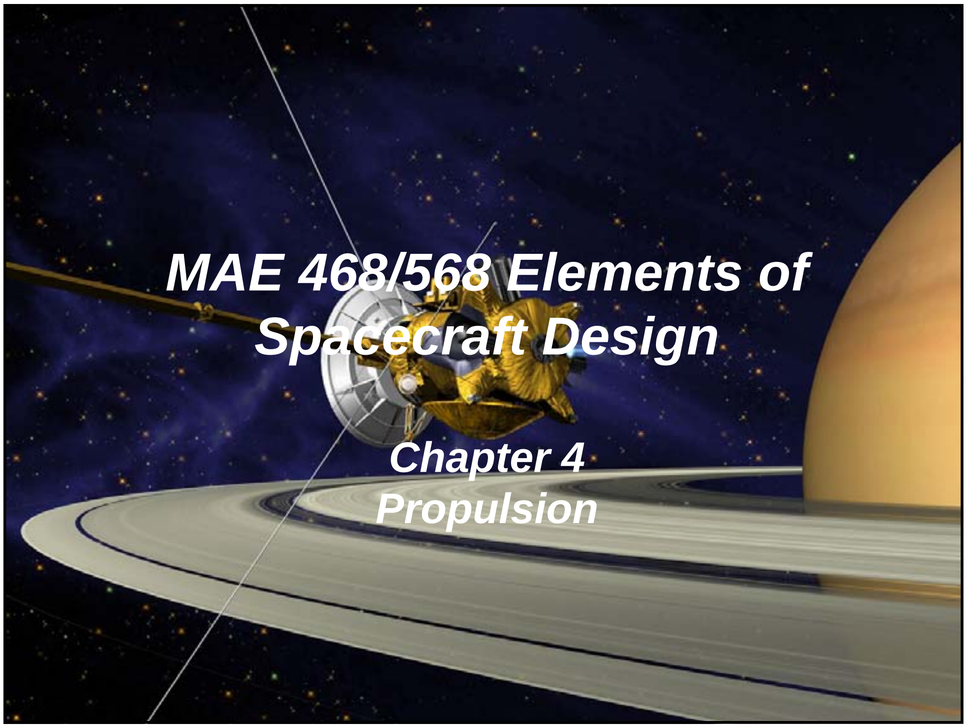

Chapter Outline

- Ch 4 – Propulsion
  - Introduction
  - Theoretical Rocket Performance
  - Propulsion Requirements
  - Monopropellant Systems
  - Bipropellant Systems
  - Dual-Mode Systems
  - Solid Rocket Systems

2

Introduction

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Introduction

- Tsiolkowski – observed that rocket propulsion was a prerequisite for space exploration
- 1883 – noted that gas expulsion could create thrust – rocket could operate in vacuum
- 1903 – published paper describing how space flight could be accomplished with rockets
  - Described staged rockets and showed mathematically that space exploration would require staging
  - Created Tsiolkowski's equation

4

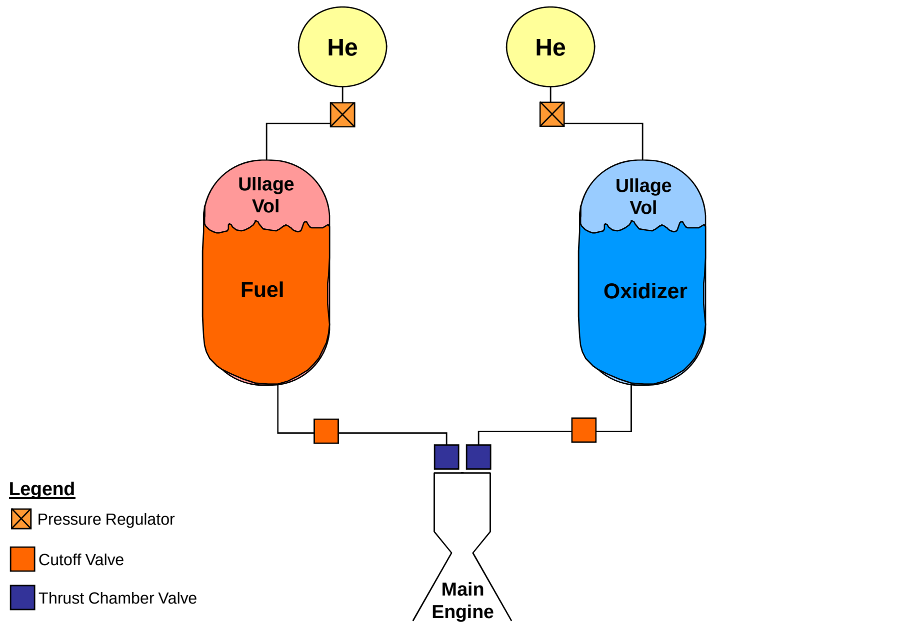
Introduction

- Goddard – observed space flight would require liquid rocket propulsion
- Goal – design, build, and fly a liquid rocket – over 200 patents in process
- 1926 – 1st liquid rocket launch, LOX/Gasoline – flew for 2.5 s, altitude of 41 ft, 63 mph
- Developed pump-fed engines, clustered stages, quick disconnects, pressurization systems, gyro stabilization
- By the time he died in 1945 – all equipment developed for Saturn V

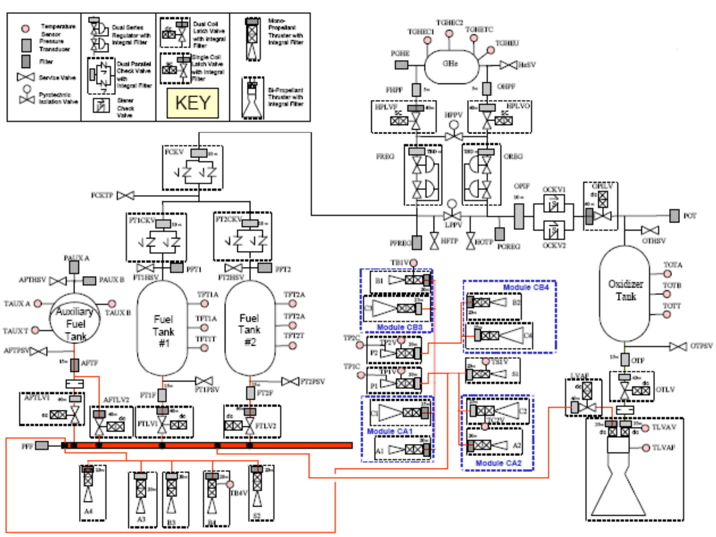
Units

- Rockets like English units
  - lbf – N
  - lbm – kg
  - psia – kPa
  - Remember
    - 1lbf = 32.2 lbm*ft/sec^2
  - Look at Table 4.1 (pg. 154) for conversions!

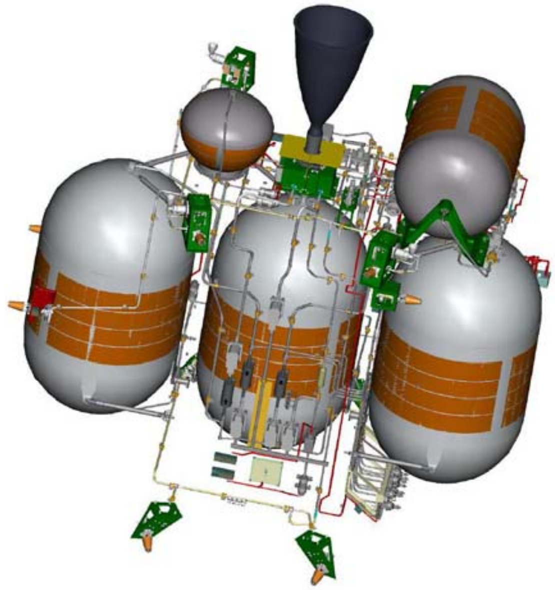
Spacecraft Propulsion System



Spacecraft Propulsion System



Spacecraft Propulsion System



Theoretical Rocket Performance

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Theoretical Rocket Performance

- Rockets generate thrust by accelerating high-pressure gas to supersonic velocities in a converging-diverging nozzle
- Most cases – high-pressure gas generated by high-temperature combustion of propellants
- Rockets
  - Combustion chamber
  - Throat
  - Nozzle

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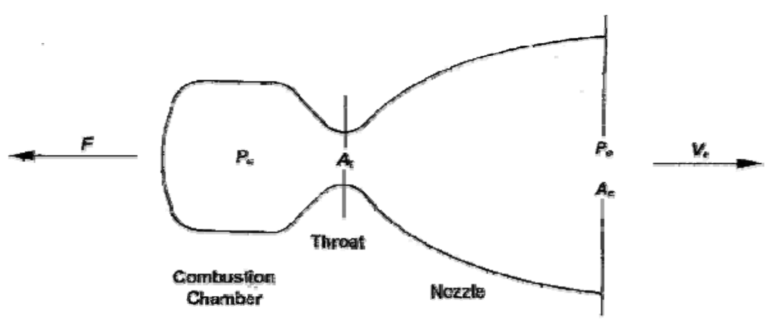
Theoretical Rocket Performance



