

16.512, Rocket Propulsion
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Lecture 24: Pressurization and Pump Cycles

Thrust Chamber Pressurization

1. Introduction

The most technically demanding part of a modern high pressure liquid propellant rocket is the chamber pressurization system. Consider for example the cluster of four RD-170 engines powering the Energia first stage. Their shared oxygen pump must deliver 1792 Kg/sec (1.63 m^3) of *LOX* at 614 *atm* pressure to the combustors, for a power of 176,000 HP. The whole engine weighs 9500 *Kg*. Similarly, the *SSME LH* pump (see Fig. 3 in lecture 25) delivers 73 Kg/sec of LH ($1.06 \text{ m}^3/\text{sec}$) at 470 *atm* pressure, for a power of 76,000 HP. The *SSME* mass is 2870 *Kg*, of which the LH turbopump is 344 *Kg*. And these flow rates are both dwarfed by those of the old F1 Saturn engine, which swallowed a small river ($3.5 \text{ m}^3/\text{sec}$) of *LOX*, albeit at a more modest pressure.

Because of the importance of this system, we will devote this lecture to engine pressurization cycles and components.

2. Gas Pressurization Systems

The simplest way to achieve the required thrust chamber pressure is to provide a small, high pressure gas reservoir, which, at firing time, pressurizes the propellant tanks. The tanks must then be thick enough to support the thrust chamber pressure, plus the injector drop. In addition, the gas reservoirs are also relatively heavy. Thus, the performance of a gas-pressurized rocket lags that of a pump-pressurized rocket (where the tanks can be much lighter), as shown in Fig. 1 (from Ref. 40). We notice here that the relative size of the gains due to a turbo pump system become overpowering for high ΔV rockets, but may not be worth the extra complexity for small ΔV

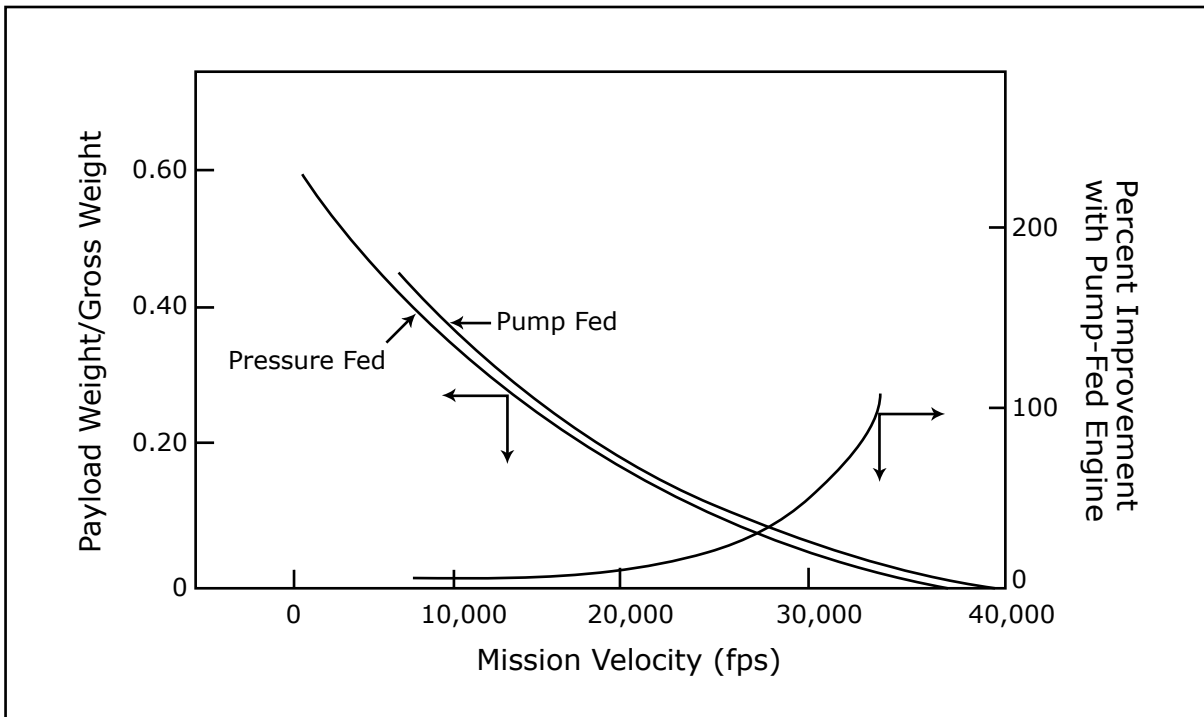


Fig. 1. Payload ratio for typical gas pressurization and pump pressurization

Thus, gas pressurization continues to play a role, particularly in small space-based engines, such as monopropellant maneuvering and attitude control thrusters. A relatively large and sophisticated example is the Shuttle OMS/RCS system.

