16.512, Rocket Propulsion
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Lecture 25: Basic Turbomachine Performance

Turbopump Pressurization Systems

1. Cycles

For higher performance, mechanical pumps must be used to feed the combustion chamber. In turn, these pumps require drive power, which is always provided by turbines using excess thermal energy in the propellants (although electrical motors have been considered for small rockets). The manner in which hot gas is provided to drive the turbines serves to distinguish among several pressurization cycles, of which the most important are summarized in Fig. 1. (From Ref. 41). The most common of these are the bi-propellant gas generator (G.G.) cycle, the expander cycle and the staged combustion cycle.

(a) The Gas Generator Cycle

The GG cycle was used in the F1 engine, and is also in use in the Delta II, Atlas and Titan rockets. In this cycle, a small fraction of the pressurized oxidizer and fuel are diverted to a medium-temperature burner (Gas Generator), which produces typically very fuel-rich gas to drive the turbine or turbines. These are designed with a large pressure ratio, and their exhaust is either dumped overboard, or injected at some point into the main nozzle to provide some extra thrust. Nevertheless, this cycle is inherently somewhat lossy, in that the turbine gas is not fully utilized in the main combustor. On the other hand, the power control is relatively straightforward, and there is little interaction of the feed system with the rest of the rocket. Any propellant combination can be used, all power levels are suitable, and any desired pressure level can be obtained although the I_{sp} loss increase with pressure (1.5-4 sec per 100 atm). The mechanical power required to drive the pumps is

$$P_P = \dot{m}_{ox} \frac{\Delta P_{OP}}{\eta_{oP}\ \rho_{ox}} + \dot{m}_F \frac{\Delta P_{FP}}{\eta_{FP}\ \rho_F} \qquad (1)$$

where ΔP_{OP} and ΔP_{FP} are the pressure rise in the oxidizer pump (OP) and fuel pump (FP), respectively which have efficiencies η_{OP}, η_{FP}. Also ρ_{ox}, ρ_F are the liquid densities. If the gas generator mass flow rate is $\dot{m}_{GG}$, the (single) turbine power is

$$P_T = \dot{m}_{GG}\ \eta_T c_P^{'} T_n \eta_{TT}^{'} \qquad (2)$$

in which η_T is the turbine isentropic efficiency (60-80%), $\eta_{TT}^{'}$ is its thermodynamic expansion efficiency

$$\eta_{TT}^{'} = 1 - \left(P_{ie}\ /\ P_n\right)^{\frac{\gamma'-1}{\gamma'}} \qquad (3)$$

which depends on pressure ratio P_{ie} / P_n and GG gas specify heat ratio, γ'. Also, $c_P^{'}$ is the specific heat of this gas, and T_{ti} its temperature, which is controlled through stoichiometry to values acceptable by uncooled turbines (700-1100K).

Ref. 41: Turbopump System for Liquid Rocket Engines. NASA SP-8107, Aug. 1974.

