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# Jet and Rocket Propulsion

## AE4451

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LECTURE 24

*a*

# Overview

- what we saw last time
  - solid rocket motors
    - grain geometry considerations, practical examples
    - calculation methods for pressure history
- today:
  - heat transfer

## Monday, March 13

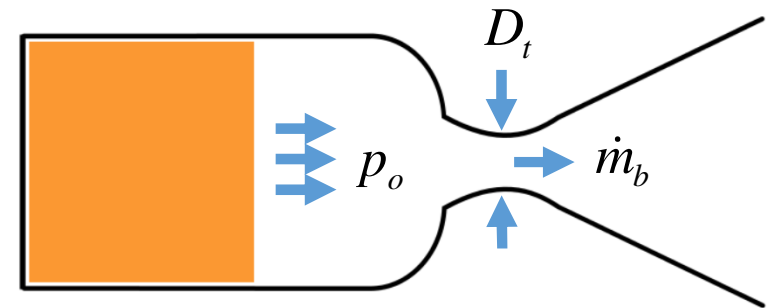
**Ryan J. McKibben:** 10 – 11 am

President's Suites C and D in the Student Success Center

GTAE grad who is now Deputy Chief of Mechanical Operations at Stennis. He is overseeing operations of both commercial and NASA programs testing components and engine assemblies there. He did a lot of work with the Artemis engine over the past few years.

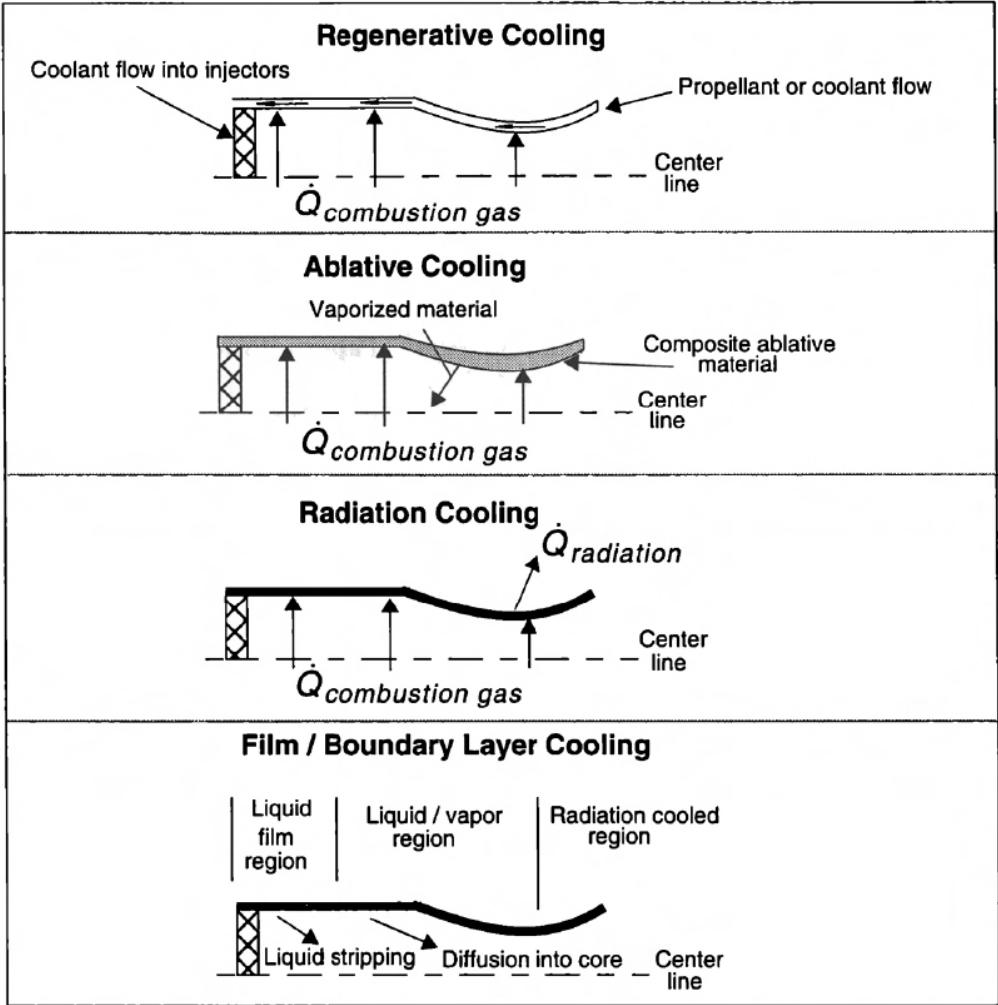
# Heat transfer

- up to this point:
  - assumed no heat transfers from the propulsion system
  
- heat transfer important issue for many parts of system
  - injectors
  - combustion chamber
  - nozzle
  
