AA 284a
Advanced Rocket Propulsion

Lecture 12
Liquid Rocket Propulsion

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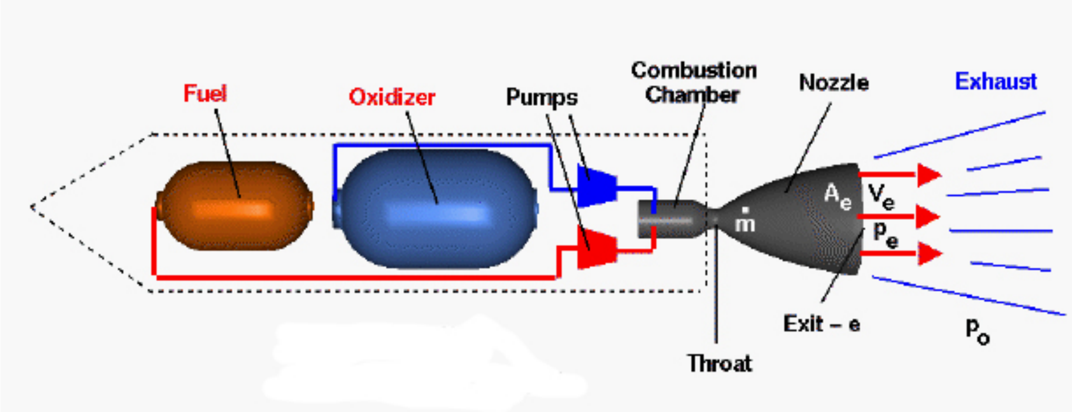
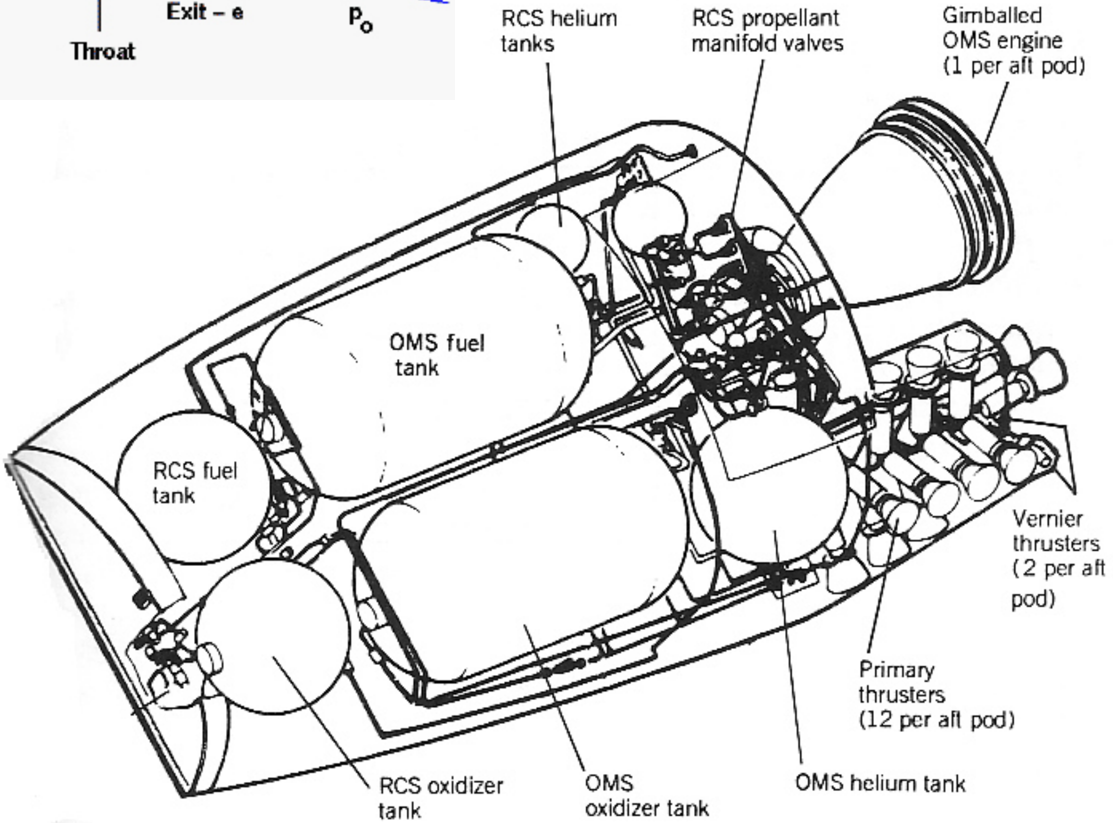
Liquid Rocket Schematic



