Introduction to Hypersonic Flow

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AGENDA

SECTION 1 WHAT DO WE MEAN BY "HYPERSONICS"

SECTION 2 HYPERSONIC AEROTHERMODYNAMICS

SECTION 3 IMPLICATIONS TO COMBAT SURVIVABILITY

- If a vehicle is moving at a velocity V and the local speed of sound is given by

$$a = \sqrt{\gamma R T} = \sqrt{\gamma \frac{p}{\rho}}$$

where γ is the ratio of specific heats, R is the gas constant for air, and T is the temperature of the air the vehicle is flying in, then the Mach number is defined as
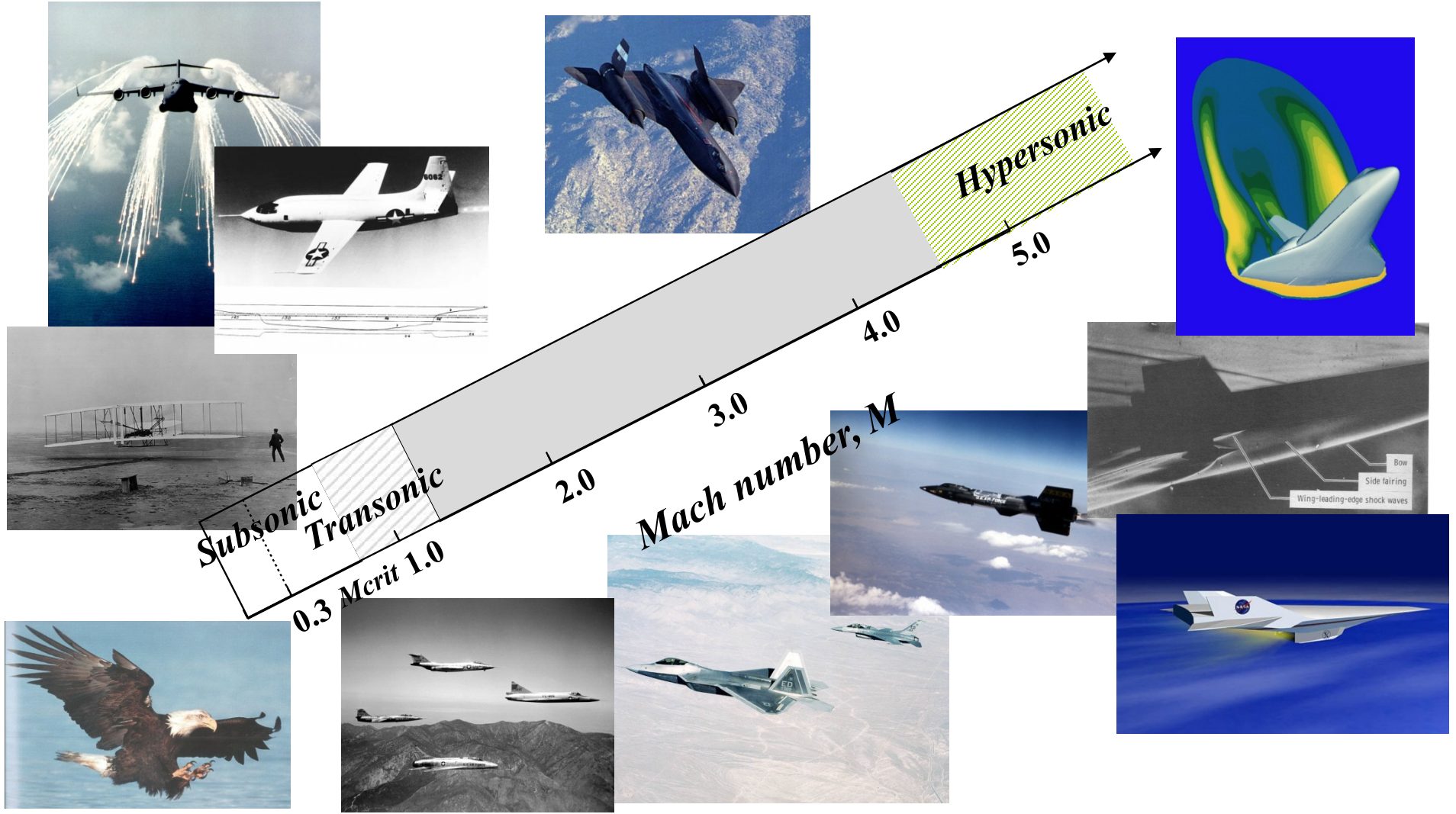
$$M = \frac{V}{a}$$

- The Mach number is the most important parameter in compressible flow theory. It relates how information propogates through the flow. We use it to explicitly define three different Mach regimes.

  - If M < 1, the flow is *subsonic*
  - If M = 1, the flow is *sonic*
  - If M > 1, the flow is *supersonic*

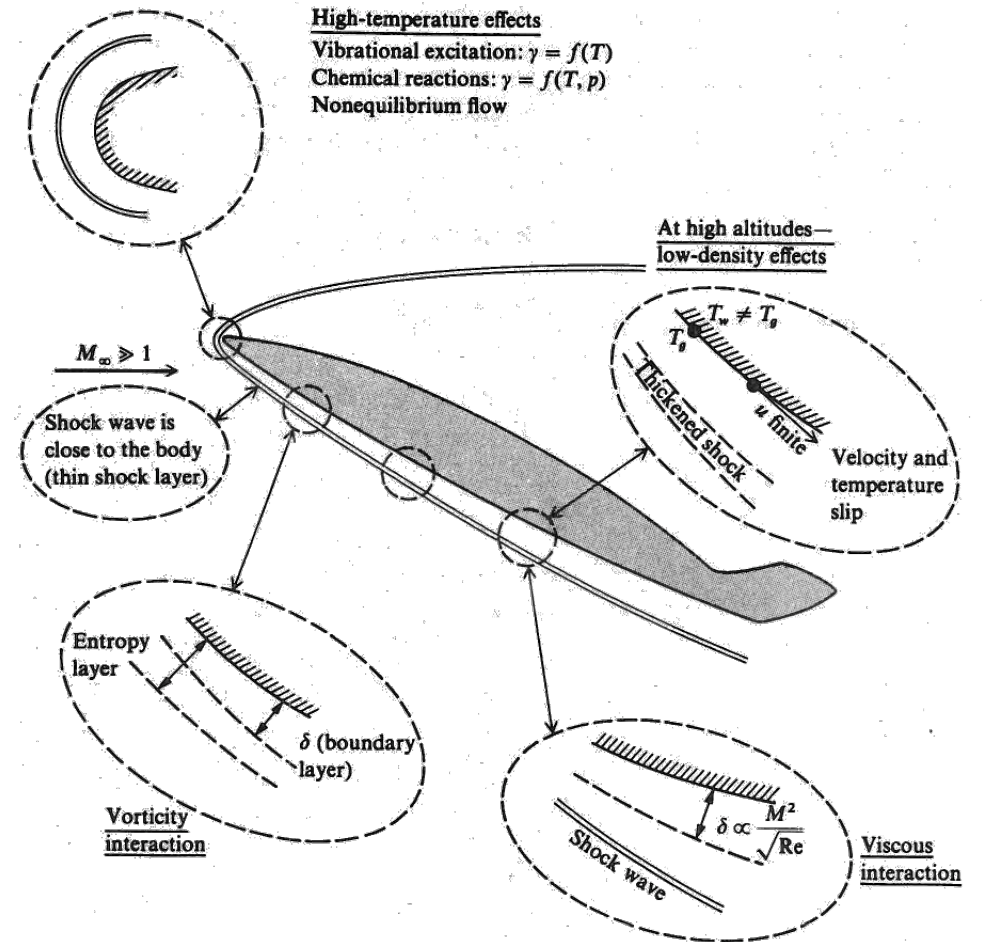- Mach number is not an inertial term. It is an *energy* term.
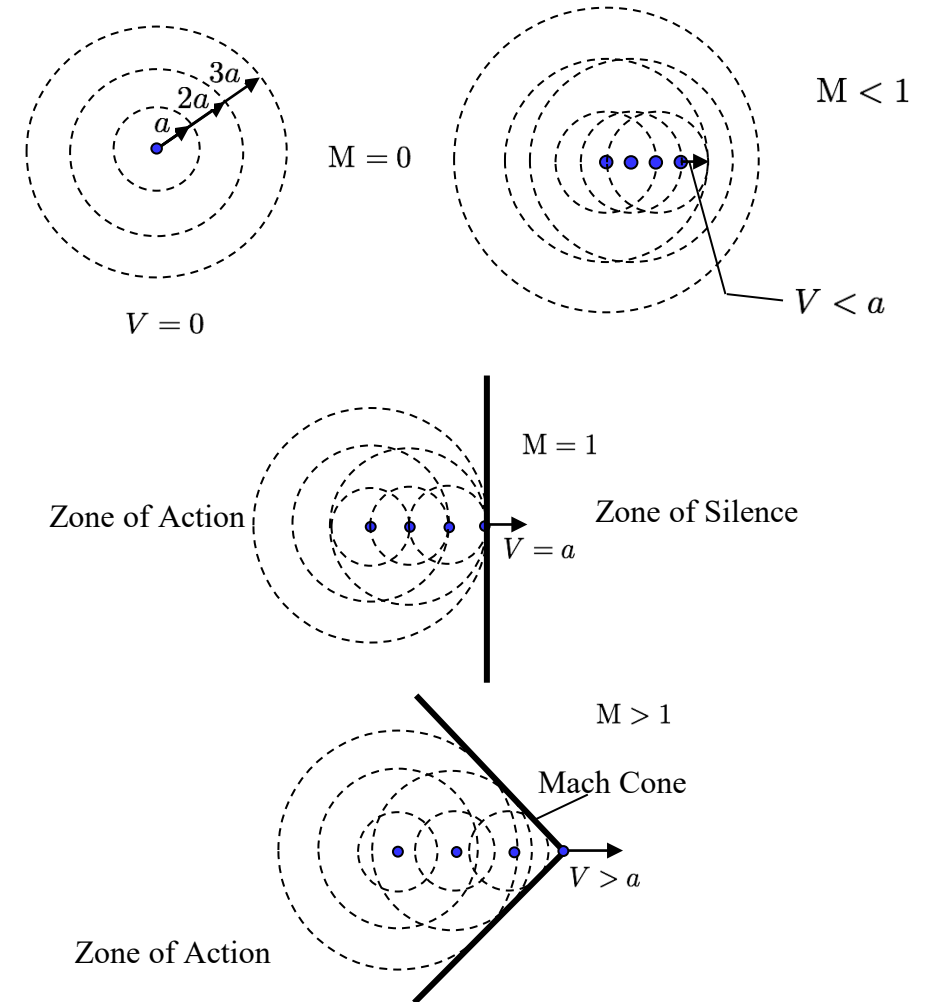
SO, WHAT MAKES A FLOW HYPERSONIC?