- material considerations
  - refractory: heat resistance, e.g. carbon, silicon carbide, alumina
  - melting point
  - thermal conductivity
  - coefficient of thermal expansion
  - specific heat capacity



# Heat transfer

## Combustion chamber cooling techniques



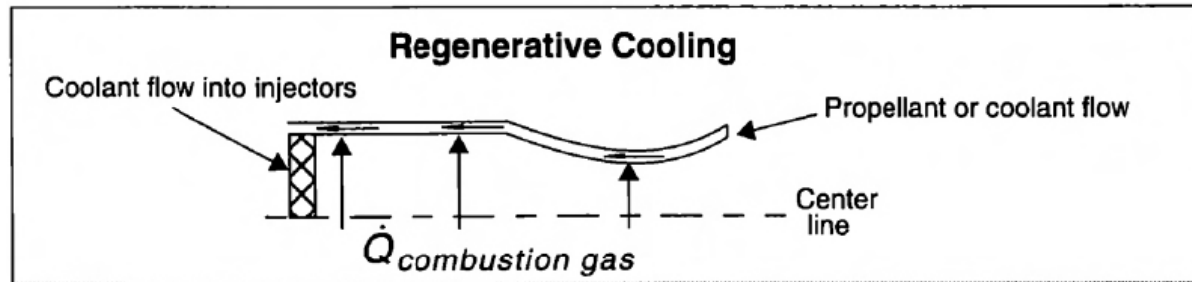
key method for liquid rocket engines

key method for solid rocket motors

Humble

# Heat transfer

## Combustion chamber cooling techniques: regenerative

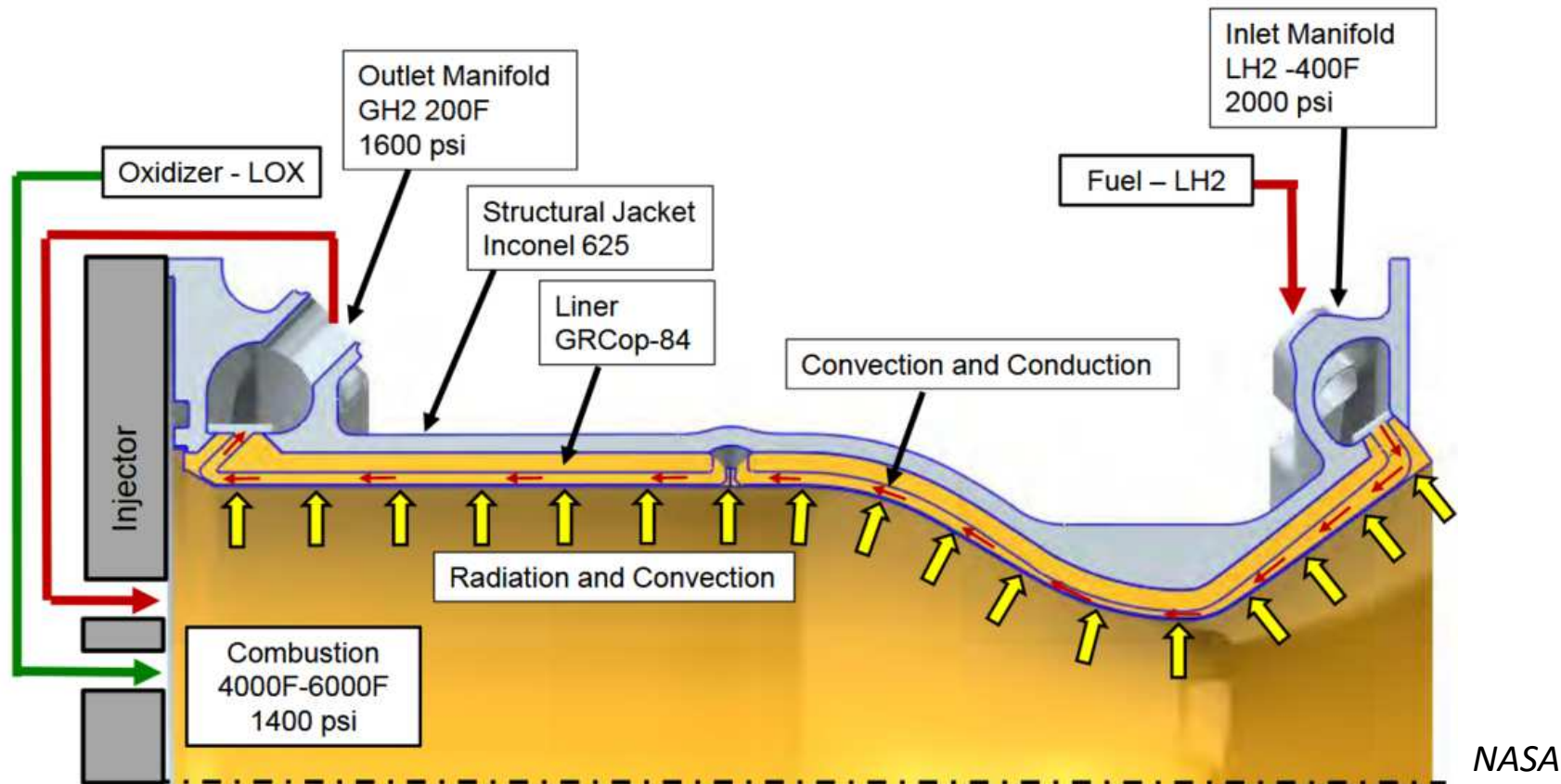


- used on large liquid rocket engines (LREs), mid to high thrust; boost, sustain, orbit transfer applications
- fuel piped through passages around combustion chamber and nozzle
  - radial conduction of heat through chamber walls, then convection by flowing propellant
  - allows recapturing of waste heat
- inter-regenerative cooling: create cool propellant film at wall with annular passages inside combustion chamber
  - mixing not optimal, loss of  $I_{sp}$
- challenge with regenerative cooling: fabrication complexity

# Heat transfer

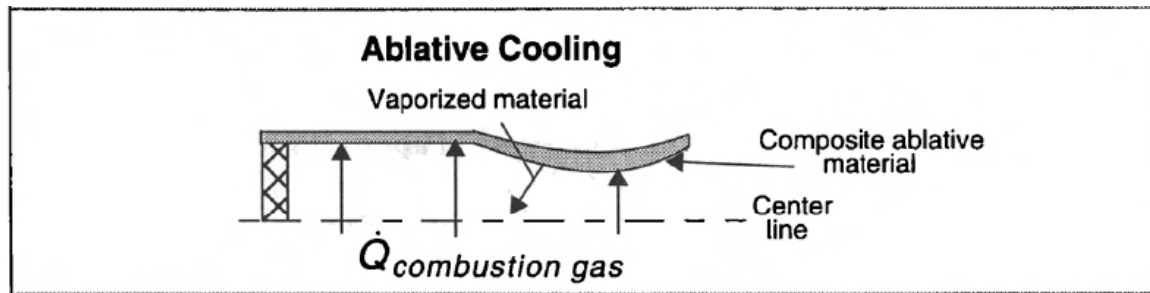
## Combustion chamber cooling techniques: regenerative

Physics of a Regeneratively Cooled Combustion Chamber



