Georgia Tech

Jet and Rocket Propulsion

AE4451

LECTURE 21

Sedina Tsikata – February 27, 2023

Overview

- what we saw last time
  - rockets
    - liquid propellant systems and components
    - pressure losses
    - cycle analysis for liquid propellant systems

- today:
  - solid rocket motors

Solid-rocket motor (SRM)

Introduction

- SRMs are the oldest rocket technology

  - first known use was for military purposes: China in 1232 – "flying fire arrows" , tube filled with gunpowder to repel invasion

  - 13th to 15th centuries: use for war, fireworks displays

  - invention of staging: German Johann Schmidlap in 16th century ("step rocket") for fireworks
    - ✸ large rocket fired and raised to high altitude
    - ✸ subsequent ignition of smaller rockets

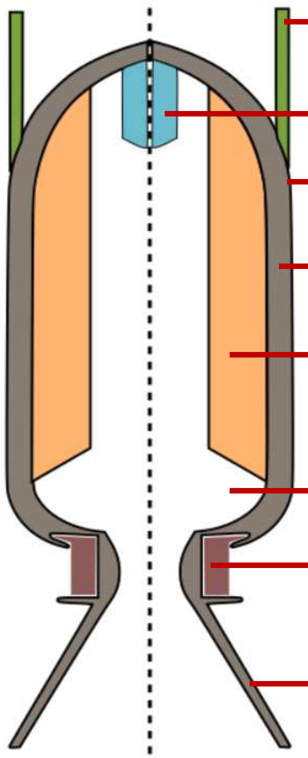
  - 18th century: "Congreve rocket" by William Congreve, Napoleonic wars etc.

  - 19th century: removal of guide stick and use thrust vectoring (holes), William Hale

US Naval Academy, from Congreve's notebook

Solid-rocket motor (SRM)

Basic structure



1. thrust skirt
   - couples motor case to vehicle
   - transfers thrust load to vehicle

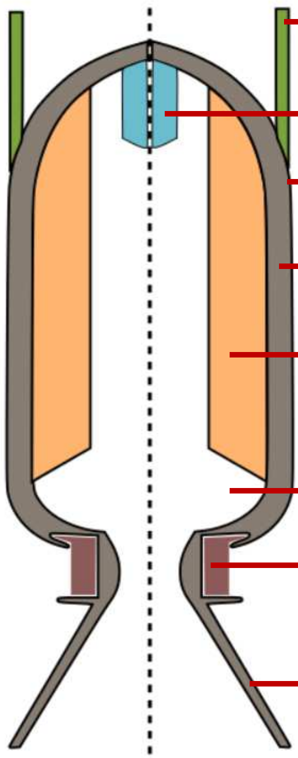
2. igniter
   - provides energy to initiate solid combustion process

3. motor case
   - contains combustion pressure
   - made from Ti / high strength steels / wound fiber

4. internal insulation
   - protects motor case from combustion temperatures
   - low thermal conductivity, reducing heat transfer to motor structure
   - ablative material to dissipate heat
   - eg. rubber mixed with fibrous material, e.g. Kevlar

Solid-rocket motor (SRM)

Basic structure



5. propellant grain
   - contains granular fuel and oxidizer in polymer matrix binder

6. port/bore
   - provides propellant surface area for combustion; increased surface area increases chamber pressure and thrust
   - number of ports chosen according to pressure, thrust needs
   - port shape sets burn surface area
   - typical shapes: star pattern or cylinder with transverse slots

7. polar boss
   - provides coupling between nozzle and motor case
   - transfers nozzle loads to motor case

8. nozzle
   - as usual, controls expansion of hot gases

Solid-rocket motor (SRM)

Comparison to liquid rocket engines

- advantages
  - simple (fewer components)
  - reliable (few moving parts), minimal maintenance
  - high density reduces storage volume
  - readily stored (in comparison to cryogenics)
  - easier to start (no pumping) and quick response
  - more readily scalable (low to high thrust)