- Hypersonic flow begins when the simplifying assumptions of supersonic flow are no longer valid
- Distinguishing features

  - Thin Shock Layer: The region between the the shock wave and the vehicle surface.
  - Entropy Layer: Strong entropy gradients leading to significant vorticity generation and propagation.
  - Viscous Interaction: Standard boundary layer transition analysis fails.
  - High Temperature Effects: The ratio of specific heats, γ, is no longer constant. Air must be treated with all the different possible species that form due to dissociation (O_2, N_2, N, O, NO) and ionization.
  - Usually low density flow due to the high altitudes that hypersonic vehicles tend to fly. However...

High-temperature effects
Vibrational excitation: $\gamma = f(T)$
Chemical reactions: $\gamma = f(T, p)$
Nonequilibrium flow

At high altitudes—
low-density effects

$T_w \neq T_\theta$

u finite

Velocity and temperature slip

$M_\infty \gg 1$

Shock wave is close to the body (thin shock layer)

Thickened shock

Entropy layer

δ (boundary layer)

Vorticity interaction

Shock wave

$\delta \propto \dfrac{M^2}{\sqrt{Re}}$

Viscous interaction

Ref: John D. Anderson, *Hypersonic and High Temperature Gas Dynamics*

WHAT WE MEAN BY HYPERSONICS

- Hypersonics

  - Always includes a high speed component ($M \gtrsim 5$)
  - Possibly lower altitude employment, which complicates the heat problem
  - Maneuverability

- Critical Flow Phenomena

  - Shock-shock and shock-boundary layer interactions
  - Non-equilibrium effects
  - Flow-structure interactions
  - Ablation
  - Flight controls
  - Atmospheric Noise

- Thermal Management, external and internal
- Multidisciplinary due to fully-coupled physics

  - Coupling effects can be beneficial or adverse
  - Systems level analysis and design optimization

- How new is the idea of maneuverable hypersonic vehicles?
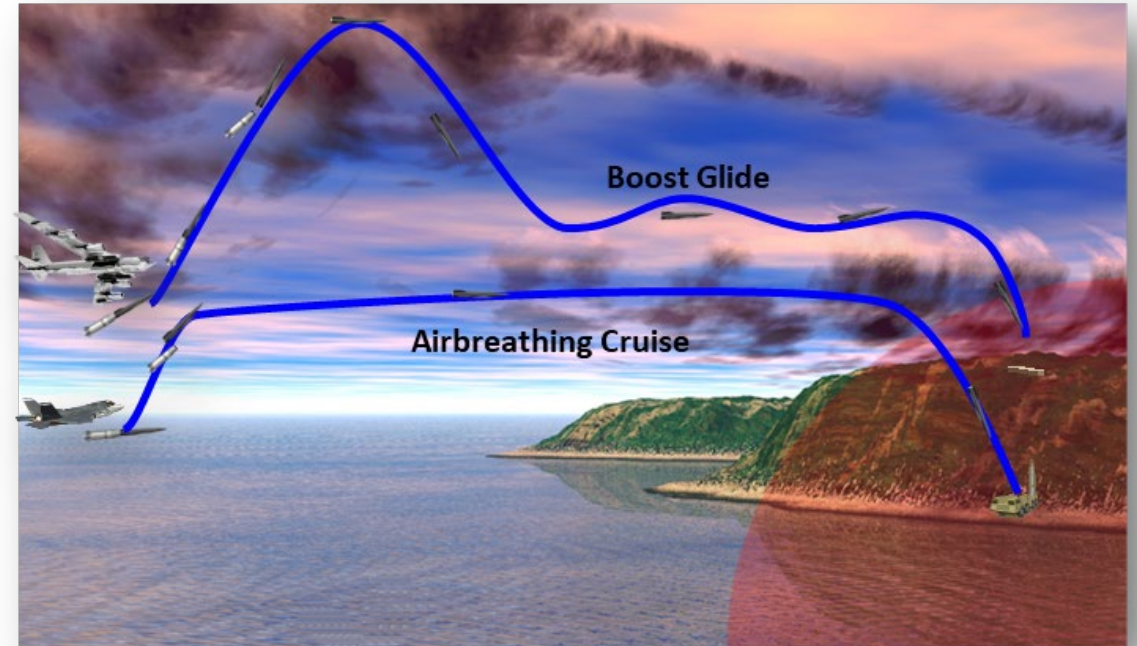
Breaking the Heat Barrier!

1959-1968 X-15

- Joint NACA, USAF, Navy program; North American Aviation selected Sep 55
- Three flight vehicles produced; 199 flights; 1 fatality
- Conventional aero controls plus reaction control system
- Heat sink structure with Inconel X skin; ablative with sealant for high Mach
- Initially 2 XLR-11 engines (16 klb thrust); later XLR-99 engine (67 klb thrust)
- First application of hypersonic theory and wind tunnel work to actual flight
- Max altitude: 354,200 feet on 22 Aug 1963
- Max Mach: 6.72 on 3 Oct 1967

- The Martin X-23 PRIME was a small uncrewed lifting body re-entry vehicle.

- PRIME was developed to study the effects of maneuvering during re-entry of Earth's atmosphere, including cross-range maneuvers.

- It was built from titanium, beryllium, stainless steel, and aluminium. It consisted of two sections: the aft main structure and a removable forward "glove section." The body of the X-23 was completely covered with a Martin-developed ablative heat shield 20 to 70 mm thick, and the nose cap was constructed of carbon phenolic material.

- At Mach 2 a drogue ballute deployed and slowed the vehicle's descent. As it deployed, its cable sliced the upper structure of the main equipment bay, allowing a 16.4 m recovery chute to deploy. It was to be recovered in midair by a specially-equipped JC-130B Hercules aircraft.

- Advanced Maneuverable Reentry Vehicle (AMaRV) was a prototype MaRV built by McDonnell-Douglas Corp.
- Four AMaRVs were made and represented a significant leap in Reentry Vehicle sophistication.
- Three of the AMaRVs were launched by Minuteman-1 ICBMs on 20 December 1979, 8 October 1980 and 4 October 1981.
- AMaRV had an entry mass of approximately 470 kg, a nose radius of 2.34 cm, a forward frustum half-angle of 10.4°, an inter-frustum radius of 14.6 cm, aft frustum half angle of 6°, and an axial length of 2.079 meters.
- Trajectory plots showing hairpin turns have been published.

Ref: Frank J. Regan and Satya M. Anadakrishnan, *Dynamics of Atmospheric Re-Entry*
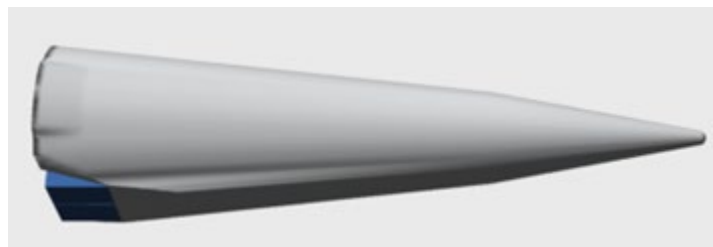
2010-2013 X-51A WAVE-RIDER

- Four powered flights over four years
- First Flight – May 26th, 2010

  - 143 seconds of scramjet operation
  - Peak Mach of 4.87; 150 nm traveled
  - Seal/nozzle breach ended flight early

- Second Flight – June 13th, 2011

  - Engine "unstarted" nine seconds after scramjet ignition
  - Post-flight investigation and ground testing yielded several scramjet operability lessons learned

- Third Flight – August 14th, 2012

  - Booster run-away control fin actuator and loss of control prior to engine light

- Fourth Flight – May 1st, 2013

  - Full duration flight: ~209 seconds of scramjet operation and 377 seconds of controlled flight
  - Peak Mach of 5.1; ~240 nm travelled in six minutes

2011 ADVANCED HYPERSONIC WEAPON

- The AHW technology demonstration programme is managed by the US Army Space and Missile Defence Command (USAS-MDC) / Army Forces Strategic Command (ARSTRAT).