# Heat transfer

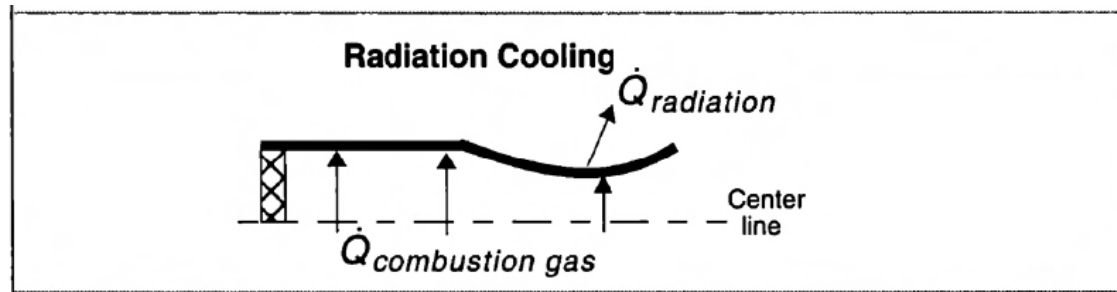
## Combustion chamber cooling techniques: ablative



- low- to mid-sized thrust levels; boost, sustain, orbit transfer applications
- hot combustion gases vaporize ablator material (silica, carbon, composites, etc.) lining wall
  - sufficient mass to carry away heat
  - limited lifetime (< 2000 s of firing)
  - can stop and restart if sufficient ablative material remains

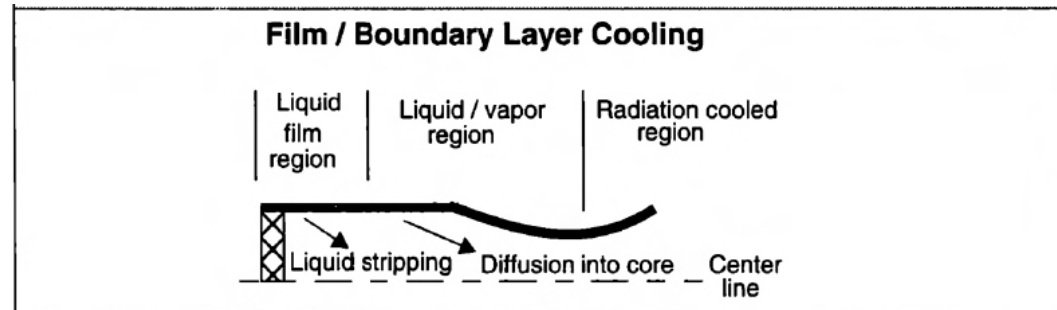
# Heat transfer

## Combustion chamber cooling techniques: radiation cooling



- station-keeping, orbit transfer applications
- hot combustion gases transfer heat by convection and radiation
  - radial conduction of heat through walls
  - radiation to surroundings
- axial variation of chamber wall temperatures exists, due to variation in flow speed
  - highest wall heat load at throat: high flow speed + stagnation values of  $T$ ,  $\rho$
  - lower wall loads in supersonic nozzle: flow speed high but lower (static) temperatures
- high melting point materials needed, e.g. columbium, + ~~coatings~~ *coatings* for oxidation protection (e.g. rhenium)

