Chemical/Nuclear Propulsion
Space System Design, MAE 342, Princeton University
Robert Stengel

- Thermal rockets
- Performance parameters
- Propellants and propellant storage

Copyright 2016 by Robert Stengel. All rights reserved. For educational use only.
http://www.princeton.edu/~stengel/MAE342.html

Chemical (Thermal) Rockets

- Liquid/Gas Propellant
  - Monopropellant
    - Catalytic ignition / chemical decomposition
    - Cold gas
  - Bipropellant
    - Separate oxidizer and fuel
    - Hypergolic (spontaneous) ignition
    - External ignition
    - Storage
      - Ambient temperature and pressure
      - Cryogenic
      - Pressurized tank
  - Throttllable
  - Start/stop cycling

- Solid Propellant
  - Mixed oxidizer and fuel
  - External ignition
  - Burn to completion

- Hybrid Propellant
  - Liquid oxidizer, solid fuel
  - Throttllable
  - Start/stop cycling
Cold Gas Thruster
*(used with inert gas)*

Moog Divert/Attitude Thruster and Valve

Monopropellant Hydrazine Thruster

- Catalytic decomposition produces thrust
- Reliable
- Low performance
- Toxic

Aerojet Rocketdyne
Bi-Propellant Rocket Motor
Thrust / Motor Weight ~ 70:1

Hypergolic, Storable Liquid-Propellant Thruster

- Spontaneous combustion
- Reliable
- Corrosive, toxic

Titan 2
Pressure-Fed and Turbopump Engine Cycles

Pressure-Fed Rocket Cycle

Gas-Generator Rocket Cycle, with Nozzle Cooling

Staged Combustion Engine Cycles

Staged Combustion Rocket Cycle

Full-Flow Staged Combustion Rocket Cycle
German V-2 Rocket Motor, Fuel Injectors, and Turbopump

Combustion Chamber Injectors
**H-1 Engine**

- **THrust (SEA Level)**
  - SA-201 thru SA-205: 200,000 LB
  - SA-206 & Subsequent: 205,000 LB

- **THrust DURation**
  - SA-201 thru SA-205: 155 SEC
  - SA-206 & Subsequent: 155 SEC

- **SPECIFIC IMPULSE (LB-SEC/LB)**
  - SA-201 thru SA-205: 260.5 MIN
  - SA-206 & Subsequent: 261.0 MIN

- **ENGINE WT DRY (INBD)**
  - SA-201 thru SA-205: 1,830 LB
  - SA-206 & Subsequent: 2,100 LB

- **ENGINE WT BURNOUT (INBD)**
  - SA-201 thru SA-205: 2,200 LB
  - SA-206 & Subsequent: 2,200 LB

- **EXIT-TO-THROAT AREA RATIO**
  - SA-201 thru SA-205: 8 TO 1
  - SA-206 & Subsequent: 8 TO 1

- **PROPELLANTS**
  - LOX & RP-1

- **MIXTURE RATIO**
  - 2.23±2%

**Contractor:** NAA/Rocketdyne

**Vehicle Application:** Saturn IB/S-IB Stage (Eight Engines)

---

**F-1 Engine**

- **THrust (SEA Level)**
  - SA-501 thru SA-505: 1,500,000 LB
  - SA-506 & Subsequent: 1,522,000 LB

- **THrust DURation**
  - SA-501 thru SA-505: 150 SEC
  - SA-506 & Subsequent: 165 SEC

- **SPECIFIC IMPULSE (LB-SEC/LB)**
  - SA-501 thru SA-505: 260 SEC MIN
  - SA-506 & Subsequent: 263 MIN

- **ENGINE WT DRY**
  - SA-501 thru SA-505: 18,416 LB
  - SA-506 & Subsequent: 18,500 LB

- **ENGINE WT BURNOUT**
  - SA-501 thru SA-505: 20,096 LB
  - SA-506 & Subsequent: 20,180 LB

- **EXIT-TO-THROAT AREA RATIO**
  - SA-501 thru SA-505: 16 TO 1
  - SA-506 & Subsequent: 16 TO 1

- **PROPELLANTS**
  - LOX & RP-1

- **MIXTURE RATIO**
  - 2.27±2.2%

**Contractor:** NAA/Rocketdyne

**Vehicle Application:** Saturn V/S-IC Stage (Five Engines)
Origins of the F-1

- Air Force legacy (1955)
  - Design undertaken before vehicle or mission were identified
- Big engine, big problems
  - 16:1 nozzle expansion
  - 6.67 MN thrust
- F-1 turbopumps
  - Oxygen: 24,811 gal/min
  - RP-1: 15,741 gal/min
- F-1 injector
- Combustion instability
  - Significant theoretical work by Luigi Crocco and David Harrje, Princeton
USSR RD-107/8 Rocket Motors