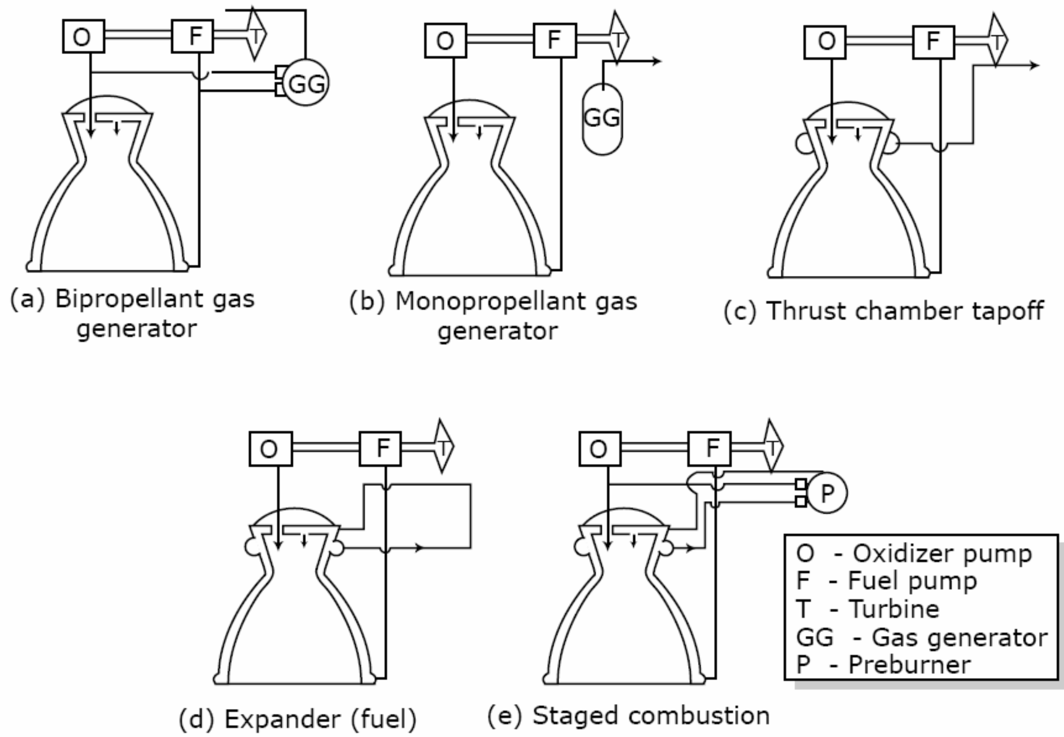

Fig. 1 Typical rocket engine cycles and turbine installations

Equating P_P and P_T yields the required $\dot{m}_{GG}$. Suppose now the turbine exhaust is expanded to the same exit pressure P_e as the main flow. The exhaust speed is then

$$c_{GG} = \sqrt{2\, c_P'\, T_{t.out}\left[1-\left(P_e\ /\ P_{te}\right)^{\frac{\gamma'-1}{\gamma}}\right]} \tag{4}$$

where

$$T_{t.out} = T_n\left(1-\eta_T\ \eta_{TT}'\right) \tag{5}$$

This speed is generally lower than the main flow exit speed,

$$c = \sqrt{2\, c_p\, T_c\, \eta} \qquad ; \qquad \eta = 1-(P_e\ /\ P_c)^{\frac{\gamma-1}{\gamma}} \tag{6}$$

where c_p and η belong to the nozzle gas, and P_c, T_c, are the chamber pressure and temperature. The relative I_{sp} loss is:

$$-\frac{\delta I_{sp}}{I_{sp}} \cong \frac{\dot{m}_{GG}}{\dot{m}}\left(1-\frac{c_{GG}}{c}\right) \tag{7}$$

and can be calculated once the turbine exhaust pressure P_{te} (and hence the turbine pressure ratio) is selected. A tradeoff is involved here: if P_{te} is not very low, too much mass must be diverted (large $\frac{\dot{m}_{GG}}{\dot{m}}$ in Eq. 7), where as if P_{te} is too low, the exhaust provides almost no additional thrust (small $\frac{c_{GG}}{c}$ in Eq. 7). An optimum can therefore be found.

As an example, consider the LOX-RP1 F-1 cycle, for which $\dot{m}_{ox} = 1844$ Kg/sec, $\dot{m}_F = 777\ \text{Kg/sec}$, $\Delta P_{OP} = 1.06 \times 10^7\ \text{N/m}^2$, $\Delta P_{FP} = 1.24 \times 10^7\ \text{N/m}^2$, $T_{ti} = 1061\ \text{K}$ and $\eta_{OP} = 0.746$, $\eta_{FP} = 0.726$, $\eta_T = 0.605$. Also, $\rho_{ox} = 1145\ \text{Kg/m}^3, \rho_F = 810\ \text{Kg/m}^3$. The GG is estimated to have $\gamma' = 1.35$ and $c_p' = 2140\ \text{J/Kg/K}$, while the main gas has $\gamma = 1.25$, $c_p = 2080\ \text{J/Kg/K}$. The chamber pressure is $P_c = 95$ atm, and at the exit, P_e = 0.68 atm. Use of the equations above yields, after some searching, an "optimum" turbine pressure ratio of 23, for which $\frac{\delta I_{sp}}{I_{sp}}$ = -0.0111. The actual F-1 engine had a turbine pressure ratio of 16.4.