- disadvantages
  - lower specific impulse
  - harder to test subcomponents
  - hard to actively throttle
  - manufacturing defects (e.g. cracks) and degradation at extreme storage conditions, also sensitivity to mechanical shocks or sharp temperature changes
  - usually no restarts
  - emissions (HCl, chlorinated compounds) and signature (smoke) for common propellants

Solid-rocket motor (SRM)

Modern applications

- strap-on boosters for space-launch vehicles, very high thrust requirement (e.g. > 5MN)

- upper stage propulsion systems for orbital-transfer vehicles (OTVs)
    - raise payload, e.g. satellite, from LEO to higher orbit
    - solid rocket OTV usually heavier than liquid rocket OTV, but more compact
    - solid: higher density impulse ($I_{sp} \times \rho$)

- spin/despin systems for aircraft

- strategic and tactical missile propulsion

- gas generators for starting liquid rocket engines and pressurizing tanks

Solid-rocket motor (SRM)

Solid propellant categories

- 2 types:

  1. Homogeneous: e.g. double-base propellant

  - typically, mixture of explosive liquid and self-burning powder
    - fuel and oxydizer not separate
    - example: nitrocellulose + nitroglycerin + additives (stabilizers, burn rate modifiers, etc.)
    - unstable molecules: a disadvantage

  - can be extruded or cast

  - used in early modern rockets
    - replaced gun/black powder
    - used in WWII JATOs (jet-assisted take-off) and early Sidewinder
    - weapons systems

  - single-base, triple-base propellants also exist

Roxel

Boeing B-47B with rocket-assisted take-off (USAF)

Solid-rocket motor (SRM)

Solid propellant categories

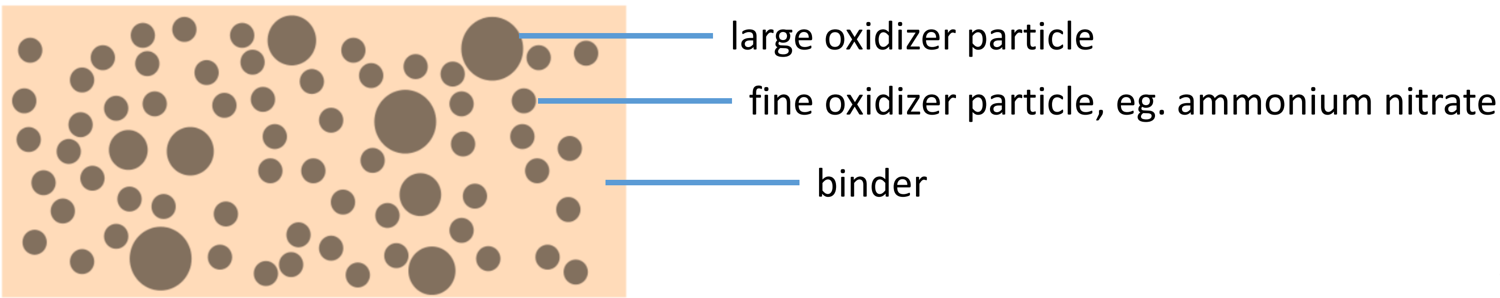
- 2 types

2. Heterogeneous: e.g. composite propellants

- separate fuel and oxidizer materials combined in grain

- oxidizer = 80% or more of propellant mix
  - typical solid oxidizers: ammonium perchlorate, ammonium nitrate, potassium chlorate
  - combined with polymer binding agent (e.g. polybutadiene)

- fuel also contributes form + rigidity to grain
  - typical fuel: metal powder, e.g. Mg, Al, Be; binder also fuel
  - minor components: catalyst, curing agents, plasticizer etc.



