Simple H_2O_2 monopropellant rocket engine

Presented by:
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Safety

A rocket engine is always dangerous due to the release of gas and energy! For this rocket engine design, the propellant is high concentration of hydrogen peroxide (H_2O_2).

- At low concentration, H_2O_2 is safe and used widely for household applications. However, at high concentrations, H_2O_2 is dangerous due to its oxidizing properties that could lead to skin burn in contact.
- H_2O_2 is not stable, may decompose into oxygen and water, and release a lot of energy. This decomposition happens rapidly and lead to damages
- H_2O_2 is not compatible with commons materials: brass (usually used for pneumatics equipments), buna rubber (usually in O-rings/sealing), oil-based grease. The construction materials should be stainless steel or aluminum, Teflon-based O-rings/sealing and grease.

Mission type

The concept mission is a small lander:

- Thrust adequate for the lander: $250N$
- Required burn time: $40s$
- Ease to ignite and shut down
- Low cost and ease to manufacture

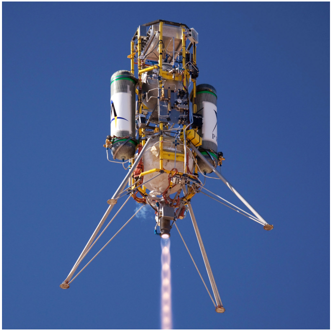
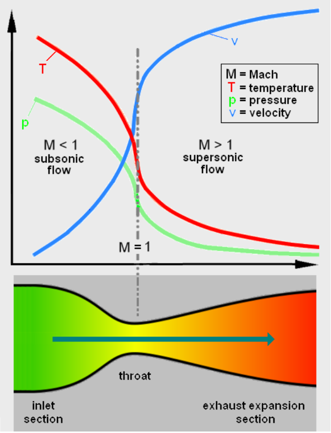


Figure: Masten Xombie test vehicle

de Laval nozzle

- At the convergence section, the flow is "choked" to sonic flow. Theoretically, any further reduction in cross-section does **not** increase the flow velocity. Theoretically, the ratio between divergence section inlet and the throat is about 1.5.
- At the divergence section, the flow is accelerated to supersonic flow. The divergence section expansion ratio depends on the ratio between inlet and ambient pressure. In case of vacuum, the limit factor of expansion ratio is surface tension and condensation of the fluid.



The values usually use to parameterize the nozzle's geometry

- Throat diameter: from characteristics thrust CF

Divergence section

- Exit diameter: from expansion ratio A_t/A_e
- Divergence half angle α, usually 15°

Convergence section

- Combustion chamber length: based on combustion, O/F mixing...
- Combustion chamber diameter: from conctraction ratio
- Convergence half angle θ, usually in the range of 20° to 45°

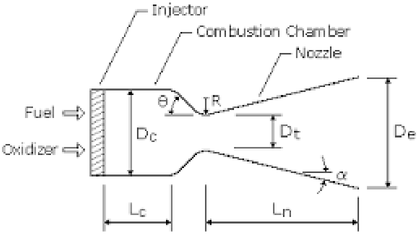
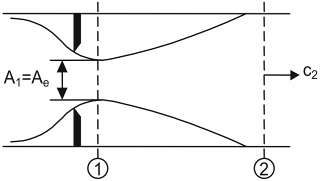


Figure 1.4

Design parameter for a de Laval nozzle

- Throat size → Depends on mass flow rate through the nozzle
- Throat expansion ratio, which affects divergence section length is A_{exit}/A_{inlet} → Depends on pressure ratio between combustion chamber and ambient, and combustion enthalpy
- Convergence and divergence half angles are calculated from real fluid's surface tension, boundary layer, Raynold number → For an ideal nozzle without any loss, the nozzle could be an orifice. In practice, these angles are usually in a consistent range