(b) <u>The Expander Cycle</u>

For engines, utilizing hydrogen (and possible methane) as fuel, the gas generator can be eliminated. Instead, the fuel is simply routed from the exit manifold of the nozzle cooling circuit to the turbine inlet. This is possible because hydrogen is supercritical at the pump exit, and it simply expands smoothly into an ordinary gas as it picks up heat. The resulting "Expander Cycle" is simple and efficient (the fuel is fully utilized in the thrust chamber). This cycle is used in the RL-10 engine and in the start-up sequence of the Japanese LE-5 engine (which then transitions to gas-generatory operation).

The principle limitation of this cycle is the relativity small amount of heat available from regenerative cooling, which limits applicability to chambe pressures under approximately 70 atm. A simple analysis can demonstrate this point. The pump power is given by Eq. (1) and the power derived from the fuel-driven turbine is

$$P_T = m_F \eta_T c_{PF} T_{ti} \left[1-\left(\frac{P_{inj}}{P_{ti}}\right)^{\frac{\gamma_F - 1}{\gamma_F}}\right] \tag{8}$$

where P_{inj}, the injector pressure, is also the turbine exhaust pressure. The turbine inlet pressure P_n, is related to the fuel pump pressure rise ΔP_{FP} and the cooling circuit loss, ΔP_{cool} by

$$\Delta P_{FP} = P_{ti} + \Delta P_{cool} - P_{TK} \tag{9}$$

while the oxidizer pump has $\Delta P_{OP} = P_{inj} - P_{TK}$. The shaft power balance then gives

$$\left(O/F\right)\left(\frac{\rho_F}{\rho_{ox}}\right)\frac{1-\delta_{TK}}{\eta_{OP}} + \frac{1/\pi_T + \delta_{cool} - \delta_{TK}}{\eta_{FP}} = \psi\left(1 - \pi_T^{\frac{\gamma_F - 1}{\gamma_F}}\right) \tag{10}$$

where

$$\delta_{TK} = P_{TK} / P_{inj}, \delta_{cool} = \Delta P_{cool} / P_{inj}, \pi_T = \frac{P_{inj}}{P_{ti}}, \text{ and } \psi = \eta_T \frac{\rho_F c_{PF} T_{ti}}{P_{inj}} \tag{11}$$

Assuming $\delta_{TK} = 0.1$, $\delta_{cool} = 0.2$, $\eta_{OP} = \eta_{NF} = 0.7$, $O/F = 5$ and $\rho_F / \rho_{ox} = 69/1140 = 0.0605$, the relationship between π_T and ψ, as given by (10) is shown plotted in Fig. 2. As the figure shows, the turbine inlet pressure increases rapidly when ψ drops below about 30 (at which point $1/\pi_T = 1.33$). In fact, as also shown in Fig. 2, the quantity ψ has a <u>minimum</u> value of approximately 17.76 when $1/\pi_T$ = 2.68, below which no solution exists. The turbine inlet temperature T_{ti} is 200K in the RL-10. Using also η_T = 0.7, $\rho_F = 69\ kg/m^3$ and C_{PF} = 14,600 J/Kg/K, we calculate from (11) $\psi \cong$ 1390 / P_{inj} (*atm*) and so the <u>maximum</u> P_{inj} is 1390/17.76=78 atm. For reference, the RL-10 has P_C = 32 atm, $P_{inj} \cong 40\ atm$, which corresponds to ψ = 35. Of course, as (11) indicates, higher P_{inj} values could be achieved if T_{ti} could be increased further.