Solid-rocket motor (SRM)

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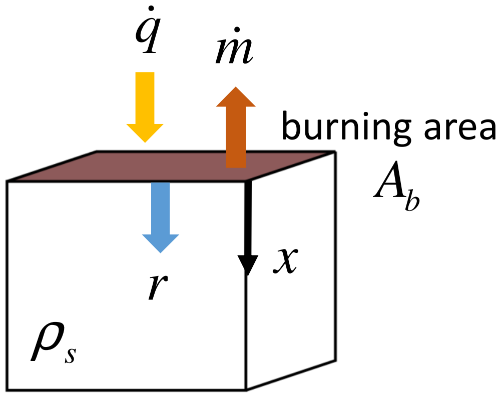
Mass production rate

- the propellant is converted to gas due to heat feedback from the flame at a rate given by $\dot{m} = r\rho_s A_b$



where *r* is the surface regression rate or burn rate

$$r = \frac{dx}{dt}$$

alternatively, regression rate = $\dot{r}_b$

- standard model used = burning rate law or St. Robert's law or Vieille's law

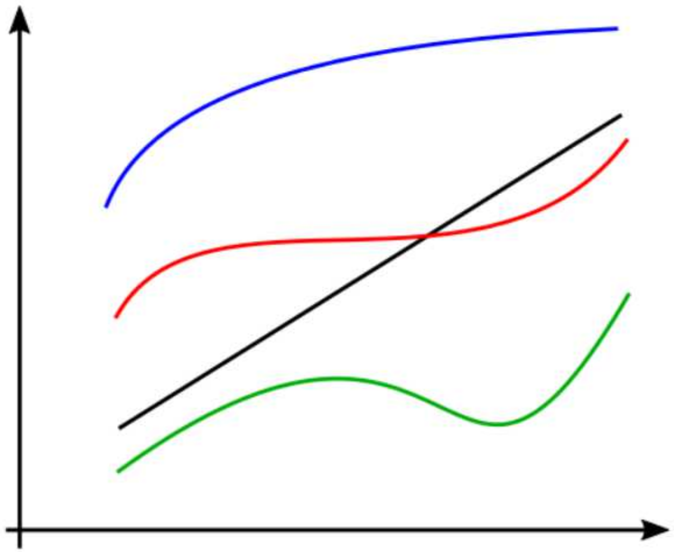
$$r = r_0 + a p_c^n \qquad a = f\,(T_{solid}, \ldots)$$

$$\ln r = \ln a + n \ln p_c \qquad (r_0 \text{ usually } 0)$$

a = burn rate coefficient
n = pressure exponent
p_c = combustion chamber pressure

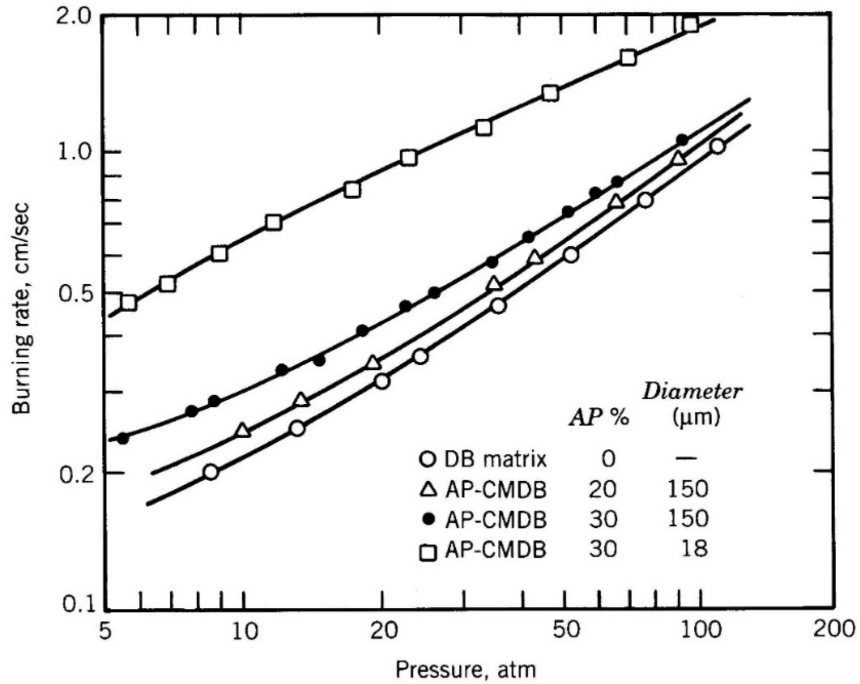
Solid-rocket motor (SRM)