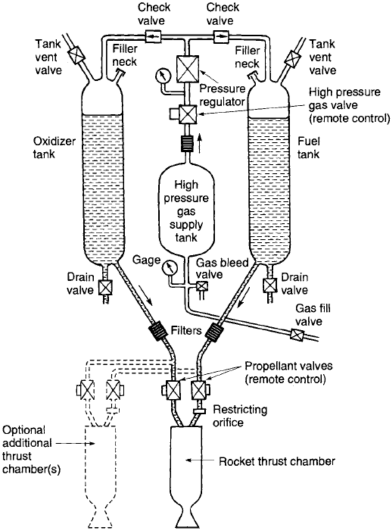
Design parameter for a de Laval nozzle

- Combustion chamber length depends on the combustion, fuel-oxidizer mixing... and usually no trivially numerical results, results are mostly from testing.
- Combustion chamber diameter depends on contraction ratio between $A_{chamber}/A_{throat}$. Ideally, this ratio is about 1.5, but due to boundary layer, turbulent flow... this value is larger than ideal value
- Convergence half angle Θ is usually in the range of 20° to 45°

TABLE II. - SUMMARY OF RESULTS

Contraction ratio	Nozzle throat area, sq in.	Chamber pressure, lb/sq in. abs		Heat-transfer rate		Characteristic exhaust velocity, c*, observed		Correction factors		Corrected c*, percent efficiency
		Theoretical	Experimental	$\frac{Btu}{sec}$	$\frac{Btu}{(sq\ in.)(sec)}$	$\frac{ft}{sec}$	Percent efficiency	Momentum pressure loss	Heat transfer	
1.5	5.33	29.3	24.9	246	1.10	7100	86.3	0.895	1.030	79.6
2	4.00	39.0	33.2	324	1.45	7310	88.9	.946	1.040	87.5
3	2.67	58.5	52.2	366	1.67	7890	95.8	.965	1.045	96.6
4	2.00	78.0	66.8	418	1.92	7720	93.7	.976	1.050	96.0
5	1.60	97.6	81.5	630	2.90	7530	91.4	.987	1.070	96.5
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

Pressure-fed cycle



Why Hydrogen peroxide?

- Hydrogen peroxide is a liquid at room temperature and pressure
  $\rightarrow$ Could be easily stored and require no thermal insulation
- Could decompose and release energy without the presence of other fuels
  $\rightarrow$ Does not require complex system compared with bipropellant
- The decomposition is autogenous with catalyst
  $\rightarrow$ Ease to ignite and restart
- Safe (with low concentration)
  $\rightarrow$ In case of malfunction, hydrogen peroxide could be easily neutralized by diluting with water

Hydrogen peroxide decomposition

The decomposition of H_2O_2

$$2H_2O_2 \rightarrow 2H_2O + O2$$

generates $\Delta H = -2884.5kJ/kg$ at standard condition $T = 25^\circ C, 1\ atm$
This decomposition usually requires a catalyst. Usually, the catalyst used is silver (both in pure form or silver coated)

Figure: Silver mesh catalyst bed

Engineering consideration

- The engine runs on a pressure-fed cycle
- High-pressure gas supply tank is limited to 300 bar (30MPa)
- The engine runs by H_2O_2 monopropellant (the energy is from the decomposition of H_2O_2 only)
- The concentration of H_2O_2 is 80%

First law of thermodynamics

Rocket engine is the heat engine, which must satisfy the first law of Thermodynamics for flow in an adiabatic nozzle

$$h_2 - h_1 = -\frac{1}{2} \times (c_2^2 - c_1^2)$$

For non-ideal gas, the enthalpy is the function of $H(pressure, Temperature)$. There are no such models universally, to determine these enthalpy values

$\rightarrow$ The enthalpy values usually come from real measurement, and complex state equations, for each kind of substance! Usually, we use numerical software to calculate the thermodynamic properties of the rocket engine.

NASA CEA demo for thermodynamics calculation

Initial conditions

- H_2O_2 80% monopropellant (20% left is water)
- Combustion chamber pressure is 30 bar, ambient pressure is 1 bar, ratio=30
- Initial H_2O_2 temperature: 293.15K

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

Sample NASA CEA result

	CHAMBER	THROAT	EXIT
Pinf/P	1.0000	1.8323	37.000
P, BAR	36.900	20.138	0.99730
T, K	782.05	681.78	364.76

PERFORMANCE PARAMETERS

Ae/At	1.0000	5.0355
CSTAR, M/SEC	824.5	824.5
CF	0.7087	1.4729
Ivac, M/SEC	1034.3	1326.5
Isp, M/SEC	584.3	1214.3

