1/40$

Sp. Fuel Consn., TSFC = $f/(F/\dot{m}) = 0.16$ kg/N-hr

If the nozzle is only a convergent nozzle:

The exit face is the throat of the nozzle.

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convergent nozzle

Pressure ratio necessary for choking : 1.893

Available pr. ratio across the nozzle= $P_{04}/P_a=3.96$

The nozzle is choked, and exit pressure, $P_e=19.1$ kPa

$T_e=T_t= T_{04}/[(\gamma+1)/2]=1280/1.2 = 1067$ K

Exit jet velocity, $V_e = V_t = \sqrt{\gamma . R . T_e} = 654$ m/s

From isentropic tables, exit (throat) area, $A_4/A_t=1.027$

Whence, exit area $A_e = 0.1809$ m^2

Thrust $F = \dot{m}[(1+f)V_e - V_1] + A_e.(P_e - P_a) = 3587$ N

Sp. Fuel Consn., $TSFC = f/(F/\dot{m}) = 0.1834$ N

Prop Efficiency $\eta_p = F.V_a/F.V_a + ½.\dot{m}.[(1+f)V_e^2 - V_a^2] = 54.8\%$

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Problem 2 – Pulsejet

An aircraft powered by a pulsejet engine is flying at 12 km altitude, at Mach 2. The engine parameters are given as : inlet area = 0.084 m^2. Combustion chamber pressure development, $P_{03}/P_{02} = 9.0$, heating value of fuel, Q = 43,000 kJ/kg, combustion efficiency = 0.96. Assuming ideal (no loss) flow through the intake, find:

(i) The air mass flow rate,
(ii) Maximum Temp.
(iii) fuel-air ratio, *f*
(iv) Exit velocity, V_e
(v) Thrust of the engine
(vi) TSFC

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At 12 km P_{0a}= 18.75 kPa ; T_{0a}= 216.65 K

The flight velocity is : $V_a = M_a\sqrt{\gamma.R.T_a}$=590 m/s

The air mass flow rate is : = $\rho_a.V_a.A_1$ = 15 kg/s

Total temp across the intake diffuser remains

constant, $T_{01}=T_{oa}= T_a[1+(\gamma-1)M^2/2]$= 390 K=$T_{02}$

Total pressure, $P_{01}=P_{oa}=P_a(T_{01}/T_{0a})^{\gamma/(\gamma-1)}$=147 kPa

In the combustion chamber pressure rises from P_{02} to P_{03} by a prescribed ratio 9.0, and the temp ratio (temp change across the CC) is same by gas laws.

The combustion delivery temp is : T_{03}=3510 K

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The fuel air ratio may be calculated

$$f = \dot{m}_f / \dot{m}_a = (c_{p\text{-}gas}.T_{03} - c_{p\text{-}air}.T_{02})/Q.\eta_{cc} = 0.0975$$

In the jet pipe:

The exhaust velocity is $V_e = \sqrt{2.c_{p\text{-}gas}.T_{03}.[1-1/9^{0.25}]}$

$= 2297$ m/s

Specific Thrust, $F/\dot{m}_a = (1+f)V_e - V_a = 1931.5$ N-s/kg

The thrust. F = 28950 N = 28.95 kN

The TSFC = $f/(F/\dot{m}_a)$ = 50 mg/N-s = 0.180 kg/N-hr

Propulsive Efficiency $\eta_p = F.V_a / F.V_a + ½.\dot{m}.[(1+f)V_e^2 - V_a^2]$

$= 0.418 = 41.8\%$

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Problem 3 – Scramjet

A scramjet powered aircraft flys at Mach 5 at 16.75 km where Ta=216.67 K and Pa=9.122 kPa. The intake has a shock structure of two oblique shocks with both deflection angles $\delta = 10^0$. By burning hydrogen fuel (Q=120,900 kJ/kg), the temp is raised to 2000 K. The fuel air ratio =0.025. The nozzle expansion ratio is $A_5/A_4 = 5.0$. The inlet and the exit areas are $A_1 = A_5 = 0.2\ m^2$. If $c_p = 1.51$ kJ/kg.K ; $\eta_{cc} = 0.8$

Calculate :

i) Mach number at combustion chamber inlet
ii) Exhaust jet velocity
iii) Overall efficiency

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Flight velocity is : $V_a = M_a\sqrt{\gamma.R.T_a} = 1475$ m/s

Mass flow through the engine $\dot{m}_a = \rho_a.V_a.A_1 = 43.3$ kg/s

Inlet total temp $T_{01} = T_{oa} = T_a[1+(\gamma-1)M^2/2] = 1300K = T_{02}$

From shock relations or tables,

Across the first shock , for $M_1=5$ & $\delta = 10^o$

Shock angle, $\beta = 19.4^o$. $M_2=4.0$ and $T_2/T_1=1.429$

Across the second shock, $M_1=4$ & $\delta = 10^o$

Shock angle, $\beta = 22.2^o$. $M_3=3.3$ and $T_3/T_2=1.33$

In the combustion chamber heat is added to air flow with supersonic speed

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Using Rayleigh Flow relations (or tables)

$M_4 = 1.26$, and $P_{04}/P_{0I} = 1.033$, $T_{04}/T_{0I} = 0.966$

Combustion chamber pressure ratio, $P_{04}/P_{03} = 0.228$

Fuel-air ratio, $f = \dot{m}_f / \dot{m}_a = (c_{p\text{-}gas}.T_{03} - c_{p\text{-}air}.T_{02})/Q.\eta_{cc}$

$= 0.01093$

<u>Nozzle Flow</u>

For $M_4 = 1.26$, $T_{04}/T_4 = 1.317$, critical area ratio $= 1.05$

Whence, $A_e/A_t = (A_e/A_4).(A_4/A_t) = 5 \times 1.05 = 5.25$

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Nozzle Flow

Nozzle outlet Mach number , M_e=3.23,
for which isentropic temp ratio T_{05}/T_5= 3.11

$$T_5 = \frac{T_5 . T_{05} . T_{04} . T_4 . T_3 . T_2}{T_{05} . T_{04} . T_4 . T_3 . T_2 . T_1} . T_1 = 654.5 \text{ K}$$

The exhaust velocity , $V_e = M_5 \sqrt{\gamma . R . T_e} =$ 1560 m/s

Specific Thrust, $F/\dot{m}_a = (1+f)V_e - V_a$ = 102 N-s/kg

TSFC = 107.15 mg/N-s = 0.385 kg/N-hr

Thrust , F =43.3 x 102 = 4416 N

Propulsive Efficiency $\eta_p = F.V_a / F.V_a + ½.\dot{m}.[(1+f)V_e^2 - V_a^2]$
=0.527 or 52.7%

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Comparison of Rotor-less Jet Engines

	M	H km	V_e m / s	F (kN)	TSFC Kg / n-hr	η_{pr}	$F / \dot{m}_a$ N / Kg / s
Ramjet (C-D noz)	1.8	16.7	916	4.09	0.16	51.2	560
Pulsejet	2	12	2297	28.95	0.18	41.8	1931
Scramjet	5	16.7	1560	4.41	0.38	52.7	102

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Next

Concluding Lecture

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