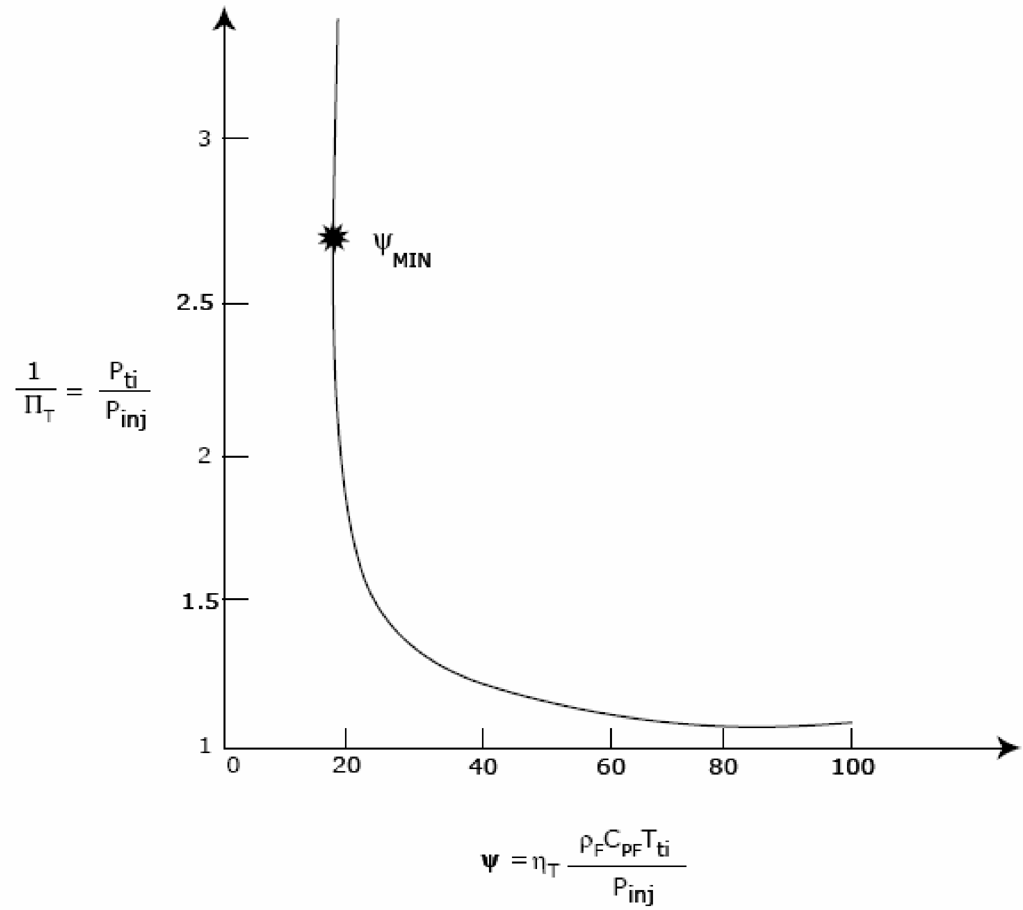

Fig. 2 Turbine Pressure Ratio for Expander Cycle

(c) Staged Combustion Cycle

For rockets where high chamber pressure as well as high efficiency is desired, the staged combustion cycle is the preferred choice. One could think of this as a modified expander cycle, in which a small amount of oxidizer is added to the fuel after the cooling circuit, thus increasing the available enthalpy for the turbine drive.

As in the expander cycle, all of the propellant is entirely used in the combustion chamber. Unlike the expander, through, any oxidizer-fuel combination can be used. Two prominent examples of this cycle are the Space Shuttle Main Engine (SSME), and the Russian RD-170 booster engine (Figs. 3 and 4 from Ref. 2). In the SSME, the pre-burners are incorporated into separate fuel and oxidizer turbopump assemblies, and process most of the fuel *(LH)* with a small fraction of the oxidizer *(LOX)*, producing a light "vitiated hydrogen" turbine driving gas. In the RD-170, the pre-burners process all of the oxidizer *(LOX)* and a fraction of the fuel (kerosene) to produce a fuel-lean gas which drives the single central turbine. In both cases, the turbine exhaust is ducted to the main combustor injectors, together with the remaining *LOX* (SSME) or kerosene (RD-170). The choice of fuel-rich preburners is precluded by carbon deposits on turbine blades and other surfaces when hydrocarbon fuel is involved.