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Theoretical Rocket Performance

- Bipropellant rocket engine – gases are generated by rapid combustion of liquid oxidizer and liquid fuel in combustion chamber (LOX/LH2)
- Monopropellant system – only one propellant is used, high-pressure, high-temperature gases are generated by decomposition of single propellant (N2H4)
- Solid system – solid fuel and oxidizer mechanically mixed and cast as a solid-propellant grain, grain occupies most of volume of combustion chamber (HTPB, AP)
- Cold gas system – no combustion involved, gas stored at high pressure and injected into chamber without combustion (He, N2)

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Thrust

- Rocket thrust generated by momentum exchange between exhaust and vehicle and by pressure imbalance at nozzle exit
- Thrust caused by momentum exchange derived from Newton's second law:

$$F_m = ma$$

$$F_m = \dot{m}_p \left(V_e - V_0\right)$$

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Thrust

- In addition to thrust caused by momentum, thrust generated by pressure-area term at nozzle exit
- If nozzle were exhausting into vacuum, pressure-area thrust would be

$$F_p = P_e A_e$$

- If ambient pressure is not zero

$$F_p = P_e A_e - P_a A_e$$

$$F_p = (P_e - P_a) A_e$$

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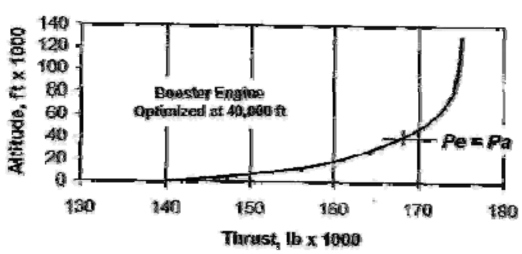
Thrust

- Total thrust – sum of thrust caused by momentum exchange and thrust caused by exit plane pressure

$$F = \dot{m} V_e + (P_e - P_a) A_e$$



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Ideal Rocket Thermodynamics

- Somewhat idealize flow of rocket engine – thermodynamics can be used to predict rocket performance parameters within a few percent of measured values
- 7 assumptions
  - Propellant gases are homogeneous and invariant in composition throughout nozzle, requires good mixing and rapid completion of combustion
  - Propellant gases follow perfect gas laws; high temperature of rocket exhaust is above vapor conditions, these gases approach perfect gas behavior
  - No friction at nozzle walls, no boundary layer
  - No heat transfer across nozzle wall
  - Flow is steady and constant
  - All gases leave engine with an axial velocity
  - Gas velocity is uniform across any section normal to nozzle axis

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Ideal Rocket Thermodynamics

- Assumptions 3,4,6,7 permit use of one-dimensional isentropic expansion relations
- Assumption 1 defines frozen equilibrium condition
- Gas composition can vary from section to section – shifting equilibrium calculation

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Specific Impulse

- Specific impulse (Isp) – describes how much thrust is delivered by an engine per unit propellant mass flow rate (engine fuel efficiency)
- Premier measurement of rocket performance
- Common to assume that an extra second of Isp was worth a million dollars in engine development

Specific Impulse

- Isp equation

$$I_{sp} = \frac{F}{\dot{m} g_O}$$

- Thermodynamic expression for theoretical vacuum Isp

$$I_{sp} g_O = \sqrt{\frac{2kRT_c}{(k-1)}\left[1-\left(\frac{P_e}{P_a}\right)^{\frac{k-1}{k}}\right]} + \frac{P_e A_e}{P_c A_t}\sqrt{\frac{RT_c}{k g_O \left(2/k+1\right)^{\frac{k+1}{k-1}}}}$$

Specific Impulse

- To fully define specific impulse, necessary to state
  - Ambient pressure
  - Chamber pressure
  - Area ratio
  - Shifting or frozen equilibrium conditions
  - Real or theoretical
- Real engine – 95-93% of theoretical
  - Want to go higher – lots of research into ways to eliminate losses

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Area Ratio

- Area ratio of rocket engine is ratio of exit area to throat area

$$\varepsilon = {}^{A_e}\!/_{A_t}$$

- Measure of gas expansion provided by engine
- Optimum area ratio provides an exit-plane pressure equal to local ambient pressure (no shock)
- Sea-level engine – near midpoint of flight – 12 common area ratio
- Spacecraft engine – optimum is infinite – largest allowed by space and weight is used – range between 50 and 300

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Mass Flow Rate

- Mass flow rate through a supersonic isentropic nozzle

$$\dot{m}_p = \frac{P_c A_t k}{\sqrt{kRT_c}} \sqrt{\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}$$

- Assumes choked flow at throat!

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Thrust Coefficient

- Useful term which first arose during rocket engine testing
- Proportionally constant between thrust and product of chamber pressure and throat area $F = P_c A_t C_f$

$$C_f = \sqrt{\frac{2k^2}{k-1}\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{k-1}{k}}\right]} + \left(\frac{P_e - P_a}{P_c}\right)\frac{A_e}{A_t}$$

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Total Impulse

- Impulse – change in momentum caused by force acting over time, for constant thrust
  - $I = Ft$
  - $I = m_p g_O Isp$
- Propulsion system size is rated based on total impulse – particularly useful in solid systems

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Mixture Ratio (MR)

- Important parameter for bipropellant systems
- Ratio of oxidizer to fuel flow rate, on a weight basis

$$MR = \frac{m_o}{m_f}$$

- Volumetric MR sometimes used in conjunction with tank sizing

$$VMR = MR\frac{\rho_f}{\rho_o}$$

- Volumetric MR – major effect on system design because it determines relative sizes of propellant tanks
- Loaded MR designed off optimum in order to have both ox and fuel tanks same volume – Isp loss is slight, two identical tanks are cheaper than two individually sized tanks

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Bulk Density

- Often convenient to use bulk density of propellant combination to expedite approximate calculations
- Bulk density – mass of a unit volume of propellant combination “mixed” at appropriate mixture ratio

$$\rho_b = \frac{MR + 1}{MR/\rho_o + 1/\rho_f}$$

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Propulsion Requirements

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Propulsion System Types

- Five different propulsion choices
  - Cold-gas system
  - Solid motor systems
  - Monopropellant systems
  - Bi-propellant systems
  - Dual-mode systems
- Selection of propulsion system type has substantial impact on total spacecraft and is key selection in early design
- Two others – ion propulsion & Hall effect thrusters – both are low thrust applications ONLY!

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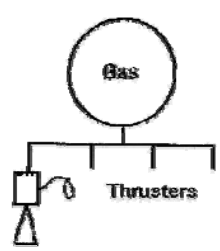
Cold-Gas Systems

- Almost all spacecraft of 1960s used this system
- Simplest choice and least expensive
- Can provide multiple restarts and pulsing
- Low Isp (40 s) and low thrust level (>1N) – major disadvantage



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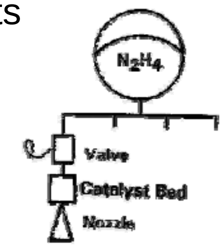
Monopropellant Systems

- Next step in complexity and cost
- Can supply pulsing or steady-state thrust
- Isp ~ 225 sec
- F ~ 0.5 – 100s N
- Common choice for attitude control and midrange impulse requirements
- Propellant – N2H4



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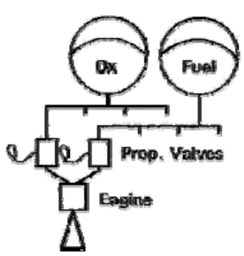
Bipropellant Systems

- Top of complexity and expense scale
- Very versatile and high performance system
- Provide Isp < 310 sec and wide range of thrust capability
- Can be used in pulsing or steady-state modes
- Most common propellant combo – N2O4/N2H4



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Dual-Mode Systems

- Designed for situations that require high-impulse burns in addition to low-impulse attitude control pulses
- High-performance high-impulse burns are provided by bipropellant N_2O_4/N_2H_4
- Fuel, N_2H_4, used as a monopropellant for low-impulse attitude control pulses
- Ideal for geosynchronous orbit and require low-impulse attitude control and station keeping
- Planetary spacecraft that required orbit insertion burns as well as attitude control are good candidates for this system as well