Burn rate behavior for solid propellants



regression rate r $\quad r = r_0 + a p_c^n$

- St. Robert's Law
- variable *n* propellant
- plateau burning
- mesa burning



Sutton

FIGURE 11–7. Measured burning rate characteristics of a double-base (DB) propellant and three composite-modified double-base (CMDB) propellants which contain an increasing percentage of small diameter (159 µm) particles of ammonium perchlorate (AP). When the size of the AP particles is reduced or the percentage of AP is increased, an increase in burning rate is observed. None of these data form straight lines.

Solid-rocket motor (SRM)

Factors affecting the burn rate *r*

- strongly dependent on pressure

$$\ln r = \ln a + n \ln p_c$$

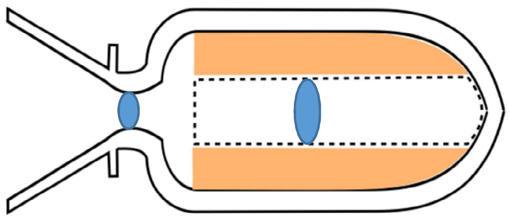
- *a* and *n* cannot be theoretically predicted for a particular propellant: empirical determination
  - also, their values are typically valid only over specific pressure range

$$r = a p_c^n$$

- initial temperature of grain influences burning rate
  - another disadvantage of SRM: sensitivity to environmental conditions

- mass flux across grain surface
  - increase in burn rate possible due to combustion gas velocity:
    "erosive burning" (increased heat flux)
  - occurs at some threshold velocity
  - rate may also be decreased: "negative erosive burning"

→ modify port area to throat area



example values:
Space Shuttle solid rocket booster

a = 0.0445289
n = 0.35
p_c = 4.3 MPa

r = 9.34 mm/s
(*a*, *n*, specified for p_c in Pa, *r* in mm/s)

Solid-rocket motor (SRM)

Factors affecting the burn rate *r*

- pressure drop across combustion chamber
  - assoc. with acceleration of products to nozzle

- motor acceleration
  - burning surfaces at 60 – 90° with respect to acceleration vector prone to increased *r*

How to modify the burn rate?

- modify particle size
  - important for some oxidizer types (ammonium perchlorate)

- modify oxidizer to fuel ratio

- add catalyst to propellant mixture

- lower chamber pressure, but will also reduce I_{sp} and so motor performance

Solid-rocket motor (SRM)

Erosive burning effect

- if the burn rate is modified by erosive burning, a modified expression can be written

$$r_b = a p_c^n + \frac{\alpha G^{0.8}}{L^{0.2}} e^{\frac{-\beta \rho_p r_b}{G}}$$

Lenoir and Robillard (1957)

G = bore mass flux (kg/m².s)
L = grain length
α,β = constants (to determine empirically)
ρ_P = density

- the above is true if the combustion gas flux is the source of the burn rate increase

- for an erosive burning effect due to compressibility,

$$r_b = a p_c^n (1 + kM)$$

Green (1954), others

k = constant (to determine empirically)
M = Mach number

Solid-rocket motor (SRM)