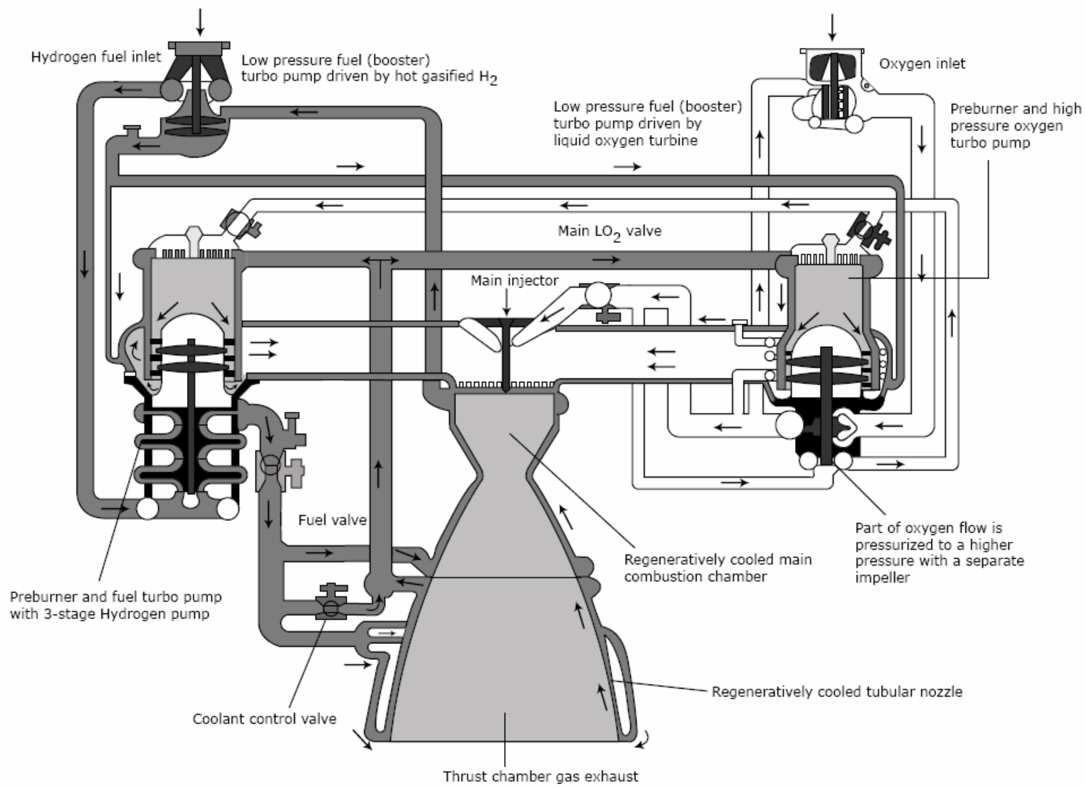

Fig. 3 Flow diagram for staged combustion cycle used in Space Shuttle Main Engine using liquid oxygen and liquid hydrogen propellants