RD-107
4 combustion chambers, 2 verniers

RD-108
4 combustion chambers, 4 verniers

R-7 Base
4-RD-107, 1-RD-108

RD-180 Engine Schematic

(used on Atlas V)
Special Shuttle Main Engine (RS-25)

SpaceX Merlin Family

Merlin 1A (ablation nozzle)  Merlin 1C (vacuum nozzle)  Merlin 1D (throttatable)

Roll control from turbine exhaust
Blue Origin BE-4

- LOX/Liquefied natural gas
- United Launch Alliance has chosen as motor for the Vulcan launch vehicle
- Thrust = 2.5 MN (550,000 lb)

RD-181 and RD-191

RD-181

- to be used on Orbital-ATK Antares

RD-191

- to be used on NPO Energomash Angara
Solid-Fuel Rocket Motor

[Diagram of a solid-fuel rocket motor with labels for different components such as retainer, aluminum chamber, propellant, and 0.01C-in. liner, as well as a graphite nozzle.]
Solid-Fuel Rocket Motor

**Thrust is proportional to burning area**

**Rocket grain patterns affect thrust profile**

**Propellant chamber must sustain high pressure and temperature**

**Environmentally unfriendly exhaust gas**

---

Hybrid-Fuel Rocket Motor

- **SpaceShipOne motor**
  - Nitrous oxide
  - Hydroxy-terminated polybutadiene (HTPB)

- **Issues**
  - Hard start
  - Blow back
  - Complete mixing of oxidizer and fuel toward completion of burn
Rocket Thrust

\[
\text{Thrust} = \dot{m}_{\text{propellant}} V_{\text{exhaust}} + A_{\text{exit}} \left( p_{\text{exit}} - p_{\text{ambient}} \right) \equiv \dot{m} c_{\text{eff}}
\]

\[
c_{\text{eff}} = \frac{\text{Thrust}}{\dot{m}} = \text{Effective exhaust velocity}
\]

\[\dot{m} \equiv \text{Mass flow rate of on-board propellant}\]

Specific Impulse

\[
I_{sp} = \frac{\text{Thrust}}{\dot{m} g_o} = \frac{c_{\text{eff}}}{g_o}, \quad \text{Units} = \frac{m}{s} = \text{seconds}
\]

\[g_o \equiv \text{Gravitational acceleration at earth's surface}\]

- \(g_o\) is a normalizing factor for the definition
- Chemical rocket specific impulse (vacuum)
  - Solid propellants: < 295 s
  - Liquid propellants: < 510 s

- Space Shuttle Specific Impulses
  - Solid boosters: 242-269 s
  - Main engines: 455 s
  - OMS: 313 s
  - RCS: 260-280 s
Specific Impulse

Specific impulse is a product of characteristic velocity, $c^*$, and rocket thrust coefficient, $C_F$

$$I_{sp} = \frac{\text{Thrust}}{\dot{m} \ g_o} = \frac{c_{\text{eff}}}{g_o}$$

$$= C_F \frac{c^*}{g_o}$$

$$= \frac{V_{\text{exhaust}}}{g_o} \quad \text{when} \ C_F = 1, \ p_e = p_{\text{ambient}}$$

- Characteristic velocity is related to
  - combustion chamber performance
  - propellant characteristics
- Thrust coefficient is related to
  - nozzle shape
  - exit/ambient pressure differential

The Rocket Equation

Ideal velocity increment of a rocket stage, $\Delta V_i$ (gravity and aerodynamic effects neglected)

$$\frac{dV}{dt} = \frac{\text{Thrust}}{m} = \frac{\dot{m} \ c_{\text{eff}}}{m} = -\frac{\frac{dm}{dt}}{m} I_{sp} g_o$$

$$V_f = I_{sp} g_o \int_{m_i}^{m_f} \frac{dm}{m} = -I_{sp} g_o \ln m |_{m_i}^{m_f}$$

$$\left( V_f - V_i \right) \equiv \Delta V_i = I_{sp} g_o \ln \left( \frac{m_i}{m_f} \right) \equiv I_{sp} g_o \ln \mu$$
Volumetric Specific Impulse