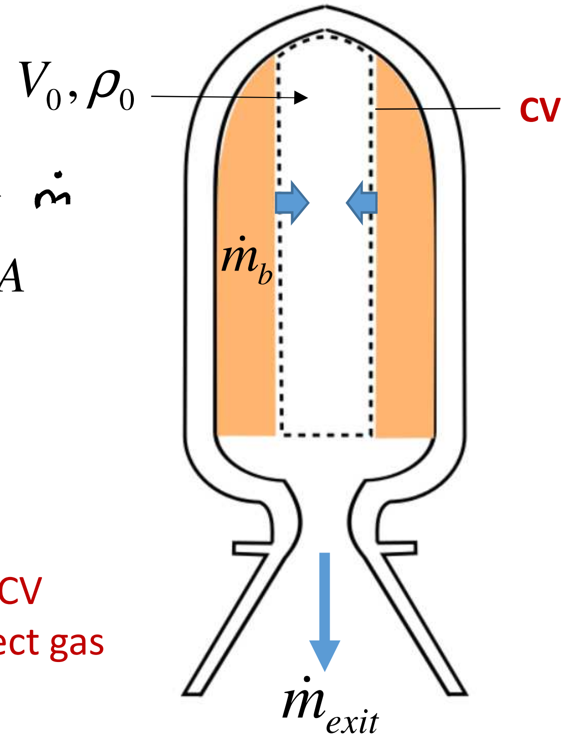
Motor internal ballistics

$$\left.\frac{dB}{dt}\right|_{CM} = \frac{d}{dt}\int_{CV} \rho\beta dV + \int_{CS} \rho\beta(\vec{u}_{rel}\cdot\hat{n})dA$$

$\rho u A = \dot{m}$

mass conservation

$$0 = \frac{d}{dt}\int_{CV}\rho dV + \int_{CS}\rho(\vec{u}_{rel}\cdot\hat{n})dA \quad \Rightarrow 0 = \underbrace{\frac{dm_{CV}}{dt}}_{\dot{m}_{store}} + \int \rho(\vec{u}\cdot\hat{n})dA$$



$$0 = \underbrace{\frac{d}{dt}(\rho_o V_o)}_{V_o \frac{d\rho_o}{dt} + \rho_o \frac{dV_o}{dt}} + \dot{m}_{exit} - \dot{m}_b$$

$$\dot{m}_b \rightarrow r\rho_s A_b$$

$$\dot{m}_{exit} \rightarrow \frac{p_o}{\sqrt{RT_o}}\sqrt{\gamma}\left(\frac{2}{\gamma+1}\right)^{\gamma+1/2(\gamma-1)} A_t$$

$$\underbrace{V_o \frac{d\rho_o}{dt}}_{\frac{1}{RT_o}\frac{dp_o}{dt}\ (\dot{m}_{store,p})} + \underbrace{\rho_o \frac{dV_o}{dt}}_{\rho_o(A_b r)\ (\dot{m}_{store,V})}$$

assumptions
- uniform gas properties in CV
- thermally, calorically perfect gas
- T_o = constant (e.g., T_{ad})
- p_o, A_b, r given at time t

$$\underbrace{\dot{m}_{store,p} + \dot{m}_{store,V}}_{\dot{m}_{cv}} = \dot{m}_b - \dot{m}_{exit}$$

Solid-rocket motor (SRM)

Motor internal ballistics

$$\dot{m}_{store,p} + \dot{m}_{store,V} = \dot{m}_b - \dot{m}_{exit}$$

- solving for rate of pressure change, using terms from previous slide

$$\frac{V_o}{RT_o}\frac{dp_o}{dt} = rA_b(\rho_s - \rho_o) - p_o A_t \underbrace{\sqrt{\frac{\gamma}{RT_o}\left(\frac{2}{\gamma+1}\right)^{\gamma+1/\gamma-1}}}_{=1/c^*}$$

for steady burning,

$$\frac{dp_o}{dt} = 0 \quad \Rightarrow p_o = r\frac{A_b}{A_t}\underbrace{(\rho_s - \rho_o)}_{\sim \rho_s \text{ often}} c^*$$

grain density >> gas density

characteristic velocity

- from burning rate law:

$r = ap_o^n$

$$p_o = ap_o^n \frac{A_b}{A_t}(\rho_s - \rho_o)c^* \quad \Rightarrow p_o = \left[aK(\rho_s - \rho_o)c^*\right]^{1/1-n}$$

where $A_b / A_t \equiv K$

for steady burning (if a, n, T_o, γ, and A_t constant) then A_b must be constant $\quad p_o \sim K^{1/1-n}$

Solid-rocket motor (SRM)

Motor stability

- from mass conservation

$$\dot{m}_{store,p} + \dot{m}_{store,V} = \dot{m}_b - \dot{m}_{exit}$$

$$\dot{m}_{store,p} = \dot{m}_b - \dot{m}_{store,V} - \dot{m}_{exit}$$

$$= \dot{m}_{incr} - \dot{m}_{exit}$$

where

$$\dot{m}_{incr} = A_b(\rho_s - \rho_o) r \propto p_o^n$$

$$\dot{m}_{exit} = \frac{p_o A_t}{c^*} \propto p_o$$

$$r = r_0 + a p_c^n$$

$$* \; c^* \equiv \frac{p_o A_t}{\dot{m}}$$

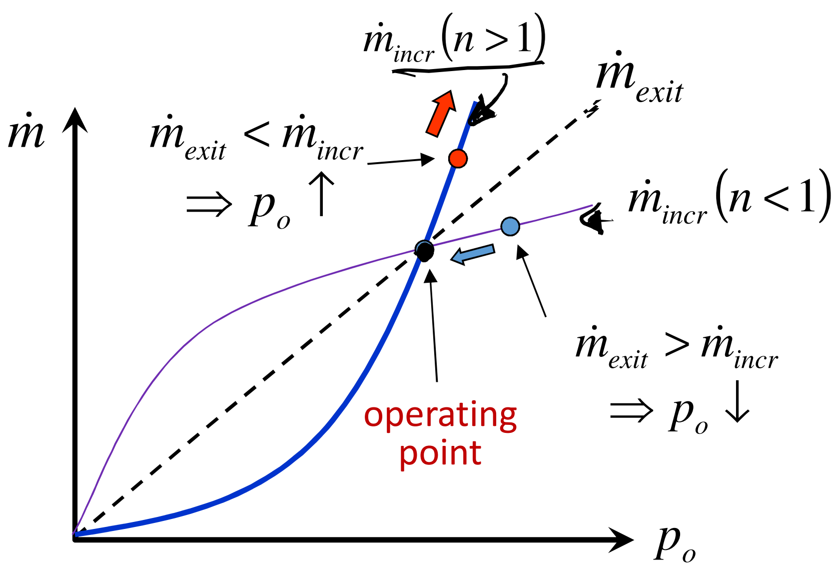
$$\frac{1}{RT_o}\frac{dp_o}{dt} \quad \rho_o(A_b r)$$

$\dot{m}_{store,p}$ $\quad$ $\dot{m}_{store,V}$

$\dot{m}_{incr} = \dot{m}_{exit}$

$\dot{m}_{incr} = c_1 p_o^n$ — $n \leq 1$

$\dot{m}_{exit} = c_2 p_o$



- for stable operation (p_o= const), need $\dot{m}_{store,p} = 0$
- stability only when:
  - $n \leq 1$
  - normally use 0.3 < n <0.7

Solid-rocket motor (SRM)

Combustion limits

- if n or p_o too low
  - unstable combustion
  - after ignition, propellant soon stops burning ($r \rightarrow 0$)

- at too high p_o
  - possibility of erratic, unpredictable burning
  - e.g., $p_o > 5000$ psi

Solid-rocket motor (SRM)

Pressure histories

- motor designer can adjust pressure profile ("history") of a solid motor by arranging how burning area changes with time (grain geometry)

- thrust given by $\tau = p_o A_t c_\tau$

  - i.e. thrust history of motor essentially follows motor's pressure history

- characterize pressure/thrust histories as generally

  - progressive: increase with time
  - neutral: constant with time
  - regressive: decrease with time
  - combinations