- In November 2011, AHW was launched from the Pacific Missile Range Facility in Kauai, Hawaii, to the Reagan Test Site on the Marshall Islands. The glide vehicle successfully hit the target, which is located about 3,700km away from the launch site.

- The test was conducted to demonstrate hypersonic boost-glide technologies and trial the capability for atmospheric flight at long-ranges.

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Ref: John D. Anderson, *Hypersonic and High Temperature Gas Dynamics*
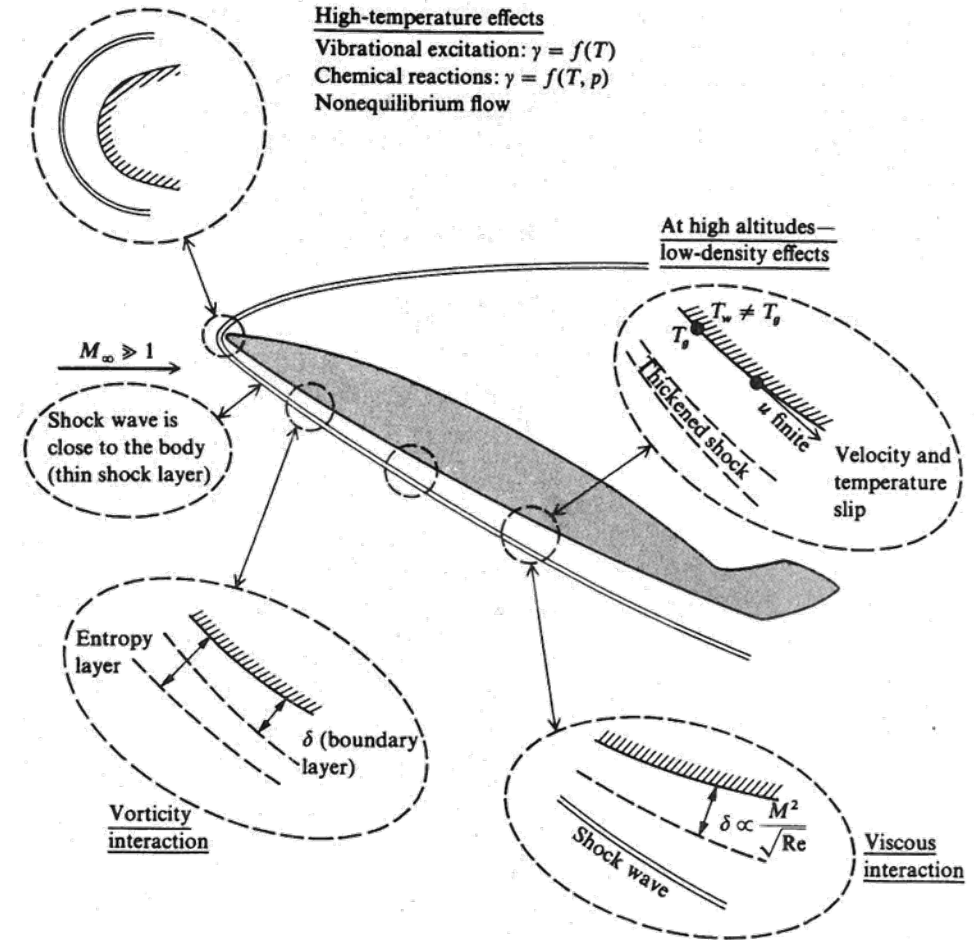
THIN SHOCK LAYERS

- Shock Layer – Flowfield region between the shock wave and the body surface
- Compare the shock wave on a wedge with a half-angle of 20° at Mach = 2 and Mach = 20
- At low Reynolds numbers, the thick boundary layer can merge with the shock wave to form a fully viscous shock layer

$M = 2$ $\theta = 53.4°$ $\delta = 20°$

$M = 20$ $\theta = 24.7°$ $\delta = 20°$

Wedge Angle $\delta = 20°$
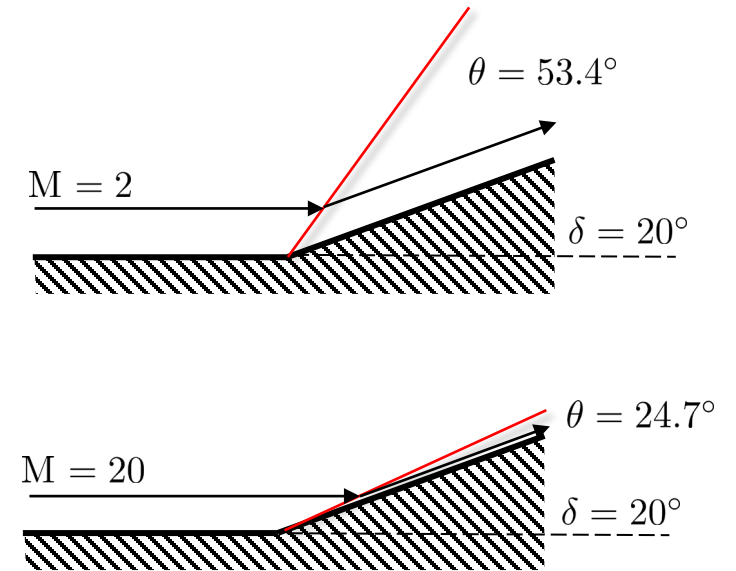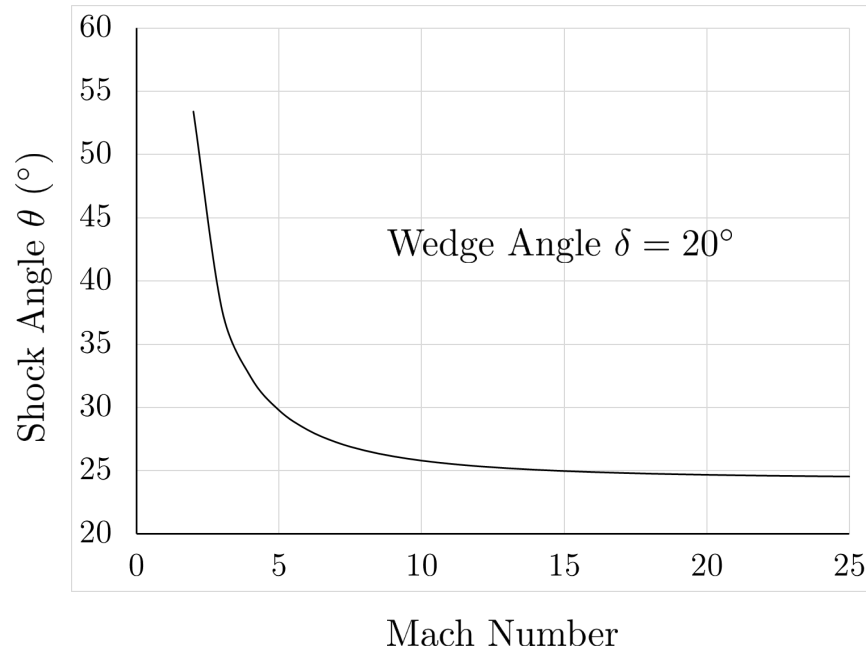
Shock angles obtained from equations in NACA-1135

- Entropy Layer – Region of strong gradients (vorticity)
- For supersonic flow, the entropy is assumed to be constant inside the boundary layer since the leading edges are assumed to be sharp
- For hypersonic flow the leading edge must be rounded or blunted both for practicality of manufacture and to ease heat fluxes (more on this later). Close to this blunt leading edge, the oblique shock becomes highly curved. Entropy increases across a shock, and the entropy increase becomes greater as the shock strength increases. Since flow near the nose passes through a nearly normal shock, it will experience a much greater change in entropy compared to flow passing through the much shallower shock angle further downstream. Strong entropy gradients exist near the leading edge generating an "entropy layer" that flows downstream along the body surface.
- In addition, the entropy layer is a region of strong vorticity that can generate large gradients in the velocity flowfield near the surface, a phenomenon called "vorticity interaction."