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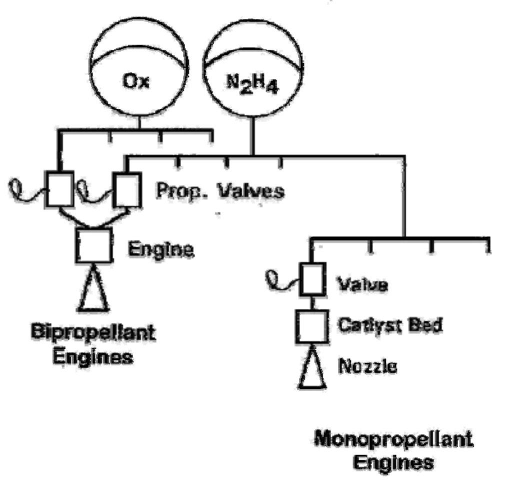
Dual-Mode Systems



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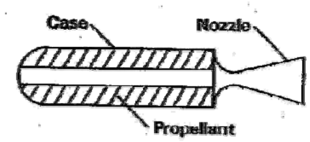
Solid Motor Systems

- Candidates when all of impulse is to be delivered in a single burn and impulse can be accurately calculated in advance
- Examples
  - Planetary orbit insertion
  - Geosynchronous apogee burns (kick motors)
- If criteria met, solids provide simplicity and reasonable performance (Isp)



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Propulsion System Summary

uirement	Cold gas	Monopropellant	Bipropellant	Dual mode	Solid
cific npulse, s	<150	230	310	310 (SS) 230 (pulse)	<300
ulse range, I-s (lb-s)	<2500 (<500)	<45,000 (<10,000)	>45,000 (>10,000)	>45,000 (>10,000) +pulsing	>45,000 (>10,000)
tart	Yes	Yes	Yes	Yes	No
ing	Yes	Yes	No	Yes	No
mmand utdown	Yes	Yes	Yes	Yes	No

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Spacecraft Propulsion Functions

Task	Description
Translational velocity change	(Usually for orbit changes)
Orbit changes	Convert one orbit to another
Plane changes	Rotate the orbit plane
Orbit trim	Remove launch vehicle errors
Stationkeeping	Maintain constellation position
Repositioning	Change spacecraft position
Rotational velocity change	
Thrust vector control	Remove vector errors
Attitude control	Maintain an attitude
Attitude changes	Change attitudes
Reaction wheel unloading	Remove stored momentum
Maneuvering	Repositioning the spacecraft axes

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Translational Velocity Change

- Main products of mission design – ΔV required for maneuvers for mission
- To convert ΔV maneuver requirements to propellant requirements – need Tsiolkowski equation

$$\Delta V = g_O I_{sp} \ln\left(\frac{m_i}{m_f}\right)$$

$$m_p = m_i\left[1 - \exp\left(\frac{-\Delta V}{g_O I_{sp}}\right)\right]$$

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Translational Velocity Change

- Assumptions
  - Thrust is only unbalanced force on vehicle, force of gravity balance by centrifugal force and drag is zero
  - Thrust is tangential to vehicle trajectory
  - Exhaust gas velocity is constant, implies a fixed nozzle throat area and sonic velocity at throat

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Derivation of Tsiolkowski Equation

- Assumption – thrust on spacecraft is exactly equal to force generated by exiting exhaust gas and opposite in sign
- Force equal to mass of vehicle times acceleration of vehicle

$$F = m\frac{dV}{dt}$$

- Thrust also equal to time rate of change of momentum (or exhaust gas velocity) times time rate of change of mass

$$F = -V_e \frac{dm}{dt}$$

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UAH

Derivation of Tsiolkowski Equation

- Setting expressions equal

$$m\frac{dV}{dt} = -V_e\frac{dm}{dt} \Rightarrow dV = -V_e\frac{dm}{m}$$

- where i and f indicate initial and final conditions and integrating yields

$$\int_i^f dV = -V_e\int_i^f \frac{dm}{m} \Rightarrow V_f - V_i = V_e\left(\ln m_i - \ln m_f\right)$$

- For $P_e = P_a$

$$V_e = g_O I_{sp}$$

$$\Delta V = g_O I_{sp}\ln\left(\frac{m_i}{m_f}\right)$$

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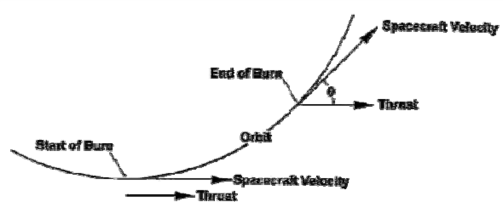
UAH

Finite Burn Losses

- Mission design calculations assume velocity is changed at a point on trajectory
  - Velocity change is instantaneous
- If assumption not valid – serious energy losses can occur
- Finite burn losses – caused by rotation of spacecraft velocity vector during burn



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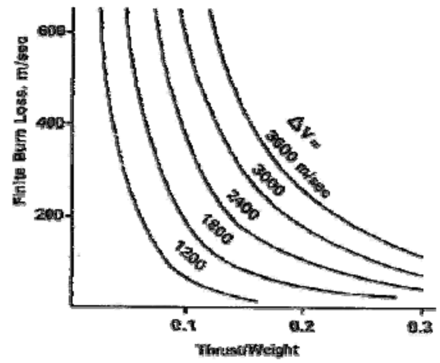
Finite Burn Losses

- Change in spacecraft velocity is not linear with time
- Acceleration is much larger at end of burn because spacecraft is lighter
- Finite burn losses can be a significant percentage of maneuver energy
- Low thrust-to-weight ratio are to be avoided; situations where thrust to weight is less than 0.5 should be analyzed
- Can be avoided by steering to hold thrust vector on spacecraft velocity vector; losses are reduced by error in steering
- Trade – saving propellant mass vs increased complexity in attitude control system



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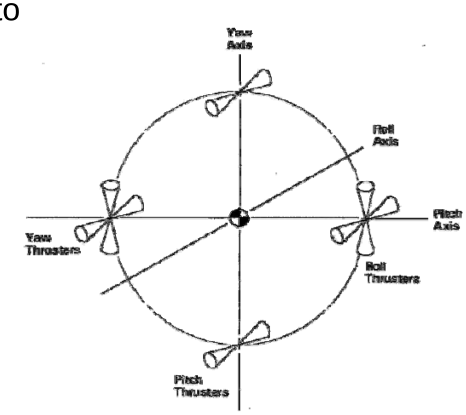
Attitude Maneuvers

- Most thrusters on spacecraft are devoted to attitude control
- Must restart frequently
- Three-axis stabilized system – attitude maneuver consists of rotation about each of spacecraft axes
- Thrusters are arranged so that applied torques are pure couples



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Applying a Torque to a Spacecraft

- Elemental action for maneuver – apply a torque to spacecraft about axis
- To apply torque, spacecraft thrusters are fired in pairs producing a torque
  - T = nFL
- From kinetics

$$\theta = \frac{1}{2}\alpha t_b^{\,2}$$

$$\alpha = \frac{T}{I_v}$$

$$\varpi = \alpha t_b$$

$$H = I_v \varpi$$

$$H = T t_b$$

Applying a Torque to a Spacecraft

- During burn, angular acceleration of spacecraft will be

$$\alpha = \frac{nFL}{I_v}$$

- When thrusters are shut down, vehicle will have turned

$$\theta = \frac{nFLt_b^{\,2}}{2I_v}$$

- At shutdown, acceleration goes to zero, spacecraft is left rotating at a velocity

$$\varpi = \frac{nFL}{I_v} t_b$$

Applying a Torque to a Spacecraft

- Angular momentum from single firing

$$H = nFLt_b$$

- Propellant consumed during a single burn

$$m_p = \frac{nFt_b}{I_{sp}} = \frac{H}{LI_{sp}}$$

- Can see long momentum arm helps propellant – constrained by launch vehicle payload fairing

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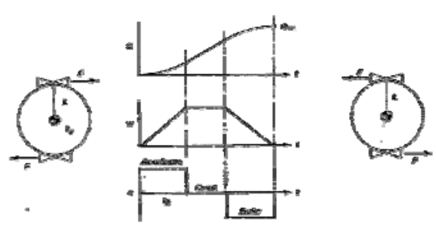
One-Axis Maneuver

- Maneuver about one axis consists of three parts
  - Angular acceleration
  - Coasting
  - Braking
- Angular acceleration produced by thruster pair firing
- Braking caused by firing opposite pair



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One-Axis Maneuver

- Total angle of rotation
  - $\theta_m = \theta(\text{accelerating}) - \theta(\text{coasting}) + \theta\ (\text{braking})$
- Rotation during coasting
  - $\theta = \omega t_c$
- Coasting rotation angle $\theta = \frac{nFL}{I_v} t_b t_c$
- Total rotation during maneuver

$$\theta_m = \frac{nFL}{I_v} t_b^{\ 2} + \frac{nFL}{I_v} t_b t_c$$

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One-Axis Maneuver

- Note that t_b is burn time of either of two burns, maneuver time is
  - $t_m = t_c + 2t_b$
- If no coast period $t_m = 2t_b = 2\sqrt{\frac{\theta_m I_v}{nFL}}$
- Propellant required for one-axis maneuver

$$m_p = 2\frac{nFt_b}{I_{sp}}$$

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One-Axis Maneuver

- Virtually all maneuvers are three-axis maneuvers
- Previous equation must be applied to each axis to determine total propellant load
- Most maneuvers come in pairs, a maneuver to commanded attitude and a maneuver back to normal attitude
- Propellant required for a complete maneuver is usually twice that obtained from a three-axis application