## Combustion chamber cooling techniques: film/boundary layer cooling



- low thrust applications, orbit-transfer, station-keeping
- rarely used alone: instead, combined with other methods
- cool film of fuel (usually) created along wall
  - increase of thermal resistance and reduces wall temperature
  - near throat: mostly vapor remaining
  - further downstream: increased area for radiation cooling

# Heat transfer

## Conductive heat transfer

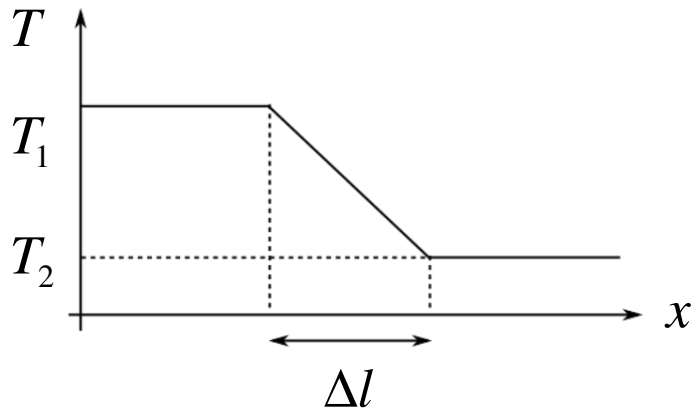
- heat transfer between molecules as a result of a temperature gradient

rate of heat transfer  $\dot{Q} = -kA \frac{dT}{dx}$

$k$  = thermal conductivity

$A$  = cross-sectional area (perpendicular to  $x$ , direction of heat flow)

$dT/dx$  = temperature change along  $x$



heat transfer per unit area

$$\dot{q} = \frac{\dot{Q}}{A} = -k \frac{(T_2 - T_1)}{\Delta l}$$

*Handwritten notes: a red arrow points to the negative sign in the numerator, and red scribbles are under the denominator.*

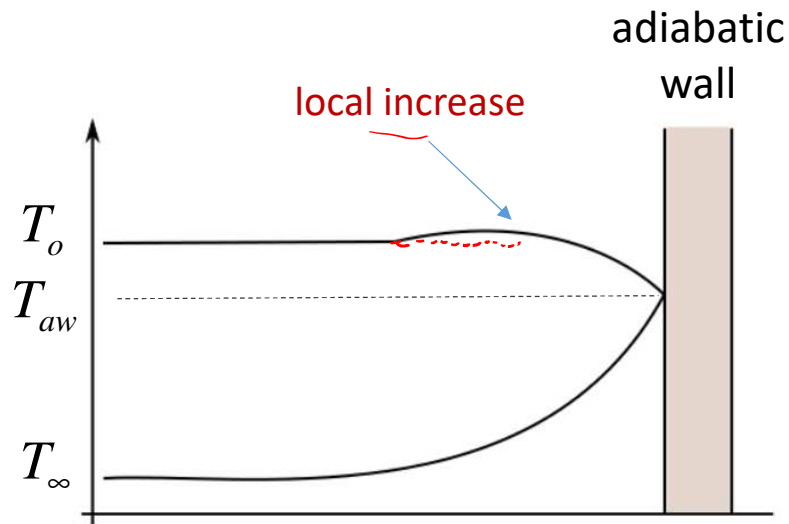
$T_2 < T_1$ :  $q$  positive in  $+x$  direction

- applied to rockets
  - high conductivity heat sink to pull heat from combustion chamber and nozzle
  - low conductivity insulator on combustion chamber and nozzle