Ref: John D. Anderson, *Hypersonic and High Temperature Gas Dynamics*
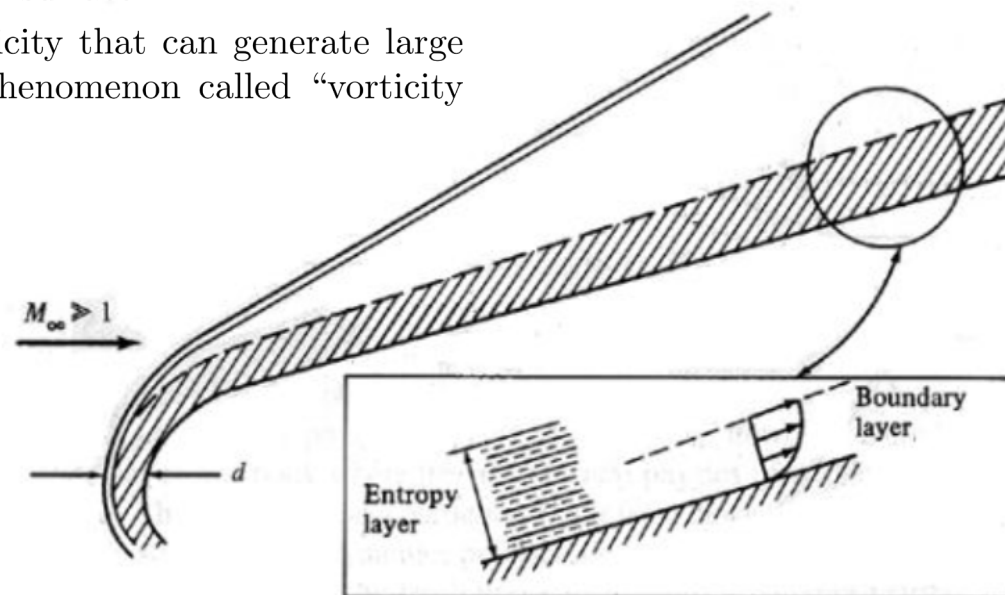
- For compressible flow, boundary layer thickness is proportional to the Mach number squared

$$\delta \propto \frac{M^2}{\sqrt{Re}}$$

- Viscous-Inviscid interaction can no longer be decoupled

  – Thick boundary layer affects flow outside boundary layer (inviscid region)
  – Changes in the flow outside the boundary layer affect the boundary layer growth

- Consequently there is an increase in surface pressure and skin friction, leading to increased drag and increased aerodynamic heating

Inviscid Region

Boundary Layer

M_∞
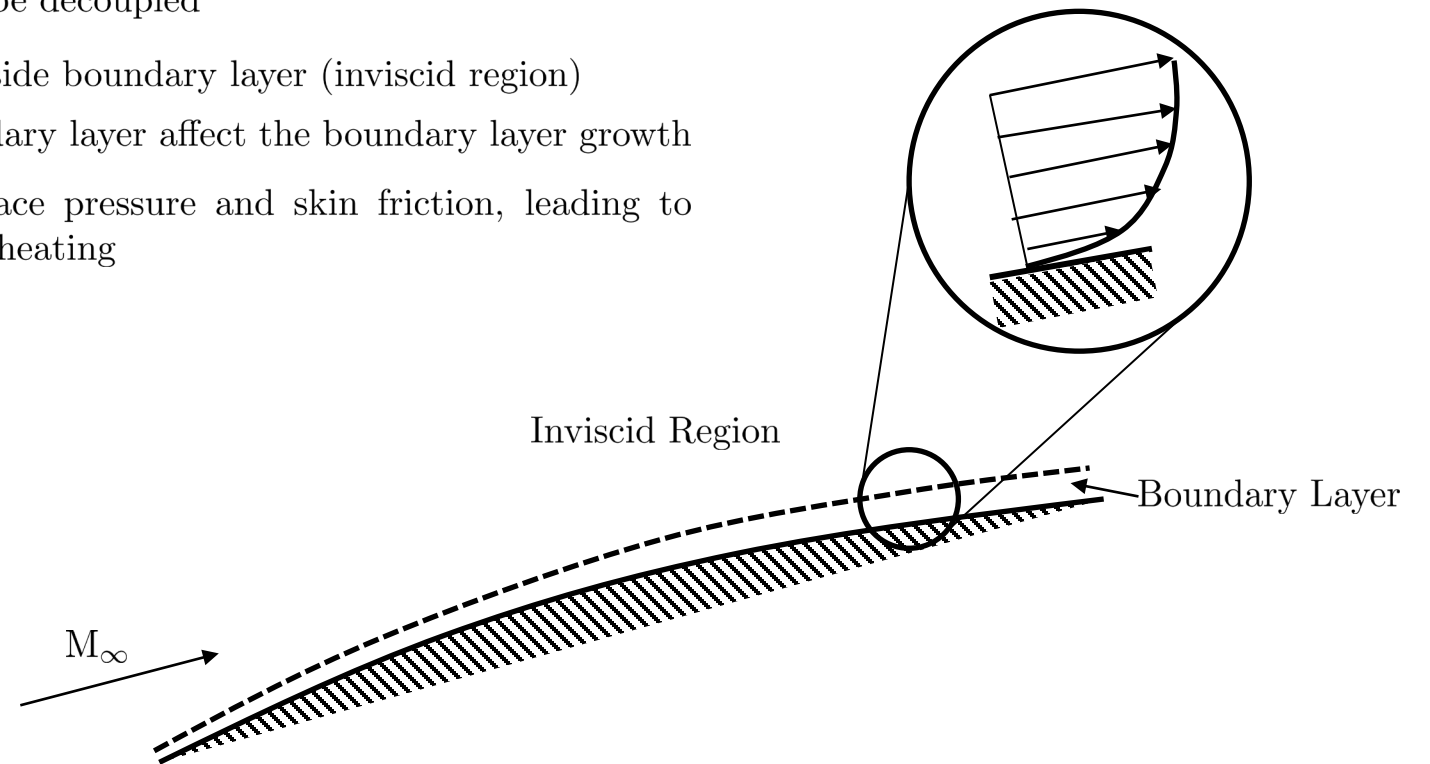
- At low speeds, the pressure distribution at the edge of the boundary layer is assumed to be the same as the pressure distribution on the wall from an inviscid flow analysis.
- At hypersonic speeds, the boundary layer influences this pressure distribution starting at the stagnation point.
- The value of $\overline{\chi}$ is used to determine when the boundary layer effects are of first order importance – a "strong interacton."

$$\overline{\chi} = \frac{M_\infty^3}{\sqrt{Re}}\sqrt{C}, \quad C = \frac{\rho_w \mu_w}{\rho_e \mu_e}$$

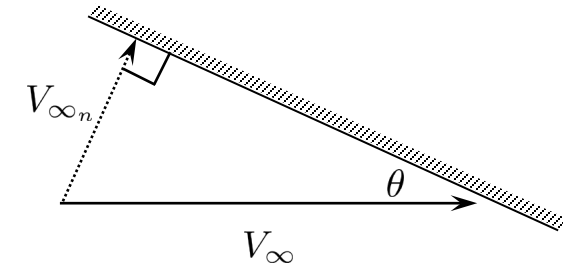$\overline{\chi} \geq 3$ a strong interaction

$\overline{\chi} < 3$ a weak interaction
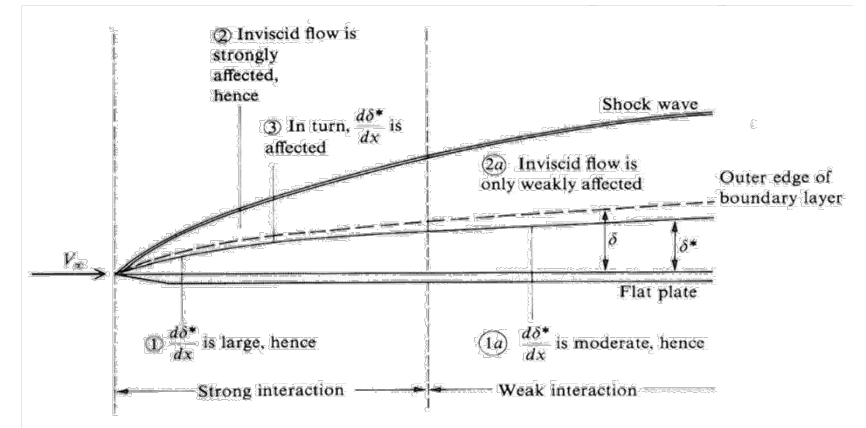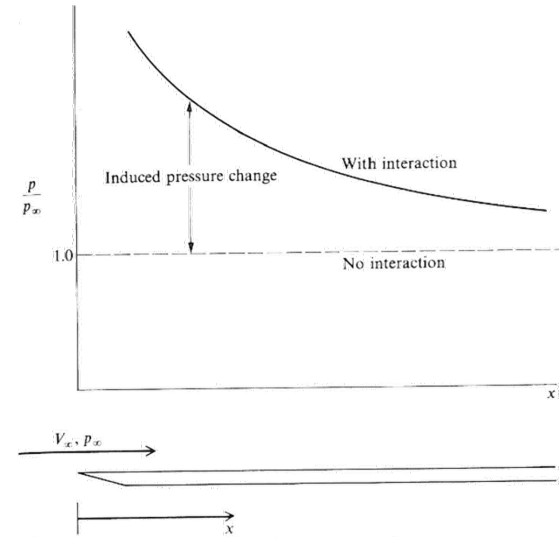
Ref: John D. Anderson, *Hypersonic and High Temperature Gas Dynamics*

- In 1687, Newton postulated the following model of fluid flow (he was actually trying to estimate forces on ships)

  - When a fluid with a velocity of V_∞ strikes a suface of area A inclined at an angle θ to the flow, the normal momentum of the fluid is totaly transfered to the surface while the tangential momentum is preserved

  - Thus the coefficient of pressure is determined from the normal portion of the flow only.