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Monopropellant

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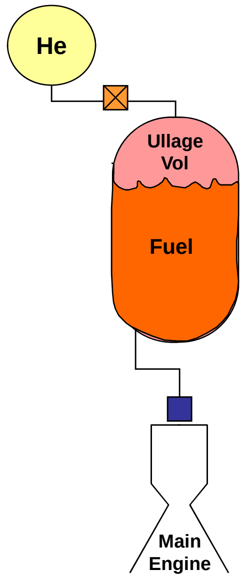
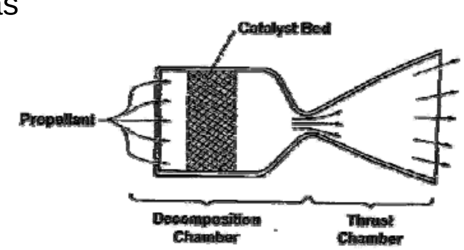
Monopropellant System



Monopropellant Systems

- Generates hot, high-velocity gas by triggering decomposition of a single propellant
- Propellant injected into catalyst bed, decomposes, resulting hot gases are expelled through converging/diverging nozzle
- Propellant slightly unstable – decomposes exothermally to produce hot gas



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Monopropellant Hydrazine Thrusters

- Propellant flow into chamber controlled by propellant valve
- Propellant injected into catalyst bed – decomposes into H2, N2, and ammonia
- Gases are expelled through converging/diverging nozzle
- Gas temperature in 1200°C range
- High temperature alloys used for converging/diverging nozzle – radiation cooling is adequate

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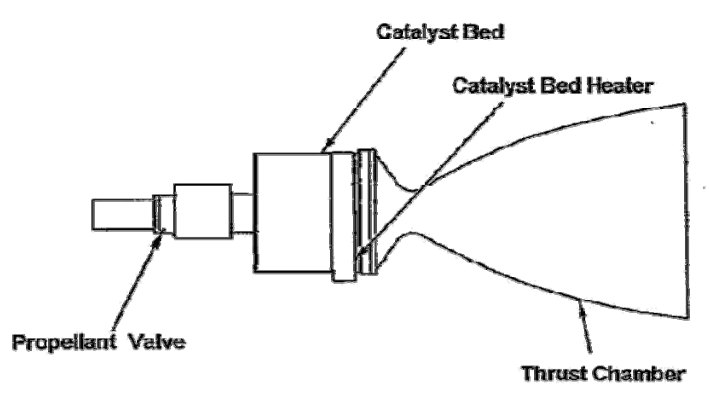
Monopropellant Hydrazine Thrusters



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Monopropellant Hydrazine Thrusters

- Catalyst bed heaters used for pulsing engines – improving first pulse performance and improve bed cycle life
- Valve performance integral of thruster performance during pulsing
- Thrust range 0.5 to 2600N
- Blow down ratios up to 6
- Pulse widths as low as 7ms
- Decomposition of hydrazine lead to H2 and ammonia – reaction is exothermic and adiabatic flame temperature is about 1700K
- Ammonia – further decomposes into H2 and N2 – reaction is endothermic and leads to reduction in flame temperature and Isp

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Steady-State Performance

- Steady-state theoretical vacuum specific impulse – 240 secs at AR of 50
- Real Isp about 93% of theoretical
- Isp at other ARs can be obtained from ratio of thrust coefficients
- Ratio of specific heats for exhaust is about 1.27

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Pulsing Performance

- ACS monopropellant thrusters require operation over wide ranges of duty cycles and pulse widths
- Pulsing Isp and minimum impulse bit very important
- Pulsing involves performance of propellant valve and feed tubing as well as chamber itself

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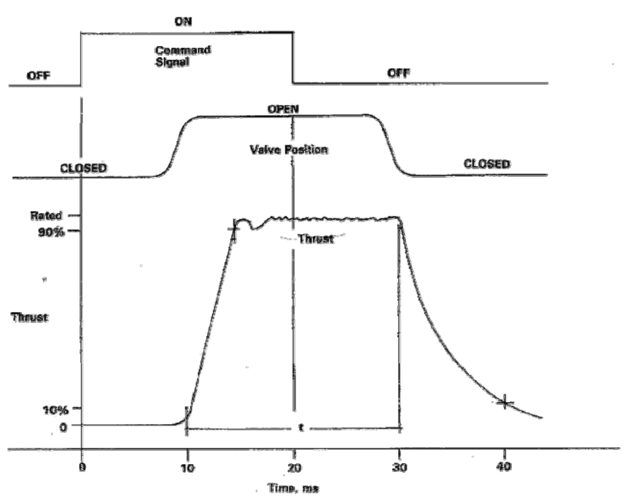
Pulsing Performance



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Pulsing Performance

- Pressure response time – time (measured from propellant valve actuation signal) required to reach arbitrary percentage of steady-state chamber pressure
- Response time affected by
  - Valve response characteristics
  - Time for propellant to flow from valve to injector
  - Ignition delay (time to wet catalyst bed and for initial decomposition heat release to bring catalyst to temperature at which ignition becomes rapid
  - Pressure rise time (time to decompose enough hydrazine to fill void space in catalyst bed and head whole bed)

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Pulsing Performance

- Valve response characteristics and feed line hydraulic delay vary for each specific design
- Valves are available with response times better than 10 ms
- Distance between valve and injector must be minimized
  - Control of heat soak back to valve sets a minimum length for injector tube
- With proper injector design, ignition delay will be approximately 10-20 ms for catalyst and propellant temperatures in range of 5-20°C
- With catalyst bed temperature at 260°C ignition delay will be approximately 1-2 ms
- Pressure rise time is much larger fraction of response time than ignition delay for most thrusters
- Response times from valve signal to 90% of steady-state chamber pressure of 15 ms have been demonstrated with tail-off time (signal to 10% thrust) of 20 ms

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Pulsing Performance

- If duty cycle is long, engine will cool between pulses – thrust will not reach rated value because of energy losses to engine heating
- If pulses are frequent, thrust will reach rated value
- Minimum impulse bit can be estimated by assuming typical system response and time, t, from ignition to first loss of thrust, equal to pulse width
- Impulse is area under thrust time curve
  - I_{min} = I(Startup) + I(Steady state) + I(Shutdown)

Pulsing Performance

- Trapezoidal approximation, minimal impulse bit

$$I_{\min} = \frac{0.005}{2}F + (t - 0.005)F + \frac{0.010}{2}F$$

- For all practical purposes impulse bit can be estimated as Ft
- For infrequent pulse the thruster will be cold, full rated thrust will not be reached
  - Thrust level corresponding to expected gas temperature should be used
- Pulsing Isp is low at low duty cycles because energy is lost reheating moto
- Short pulses deliver low Isp for same reason
- Low duty cycles and short pulses in combination deliver Isp ~115s
- Isp is limited at low duty cycles b performance of ambient temperature bed
- Performance of thruster operating with a cold bed can be estimated by assuming exhaust gas exists at bed temperature
- All hydrazine that reaches catalyst bed is decomposed – no liquid losses on start or shutdown

Negative Pulses

- Course of steady-state burn – often desirable to pulse off to create attitude control torques
- When main engine is firing, other ACS might be needed to aid in control

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Thruster Weight

- Least-squares curve fit of actual monopropellant thruster/valve designs

$$m_t = 0.4 + 0.0033F$$

- For low thrust levels thruster weight approaches valve weight
- Use 0.3 kg as minimum thruster/valve weight for low thrust levels
- Estimated weight must be increased if redundant valves are used

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Propellant Systems

- Purpose – to contain propellants and to serve them to engine on demand at proper pressure, quality, and cleanliness
- Anhydrous hydrazine
  - Clear, colorless, hygroscopic liquid
  - Distinct ammonia-like odor
  - Stable chemical that can be stored for long periods without loss of purity
  - Relatively insensitive to shock
  - Strong reducing agent
  - Toxic
  - Special preparation, special equipment, and special procedures required for handling it

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Propellant Inventory

- Propellant inventory subdivided tabulation of loaded propellant weight
- Usable propellant – portion of propellant loaded, which is actually burned
  - Quantity required for all maneuvers and all attitude control functions
  - Important that usable propellant quantity be calculated under worst-case conditions
- Trapped propellant – propellant remaining in feed lines, tanks, valves, hold-up in expulsion devices, and retained vapor left in system with pressurizing gas
  - About 3% of usable propellant
- Uncertainty – added to load to ensure that usable propellant can be no less than worst-case requirement
  - About 0.5% of usable propellant