Liquid Systems:

- Monopropellant vs. Bipropellant
- Pressure fed vs pump fed
- Propellants: Hypergolic

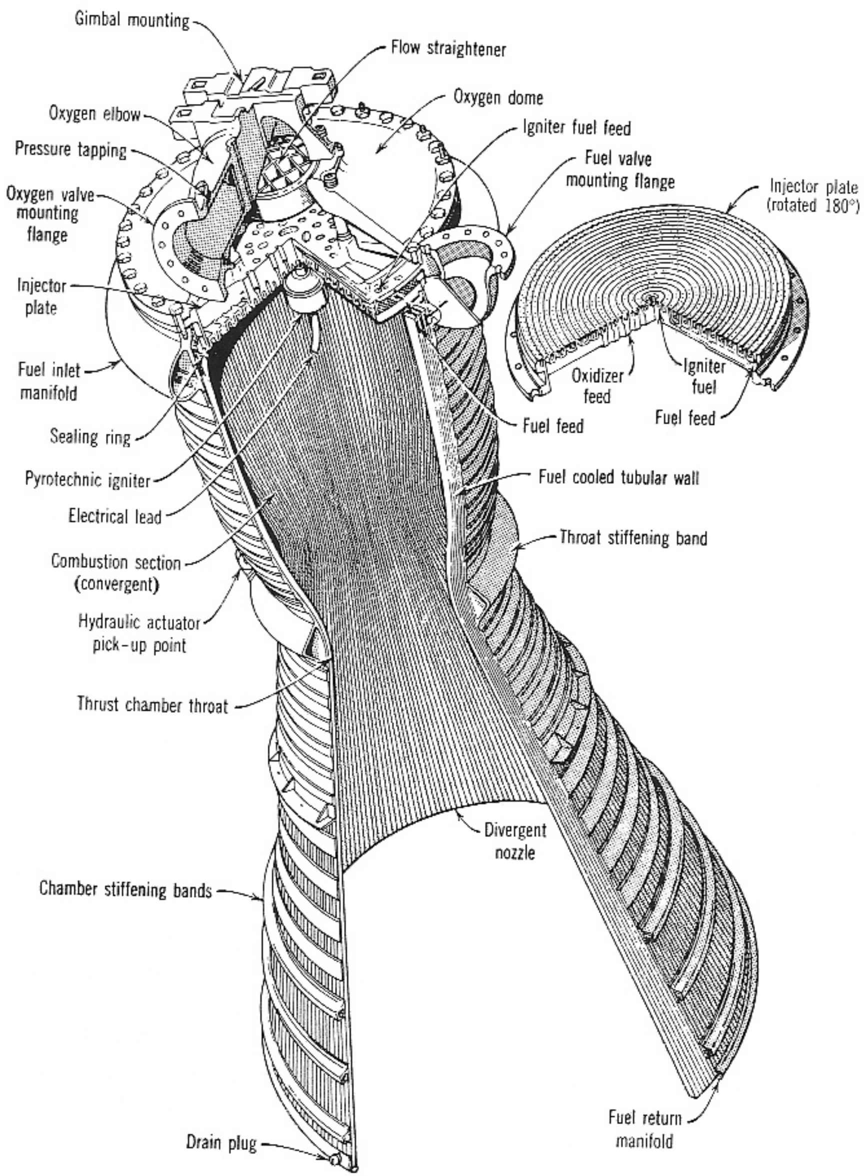


Liquid Rocket Types –Based on Propellants

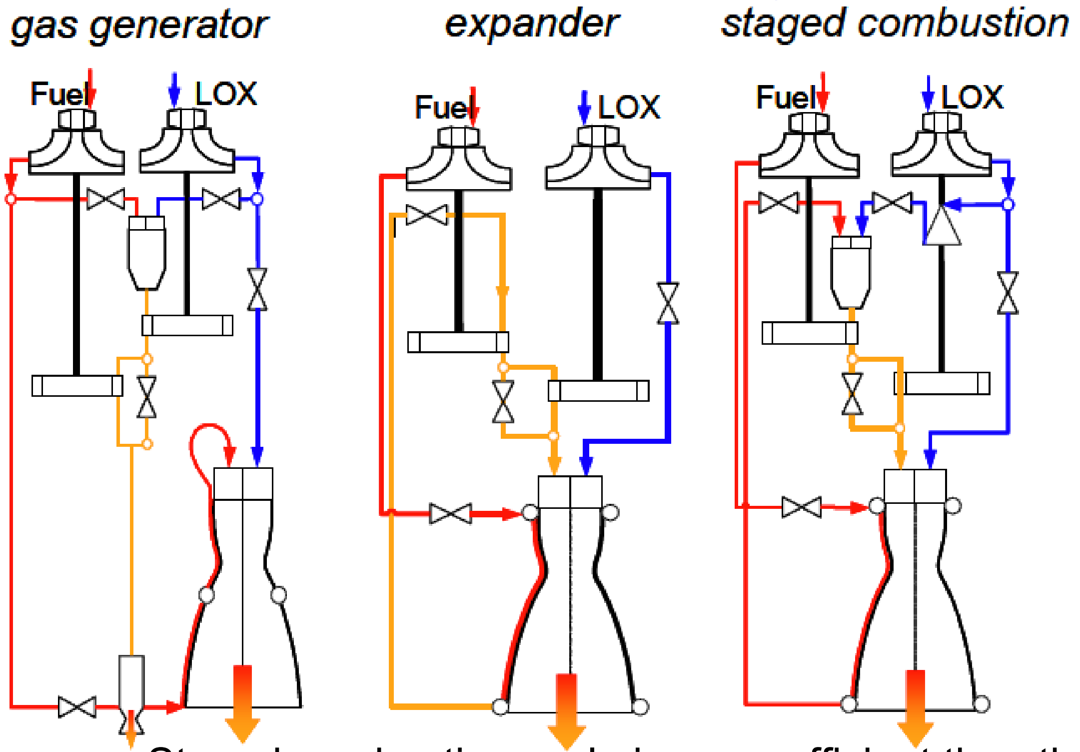
- Monopropellant systems:
  - Single liquid, simpler system
  - Decompose over a catalyst bed
  - Low Isp performance
  - H_2O_2, hydrazine, N_2O
  - Used in satellite propulsion, RCS
- Cryogenic Engines
  - LOX/LH2
  - Expensive, but very high Isp
  - Upper states
- LOX/Kerosene
  - Decent Isp and density
  - Hard to stabilize
  - A lot of launch vehicles use LOX/Kerosene engines
- Storable
  - NTO/Hydrazine or derivative
  - Toxic, not favored in modern systems
- LOX/Methane
  - Up and coming technology
  - Good compromise between kerosene and H2

Liquid Rocket Thrust Chamber

- Thrust chamber includes
  - Injector
  - Combustion chamber
  - Nozzle
  - Ignition system
- Introduce the oxidizer and fuel in liquid or gaseous phase
- Control the mass flow rate of oxidizer and fuel
- Vaporize mix and react the components
- Expel though the nozzle



Pump Fed Liquid Rocket Cycle Types



- Need a working gas to drive the turbine of the turbopump system
- Vaporize the propellant to drive the turbine by
  - Combustion (very lean or very rich to limit temperature)
  - Use the fuel from regenerative cooling
- Turbine inlet temperature is typically around 800-850 K