# Heat transfer

## Convective heat transfer

- heat transfer between moving fluid and solid surface



- initial high speed, low temperature free stream:  $T_\infty$
- velocity decreases approaching wall, increase in T

$$T_{aw} \neq T_o$$

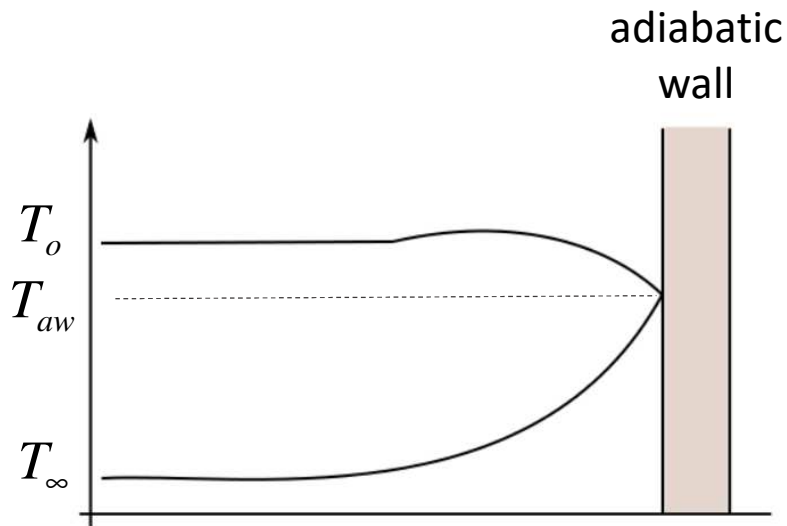
- large difference between wall and free stream
- heat transfer back into free stream from hot, slow moving fluid near wall
- hence  $T_{aw} < T_o$

- from energy conservation: near-wall region has to have higher stagnation temperature than free stream

$T_{aw}$  = adiabatic wall temperature  
 $T_o$  = chamber stagnation temperature  
 $T_\infty$  = free stream static temperature

# Heat transfer

## Convective heat transfer



$h_g$  = heat transfer coefficient of hot gas (W/m<sup>2</sup>.K)  
 $T_{aw}$  = adiabatic wall temperature  
 $T_{wh}$  = actual wall temperature on hot side

- cannot directly apply stagnation and isentropic relations to find free stream properties

- define recovery factor  $r$  
$$r = \frac{T_{aw} - T_{\infty}}{T_o - T_{\infty}}$$

- typical rocket propellants, compressible boundary layers up to  $M = 4$

$$r = 0.91 \quad (\text{from experiments})$$

$$T_{aw} = rT_o + T_{\infty}(1 - r) \quad \text{find adiabatic wall temperature}$$

$$\dot{q} = h_g (T_{aw} - T_{wh})$$

assumption: 1D flow

# Heat transfer

## Convective heat transfer

$$* T_{aw} = rT_o + T_\infty (1 - r)$$

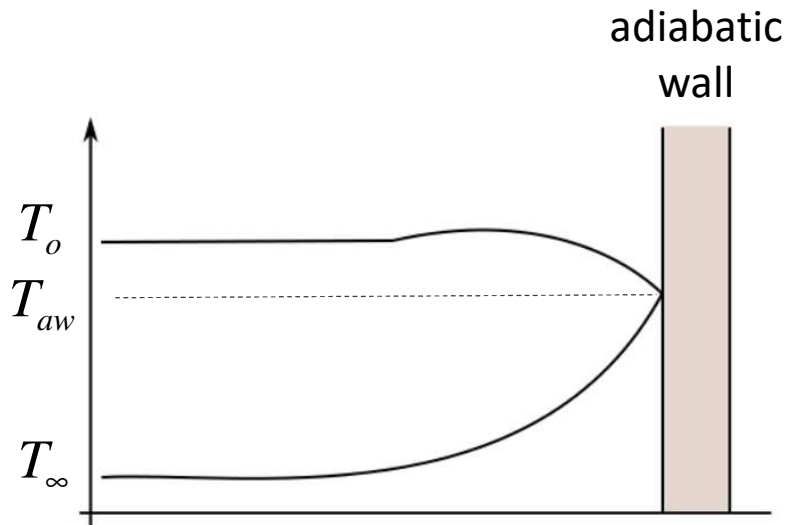
- special conditions at throat

$T_{aw} \approx T_{og}$  i.e. adiabatic wall temperature ~ hot (chamber) gas stagnation temperature

$$r = 1$$

- for  $T_{aw} = T_{wh}$

no heat transfer  $\dot{q} = 0$



$h_g$  = heat transfer coefficient of hot gas (W/m<sup>2</sup>.K)

$T_{aw}$  = adiabatic wall temperature

$T_{wh}$  = actual wall temperature on hot side

# Heat transfer

## Convective heat transfer

- application in regenerative cooling for liquid rocket engines
  - methods:
    - (i) pipe propellant through tubes around nozzle, or
    - (ii) create thin film of fuel/cool gas along nozzle wall
  - absorption of heat from nozzle wall
  - low speed flow