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Propellant Inventory

- Propellant reserves are very valuable commodity
- More of project reserves placed in propellant – the better
- 3 primary reasons
  - Propellant is usually life-limiting expendable on spacecraft
  - Often desirable to make unplanned maneuvers in response to unexpected results or emergency conditions
  - Usually extended mission objectives that can be achieved after primary mission

Propellant Loading

- Spacecraft systems use weight method to measure propellant loaded
- System is moved to remote area and weighed empty
- Propellant is then loaded using special clothing and procedures to protect personnel
- Loading is hazardous event – practice
- After loading system – weighed again
- Weight change is propellant load
- Because of loading – desirable to design any liquid propellant system to be readily removable from spacecraft
  - Remote area loading; parallel processing hazard avoidance
- Loading close to launch site near launch date

Pressurization Subsystems

- Purpose – to control gas pressure in propellant tanks
- Tank pressure must be higher than engine chamber pressure by amount equal to system losses
- Significant delta pressure must be maintained across injector for combustion stability

Pressurization Subsystems

- Pressurants
  - Must be inert in presence of propellants
  - Low molecular weight desirable
  - Two pressurants – N2 and He
  - He – provides lightest system – difficult to prevent He leakage
  - N2 – used if weight situation will allow it
- Ullage
  - Volume that pressurant occupies above propellant

Blowdown System

- Two systems
  - Regulated
  - Blowdown
- Blowdown – tank is pressurized to an initial level and pressure is allowed to decay as propellant is used
- Advantages
  - Simplest method – more reliable
  - Less expensive – fewer components
- Disadvantages
  - Tank pressure, thrust, and propellant flow rate vary as function of time
  - Isp is 2nd order function of chamber pressure and drops as function of time
- Variability of flow rate and engine inlet pressure make blowdown system difficult to use with bipropellant systems

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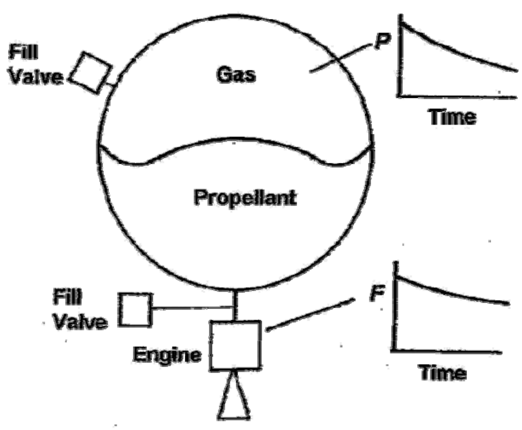
Blowdown System



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Tank Gas Thermodynamics

- Blowdown ratio – ratio of initial pressure to final pressure

$$B = \frac{P_{gi}}{P_{gf}} = \frac{V_{gf}}{V_{gi}}$$

$$V_{gi} = \frac{V_u}{B-1}$$

$$V_{gi} = \frac{W_u}{\rho(B-1)}$$

- Maximum blowdown ratio is determined by inlet pressure range engines can accept
- Ratios of 3 or 4 common – can be as high as 6

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Tank Gas Thermodynamics

- Equation of state
  - An ideal gas at any state point – product of tank pressure and ullage volume

$$PV = mRT$$

$$m = \frac{PV}{RT}$$

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Tank Gas Thermodynamics

- Isothermal expansion
  - Outflow of propellant is usually slow and heat transfer will keep gas temperature fixed at or near propellant temperature – process is isothermal

$$P_2 = P_1 \frac{V_1}{V_2}$$

  - During blowdown process, tank pressure at time t

$$P(t) = P_{gi}\left(\frac{V_{gi}}{m_p(t)/\rho + V_{gi}}\right)$$

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Tank Gas Thermodynamics

- Isentropic expansion
  - If propellant withdrawn rapidly – translational burn – gas expansion in ullage will be isentropic and temperature will drop during process

$$P_1 V_1^k = P_2 V_2^k$$

$$P(t) = P_{gi}\left(\frac{V_{gi}}{m_p(t)/\rho + V_{gi}}\right)^k$$

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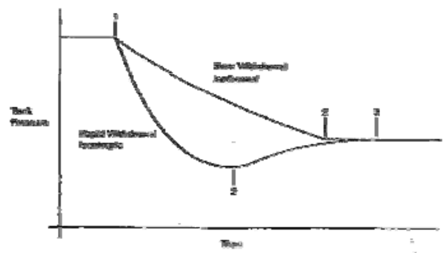
Tank Gas Thermodynamics

- Isentropic expanison – temperature drops more rapidly and drops below propellant temperature
- After engine shutdown – gas temperature warms up to propellant temperature
- Tank pressure and thrust will be lower during portion of burn than isothermal calculations would predict



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Pressurization

- Pressurant mass

$$m_{gi} = \frac{P_p V_p}{R T_i}\left[\frac{k}{1-\left(P_g / P_i\right)}\right]$$

- k = 1.67 for He
- R = 2077.3 J/(kg-K) for He

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Tankage Weight

- Major dry weight component in liquid propellant propulsion system is tankage
- 1st – volume of tank must be established and maximum tank pressure set
- Tank weight can then be estimated
- Four components to volume
  - Initial ullage
  - Useable propellant volume
  - Unusable propellant volume – 3-4% of usable
  - Volume occupied by zero-g device

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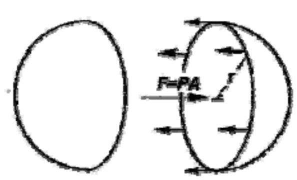
Spherical Tanks

- Common tank configurations
  - Spherical
  - Barrel with hemispherical domes
- Tank weights are byproduct of structural design of tanks
- For spheres – load in walls is pressure times area

$$PA = P\pi r^2$$



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Spherical Tanks

- Stress is load divided by area carrying load

$$\sigma = \frac{P\pi r^2}{2\pi r t} = \frac{\Pr}{2t}$$

$$t = \frac{\Pr}{2\sigma}$$

- Knowing wall thickness – weight of tank membrane calculated

$$R = r + t$$

$$W = \frac{4}{3}\pi\rho\left(R^3 - r^3\right)$$

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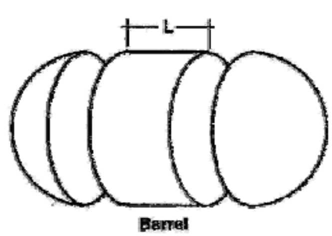
Cylindrical Tanks

- Hoop stress is twice that in spherical shell

$$t = \frac{\Pr}{\sigma}$$

- Weight of barrel

$$W = \pi L\rho\left(R^2 - r^2\right)$$



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Penetrations and Girth Reinforcements

- Membrane weight is a perfect pressure shell
- Areas of reinforcement (land areas)
  - Girth welds – weld between two hemispheres
  - Penetration welds – welds for inlet and outlets
  - Bladder attachment – attachment between bladder and tank wall
  - Structural mounting of tank
- Weight must be added to membrane weight to account for these reinforcements
- Weld areas must be reinforced because of reduction in material properties near welds
- For girth weld two reinforcing bands with thickness, t, a width of 5 cm each are added

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Penetrations and Girth Reinforcements

- Penetrations for attachments, inlet and outlet tubes weaken shell and require reinforcement rings
- 15 cm in diameter centered on penetration, with thickness, t
- Mounting pads on tank add about 2% of supported weight
- Penetrations, girth land areas, and mounting pads normally add about 25% to shell weight

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Tank Mass Estimating Equation

$$m_{PT} = 0.0116 * P(kPa) * V(m^3)$$

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Monopropellant System Design Steps

- Define requirements
- Calculate propellant required, add margin
- Select propellant control device
- Decide dual-vs single-propellant tanks
- Decide propellant tank type, sphere, barrel, conosphere
- Select pressurant – He is mass is critical
- Select pressurization system type and set performance parameters – max tank pressure and blowdown ratio
- Design propellant tank
- Design engine modules and general arrangement
- Design system schematic; plan redundancy
- Calculate system mass
- Conduct trade studies of system alternatives; repeat process