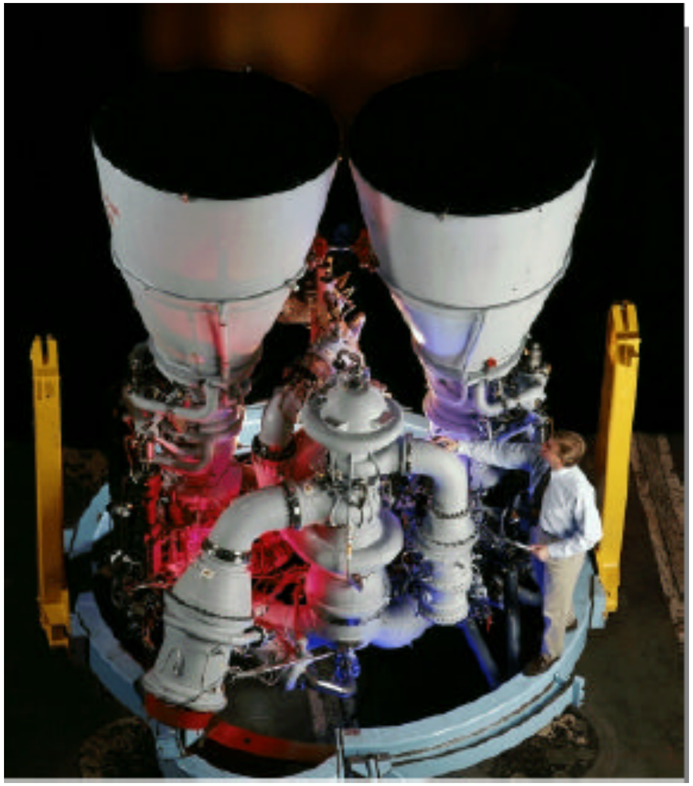
- Staged combustion cycle is more efficient than the gas generator cycle (older systems such as F1 use the gas generation cycle).
- Fuel rich pre burners are easier to develop but not as desirable as the oxidizer rich burners
  - Soot deposit on turbine blades
  - Much more oxidizer than fuel (O/F >1)
- Only LOX rich staged combustion systems are Russian engines
- LOX/H2 systems typically utilize fuel rich pre burners

Liquid Rocket Engines

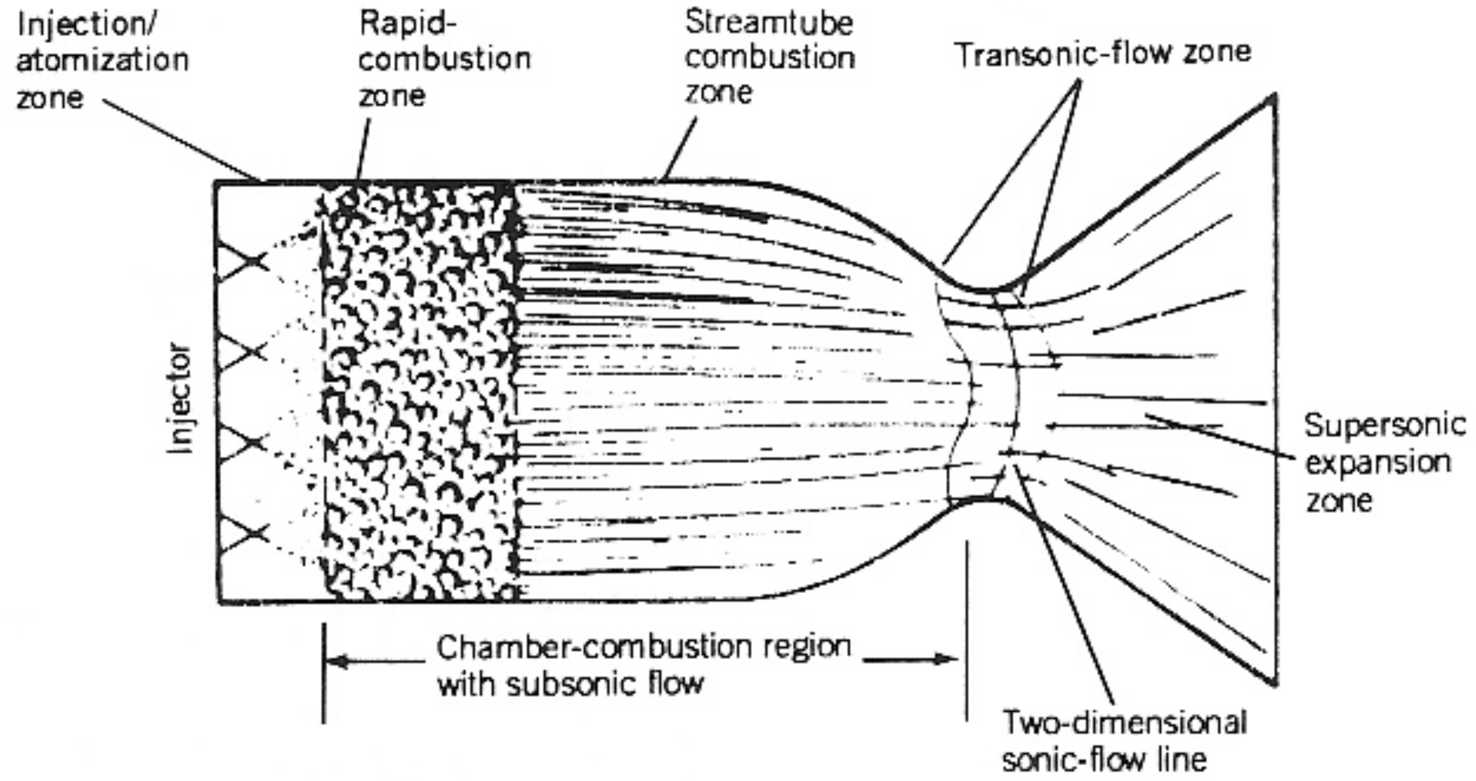
Engine	Thrust, klb	Isp, sec	Propellants	Design Year
F-1 (Saturn V- First Stage)	1,500 (SL)	265 (SL)	LOX/RP-1	1959
J-2 (Saturn V-Upper stages)	230 (vac)	425 (vac)	LOX/H2	1960
RS-27A (Old Delta Booster)	200 (SL)	255 (SL)	LOX/RP-1	1987
MA-5A (Old Atlas Booster)	430 (SL)	265 (SL)	LOX/RP-1	1988
SSME (RS-24) (Space Shuttle Main Engine)	512 (vac)	453 (vac)	LOX/H2	1972
SE-10 (Lunar Module Descent)	Max 10.5 (vac)	305 (vac)	N2O4/N2H4+ UDMH	1963
RD-180 (Atlas V Booster)	900 (SL)	311 (SL)	LOX/Kerosene	Late 1970’s