- The normal force can be equated to the pressure difference on the surface. Dividing by dynamic pressure yields the desired result for the coefficient of pressure.

$$N = \dot{m}_n V_{\infty_n}$$
$$= (\rho_\infty A V_\infty \sin\theta)(V_\infty \sin\theta)$$
$$= \rho_\infty A V_\infty^2 \sin^2\theta$$
$$N = (p - p_\infty) A$$

$$(p - p_\infty) A = \rho_\infty A V_\infty^2 \sin^2\theta$$
$$(p - p_\infty) = \rho_\infty V_\infty^2 \sin^2\theta$$
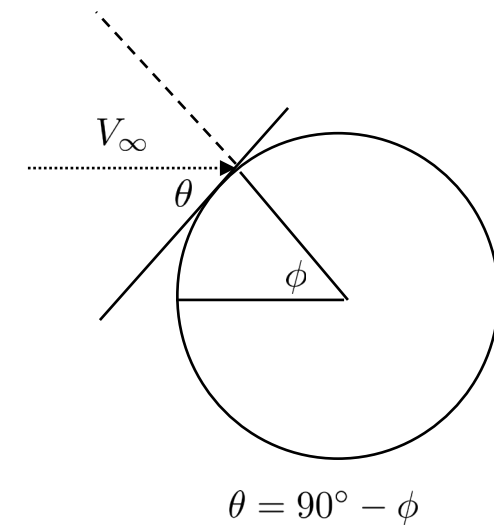
$$c_p \equiv \frac{p - p_\infty}{\overline{q}_\infty}$$
$$= \frac{\rho_\infty V_\infty^2 \sin^2\theta}{\frac{1}{2}\rho_\infty V_\infty^2}$$

$$\boxed{c_p = 2\sin^2\theta}$$

$$\theta = 90° - \phi$$

The same results are obtained if you use the shock relations and assume Mach number approaches infinity and γ approaches 1

- Newtonian Flow:

$$c_p = 2\sin^2\theta$$

- Newtonian flow indicates that performance is independent of Mach number!
- We can modify the Newtonian flow by limiting the value of 2 with a physics model. If we assign p_{0_2} as the total pressure behind the normal shock determined by the freestream Mach number, then
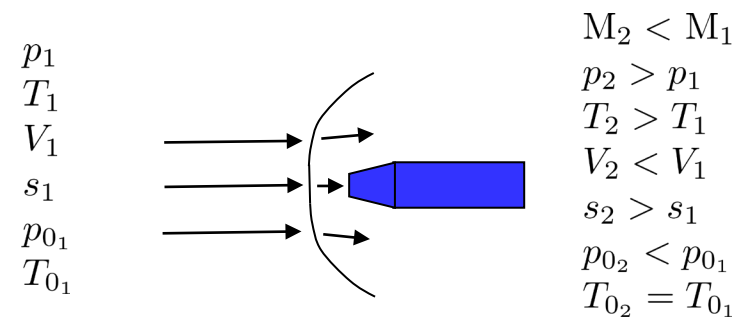
$$c_{p_\mathrm{max}} = \frac{p_{0_2} - p_\infty}{\frac{1}{2}\gamma p_\infty M_\infty^2}$$

and the modified Newtonian flow becomes

$$\boxed{c_p = c_{p_\mathrm{max}} \sin^2\theta}\,.$$

- It turns out, for hypersonic flow, $c_{p_\mathrm{max}} \approx 1.83$, so still Mach number independent!
- Recall, Mach number relates information propogation. Above about Mach 5, only so much information about the flow can be transmitted.
- Modified Newtonian flow gives more accurate results for the c_p calculation around blunt bodies.
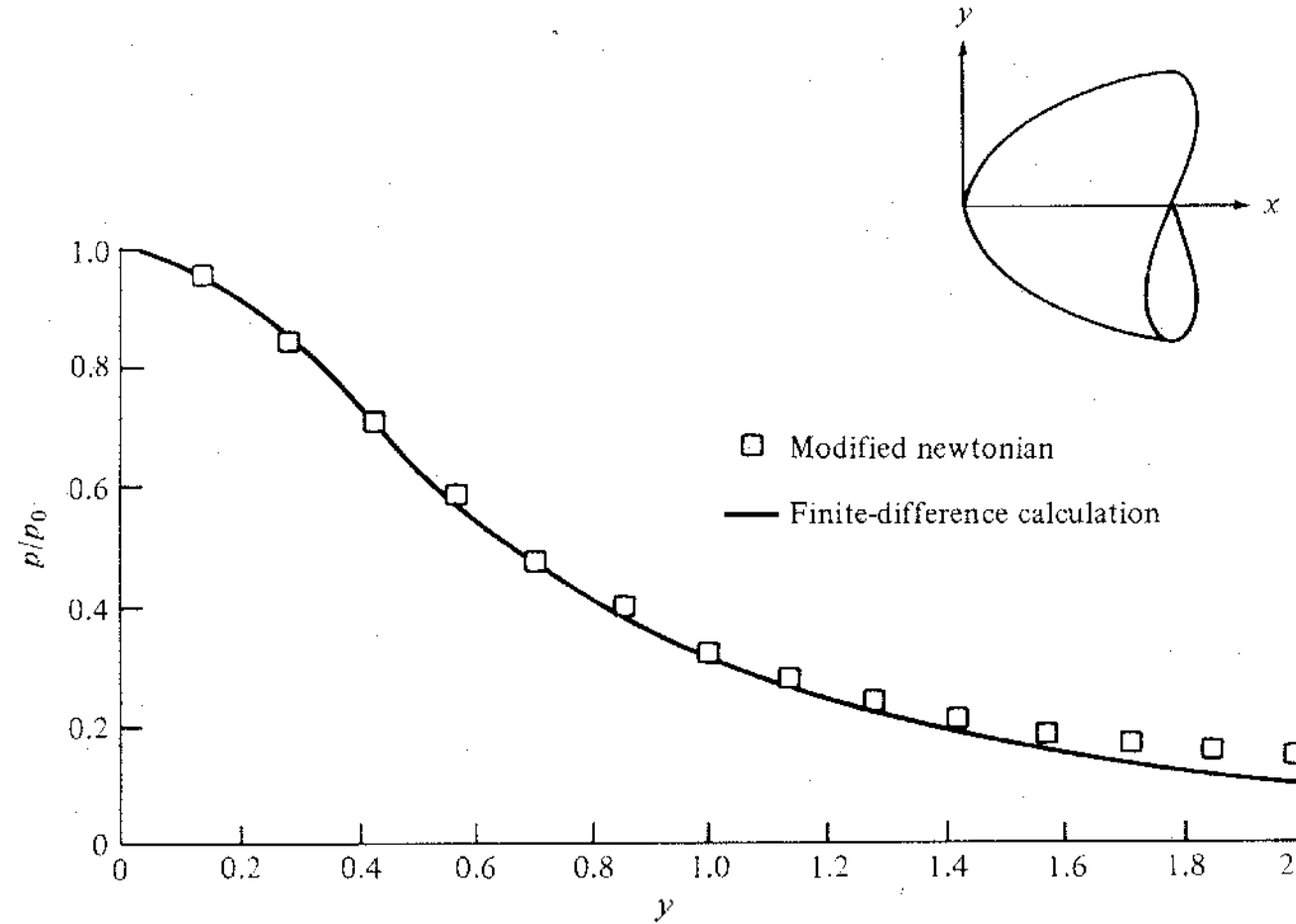
$M_1 > 1$

Immediately behind the shock

p_1
T_1
V_1
s_1
p_{0_1}
T_{0_1}

$M_2 < M_1$
$p_2 > p_1$
$T_2 > T_1$
$V_2 < V_1$
$s_2 > s_1$
$p_{0_2} < p_{0_1}$
$T_{0_2} = T_{0_1}$

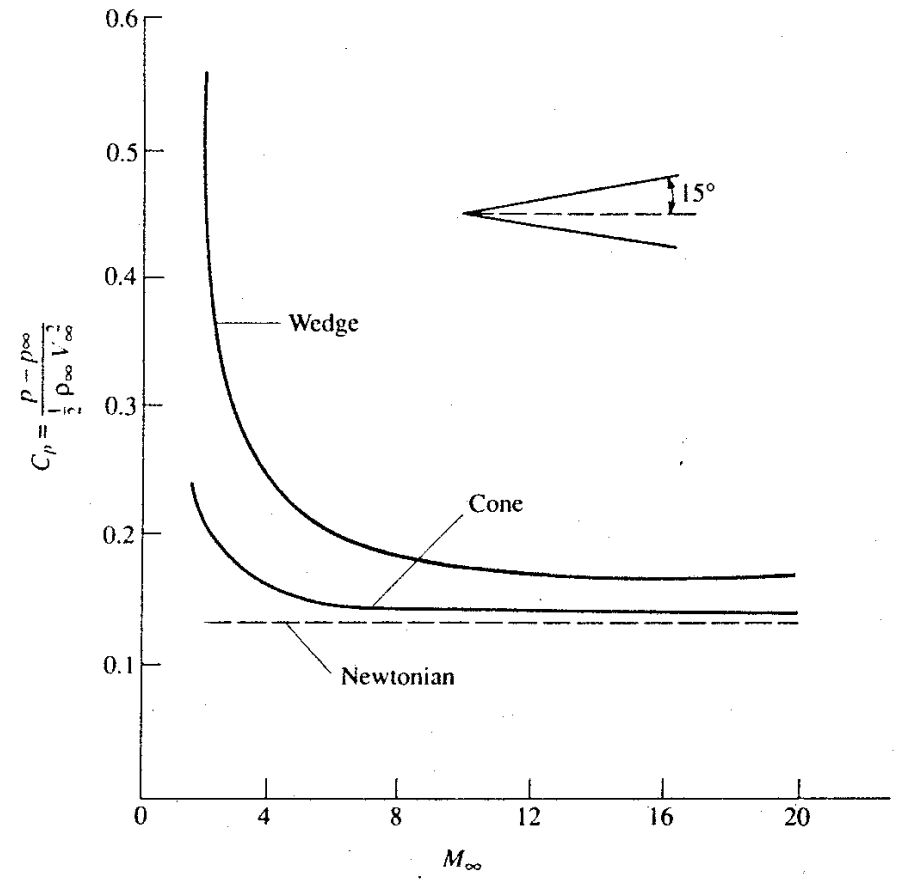The properties at $()_1$ are the same as the freestream properties $()_\infty$