Specific impulse

\[ \Delta V_I = I_{sp} g_o \ln \mu = I_{sp} g_o \ln \left( \frac{m_{\text{final}} + m_{\text{propellant}}}{m_{\text{final}}} \right) = I_{sp} g_o \ln \left( 1 + \frac{m_{\text{propellant}}}{m_{\text{final}}} \right) \]

\[ = I_{sp} g_o \ln \left( 1 + \frac{\text{Density}_{\text{propellant}} \cdot \text{Volume}_{\text{propellant}}}{m_{\text{final}}} \right) \]

\[ \approx g_o I_{sp} \left( \frac{\rho_{\text{propellant}} \cdot \text{Vol}_{\text{propellant}}}{m_{\text{final}}} \right) = g_o \left( I_{sp} \rho_{\text{propellant}} \right) \frac{\text{Vol}_{\text{propellant}}}{m_{\text{final}}} \]

Volumetric specific impulse

\[ I_{sp_{\text{vol}}} \triangleq VI_{sp} = I_{sp} \rho_{\text{propellant}} \]

Volumetric Specific Impulse

- For fixed volume and final mass, increasing volumetric specific impulse increases ideal velocity increment

<table>
<thead>
<tr>
<th></th>
<th>Density, g/ cc</th>
<th>Vlsp, s (g/cc), Isp, s, SL</th>
<th>Vlsp, s (g/cc), Isp, s, vac vac</th>
</tr>
</thead>
<tbody>
<tr>
<td>LOX/Kerosene</td>
<td>1.3</td>
<td>265</td>
<td>345</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>LOX/LH2 (Saturn V)</td>
<td>0.28</td>
<td>360</td>
<td>101</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>424</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>119</td>
</tr>
<tr>
<td>LOX/LH2 (Shuttle)</td>
<td>0.28</td>
<td>390</td>
<td>109</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>455</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>127</td>
</tr>
<tr>
<td>Shuttle Solid Booster</td>
<td>1.35</td>
<td>242</td>
<td>327</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>262</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>354</td>
</tr>
</tbody>
</table>

- Saturn V Specific Impulses, vacuum (sea level)
  - 1st Stage, 5 F-1 LOX-Kerosene Engines: 304 s (265 s)
  - 2nd Stage, 5 J-2 LOX-LH2 Engines: 424 s (~360 s)
  - 3rd Stage, 1 J-2 LOX-LH2 Engine: 424 s (~360 s)
Typical Values of Chemical Rocket Specific Impulse

- Chamber pressure = 7 MPa (low by modern standards)
- Expansion to exit pressure = 0.1 MPa

<table>
<thead>
<tr>
<th>Liquid-Fuel Rockets</th>
<th>V\text{Isp}, \text{kg-s/m}^3 \times 10^3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Monopropellant</td>
<td></td>
</tr>
<tr>
<td>Hydrogen Peroxide</td>
<td>165, 238</td>
</tr>
<tr>
<td>Hydrazine</td>
<td>199, 201</td>
</tr>
<tr>
<td>Nitromethane</td>
<td>255, 290</td>
</tr>
<tr>
<td><strong>Bipropellant</strong></td>
<td></td>
</tr>
<tr>
<td>Fuel</td>
<td>Oxidizer</td>
</tr>
<tr>
<td>Kerosene</td>
<td>Oxygen</td>
</tr>
<tr>
<td></td>
<td>Flourine</td>
</tr>
<tr>
<td></td>
<td>Red Fuming</td>
</tr>
<tr>
<td></td>
<td>Nitric Acid</td>
</tr>
<tr>
<td>Hydrogen</td>
<td>Oxygen</td>
</tr>
<tr>
<td></td>
<td>Flourine</td>
</tr>
<tr>
<td></td>
<td>Nitrogen</td>
</tr>
<tr>
<td>UDMH</td>
<td>Tetroxide</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Solid-Propellant Rockets</th>
<th>V\text{Isp}, \text{kg-s/m}^3 \times 10^3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Double-Base</td>
<td></td>
</tr>
<tr>
<td>AFU</td>
<td>196, 297</td>
</tr>
<tr>
<td>ATN</td>
<td>235, 376</td>
</tr>
<tr>
<td>JPN</td>
<td>250, 405</td>
</tr>
<tr>
<td><strong>Composite</strong></td>
<td></td>
</tr>
<tr>
<td>JPL 540A</td>
<td>231, 383</td>
</tr>
<tr>
<td>TRX-H609</td>
<td>245, 431</td>
</tr>
<tr>
<td>PBAN (SSV)</td>
<td>260, 461</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Hybrid-Fuel Rocket</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuel</td>
<td>Oxidizer</td>
</tr>
<tr>
<td>HTPB</td>
<td>N2O</td>
</tr>
</tbody>
</table>