A High Performance Engine: RD-180

- High performance LOX/kerosene engine
- Built and marketed by RD AMROSS
  - 50% NPO Energomash (Russian)
  - 50% Pratt and Whitney (US)
- Derived from the Russian engine RD-170 (developed for the Energia/Buran system)
- Used in Atlas III and Atlas V launchers
- Staged combustion cycle – Oxidizer rich pre burner
- Vacuum Isp: 337.8 sec
- Nozzle expansion ratio: 36.4
- Chamber pressure: 257 atm
- O/F: 2.71
- Thrust: 0.9 Mlbf
- Throttling range: 47% to 100%



Liquid Rocket Combustion

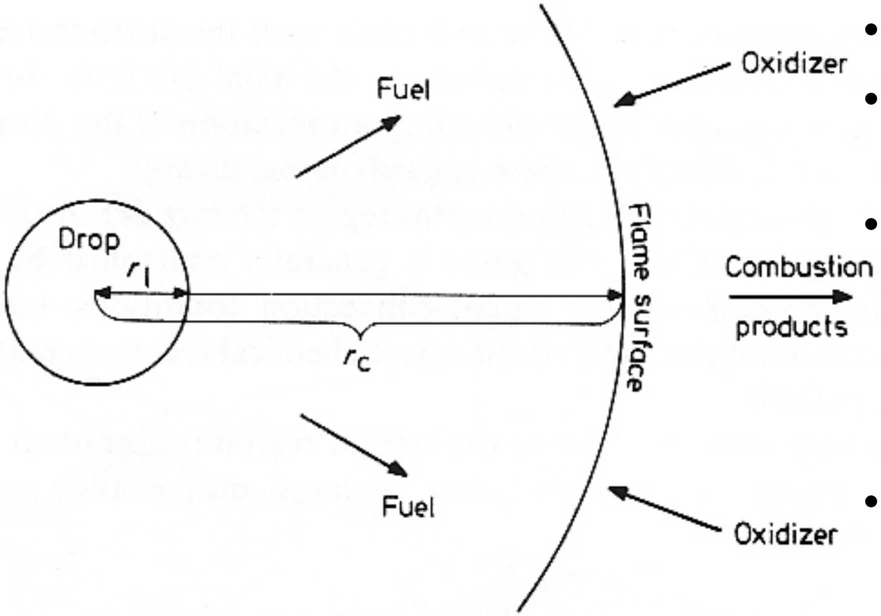


- Monopropellant vs Bipropellant
- Low Residency Times: <10 msec
- High Volumetric Heat Release: 370 Mega-Watt/m^3