If a gas pressurization system is adopted, a choice must be made between regulated or blow-down gas delivery. The regulated type is used most often, because it avoids chamber operation over an extended pressure range, which may lead to stability problems. On the other hand, blow-down operation does result in a simpler and lighter system, with less gas inventory, and is standard in hydrazine monopropellant applications.

Consider a regulated gas feed system, schematized in Fig. 2. Let $P_G(t)$ be the decreasing pressure in the gas reservoir, of constant volume V_G and P_p the regulated tank pressure, where $V_p(t)$ is the increasing ullage volume in both propellant tanks. The internal energy of the gas at some times t is $U_G + U_p$ and the work of gas expansion in dt is $P_p dV_p$. Therefore, integrating in $(0,t)$, (assuming adiabatic expansion)

$$U_G + U_p + P_p V_p = U_{G0} \quad (1)$$

NOTE: If isothermal exp., $U_G + U_p = U_{Go}$ and then $P_G(t) V_G + P_p V_p(t) = P_{Go} V_G$

which can be re-written for an ideal gas as

$$\left(\text{Using } U = \frac{P_v}{\gamma - 1} \right) \quad P_G(t) V_G + \gamma P_p V_p(t) = P_{Go} V_G \quad (2)$$

This relates $P_G(t)$ to $V_p(t)$ at any time during operation. If the final gas pressure is P_{Ge} (large enough to still overcome the regulator and injector drops), and if V_{TK} represents the whole tankage volume, we obtain for the gas reservoir volume

$$V_G = \frac{\gamma P_p}{P_{Go} - P_{Ge}} V_{TK} \quad (3)$$

(no γ for isothermal case) (more generally in between)

The minimum mass of this gas reservoir, assumed spherical of radius R_G , can be estimated by noting that its wall thickness t_G must be

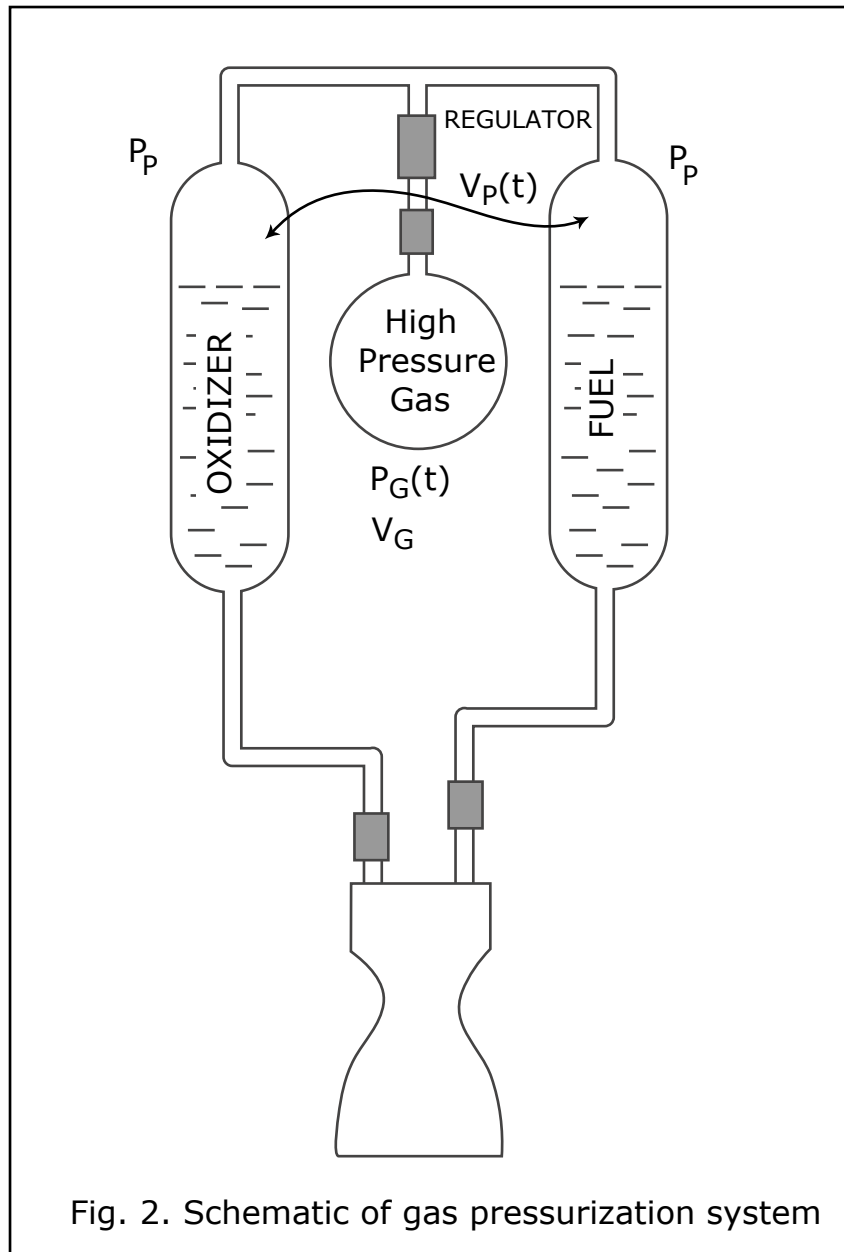
$$t_G = \frac{P_{go}}{2\sigma_{GW}} R_G \quad (4)$$