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Bipropellant

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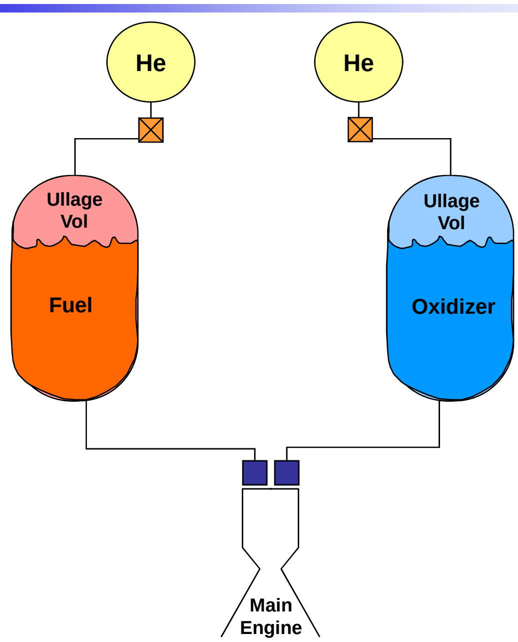
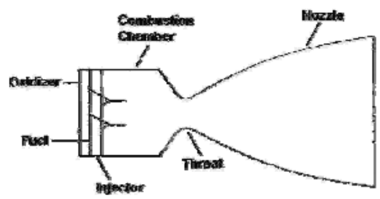
UAH

Bipropellant System



Bipropellant Systems

- Offer most performance
- Most versatility
  - Pulsing
  - Restart
  - Variable thrust
- Offer most failure modes
- Highest price tags
- Major parts
  - Injector
  - Nozzle
  - Cooling system
  - Thrust-chamber valves



Injector

- Introduces oxidizer and fuel into combustion chamber
  - Promotes stable, efficient combustion without overheating injector face or chamber walls
- Injector design is single most important contributor to engine performance
- Determines whether combustion will be stable
- Most sensitive area – combustion zone of thrust chamber just downstream of injector
- Propellant enters high-pressure zone as room-temperature liquids and leave as supersonic gases at velocities > 600 m/sec and temperatures as high as 3500°C

Injector

- Mixing, ignition, and combustion occur in milliseconds
- Propellants introduced through injector plate at high velocity so that fuel and oxidizer are shattered into droplets small enough to complete combustion upstream of throat
- Continuous and rapid combustion must occur near injector face so that pockets of mixed but unreacted propellants do not form and explode
- Thrust developed by single pair of injector orifices can be as high as 2000 N

Cooling

- Adiabatic flame temperature for NTO/MMH is 3414K
- Heat transfer rates at throat can be as high as 3200 J/s-cm^2 during operation
- Throat and nozzle cooling important design problem
- Methods
  - Regenerative cooling
  - Film cooling
  - Ablative cooling
  - Radiation cooling

Cooling

- Regenerative cooling
  - Used on large, launch vehicle engines
  - Nozzle construction is a bundle of shaped cooling tubes brazed together and radially reinforced
  - Fuel is routed through tubes from exit plane up to injector
  - Energy collected from gas stream in cooling is returned to combustion process by hot fuel as it enters chamber – process is regenerative

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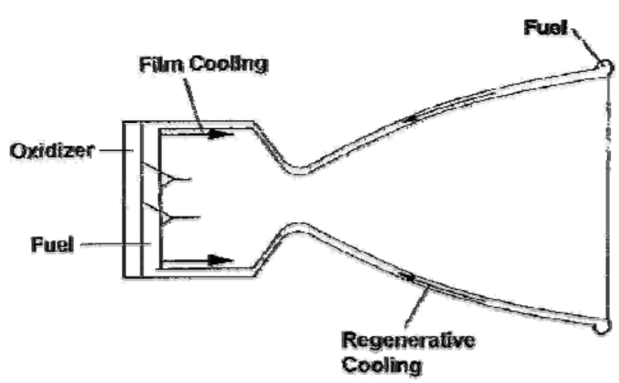
Regenerative Cooling Flow



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Cooling

- Film cooling
  - Used in all engines to supplement primary cooling method
  - Cylindrical film of fuel injected into chamber such that it washes down wall
  - Flow is designed so that some liquid will reach throat
  - Cooling comes from heat of vaporization of fuel
  - Combustion near wall is fuel rich and inefficient – Isp is reduced
  - Amount of fuel used is minimized in engine design

Cooling

- Ablative cooling
  - Chamber wall is constructed of material that will evaporate and erode away during firing
  - Throat will be a machined carbon block to prevent throat erosion
  - Ablative cooling takes place primarily in combustion chamber and diverging section – usually made of fiberglass epoxy
  - Material thickness is sized so that expected burn time will remove allowable amount of material

Cooling

- Radiation cooling
  - Chamber is made of refractory material and radiation heat rejection to space
  - Enhanced film cooling keeps chamber walls within acceptable limits

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Bipropellants

- Goddard
  - LOX/Kerosene
- Germans
  - LOX/Alcohol
- Post WWII
  - Kerosene – RP-1
  - LOX not used – storability issue
  - Storable propellants
    - Nitric acid – corrosive, still used by Russia
    - NTO developed
  - Fuels
    - N2H4 & UDMH
    - 50/50 mixture of N2H4 & UDMH – Aerozine 50
    - MMH – properties of Aerozine, but with mixing problems

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Bipropellants

- Saturn series
  - LOX/LH2
  - Fluorides – too energetic for metal tank containment
- F2, O2, & H2 – cryogenics
  - Extremely low temperatures at which they are liquids
- Cryogenics never used in spacecraft
  - Only in launch vehicles to date
  - Ongoing studies to show benefits of systems

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Bipropellants

- Nitrogen tetroxide (N2O4)
  - Equilibrium solution of NO2 and N2O4
  - Equilibrium state varies with temperature
  - Relative of nitric acid – smells like it
  - Reddish brown and very toxic
  - Hypergolic with N2H4, Aerozine 50, MMH
  - Pulsing performance practical
  - Compatible with stainless steel, aluminum, and Teflon
  - Incompatible with all elastomers

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Bipropellants

- Monomethylhydrazine (MMH)
  - Clear water-white toxic liquid
  - Sharp decaying fish smell
  - Not sensitive to impact or friction
  - More stable than hydrazine when heated
  - Decomposes with catalytic oxidation
  - N2O4/MMH – dominant spacecraft propellant combination for bipropellant spacecraft propulsion

Bipropellants

- Hydrazine (N2H4)
  - Dual-mode propulsion systems made N2H4 propellant of choice
  - Provides slightly higher Isp with N2O4 than MMH
  - Can be used as monopropellant for pulsing

Propellant Inventory

- Subdivided tabulation of loaded propellant weight
- Usable propellant first subdivided into oxidizer and fuel
- NTO/MMH ~ 1.6
- NTO/N2H4 ~0.9

$$MR = \frac{m_o}{m_f}$$

$$m_f = \frac{m_u}{1 + MR}$$

$$m_o = m_u - m_f$$

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Propellant Inventory

- Usable propellant – portion of propellant loaded that is actually burned
- Trapped propellant – propellant remaining in feed lines, tanks, valves, hold up in expulsion devices and related vapor left in system with pressurizing gas
  - About 3% of usable propellant in small bipropellant systems
- Two primary activities for required propellant
  - Propellant required for velocity change maneuvers as dictated by mission design
  - Propellant required for control attitude
- Tsiolkowski equation used to determine propellant requirements

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Propellant Inventory

- Outage – caused by difference between mixture ratio loaded and mixture ratio at which engine actually burned
- Always some of one propellant left when other propellant is depleted – remaining propellant called outage
- Outage depends on loading accuracy and on burn-to-burn repeatability of engine MR
- 1% used for initial estimates
- 5% JPL requirement
- Calculate propellant requirements under worst-case conditions – show propellant inventory as stated

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Propellant Inventory

- Reserves shown as separate item in propellant inventory
- Reserves are valuable commodity
- 3 reasons
  - Propellant is usually life-limiting expendable
  - Often desirable to make unplanned maneuvers in emergency conditions
  - Usually extended mission objectives that can be achieved after primary mission
- Measurement uncertainty in propellant loading – 0.5%
- Added to load to ensure usable propellant can be no less than worst-case requirement

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Propellant Control and Tankage

- Titanium, aluminum, and stainless steel all compatible with NTO and N2H4
- Titanium – lightest and most common
- Usually use identical tank shells for oxidizer and fuel as cost-saving measure
- NTO – not compatible with elastomers – propellant control devices limited to metals and Teflon

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Pressurization

- Blowdown pressurization not used with bipropellants
  - Difficulty in keeping both tanks at same pressure
  - Difficulties with varying inlet pressure to bipropellant engines
- He or N2 systems used
  - He most frequent choice
- Regulated system controls pressure in propellant tanks at preset pressure
- Pressurant stored at high pressure (3000-5000 psia)
- Engine supplied propellant at tightly controlled lower pressure and flow rate
- Thrust does not vary during propellant consumption
- Regulation is essential in order to keep flow rate of each propellant constant and at correct mixture ratio