Liquid Rocket Combustion

- Combustion Zones:
  - Injection Atomization Zone:
    - Heterogeneous mixture (liquid/gas)
    - Low velocities
    - Relatively cool
    - Main process evaporation of the droplets
  - Rapid Combustion Zone:
    - Intensive fast combustion reactions
    - Large increase in velocity (due to gas/liquid volume increase)
    - Combustion is an inherently unsteady process (small explosions)
  - Stream Tube Combustion Zone:
    - High gas velocity, small residence time
    - Combustion reactions at a slow rate
    - Stream tubes are formed
    - Limited transport across the stream tubes
- Boundaries between zones are fuzzy
- Combustion models and design tools are incomplete

Droplet Evaporation Model



- Diffusion flame
- Transport through molecular diffusion + convection
- Diffusion flame, O/F stoichiometric
- Evaporation of a burning droplet in quiescent environment
- D^2 law for droplet evaporation

$$\frac{dD^2}{dt} = -K = -\frac{8\lambda}{C_p \rho}\ln(1+B)$$

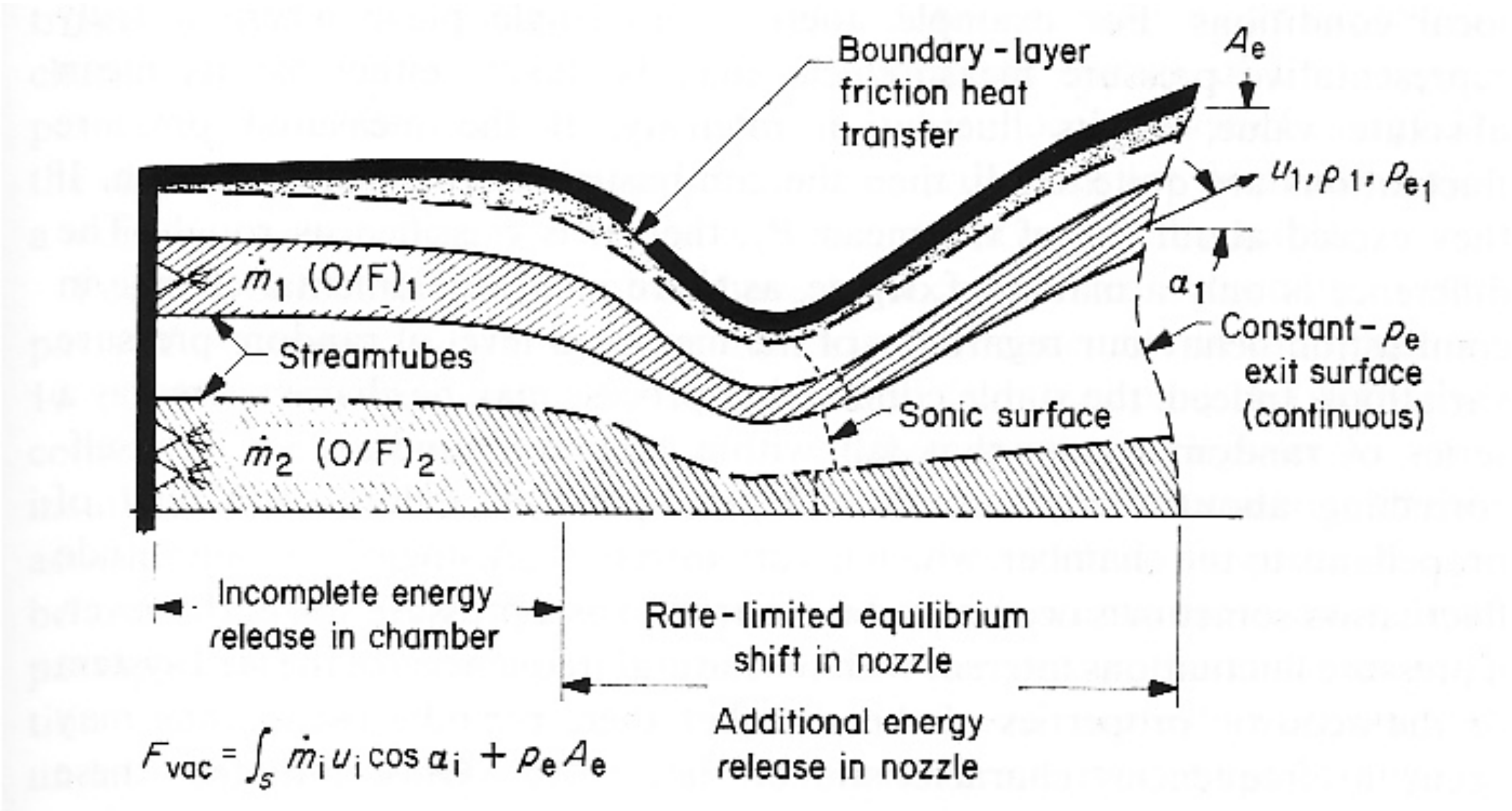
- The Spalding number is

$$B = \frac{1}{h_v}\left[C_p\left(T_\infty - T_l\right) + \frac{Q_r Y_{ox,\infty}}{(O/F)_{stoic}}\right]$$

- As a first order approximation droplet surface temperature can be taken as the boiling temperature

- See “Combustion” by I. Glassman for a rigorous derivation

Liquid Rocket Combustion-Stream Tube Model



- One dimensional model
- Different steam tubes could be at different O/F ratios. Introduces a loss
- Boundary layer heat transfer can be modeled by this model

Liquid Engine Design Issues

- Must consider **performance, stability and compatibility** simultaneously
- Performance (Isp):
  - Theoretical value (for a given average O/F)
  - Losses:
    - Thermal (heat transfer)
    - Combustion efficiency (mixing/kinetics)
    - Nonuniform propellant distribution (mixing)
    - Boundary layer (friction)