$$\dot{q} = h_g (T_{aw} - T_{wh}) \quad \text{not applicable}$$

- convection equation in regenerative cooling

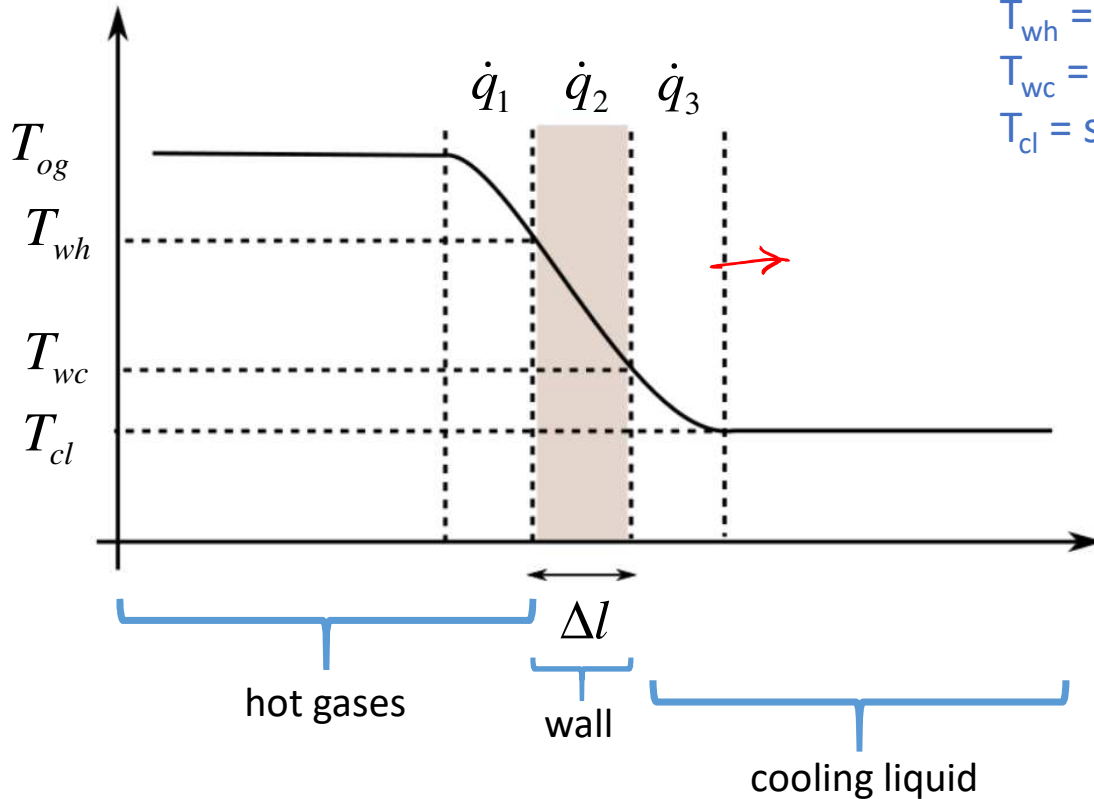
$$\dot{q} = h_l (T_{wc} - T_l)$$

$h_l$  = heat transfer coefficient of cooling liquid  
 $T_{wc}$  = wall temperature on cooled wall side  
 $T_l$  = temperature of cooling liquid

*Handwritten notes:*  
 |  
 Cooled side wall temp  
 liquid temp

# Heat transfer

## Convective heat transfer



$T_{og}$  = stagnation temperature of gases  
 $T_{wh}$  = static temperature on hot side of wall  
 $T_{wc}$  = static temperature on cold side of wall  
 $T_{cl}$  = static temperature of cooling liquid