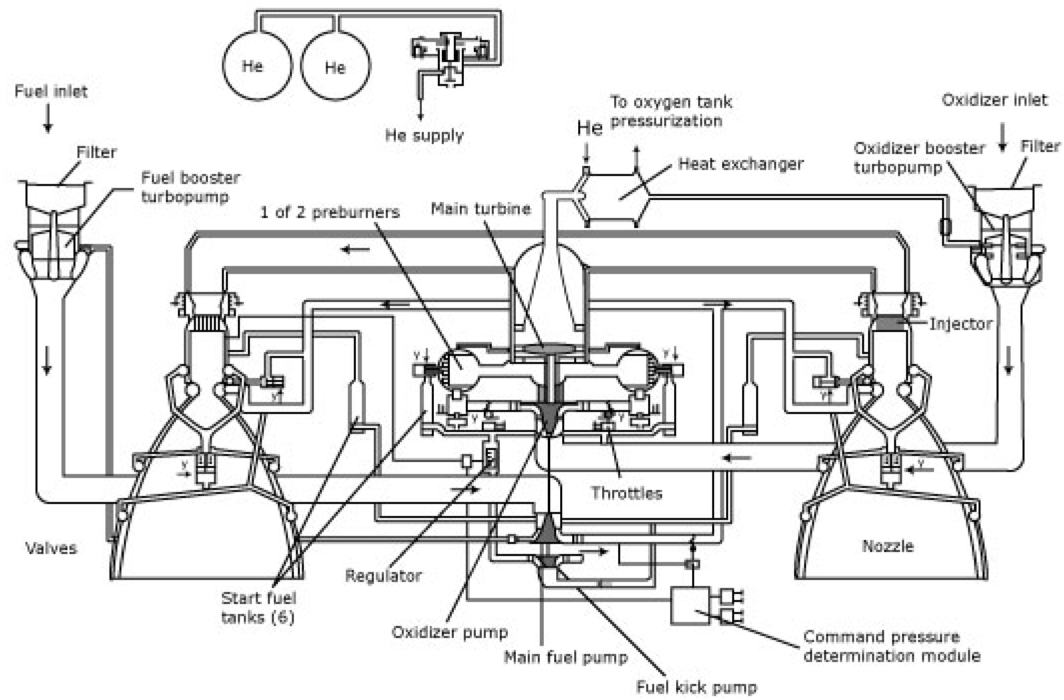

Fig. 4 Simplified flow diagram of the high-pressure RD-170 rocket engine. The single-shaft large turbopump has a single-stage reaction turbine, two fuel pumps, and an oxygen pump with an inducer. It supplies propellant to two oxidizer rich preburners and four thurst chambers (only two or shown in this diagram). It has a separate low-pressure fuel turbopump and a low-pressure oxygen turbopump. The pressurized helium supplied to various actuators are indicated by the symbol y. See also Table 7-7 and Figure 7-9. Adapted from NPO Energomash, Moscow.

The staged combustion cycle provides the highest levels of rocket performance, but at the cost of greatly increased complexity. This is both, because of the many ducts and valves involved, and because of the very high pump exit pressures. A secondary potential difficulty, which is shared by the expander cycle, is that the turbines are unchoked and there is a possibility of low-frequency instability developing.

Consider a simplified schematic (Fig. 5) of a fuel-rich cycle analogous to that of the SSME. The injector pressure, P_i, the turbine inlet temperature, T_{ti}, and the overall *O/F* ratio, r, are prescribed. The heat input and pressure drop in the cooling circuit (Q_{cool}, ΔP_{cool}) are also assumed given. We wish to determine the required pressure rise ΔP_{FP} by the fuel pump, as well as the *O/F* in the pre-burners (r_{PB}) and the fuel split S_F between them. The pressure rise in the oxidizer pump is simply $\Delta P_{OP} = P_i - P_{TK}$.

The analysis is best done iteratively. If ΔP_{FP} is temporarily assumed known, then the turbine inlet pressure is

$$P_{ri} = P_{TK} + \Delta P_{FP} - \Delta P_{cool} - \Delta P_{PB} \tag{12}$$

where ΔP_{PB} is the pressure drop in the pre-burner (mainly associated with injection). The required rise in the oxidizer booster pump is then

$$\Delta P_{OBP} = P_{ti} + \Delta P_{PB} - P_i \tag{13}$$

The energy balance in one of the pre-burners is written as

$$r_{PB}\left(h_{OTK} + \frac{\Delta P_{OP}}{\rho_0\,\eta_{OP}} + \frac{\Delta P_{OBP}}{\rho_0\,\eta_{OBP}} \right) + h_{FTK} + \frac{\Delta P_{FP}}{\rho_F\,\eta_{FP}} + Q_{cool} + \frac{r_{PB}}{r_{st}} h^f = \left(1 + r_{PB}\right) C_{Pti}(T_{ti} - T_{ref}) \tag{14}$$

where h_{OTK} and h_{FTK} are the enthalpies of oxidizer and fuel in their tanks (this ignores the low pressure booster pumps) r_{st} is the stoichiometric *O/F* ratio and *hf* the fuel heat value at the reference temperature T_{ref}. The specific heat of the fuel-rich burnt gas is c_{pti}. Eq. (14) can be solved for r_{PB} if ΔP_{FP} is assumed known.