    - Geometry (nozzle divergence)
    - Particle lag (two phase flow)
  - The Isp efficiencies of the modern engines are fairly high: 95-98 %
  - Operational conditions are critical
    - Low pressures: Kinetics
    - Low Thrusts: Boundary layer
    - Small Volume: Low combustion efficiency

Liquid Engine Design Issues

- Stability:
  - Liquid engine combustion is an inherently unstable process
  - Metastable (minimize the amplitude of the fluctuations)
  - Stability fixes:
    - Chemical
    - Aerodynamic
    - Mechanical
- Compatibility
  - Environmental components
    - Thermal (Heat Transfer)
    - Chemical (Reactions)
    - Gas dynamic (Erosion)

Liquid Engine Design Issues

- Control Variables:
  - Feed system dynamics
  - Injector design, injector pressure drop
  - Combustion chamber geometry
    - Combustion chamber volume, Vc (includes the convergent part of the nozzle). Define L*

$$L^* = V_c / A_t$$

    - Residence time in the chamber is

$$\tau = L^* / c^*$$

    - Efficiency increases with L*. More time for atomization, vaporization, mixing and reacting
    - For typical liquid systems L* ranges 0.8-3 meters
    - Baffles
    - Absorption cavities
  - Propellant additive selection (i.e. Hypergolic propellants)

Liquid Engine Design Issues

- Combustion chamber and nozzle walls must be cooled
  - Regenerative cooling
    - Cooling jacket, some of the heat is used to warm the fuel
  - Ablative cooling
    - Carbon graphite, phenolic
  - Film cooling
    - Injector face cooling
  - Radiation cooling
    - Used in small engines and monopropellant systems
    - Niobium, Rhenium coated Inconel
  - Combination
- As the combustion chamber size reduces
  - Heating intensity increases
  - Surface area decreases
- Nozzle throat has the maximum heat transfer
- Total pressure loss in the chamber
  - Ac/At must be high to minimize the total pressure loss