Ref: Anderson, Fundamentals of Aerodynamics, Figure 10.10. Surface-pressure distribution on an axisymmetric body of parabolic shape, M=4.
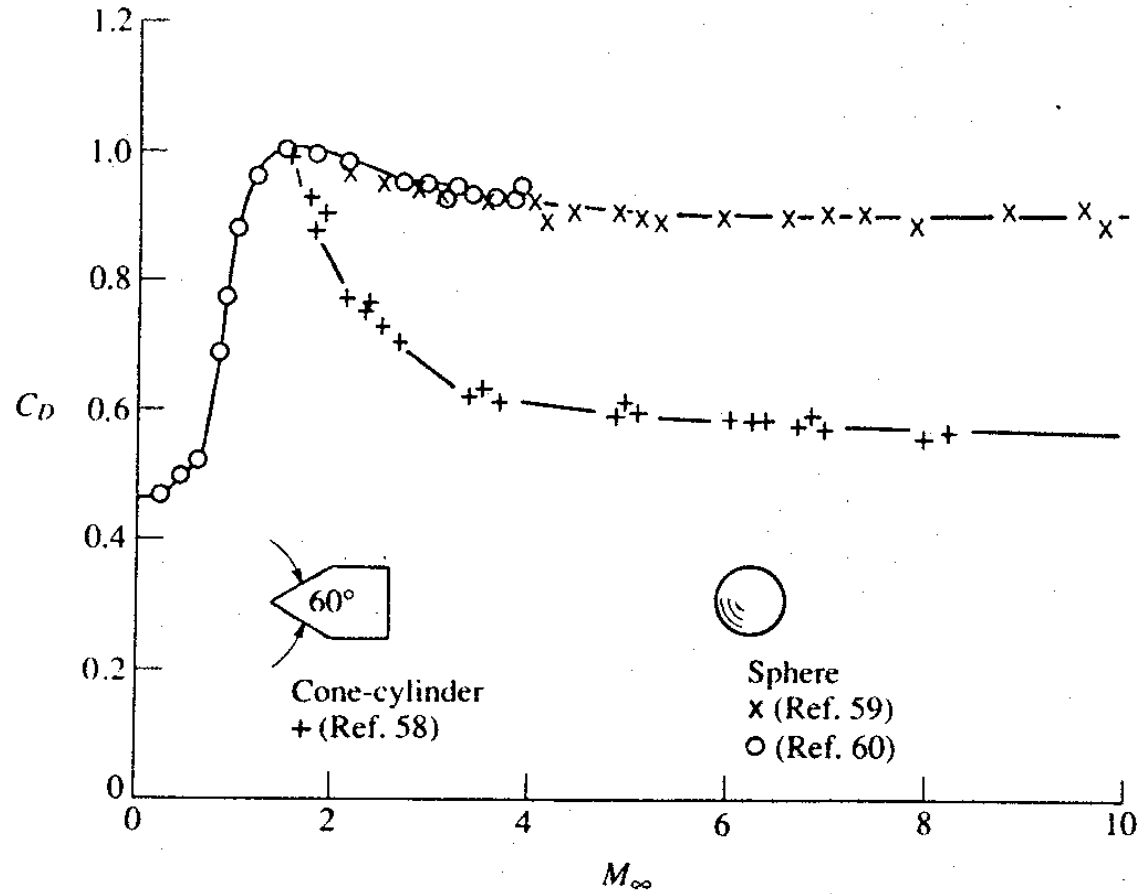
Ref: Anderson, Fundamentals of Aerodynamics, Figure 14.13.Comparison between Newtonian and exact results for the pressure coefficient on a sharp wedge and a sharp cone.

Ref: Anderson, Fundamentals of Aerodynamics, Figure 14.14. Drag Coefficient for a sphere and a cone cylinder from ballistic range measurements.

- Consider the hypersonic flow over a flat plate where

$$c_n = c_p$$
$$\theta = \alpha \, .$$

- We can write the lift and drag coefficients as

$$c_l = c_n \cos \alpha$$
$$c_d = c_n \sin \alpha \, .$$
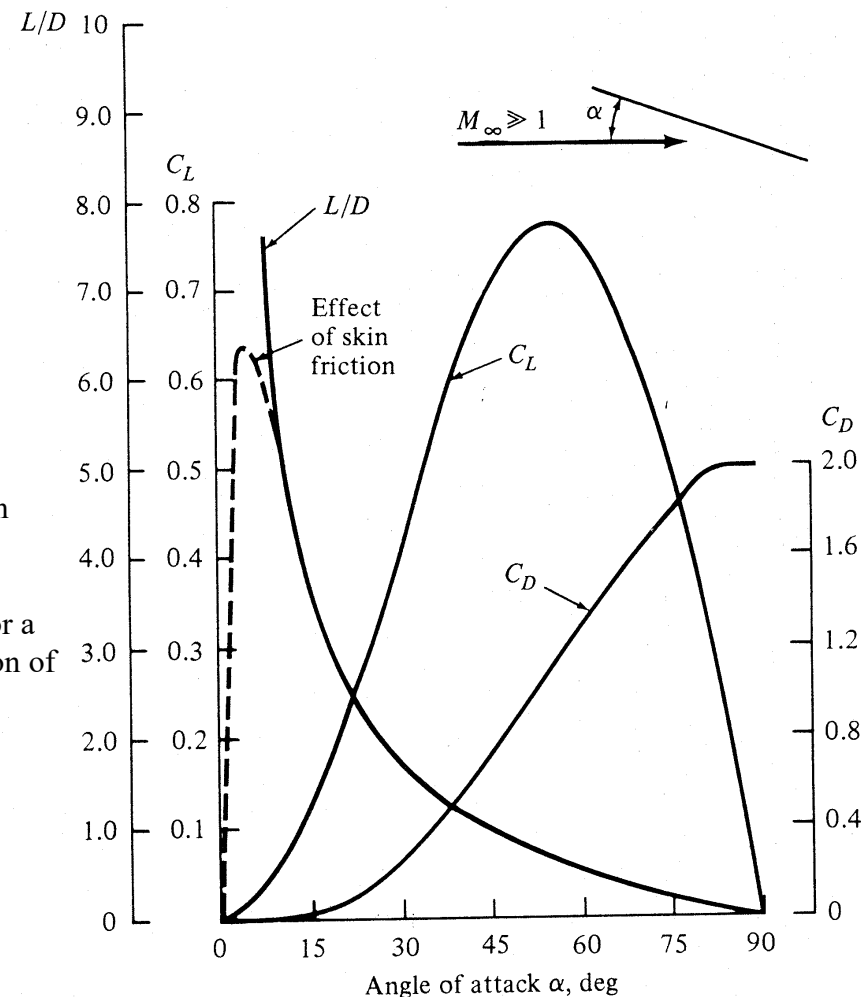
- According to Newtonian flow, we then have

$$\boxed{c_l = 2 \sin^2 \alpha \cos \alpha}$$

$$\boxed{c_d = 2 \sin^3 \alpha}$$

$$\boxed{l/d = \cot \alpha}$$

- We often speak of the hypersonic lift to drag ratio, because the ratio is no longer dependent on Mach number, just on angle of attack.

Ref: Anderson, Hypersonic and High Temperature Gas Dynamics, Fig. 3.6, Newtonian results for a flat plate as a function of angle of attack.
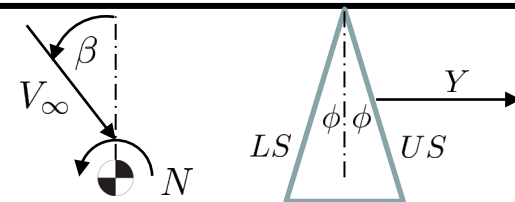
DIRECTIONAL STABILITY

- Consider the directional stability problem. The yawing moment due to the vertical tail is given by

$$C_{N_{\beta_{VT}}} = V_{VT} \frac{\overline{q}_{VT}}{\overline{q}_{\text{ref}}} C_{Y_{VT}} .$$

The first term is the vertical tail volume coefficient. The second term is the ratio of dynamic pressures. The third term is given by $C_{Y_{VT}} = c_{p_{LS}} - c_{p_{US}}$, with the correct interpretation of the lower surface and upper surface.
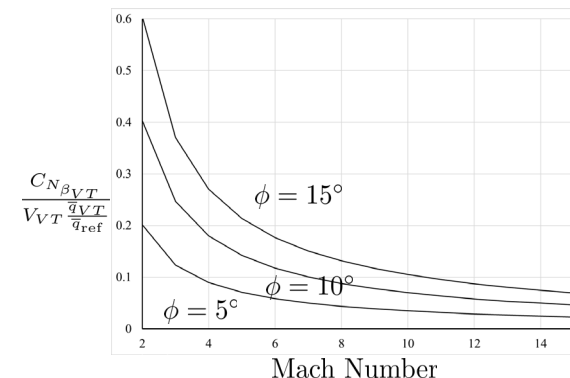
$$\theta_{LS} = \beta + \phi$$
$$\theta_{US} = \beta - \phi$$