Exhaust Velocity vs. Thrust Acceleration
Rocket Characteristic Velocity, $c^*$

$$c^* = \frac{1}{\Gamma} \sqrt{\frac{R_o T_c}{M}}$$

where $\Gamma = \sqrt{\gamma \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma + 1}{2(\gamma - 1)}}}$

- $R_o$ = universal gas constant = $8.3 \times 10^3$ kg m$^2$/s$^2$ °K
- $T_c$ = chamber temperature, °K
- $M$ = exhaust gas mean molecular weight
- $\gamma$ = ratio of specific heats (~1.2-1.4)

Rocket Characteristic Velocity, $c^*$

$$c^* = \frac{p_c A_t}{\dot{m}} = \text{exhaust velocity if } C_F = 1$$
Rocket Thrust Coefficient, $C_F$

$$C_F = \frac{\text{Thrust}}{p_c A_t} = \lambda \Gamma \left[ \frac{2\gamma}{\gamma - 1} \left( 1 - \left( \frac{p_e}{p_c} \right)^{(\gamma - 1)/\gamma} \right) + \left( \frac{p_e - p_{\text{ambient}}}{p_c} \right) \right] \frac{A_e}{A_t}$$

Thrust = $\lambda \dot{m} v_e + A_e \left( p_e - p_{\text{ambient}} \right)$

$\lambda$: reduction ratio (function of nozzle shape)

$C_F$ typically 0.5 - 2

Thrust Coefficient, $C_F$, vs. Nozzle Expansion Ratio
Mixture Ratio, $r$

$$r = \frac{\dot{m}_{\text{oxidizer}}}{\dot{m}_{\text{fuel}}}; \quad \dot{m}_{\text{fuel}} = \frac{\dot{m}_{\text{total}}}{1 + r}; \quad "\text{leaner}" < r < "\text{richer}"
$$

- **Stoichiometric mixture:** complete chemical reaction of propellants
- **Specific impulse maximized** with lean mixture ratio, $r$ (i.e., below stoichiometric maximum)

**Effect of Pressure Ratio on Mass Flow**

In choked flow, mass flow rate is maximized

$$\dot{m} = \frac{T_p A_r}{\sqrt{R_o T_c} / M}$$

Choked flow occurs when

$$\frac{p_e}{p_c} \leq \left(\frac{2}{\gamma + 1}\right)^{\gamma/\gamma - 1} \approx 0.53$$
Combustion Instability

- Complex mix of species, phases, pressures, temperatures, and flows
- Cavity resonance

Harrje, NASA SP-194, 1972

Combustion Instability
Shock Diamonds

When $p_e \neq p_a$, exhaust flow is over- or underexpanded. Effective exhaust velocity is less than the maximum value.

Rocket Nozzles

Types of rocket nozzles:

- a. Conventional conical nozzle
- b. Conventional bell-type nozzle
- c. Spike-type nozzle
- d. Plug-type nozzle
- e. Expansion-deflection type nozzle

https://www.youtube.com/watch?v=qiMSko4HBe8
Rocket Nozzles

- Expansion ratio, $A_e/A_t$, chosen to match exhaust pressure to average ambient pressure
  - Ariane rockets: Viking V for sea level, Viking IV for high altitude

- Rocket nozzle types
  - DeLaval nozzle
  - Isentropic expansion nozzle
  - Spike/plug nozzles
  - Expansion-deflection nozzle
Linear Spike/Plug Nozzles

Aerospike/Bell Nozzle Exhaust Plume Comparisons

Sea Level
- \( P_a = P_{\text{design}} \)

Design Altitude
- \( P_a = P_{\text{design}} \)

High Altitude
- \( P_a < P_{\text{design}} \)

Throttling, Start/Stop Cycling

CECE Demonstrator Pintle Injector

Pintle Injector Operation
Reaction Control Thrusters

- Direct control of angular rate
- Unloading momentum wheels or control-moment gyros
- Reaction control thrusters are typically on-off devices using
  - Cold gas
  - Hypergolic propellants
  - Catalytic propellant
  - Ion/plasma rockets