Sample chamber-nozzle geometry calculation

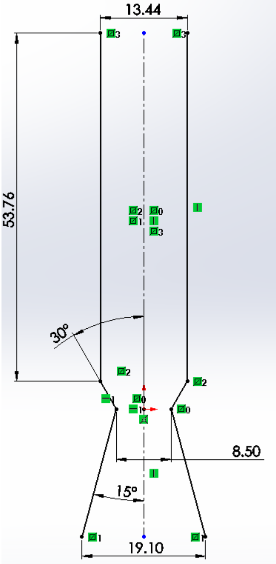
- The combustion temperature is 782.05K (508.9$^\circ C$. It is a relatively low value for metal materials. Hence, it does not require to cool down the engine, and no need to calculate heat transfer
- For the required thrust of 250N, the required mass flow rate is
$$\dot{m}[kg] = \frac{F_{thrust}[N]}{Isp[m/s]} = \frac{250N}{1214.3m/s} = 0.206kg/s$$
- Throat area
$$A_t[in^2] = \frac{F_{thrust}[lbf]}{C_f \times p_{chamber}[psi]} = \frac{56.2lbf}{1.4729 \times 435.1psi} = 0.0877in^2 = 56.59mm^2$$
$\rightarrow$ The throat diameter $D_{throat} = 8.5$mm
- Nozzle exit diameter
$$D_{exit} = D_{throat} \times \sqrt{\frac{A_{exit}}{A_{throat}}} = 8.5mm \times \sqrt{5.0355} = 19.1mm$$

Sample chamber-nozzle geometry calculation

For this simple engine, some characteristics properties are

- Geometry is linear (which is about 93% efficiency)
- Combustion chamber-throat concentration: 2.5
  $d_{chamber} = d_{throat} \times \sqrt{contraction\ ratio} = 8.5mm \times \sqrt{2.5} = 13.44mm$
- Combustion chamber length/diameter ratio: 4
  $L_{chamber} = d_{chamber} \times \frac{L}{D} = 13.44mm \times 4 = 53.76mm$
- Converging half angle: $\Theta = 30^\circ$
- Diverging half angle: $\alpha = 15^\circ$

Sample chamber-nozzle geometry calculation



Sample chamber-nozzle thickness calculation

Due to manufacturing constraints, the nozzle assembly is 3D-printed, AlSi10Mg alloy. This material has tensile strength of 210MPa. Safety factor for this engine is at least $SF = 2$. There are 3 consumptions

- The material loss 30% percent of its strength at 782.05K
- The thickness is the same for the whole assembly. The 3D-printed part strength is homogeneous in all directions
- The stress is highest at combustion chamber

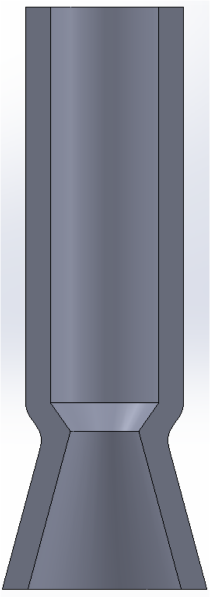
Hoop (Circumferential) Stress for thin-walled vessel

$$\sigma_h \times SF = \frac{pd}{2t}$$

Longitudinal (Axial) Stress for thin-walled vessel

$$\sigma_l \times SF = \frac{pd}{4t}$$

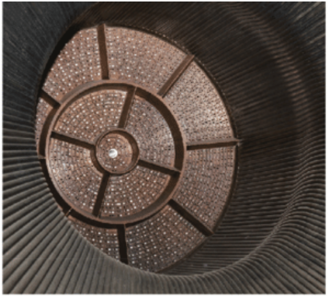
$\Rightarrow t = 2.5mm$ with SF $= 2.3$



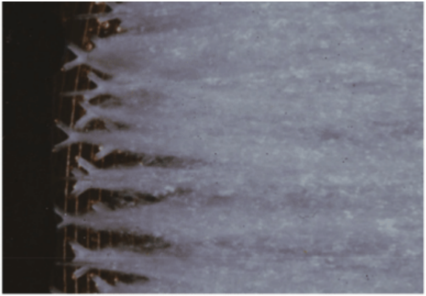
Simple injector

A rocket engine injectors has two main contributions

- Mix the fuel and oxidizer
- Act as a flow-restricting orifice to prevent the combustion gas to blow-back into feed line. This is different from normal injector that used in internal-combustion engine as rocket engine is a continuously flow engine