- Lineraized supersonic thin airfoil theory predicts

$$c_p = \frac{2\theta}{\sqrt{\text{M}_\infty^2 - 1}} ,$$

which would result in a restoring moment of

$$C_{N_{\beta_{VT}}} = V_{VT} \frac{\overline{q}_{VT}}{\overline{q}_{\text{ref}}} \frac{4\phi}{\sqrt{\text{M}_\infty^2 - 1}}$$
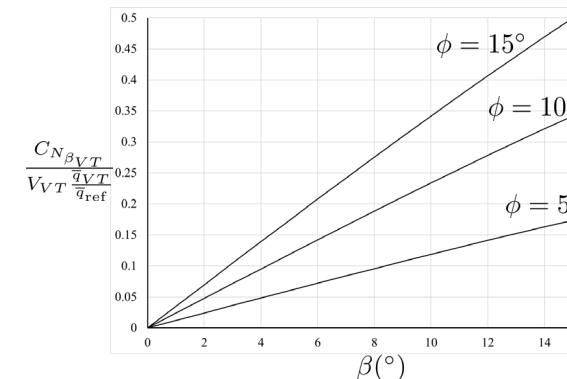
Thus, supersonic flow theory predicts that the restoring moment goes to zero as the Mach number increases!

- However, Newtonian flow predicts

$$c_p = 2 \sin^2 \theta$$

and the resulting restoring moment is

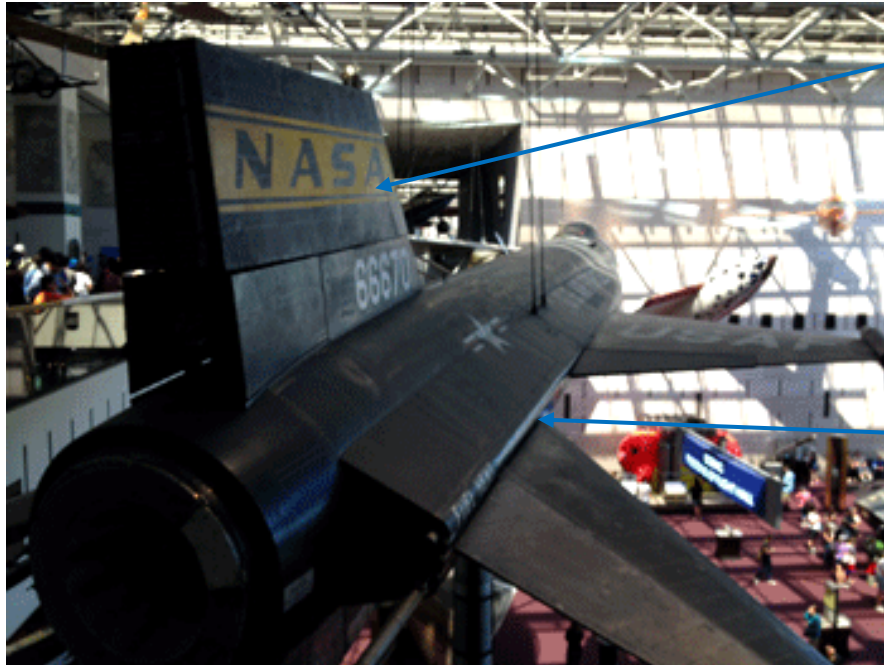$$C_{N_{\beta_{VT}}} = 2 V_{VT} \frac{\overline{q}_{VT}}{\overline{q}_{\text{ref}}} \sin(2\beta) \sin(2\phi)$$

Now the restoring moment increases with increasing wedge angle and increasing sideslip!
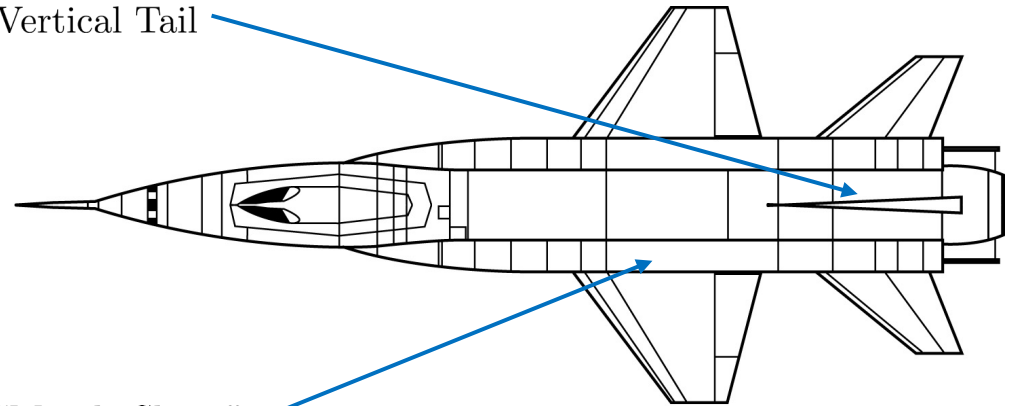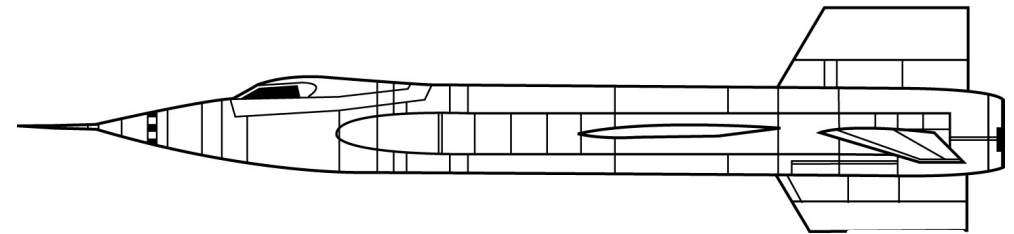
- Academic exercise?

Vertical Tail

"Missile Skirts"

- The hypersonic challenge of the 1950's: Ballistic Missile Atmospheric Re-Entry
- Based of supersonic theory, hypersonic vehicles would need to be even more slender and sharp
- The major theoretical advance came with the publication of

  - H. Julian Allen and A.J. Eggers, Jr., "A Study of the Motion and Aerodynamic Heating of Ballistic Missiles Entering the Earth's Atmosphere at High Supersonic Speeds," NACA R 1381, 1953

- Allen and Eggers showed

$$\dot{q}_{\mathrm{max},laminar} \sim \frac{1}{\sqrt{R_N}}$$

- They concluded *a blunt nose forces a detached shock and most of the heat goes off the surface and into the flowfield, not the vehicle, and enables practical re-entry vehicles.*
- Experimental evidence suggests the heat flux (heat transfer per unit area) can be approximated based on the following relationship:

$$\dot{q}_{\mathrm{max},laminar} \sim \sqrt{\frac{\rho_\infty}{R_N}} V_\infty^3$$

- The three modes of heat transfer are given by

  - *Conduction* - heat transfer through a substance due to a temperature gradient:

  $$\dot{q}_{\text{cond}} = k\frac{T_2 - T_1}{x_2 - x_1} \ ,$$

  where k is the coefficient of thermal conductivity.

  - *Forced Convection* - heat transfer due a moving fluid over a solid body:

  $$\dot{q}_{\text{conv}} = h\left(T_\infty - T_w\right) \ ,$$

  where h is the convective heat transfer coefficient.

  - *Radiative Cooling* - heat transfer by electromagnetic waves:

  $$\dot{q}_{\text{rad}} = \epsilon\sigma T_w^4 \ ,$$

  where ϵ is the emmisivity of the material and σ is the Stefan-Boltzmann constant.

- The heat flux balance is

$$\dot{q}_{\text{cond}} + \dot{q}_{\text{conv}} - \dot{q}_{\text{rad}} = 0$$

David E. Glass, "Ceramic Matrix Composite (CMC) Thermal Protections Systems (TPS) and Hot Structures for Hypersonic Vehicles," AIAA-2008-2682

ESTIMATING HEAT FLUX

- If we assume the thermal protection system has low conductivity (a pretty good assumption), then we can neglect $\dot{q}_{\text{cond}}$. *Sans shock-to-shock interactions*, the maximum heat flux is typically at the stagnation point, so we focus our approximation there.

$$\dot{q}_s = (\dot{q}_{\text{conv}})_s = (\dot{q}_{\text{rad}})_s$$

- There are many empirical approaches for estimating the convective stagnation point heat flux, such as Fay-Riddell, Chapman, Hildago, and Sutton & Graves. The latter is the simplest to use and still gives good engineering predictions.

$$\boxed{\dot{q}_s = 1.74153 \times 10^{-4} \sqrt{\frac{\rho_\infty}{R_N}} V_\infty^3 \ \text{W/m}^2}$$

- With the defintion of $\dot{q}_{\text{rad}}$, the wall temperature at the stagnation point can also be estimated.