- Thrusters commanded in pairs to cancel velocity change

Apollo Lunar Module RCS  Space Shuttle RCS  RCS Thruster

Divert and Attitude Control Thrusters

https://www.youtube.com/watch?v=W8efpDBvTDE
https://www.youtube.com/watch?v=71qgl6bddM8
https://www.youtube.com/watch?v=KBMU6l6GsdM
https://www.youtube.com/watch?v=JURQYt669_g
Nuclear Propulsion

- Nuclear reaction produces thermal energy to heat inert working fluid
  - Solid core
  - Liquid core
  - Gaseous core
- High propellant temperature leads to high specific impulse
- Working fluid chosen for low molecular weight and storability

Solid-Core Nuclear Rocket

- Operating temperature limited by
  - melting point of reactor materials
  - cracking of core coating
  - matching coefficients of expansion
- Possible propellants: hydrogen, helium, liquid oxygen, water, ammonia
- $I_{sp} = 850 - 1,000$ sec
- $T/W \sim 7:1$
Project Rover, 1955-1972

NERVA Rocket, $I_{sp} \sim 900$ sec

NERVA-Powered Mars Mission

Kiwi-B4-A Reactor/Rocket
Liquid/Particle-Core Nuclear Rocket

- Nuclear fuel mixed with working fluid
- In principle, could operate above melting point of nuclear fuel
- \( I_{sp} \approx 1,300 – 1,500 \) sec
- Conceptual
- Massive radioactive waste

Open-Cycle Gas Core Nuclear Rocket

- Toroidal circulation of working fluid confines nuclear fuel to center
- Fuel does not touch the wall
- Conceptual
- Massive radioactive waste
- \( I_{sp} \approx 3,000 – 5,000 \) sec
Closed-Cycle Gas Core Nuclear Rocket

- “Nuclear light bulb”
- Nuclear fuel contained in quartz container
- $I_{sp} \sim 1,500 - 2,000$ sec
- Conceptual

Nuclear-Pulse ("Explosion") Rocket - Project Orion

“Physics packages” ejected behind the pusher plate

https://en.wikipedia.org/wiki/Project_Orion_(nuclear_propulsion)
Next Time:
Launch Vehicles

SUPPLEMENTAL MATERIAL
**Propellant Tanks**

Propellant must be kept near the exit duct without bubbles during thrusting.

---

**Ion/Plasma Thrusters**

<table>
<thead>
<tr>
<th>Engine</th>
<th>Propellant</th>
<th>Required power</th>
<th>Specific impulse</th>
<th>Thrust</th>
</tr>
</thead>
<tbody>
<tr>
<td>NSTAR</td>
<td>Xenon</td>
<td>2.3 kW</td>
<td>3,300 to 1,700 s</td>
<td>92 max</td>
</tr>
<tr>
<td>NEXT</td>
<td>Xenon</td>
<td>6.9 kW</td>
<td>4,300 mN</td>
<td>92 max</td>
</tr>
<tr>
<td>HiPEP</td>
<td>Xenon</td>
<td>20–50 kW</td>
<td>6,000–9,000 s</td>
<td>460–670 mN</td>
</tr>
<tr>
<td>Hall effect</td>
<td>Xenon</td>
<td>25 kW</td>
<td>3,250 mN</td>
<td>950 mN</td>
</tr>
<tr>
<td>FEEP</td>
<td>Liquid Cesium</td>
<td>6×10–5–0.06 kW</td>
<td>6,000–10,000 s</td>
<td>0.001–1 mN</td>
</tr>
<tr>
<td>VASIMR</td>
<td>Argon</td>
<td>200 kW</td>
<td>3,000–12,000 s</td>
<td>~5,000 mN</td>
</tr>
<tr>
<td>DS4G</td>
<td>Xenon</td>
<td>250 kW</td>
<td>19,300 mN</td>
<td>2,500 max</td>
</tr>
</tbody>
</table>
Variable Specific Impulse Magnetoplasma Rocket (VASIMR)

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Required power kW</th>
<th>Specific impulse s</th>
<th>Thrust mN</th>
</tr>
</thead>
<tbody>
<tr>
<td>Argon</td>
<td>200</td>
<td>3,000–12,000</td>
<td>~5,000</td>
</tr>
</tbody>
</table>

DAWN Spacecraft

<table>
<thead>
<tr>
<th>Engine</th>
<th>Propellant</th>
<th>Required power kW</th>
<th>Specific impulse s</th>
<th>Thrust mN</th>
</tr>
</thead>
<tbody>
<tr>
<td>NSTAR</td>
<td>Xenon</td>
<td>2.3</td>
<td>3,300 to 1,700</td>
<td>92 max</td>
</tr>
</tbody>
</table>