heat transfer from gas to wall

$$\dot{q}_1 = h_g (T_{og} - T_{wh})$$

*coefficient for gas*

heat transfer through wall

$$\dot{q}_2 = \frac{-k}{\Delta l} (T_{wc} - T_{wh})$$

*conductivity*  
*final*      *initial*

heat transfer from wall to coolant

$$\dot{q}_3 = h_l (T_{wc} - T_{cl})$$

*coeff. liquid*

$h_l, h_g$  = heat transfer coefficients of liquid, gas

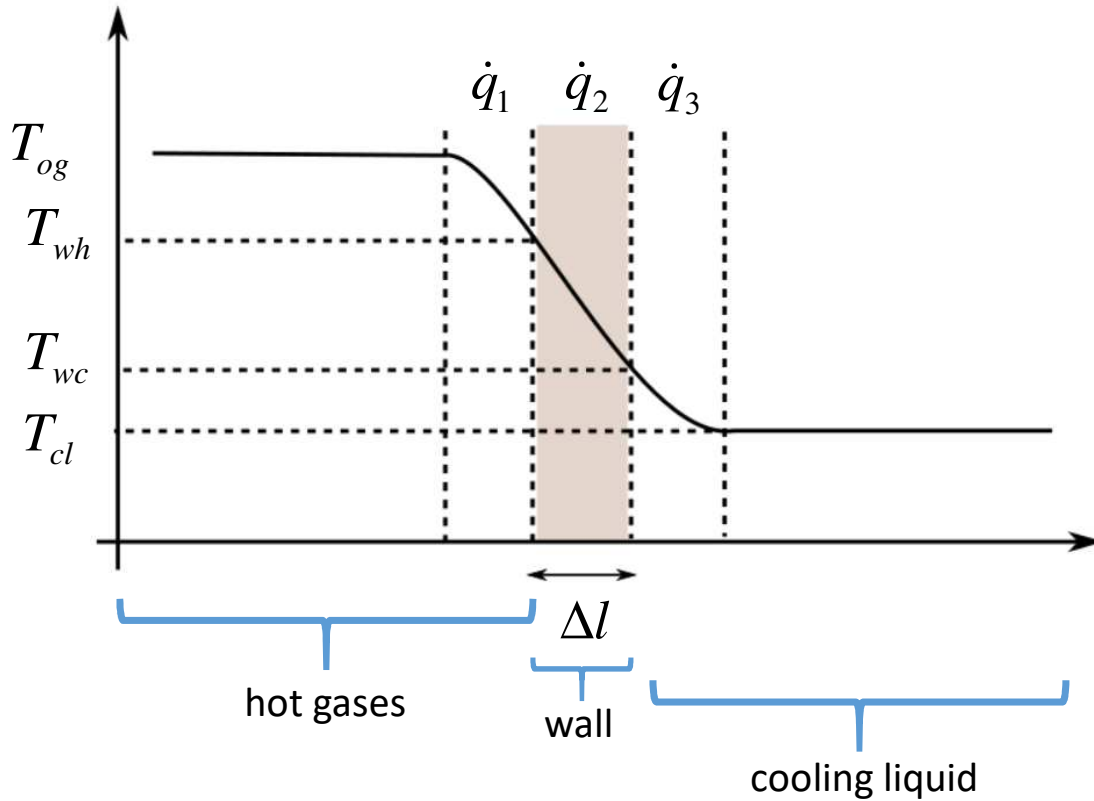
# Heat transfer

## Convective heat transfer

- assuming steady state conditions,

$$\dot{q}_1 = \dot{q}_2 = \dot{q}_3$$

$$\dot{q} = \frac{T_{og} - T_{cl}}{\frac{1}{h_g} + \frac{\Delta l}{k} + \frac{1}{h_l}}$$



# Heat transfer

## Convective heat transfer: heat transfer coefficients

- heat transfer coefficients a function of:  $h_g, h_L$

### fluid properties

specific heat, conductivity, density, viscosity

### flow conditions

velocity, channel dimensions

$$h_g = 0.026k \left( \frac{\rho v}{\mu} \right)^{4/5} \left( \frac{1}{D} \right)^{1/5} \left( \frac{c_p \mu}{k} \right)^{2/5} \quad (\text{empirically-determined})$$

### assumptions:

- fully-developed flow
- circular pipe

$k$  = thermal conductivity of gas

$v$  = gas flow velocity

$D$  = flow channel diameter

$\mu$  = dynamic viscosity of gas

$\rho$  = gas density

$c_p$  = specific heat capacity

# Heat transfer

## Convective heat transfer: heat transfer coefficients

- heat transfer coefficient for slow-moving coolant

$$h_l = 0.023 \left( \frac{c_p \dot{m}}{A} \right) \left( \frac{\mu}{D \rho v} \right)^{1/5} \left( \frac{k}{\mu c_p} \right)^{2/3} \quad \text{(empirically-determined)}$$

assumptions:

- fully-developed flow
- circular pipe

$k$  = thermal conductivity of coolant

$v$  = coolant flow velocity

$D$  = flow channel diameter

$\mu$  = dynamic viscosity of coolant

$\rho$  = coolant density

$c_p$  = specific heat capacity

- how to increase transfer coefficient?  
e.g. increase mass flow rate, decrease  $D$

# Heat transfer

## Convective heat transfer: overheating coolant

- what happens if coolant overheats?
  - liquid can vaporize
  - degradation of ~~can~~ multi-component liquids (RP-1), "cracking"
  - potential to produce solid residues, (RP-1, CH<sub>4</sub>), "coking"
  - autoignition (monopropellant, e.g. N<sub>2</sub>H<sub>4</sub>)
  
- if coolant below its critical point, can lead to boiling
  - nucleate boiling: gas bubbles form near wall, collapse as they reach cooler liquid
  - phase change increases heat transfer, however:
    - if significant, gas film stays near surface, drastically reduces heat transfer
  
- if above critical point, no boiling
  - H<sub>2</sub> ( $p_c=13$  bar,  $T_c= -240$  °C)

# Heat transfer

## Radiative heat transfer

- heat transfer from hot gas (e.g. to nozzle and chamber)

$$\dot{q} = \epsilon \sigma T^4$$

$\epsilon$  = emissivity (= 1 for perfect radiator)

$\sigma$  = Stefan-Boltzmann constant,  $5.669 \times 10^{-8} \text{ W/m}^2 \cdot \text{K}^4$

- nozzle radiative absorption: a function of reflectivity and absorptivity
- typical values: 5 – 35% of total heat transfer