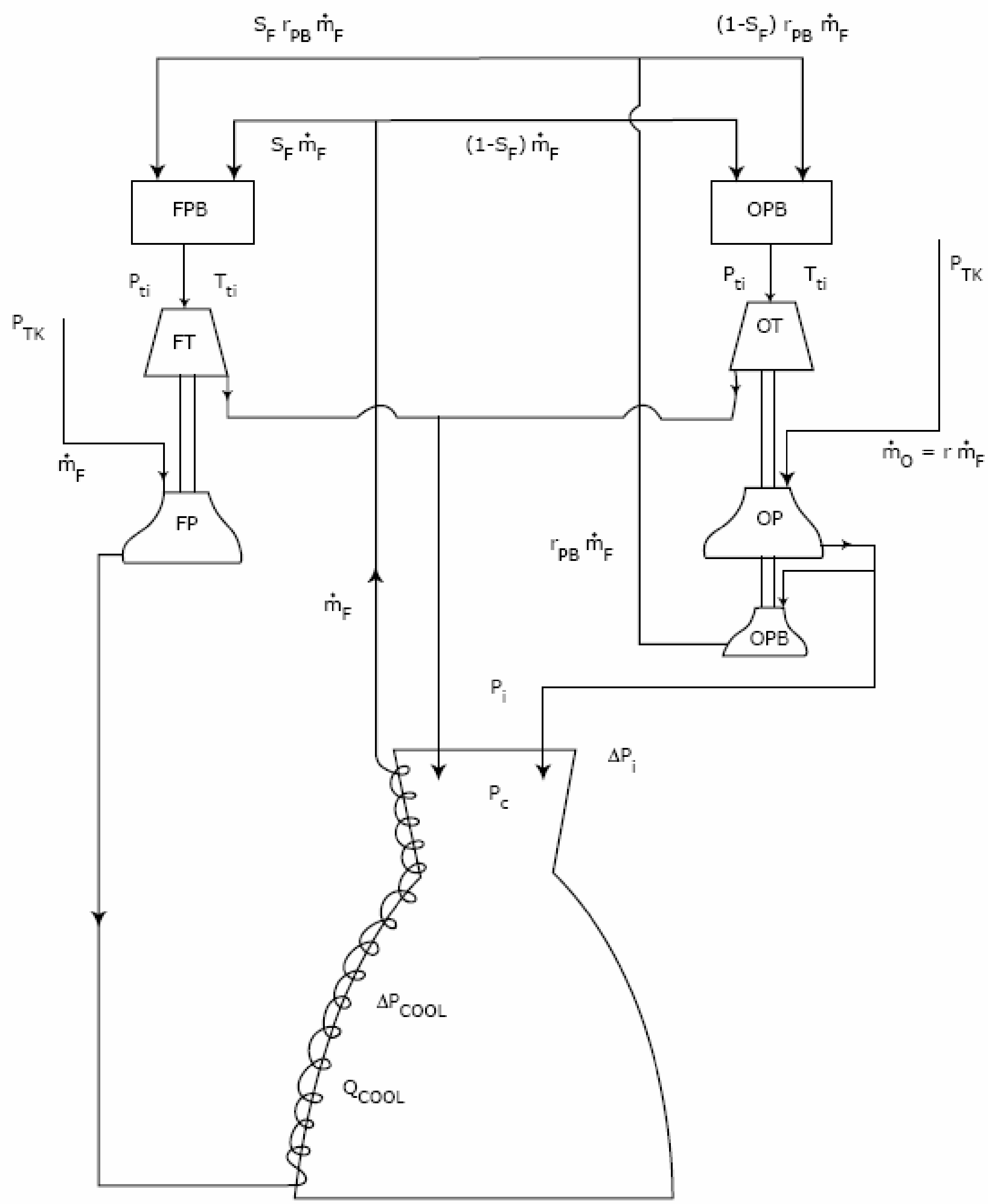

Fig. 5: Schematic of fuel-rich staged-combustion cycle

The shaft power balance for the fuel turbopump is

$$\frac{\Delta P_{FP}}{\rho_F \eta_{FP}} = S_F \left(1+r_{PB}\right) c_{pti} T_n \eta_{FT} \left[1-\left(\frac{P_i}{P_{ti}}\right)^{\frac{r'-1}{r'}}\right] \quad (15)$$

where η_{FT} is the turbine efficiency and $\gamma' = \gamma_{ti}$ belongs to the pre-burner gas. A similar balance can be written for the oxidizer turbopump, and, by division, we can solve for the fuel split S_F.

$$S_F = \left[1+\frac{\eta_{FT}}{\eta_{OT}} \frac{\rho_F}{\rho_0} \frac{\eta_{FP}}{\Delta P_{FP}}\left(r\frac{\Delta P_{OP}}{\eta_{OP}} + r_{PB}\frac{\Delta P_{OPB}}{\eta_{OPB}}\right)\right]^{-1} \quad (16)$$

After r_{PB} has been calculated from (14), S_F is given by (16), and then (15) can be used as a check on the assumed ΔP_{FP}. In reality, an outer iteration loop is required by the fact that γ' and c_{pti} themselves depend sensitively upon the preburner stoichiometry, r_{PB}. With the approximations $c_{pH2} \cong 7.67$ cal / mol / K , and $c_{pH_2o} \cong 10.63$ cal / mol / K

$$\gamma' = \frac{7.67 + 0.37 r_{PB}}{5.68 + 0.37 r_{PB}} \quad (17)$$

and then (for H_2-O_2),

$$c_{\rho ti} = \frac{\gamma'}{\gamma'-1}\frac{R}{M^1} \; ; \; M' = 2\,(1+r_{PB})\ g\ /\ mol \quad (18)$$