We can define a web distance *W*

= linear amount of propellant consumed as measured normal to local burn surface

- integral of burning rate history, i.e.

$$\mathcal{W} = \int_0^{t_b} r(t)\,dt$$

t_b = burn time

Solid-rocket motor (SRM)

Grain geometries and thrust history

- using grain geometry to modify thrust history

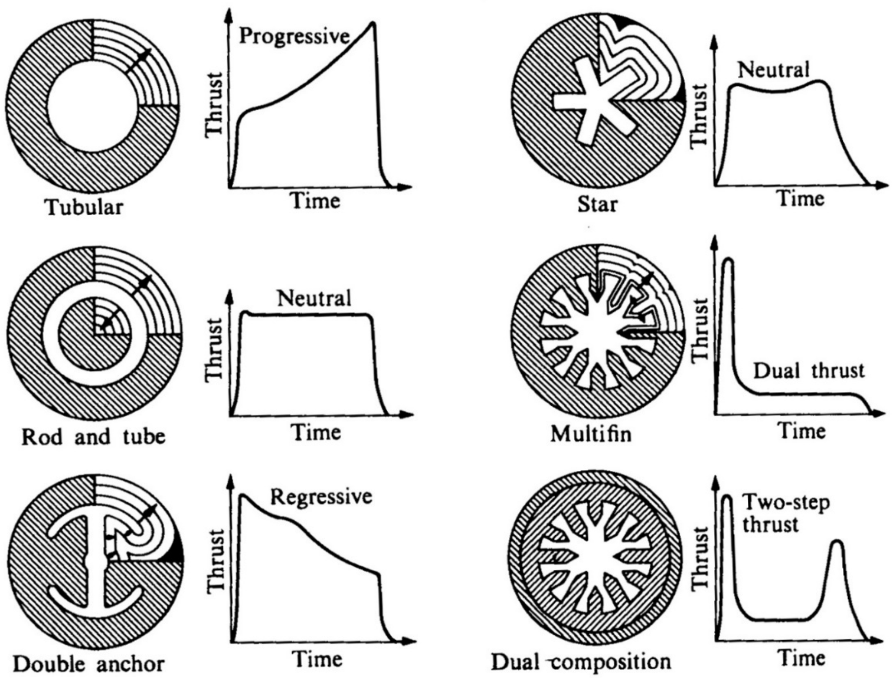


FIGURE 12.17 Internal-burning charge designs with their thrust-time programs. (Courtesy Shafer [18].)

from Hill and Peterson

Solid-rocket motor (SRM)