where σ_{GW} is the working stress of the wall material. If the density of this wall material is ρ_{GW} , we obtain for the gas tank mass

$$M_{GTK} = \frac{3}{2} \frac{\rho_{GW}}{\sigma_{GW}} \frac{\gamma P_p V_{TK}}{1 - \left(\frac{P_{Ge}}{P_{Go}} \right)} \quad (5)$$

A similar calculation can be made for the mass of each of the propellant tanks. If, for simplicity, both, oxidizer and fuel tanks are assumed to be geometrically similar and made of the same material (a cylindrical body of length L_p , capped by hemispheres of radius R_p , with equal L_p/R_p for both), then the propellant tank mass is

$$M_{PTK} = \frac{3}{2} \frac{\rho_{PW}}{\sigma_{PW}} \frac{1 + L_p / R_p}{1 + \frac{3}{4} L_p / R_p} P_p V_{TK}$$



We notice from (5) that the gas tank mass is nearly independent of its initial pressure: a higher P_{G_0} allows a smaller volume V_G (Eq. 3), but requires thicker walls (Eq. 4). Values of P_{G_0} 5-10 times P_p are common. To compare the mass of the gas tank to that of the main propellant tanks, assume $P_{G_0}/P_{G_0} = 0.2$, $\gamma = 5/3$ (He gas) and $L_p/R_p = 6$. For equal material ratio (ρ/σ), Eqs. (5) and (6) then give $\frac{M_{GTK}}{M_{PTK}} = 1.64$, indicating that the gas reservoir is likely to be the heavier component (although adding anti-slosh baffles, insulation, etc. may modify this conclusion).

The mass of the propellant itself is $M_p = V_{TK} \bar{\rho}_p$, where the densities of oxidizer and fuel have been averaged to $\bar{\rho}_p$. To compare to the minimum tankage mass, assume steel construction, with $\sigma/\rho \cong 3.4 \times 10^4 (\text{m/sec})^2$ and $P_p = 20 \text{ atm}$. Eq. (5) then gives $M_{GTK} / V_{TK} = 184 \text{ Kg/m}^3$, and Eq. (3.6) gives $M_{GTK} / V_{TK} = 112 \text{ Kg/m}^3$, for a total of 296 Kg/m^3 . This is to be compared to the mean propellant density; for N_2O_4 -UDMH this is of the order of 1000 Kg/m^3 . Thus, not counting flanges valves, regulators, etc., the tankage mass is 30% of the propellant mass, a figure which is excessive for any ambitious mission. Advanced composite materials can reduce this mass significantly, however, Titanium is used mostly.

An alternative method to reduce the required gas reservoir mass is to heat the pressurizing gas prior to injection into the propellant tank. This can be accomplished through heat exchanging, perhaps using the gas as a nozzle coolant, or by including in the gas a very small amount of oxygen and hydrogen, below the flammability limit, and then passing the gas through a catalytic bed reactor. In either case, assuming a gas temperature rise ΔT is accomplished, a term $M_G c_v \Delta T$ needs to be added on the right hand side of Eq. (1), where $(M_G(t))$ is the gas mass that has flowed through the heater. This yields eventually a modified expression for the gas reservoir volume:

$$V_G = \frac{\gamma P_p}{P_{G_0} \left(1 + \frac{\Delta T}{T_{G_0}} \right) - P_{G_e}} V_{TK} \quad (7)$$

As an example, 1% O_2 plus 2% H_2 (by mole) in He yields upon reaction $\Delta T \cong 230\text{K}$, which nearly cuts in half the required gas tank volume (and mass). One problem with heating methods is the potential for erratic variations in feed pressure if propellant sloshing cools the pressurizing gas in an unsteady manner.