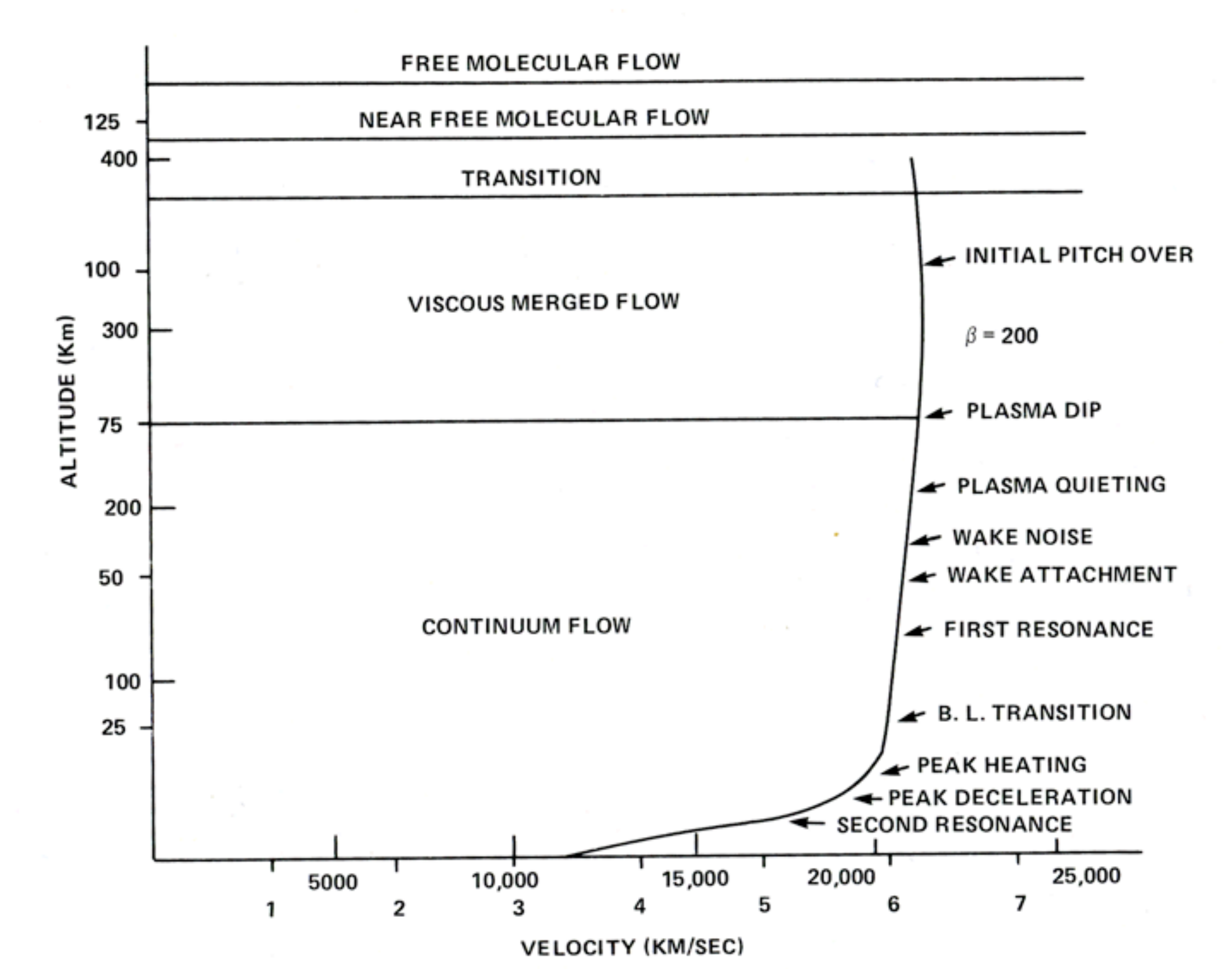
Entry Aerothermodynamics

- Basic flow regimes
- Heating rates

Five Basic Flow Regimes

- Free molecular regime
- Near-free molecular regime
- Transition regime
- Viscous merged boundary layer
- Continuous regime