# Heat transfer

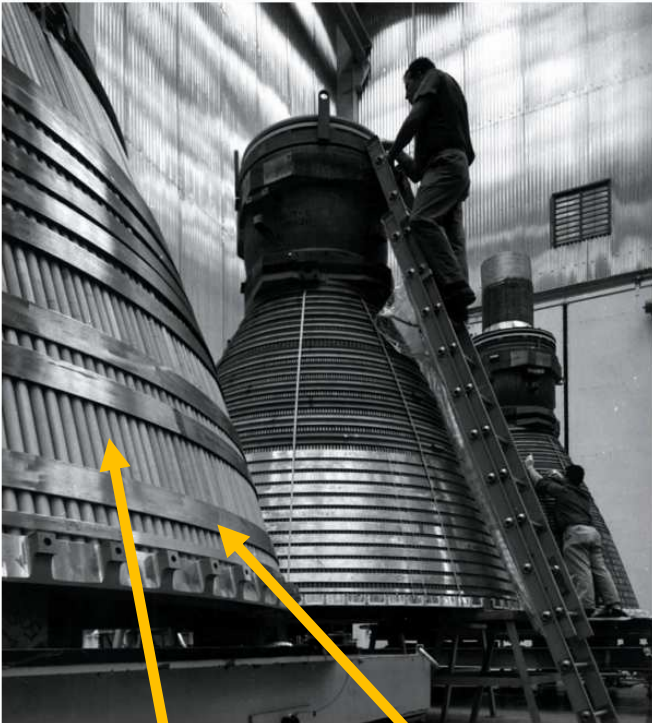
## Regenerative cooling example: F-1 thrust chamber

- H-1, F-1 LRE: NASA's Apollo space program in the 1960s, 1970s
- previous fabrication technique for tubular-wall regeneratively-cooled engines
  - pure Ni tube bundles hand-brazed with low-temperature silver based alloys
- F-1 thrust levels up to 10x those of previous liquid rocket engines
  - 6 670 000 N normal thrust
  - necessity for materials to withstand high temperatures, pressures, stresses
- new materials and processes developed for F-1
  - furnace brazing of Inconel X-750 (Ni alloy) with high-temperature alloys
  - world's largest brazing furnace of its type constructed

# Heat transfer

## Regenerative cooling example: F-1 thrust chamber

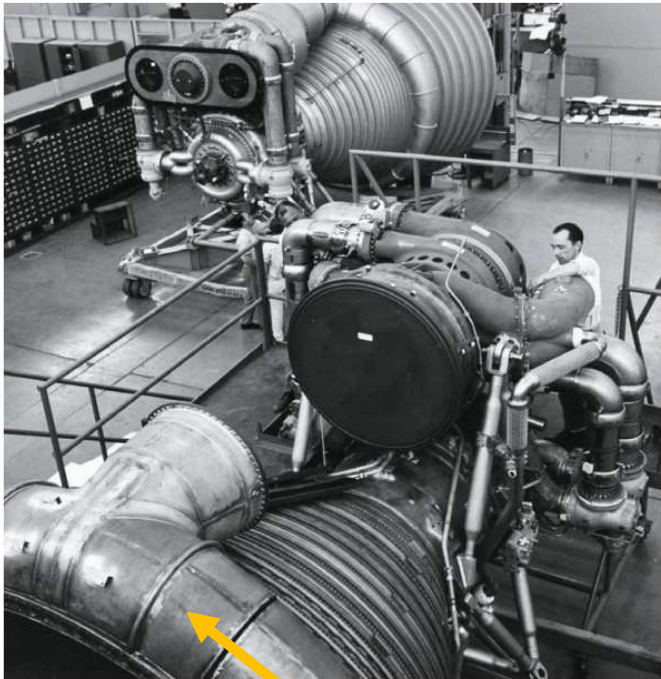
- 178 primary tubes and 356 secondaries
- 900 meters of brazed joints to keep the combustion gases contained



tube bundle and bands around thrust F-1 thrust chamber (Rocketdyne)



turbopump

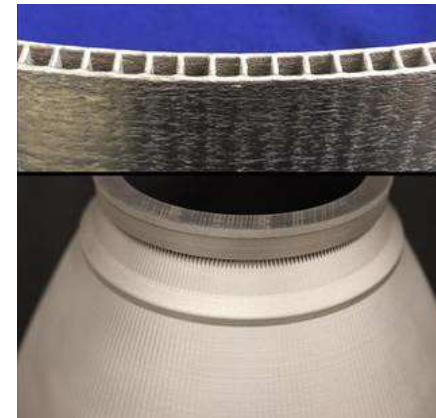


turbine exhaust manifold

# Heat transfer

## Regenerative cooling

- modern technologies can advance how regenerative cooling is implemented
- example:
  - NASA's Rapid Analysis and Manufacturing Propulsion Technology project (RAMPT)
  - being implemented for Artemis manned moon mission
  - additive manufacturing technique to 3D print rocket engine parts, e.g. with cooling integrated



NASA, 2020