$$\boxed{T_s = \left(\frac{\dot{q}_s}{\epsilon\sigma}\right)^{1/4}}$$
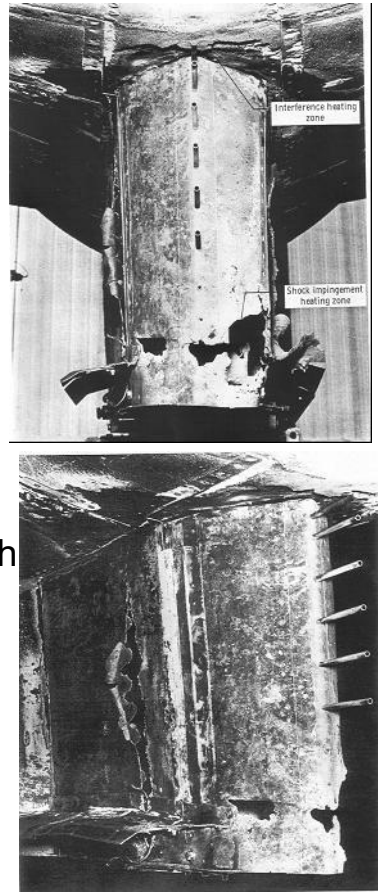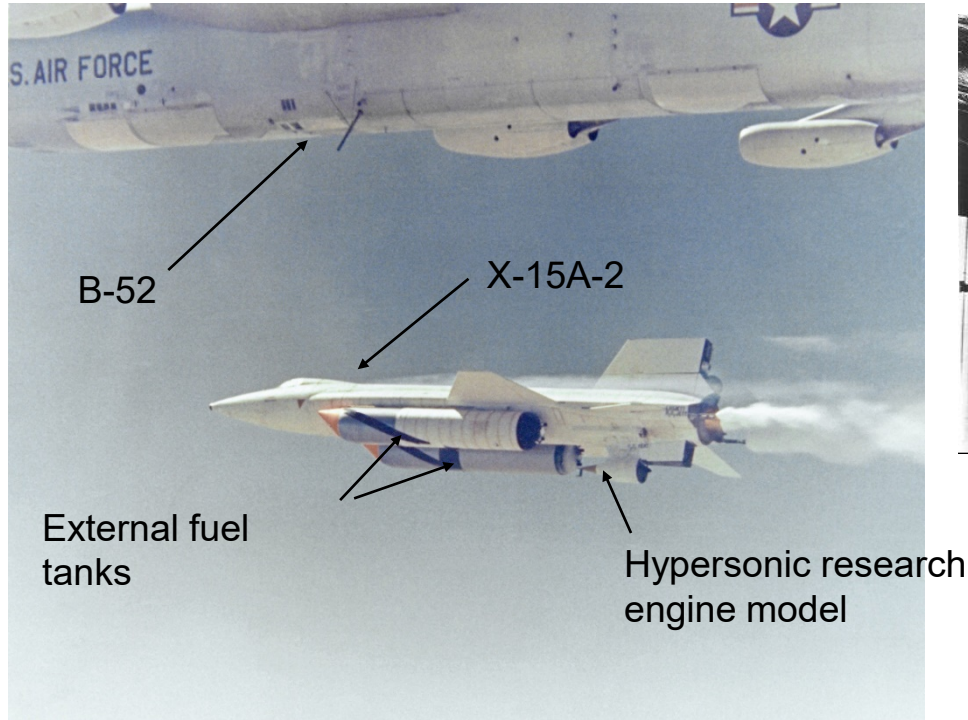
where ϵ is typically in the range of 0.8 to 0.9, and $\sigma = 5.6704 \times 10^{-8}$ W/m^{2}K^4 is the Stefan-Boltzmann constant.

- These types of estimations are very important when, in the absence of flight test data, deciding if high fidelity simulations, such as computational fluid dynamics, are providing a realistic answer.
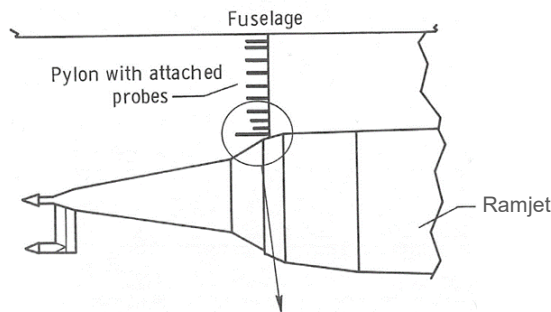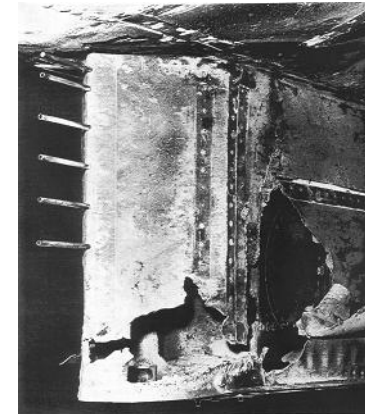- How do shock-to-shock interactions affect the heat flux?

B-52

X-15A-2

External fuel tanks

Hypersonic research engine model

Interference heating zone

Shock impingement heating zone

If there had been any question that the airplane was going to come back in that shape, we never would have flown it.
Jack Kolf
X-15 Project Engineer

As a point of reference, the entire output of a moderate-size nuclear power plant would be required to provide this heating rate to a 1-m^2 piece of material.
van Wie *et al*

Fuselage

Pylon with attached probes

Ramjet

Structure: Inconel X (a nickel-chromium alloy) plus an ablative cover.
From Iliff and Shafer, AIAA Paper 93-0311 and NASA TM X-1669

- High temperature flows are fundamentally different than classical thermo-dynamics

  - The thermodynamic properties (p, ρ, T, e, h, s, etc.) behave differently

  - The transport properties (μ and k) also behave differently. Diffusion becomes important.

  - High heat transfer rates are usually a dominant aspect of any high-temperature application.

  - The ratio of specific heats is no longer constant.

  - Due to the items listed above, virtually all analyses of high temperature gas flows require some type of numerical simulation rather than closed-form solutions.

  - If the temperature is high enough to cause ionization, the gas becomes a partially ionized plasma, which has a finite electrical conductivity. In turn, if the flow is in the presence of an exterior electric or magnetic field, then electromagentic body forces act on the fluid elements. This the purview of an area called magnetohydrodynamics.

  - If the gas temperature is high enough, there will be nonadiabatic effects due to radiation to or from the gas.
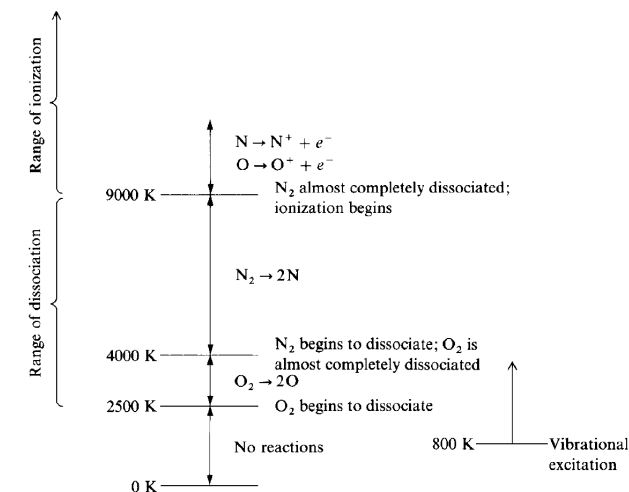

Fig. 9.12   Ranges of vibrational excitation, dissociation, and ionization for air at 1 atm pressure.
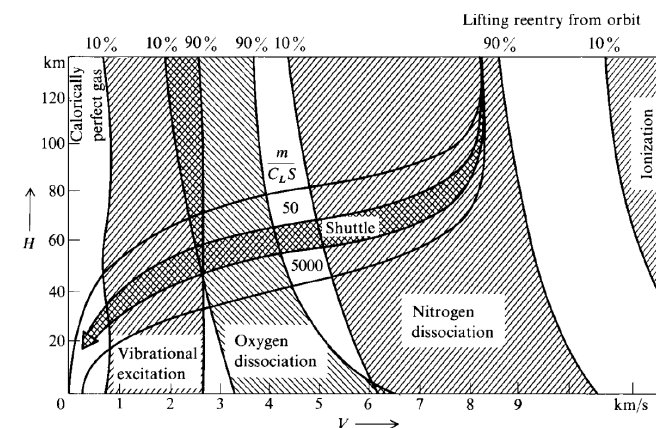

Fig. 9.13   Velocity-amplitude map with superimposed regions of vibrational excitation, dissociation, and ionization (from [79]).

Ref: John D. Anderson, *Hypersonic and High Temperature Gas Dynamics*

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SECTION 3 IMPLICATIONS TO COMBAT SURVIVABILITY

- Physics drives the design
  - Glide weapons designed for minimum wave drag
  - Both glide weapons and scramjets require large boosters
  - Design is difficult due to
    - viscous interactions
    - high temperatures
    - trajectory
    - predicting shock/fuselage interaction shape
  - Airbreathing (scramjet) vehicles have a highly integrated airframe and propulsion system
- Materials challenge
  - Large heat flux and integrated heat loads requires thermal protection systems
  - Use of windows for sensors becomes a challenge
- Maneuverability
  - Maneuvering comes at the cost of speed, particularly for glide weapons

IMPLICATIONS FOR DEFENSIVE SYSTEMS

- Hypersonic defensive weapons have all the same challenges as defensive weapons.
- The issue comes down to decreased decision time for engagement
  - Employment altitudes and maneuverability complicate find, fix, track, and target, engage, assess