Entry Flow Regimes



ref: Frank J. Regan, Reentry Vehicle Dynamics AIAA Education Series, NY, NY 1984

Flow Regime Definitions

Knudsen number in rarified flow

$$K = \frac{\lambda}{R_N}$$

Mean free path after collision

$$\lambda_c = \frac{4}{\sqrt{\pi\gamma}} \left(\frac{T_w}{T_\infty} \right)^{\frac{1}{2}} \frac{\lambda_\infty}{M_\infty}$$

If $T_w \sim T_\infty$

$$\lambda_c \cong 1.9 \frac{\lambda_\infty}{M_\infty}$$

Free Molecular Regime

- Orbital flight
- $\lambda \gg \ell$
- Molecule encountering a boundary (e.g., surface of vehicle) attains the state of the boundary after a single collision
- $K_c \geq 10$ or $K_\infty > 5.24 M_\infty$

Near Free Molecular Flow Regime

- Also known as "slip region"
- Gas molecule only attains state of moving boundary after several collisions
- Molecules near the wall will have a different velocity from the wall
- Temperature will be nearly discontinuous function of separation from wall
- $10 \leq K_c \leq \frac{1}{3}$ or $5.4M_\infty \leq K_\infty \leq 0.175M_\infty$

Transition Region

- Very difficult to treat analytically
- For engineering purposes, usually treated as interpolation between slip and viscous flow
- $0.175M_\infty \leq K_\infty \leq 1$

Viscous Merged Layer Regime

- Viscous effects in forming shock and boundary layer must be treated in a unified manner
  - Boundary layer on the wall alters the conditions for the forming shock wave
  - Large pressure gradients across the shock wave significantly alter the boundary layer
- Neither shocks nor boundary layers can be treated as discontinuities
- $1 \leq K_\infty \leq \dfrac{0.1}{\rho_s/\rho_\infty}$

Continuous Regime

- Classical fluid mechanics of high Reynolds number
- Shock waves and boundary layer treated as discontinuities
- $K_\infty > \dfrac{0.1}{\rho_s/\rho_\infty}$
- Subdivided based on Mach number
  - Incompressible (subsonic) $(M \leq \sim 0.8)$
  - Transonic $(\sim 0.8 \leq M \leq \sim 1.3)$
  - Supersonic $(\sim 1.3 \leq M \leq \sim 5)$
  - Hypersonic $(\sim 5 \leq M)$

Aerothermodynamic Heating

$\dot{q} \equiv$ heating rate per unit area $\langle J/sec/m^2 \rangle$

$$\dot{q} \equiv \frac{dq}{dt} = k(T_r - T_w)$$

$k \equiv$ convective heat transfer coefficient$\langle \frac{J}{m^2 secK} \rangle$

$T_r \equiv$ recovery temperature$\langle K \rangle$

$T_w \equiv$ wall temperature$\langle K \rangle$

$$T_r = T_\infty \left(1 + \frac{\gamma - 1}{2} M^2 \right)$$

Wall Temperature

$T_{w\ell} \equiv$ local wall temperature

$$(T_r - T_w)_\ell = (T_\infty - T_{w\ell}) + T_\infty \frac{\gamma - 1}{2} M^2$$

For high Mach numbers,

$$(T_r - T_w)_\ell = T_r - T_{w\ell} \cong T_\infty \frac{\gamma - 1}{2} M^2$$

Mach Number Manipulation

By definition,

$$M^2 = \frac{V^2}{a^2} = \frac{V^2}{\gamma RT}$$

$$M^2 T = \frac{V^2}{\gamma R} = \frac{V^2}{(c_p/c_v)(c_p - c_v)} = \frac{V^2}{c_p(\gamma - 1)}$$

$$T_r - T_{w\ell} = \frac{V^2}{2c_p}$$