As an example, Fig. 3.8 shows some results in which we have used

$$T_{ref} = 0\,K \quad h_{OTK} \cong h_{FTK} \cong 0$$

and also

$r = 6$, $r_{st} = 8$, $P_{OTK} = 2$ atm $P_{FTK} = 1$ atm, $h^f = 1.21\text{x}10^8$ J/Kg, $T_{ti} = 1100$ K, $\eta_{FP} = 0.741$
$\eta_{op} = 0.781$, $\eta_{OBP} = 0.696$, $\eta_{FT} = 0.790$, $\eta_{OT} = 0.729$,
$\Delta P_{cool} = 0.15\,P_c$, $\Delta P_{PB} = 0.05\,P_c$,
$Q_{cool} = 7.40\times10^5$ J/kg, $\Delta P_i = P_i - P_c = 0.1\,P_c$

As shown in Fig. 6, the pressure rise in the fuel pump increases more steeply than the chamber pressure, just as it did in the expander cycle. However, very high pressures are now attainable. The ΔP_{FP} for the SSME is shown for reference. Notice how the pre-burner stoichiometry is essentially fixed by T_{ti} , and does not vary much over the pressure range.

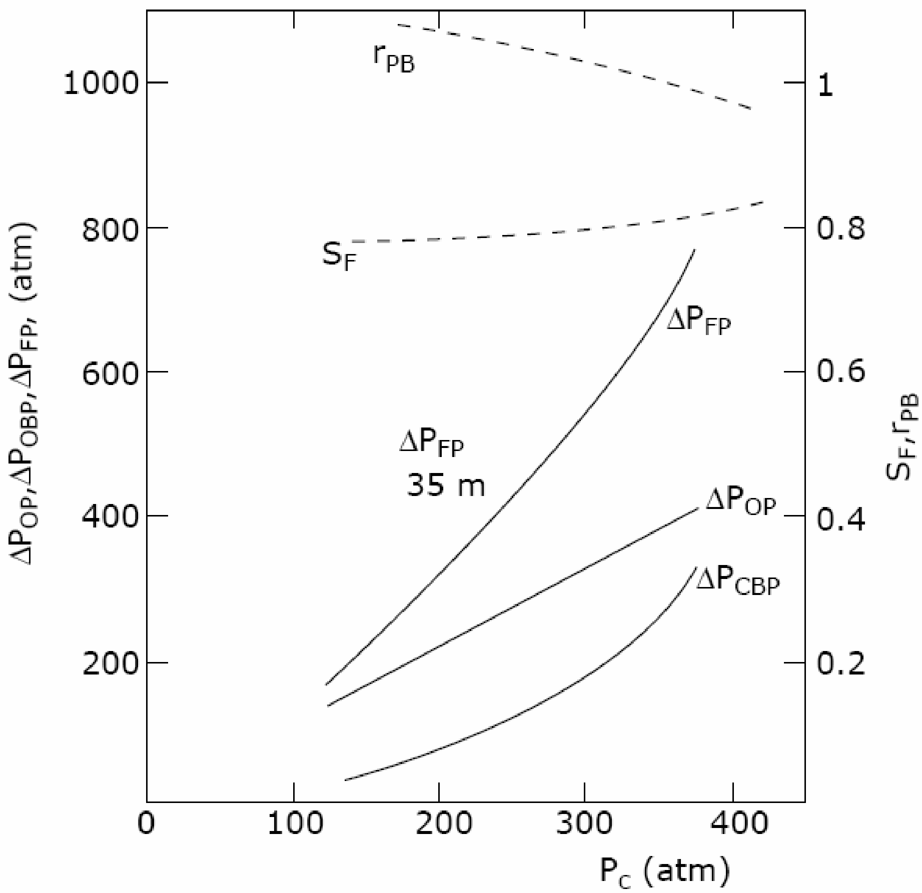

Fig. 6 Pump ressure rises, pre-burner stoichiometry and fuel split (see text) for a LH-LOX staged combustion cycle.