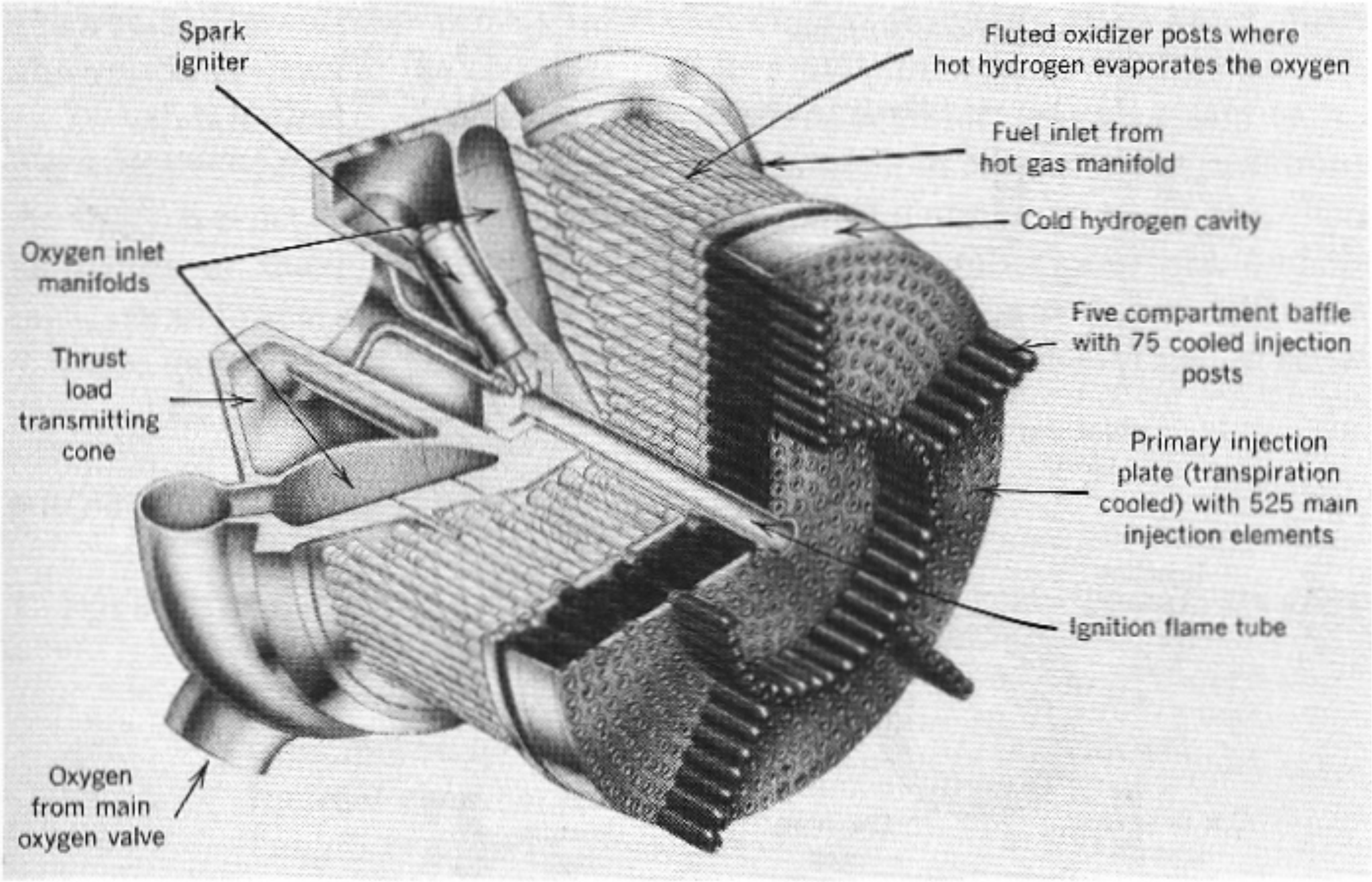
Liquid Engine Injector Design

- Injector design is critical for stable and efficient operation
  - Meter the oxidizer and fuel flow rates
  - Atomize the liquids
- Types
  - Impinging stream: doublet, triplet, self impinging
  - Shower head (V2 rocket injector)
  - Hollow post sleeve element
  - Splash plate
  - Pintle
- Flow rate expression

$$\dot{m} = C_d A_i \sqrt{2 \rho_l \Delta P}$$

- Typically Cd varies from 0.60 to 0.84 (0.61 for square edge orifice)
- Large injector pressure drop is important for stability and efficiency
  - Increasing Delta P increases the tank weight or pump requirements
- Momentum matching for the oxidizer and fuel streams. Makes the throttling more difficult

Liquid Rocket Injector Design - SSME Injector



- Liquid rocket injectors are very complex devices

Liquid Rockets Summary

Summary

- Complicated design
- Expensive manufacturing
  - Exotic materials
  - Complex parts with tight tolerance requirements
- Very good Isp performance
- Multiple liquids in the system
- Fire hazard
- Mature technology

Challenges

- Cost effective systems
- LOX/RP-1: Oxygen rich pre-burner
- Replacement for hydrazine and its derivatives