Other grain geometries

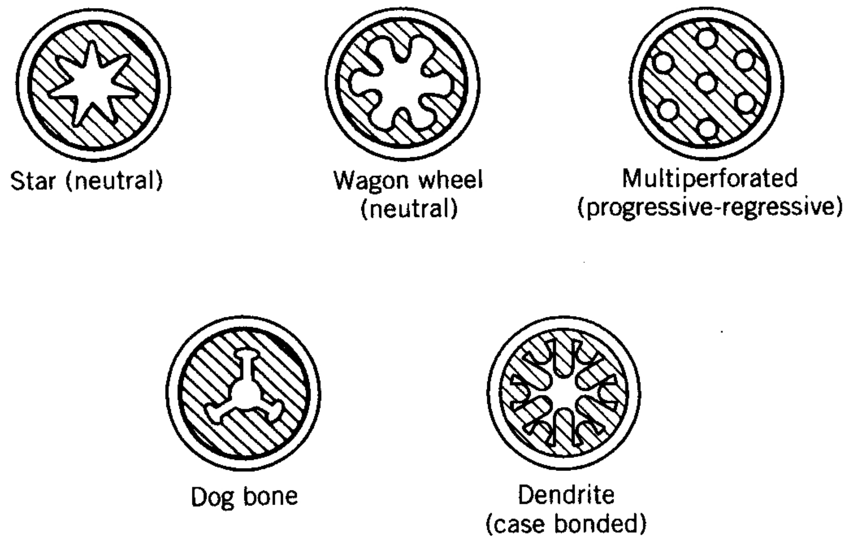
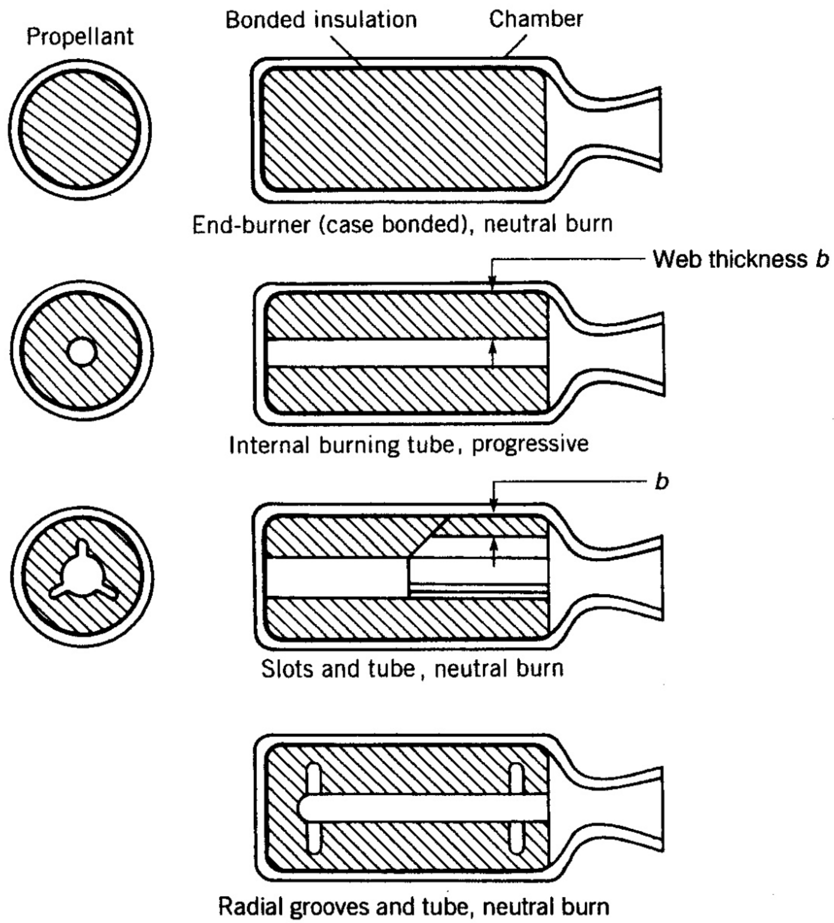


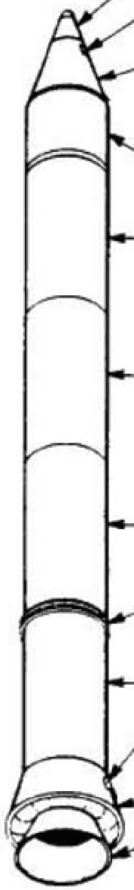
FIGURE 11–16. Simplified diagrams of several grain configurations.



from Sutton

Solid-rocket motor (SRM)

Space shuttle rocket booster



NASA

- largest SRM flown and first designed for reuse
  - diameter = 3.71 m, length = 45.5 m
- sea level thrust: 14.7 MN
- mass: 5.9 x 10^5 kg (inert: 8.7 x 10^4 kg)
  ~ 71% of thrust at lift-off and ascent
- propellant composition (mass fractions)
  AP: 69.6%, Al: 16%,
  Fe_2O_3 (catalyst): 0.4%,
  HTPB (binder): 12.04%
  epoxy (curing agent): 1.96%
- four segments
  - 11 point star (neutral) in forward segment
  - double truncated cone (regressive) in 3 segments

Solid-rocket motor (SRM)

Space shuttle rocket booster