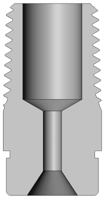
(a) Injection faceplate of F-1 engine



(b) Water flow test of main combustion chamber of F-1 engine

- For our simple H_2O_2 engine, there is no mixing as our engine is monopropellant.
  $\rightarrow$ The nozzle could be a simple spray nozzle.
- For combustion stability, the pressure drop through the injector is 15% to 20%
  $\rightarrow$ For combustion pressure of 30 bar, the pressure drop is 6 bar.
- Also, as the mass flow rate is $\dot{m} = 0.206 kg/s$ and $\rho_{H_2O_2} = 1.4 kg/l$, the volumetric flow rate is $\dot{V} = 0.2884 \mathrm{l/s} = 4.57 gpm$ (gallon per minute)

From volumetric flow rate and pressure drop, we could find the spray nozzle orifice diameter (ISO 5167 standard for spray nozzle). An online tool: `http://www.pressure-drop.com/Online-Calculator/index.html`

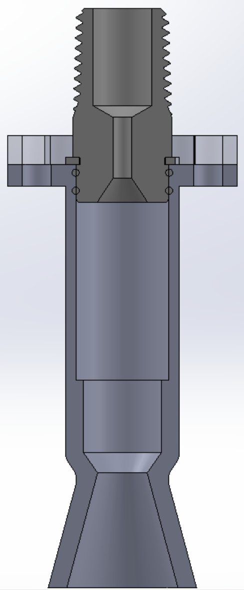


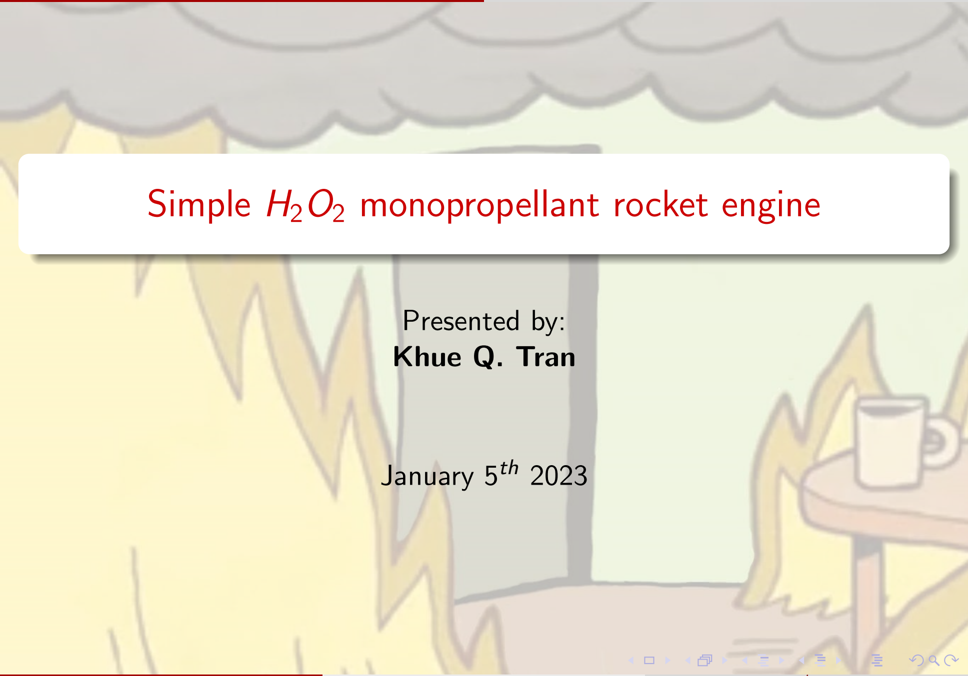
McMaster-Carr nozzle

Instead of calculating, the flowrate and pressure drop could be supplied by the spray nozzle manufacture. In this case, we chose Mcmaster-Carr 32885K571 spray nozzle (`https://www.mcmaster.com/32885K571/`)

- This spray nozzle is stainless steel
- This spray nozzle is 120°, full-cone type
- Flow rate at 100psi pressure drop is 4.6gpm

Overall CAD file





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