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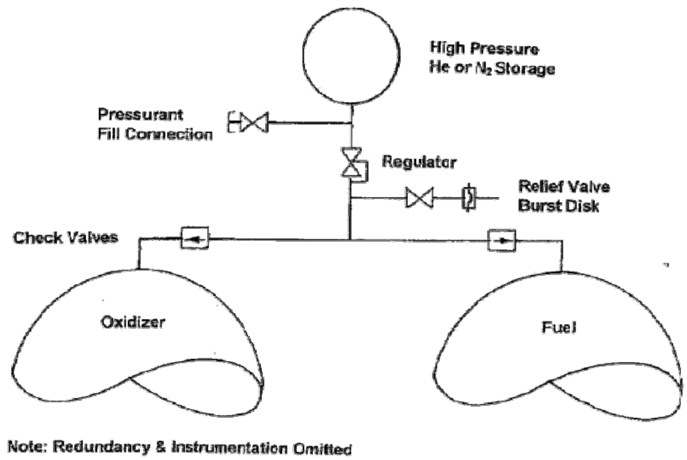
Pressurization



Pressurization

- Pressurant tank weight is essentially constant for any initial pressure for a given gas weight
  - Takes 3.2 kg of tank to contain 1 lb of He
- Initial storage pressure selected is 2nd order trade
- Regulator requires inlet pressure around 100 psi higher than outlet pressure in order to operate properly
- Unusable pressurant is trapped in pressurant tank at or above minimum regulator inlet pressure

Pressurization

- Typical pressure profile for a regulated system
  - Maximum pressurant tank pressure – 4500 psia
  - Minimum pressurant tank pressure – 650 psia
  - Minimum regulator inlet pressure – 600 psia
  - Nominal propellant tank pressure – 500 psia
  - Minimum engine valve inlet pressure – 450 psia
  - Nominal chamber pressure – 350 psia
- Minimum ullage volume must be about 3% of propellant tank volume in order for regulator to have stable response when outflow starts
- Relief valves necessary to protect tank in case regulator fails open
- Good practice to place burst disk upstream of relief valve so that pressurant is not lost overboard at valve seat leakage rate
  - Particularly important if He is pressurant
- Each propellant tank ullage must be isolated to prevent vapor mixing

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Pressurization

- For constant temperature conditions volume of pressurant sphere

$$V_s = \frac{P_r V_u}{P_1 - P_2}$$

- Once pressurant sphere volume calculated – calculate initial weight of stored pressurant from equation of state
- Total weight of pressurant on board is initial weight of pressurant in sphere plus initial weight of pressurant in each tank ullage

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Pressurization

- Careful considerations should be given to propellant vapor mixing between check valves after permeation of check valve seals
- Propellant vapor mixing between check valves have been issues with previous missions
  - Viking I – still successfully landed
  - Mars Observer – did not enter Mars orbit
- Positive mixing prevention particularly important on long duration missions

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Unusable Propellant

- Unusable propellant greater with bipropellant systems than monopropellant systems
- Difference between loaded mixture ratio and burned mixture ratio results in residual of one or the other propellants
- Residual – called outage
- Unusable propellant is about 4% of total load by weight

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Pulsing Performance

- Isp of 50% of theoretical can be expected with bipropellant engine producing 0.01 s pulses
- With 0.1 pulses – 75-85% can be expected
- Pulsing Isp of 170 sec demonstrated
- Isp of 128 sec for 25-lbf engine
- Isp of 191 sec for 45-lbf engine
- Impulse repeatability of ± 10% pulse-to-pulse and ± 30% engine to engine

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Bipropellant Thruster Mass

- Thruster mass

$$M_T = 0.0073 * F(N) + 1.95$$

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Dual-Mode Systems

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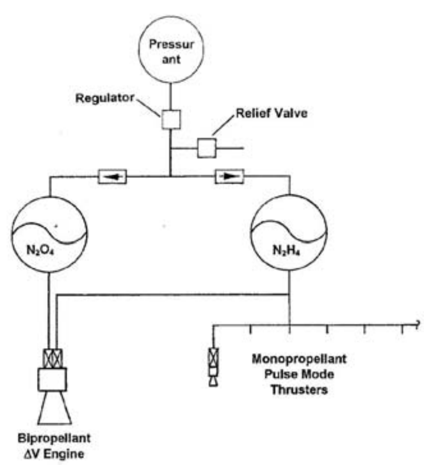
Dual-Mode Systems

- Many missions require high-velocity change burn as well as pulse mode attitude control operation
- Use N2H4 as bipropellant fuel
  - NTO/N2H4 as main propulsion
  - N2H4 as monopropellant ACS



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Dual-Mode Systems

- Advantages
  - Ability to use N2H4 as a monopropellant in attitude control thrusters and as fuel in bipropellant main engines with resulting system simplification
  - Significant increase in Isp for orbit insertion burn
    - Solids are around 290 sec
    - NTO/N2H4 ~320s sec

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Solid Rocket Systems

122

UAH

Solid Rocket Systems

- Oxidizer and fuel are stored in combustion chamber as a mechanical mixture in solid form
- When propellants are ignited, they burn in place
- Used extensively where
  - Total impulse requirement is known accurately in advance
  - Restart is not required
- Kick stage of geosynchronous spacecraft

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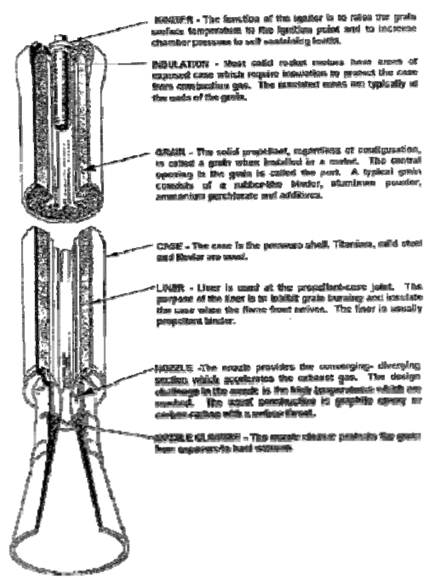
UAH

Solid Rocket Systems



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Propellants

- Solid rockets over 700 years old
  - Chinese invented in 1232
- Basis of Chinese rockets – black powder
- Smokeless powder invented by Alfred Nobel in 1879
  - Not used as rocket fuel until 1918 by Goddard
- Double-base propellant
  - Combination of nitrocellulose dissolved in nitroglycerin
- Composite propellant
  - Modern, high-performance propellants made in 1942 by JPL

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Propellants

- Composite propellants
  - Composed of one of several organic binders, aluminum powder, and an oxidizer – usually AP
  - Binders – rubber-like polymers – serve dual purpose as fuel and bind aluminum with AP
  - Common binders
    - HTPB, PBAN
  - Contain small amounts of chemical additives to improve physical properties
    - Burn rate, smooth burning, casting characteristics, structural properties, absorb moisture during storage

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UAH

Propellants

- Solid propellants produce vacuum Isp of 300 sec or lower
  - Somewhat lower than bipropellants
- Solid propellant densities are higher
  - Smaller systems for a given impulse
- Higher Isp – less stable ingredients
  - Boron
  - Nitronium perchlorate

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UAH

Hazard Classification

- Propellants divided into two explosive categories by DoD and DoT
  - Class 2 – catastrophic failure produces burning or explosion
  - Class 7 – catastrophic failure produces detonation
- Motor classification dictates safety requirements for using and handling motor

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Burning Rate

- Control of exhaust gas flow by precise control of exposed grain surface area and burning rate of propellant mixture
- Solid transformed into combustion gases at grain surface
- Surface regresses normal to itself in parallel layers
- Rate of regression – burning rate
- Burning rate – measured by measuring flame front velocity on a strand

$$\dot{m}_p = \rho A_p \dot{r}$$

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Burning Rate

- Pressure effect
  - Grain density is constant during burn, gas flow rate is controlled by grain area and burning rate
  - Gas flow rate, thrust, and chamber pressure go up with an increase in burn area
  - Burning area goes up exponentially with pressure

$$\dot{r} = cP^n$$

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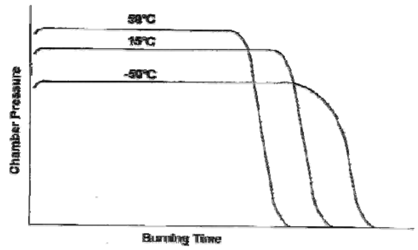
UAH — *Burning Rate*

- Temperature effect
  - Chamber pressure increases nonlinearly with grain temperature because burning rate increases with temperature and pressure
  - Sensitivity of chamber pressure to temperature is typically 0.12 to 0.50% per °C



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UAH — *Burning Rate*

- Temperature effect
  - Two reasons to control temperature
    - Limit variation in performance
    - Minimize thermal stress in grain and minimize possibility of grain cracks
  - Typical temperature control is 15 - 38°C
  - Control range can be tightened before motor firing

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Burning Rate

- Acceleration effect
  - Burning rate increased by acceleration perpendicular to burning surface
  - Acceleration of 30 g can double burn rate
  - Enhancement of burning rate has been observed for acceleration vectors at any angle of 60 to 90 deg with burning surface
  - Increase in burn rate is attributed to presence of molten metal and metal oxide particles that are retained against grain surface by radial acceleration with attendant increase in heat transfer at grain surface
  - Effect is enhanced by increased aluminum concentration and particle size

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Space Storage

- Some controversy and little data regarding ability of solid motor grain to withstand long periods of space exposure with open port
- Concern – evaporation could affect chemical composition of propellant or bond line and affect performance
- Two options
  - Conduct adequate testing to establish post vacuum performance
  - Design a sealed nozzle closure

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Grains

- Solid motor – propellant tank and combustion chamber are same vessel
- Viscous propellant mixture is cast and cured in mold to achieve desired shape and structural strength
- After casting – propellant called grain
- Most frequent method of casting is done in place in motor case using a mandrill to form central port
  - Grain called case bonded

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Grains

- Grains also cast separately and loaded into case at later time
  - Called cartridge loaded
- Large grains cast in segments – then stacked to form motor
  - SRBs on shuttle and Titan IV
- Segmented grains solve transportation issue, but require high-temperature case joints
  - Serious failure source
  - *Challenger* accident
- Liner used in grain to case interface to inhibit burning as flame front arrives at case wall
- Liner composition usually binder without added propellants
- Areas in case where no grain interface (ends of motor), insulation used to protect case for full duration of firing

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Grains

- Grain shape
  - Primarily cross section
  - Determines surface burning area as function of time
  - Burning area, along with burning rate, determines thrust
  - For given propellant, surface area vs time determines shape of thrust time curve
  - Cylindrical grain inhibited on sides – called an “end burner” would have constant burning area and constant thrust
  - Grain with essentially constant burning surface area and thrust time curve is called neutral burning

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Grains

- Grain shape
  - Progressive – grain that has increasing surface area with time
  - Primary disadvantage – as mass decreases thrust increases – higher acceleration loads
  - Cylindrical grain inhibited on ends and burning from sides would be regressive
  - Neutral burning is most common design for spacecraft

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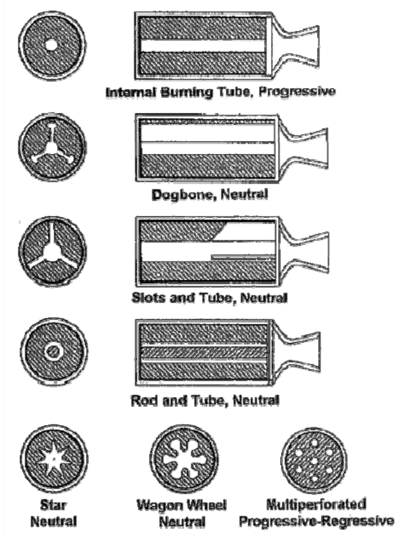
Grain Configurations



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Grains

- Crack in grain provides unplanned increase in exposed area with resultant sudden increase in pressure – can lead to failure
- Numerous precautions
  - Limits on grain temperature extremes
  - Limits on shock loads on grain and bond line
  - Final set of grain x rays at launch site just before launch
- Star grain most commonly used shape and nearly neutral
  - Flame does not reach liner in all locations at same time
  - When flame front reaches liner, chamber pressure starts to decay
  - At given pressure aggressive burning or deflagration can no longer occur
  - Residual propellant (remains after combustion) – called sliver
  - Sliver – about 2% of grain weight

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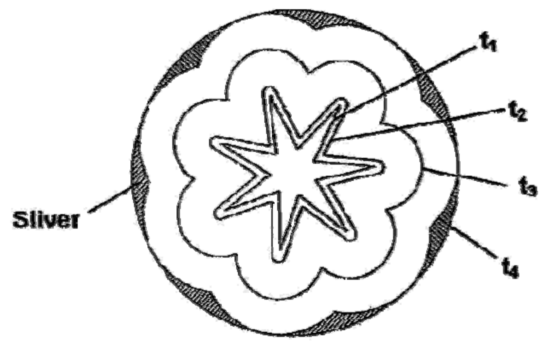
Grains



Shelf Life

- Shelf life of motor is conservative estimate of acceptable length of time from pour date to firing date
- Typical shelf life is 3 yrs
  - Pour date must not be more than 3 yrs from on-mission firing
- Pour date becomes major program milestone

Offload Capability

- Many solid motors demonstrated proper performance over range of propellant loads
  - Called off-load capability
- Useful in matching required total impulse with available total impulse
- Off loading accomplished by pouring motor case less than full

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Performance

- Burning time and action time
  - Estimated from internal ballistics using strand burning rates or measures on test stand
  - Two times of interest
    - Burning time – measured starting from 10% thrust at ignition to 90% thrust during shutdown
    - Action time – measured starting at 10% thrust during ignition to 10% thrust during shutdown
  - Thrust level from which 10% and 90% points are calculated is maximum thrust
  - 90% thrust point on shutdown is approximate time when flame front reaches inhibitor at case wall
  - From 90% to 10% slivers are being consumed (less efficiently than combustion at full chamber pressure)

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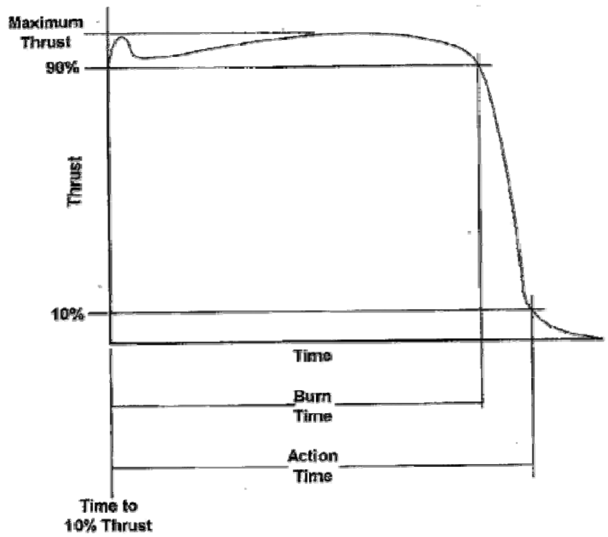
Performance



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Performance

- Motor firing in vacuum – thrust will continue for substantial period of time after 10% thrust point on shutdown
- During tail-off period – low chamber pressure maintained by burning of sliver, outgassing of inert materials, and low level smoldering of hot insulation, nozzle materials, and liner
- Impulse, thrust level, and time-to-zero are nearly impossible to measure in atmosphere because hard vacuum cannot be maintained with motor outgassing
- Important to design spent case staging sequence to provide an adequate period before release in order to make sure case will not follow spacecraft

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Effective Propellant Mass

- Propellant actually consumed by motor
- Difference between initial total weight of motor and post burn total weight

$$M_{eff} = M_i - M_f$$

- Can be either more or less than total propellant load
- Two effects
  - Sliver, tends to reduce effective propellant
  - Consumed inert parts, tend to increase effective propellant
    - Consumed inert weights – nozzle materials and volatiles forced out of insulation and liner

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Impulse

- 2 values result from two thrust times
- Thrust integrated from time zero to time, t_b, called burning time impulse

$$I_b = \int_0^{t_b} F dt$$

- If action time, action time impulse

$$I_a = \int_0^{t_a} F dt$$

- Action time – often called total impulse – true in sense that tail-off impulse is essentially unusable

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Thrust

- 2 versions of time-average thrust

$$\overline{F}_b = \frac{I_b}{t_b} \qquad \overline{F}_a = \frac{I_a}{t_a}$$

- Burn time average $\overline{F}_b$ most often used
- Frequently referred to as average thrust
- Maximum thrust is critical to design of case
  - Calculated from internal ballistics for hot grain using maximum burning rate

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Specific Impulse

- Propellant Isp obtained from thrust time test data

$$I_{sp} = \frac{I_a}{m_p g_o}$$

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Motor Weight

- Total weight can be expressed as

$$m_{SRM} = \frac{m_p}{\eta}$$

- Average propellant mass fraction of 23 current motors is 0.93 with correlation of 0.954
- Useful equation

$$m_{SRM} = \frac{m_p}{0.93}$$

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Selecting a Solid Motor

- Highly unusual to develop a custom solid motor for spacecraft
- Currently motor is selected from wide array of off-the-shelf motors offered by industry
- "Off the shelf" – design, manufacturing process, tooling, and test equipment exist, have been qualified, and motors have flight history
  - Does not mean motors are immediately available from warehouse
- Key parameter – total impulse mission requirement
- Many motors are designed to be offloaded slightly – can tailor desired total impulse

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