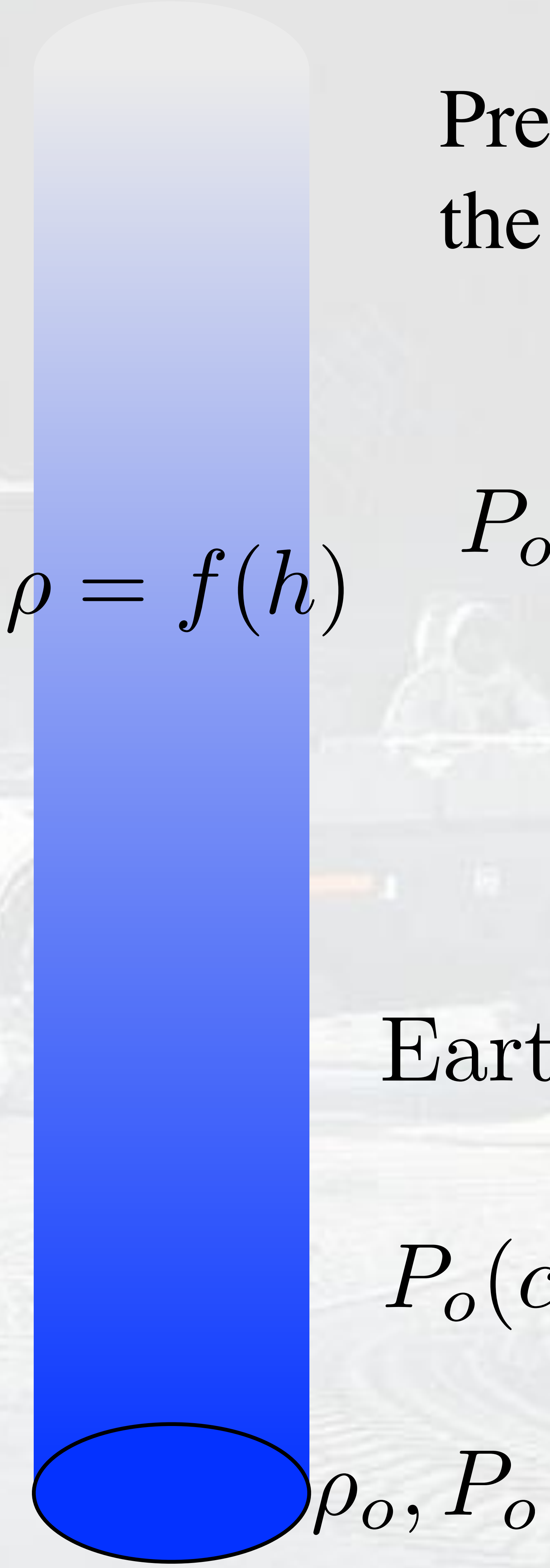
Robotic Mobility – Atmospheric Flight

- Gaseous planetary environments (Mars, Venus, Titan)
- Entry, descent, and landing
- Lighter-than-"air" (balloons, dirigibles)
- Heavier-than-"air" (aircraft, rotorcraft)

Atmospheric Density with Altitude



Pressure=the integral of the atmospheric density in the column above the reference area

$$P_o = \int_o^\infty \rho g dh = \rho_o g \int_o^\infty e^{-\frac{h}{h_s}} dh = -\rho_o g h_s \left[e^{-\frac{h}{h_s}} \right]_o^\infty$$

$$= -\rho_o g h_s [0 - 1]$$

$$\boxed{P_o = \rho_o g h_s}$$

$$\text{Earth: } \rho_o = 1.226 \frac{kg}{m^3}; \ h_s = 7524m;$$

$$P_o(calc) = 90,400 \ Pa; \ P_o(act) = 101,300 \ Pa$$

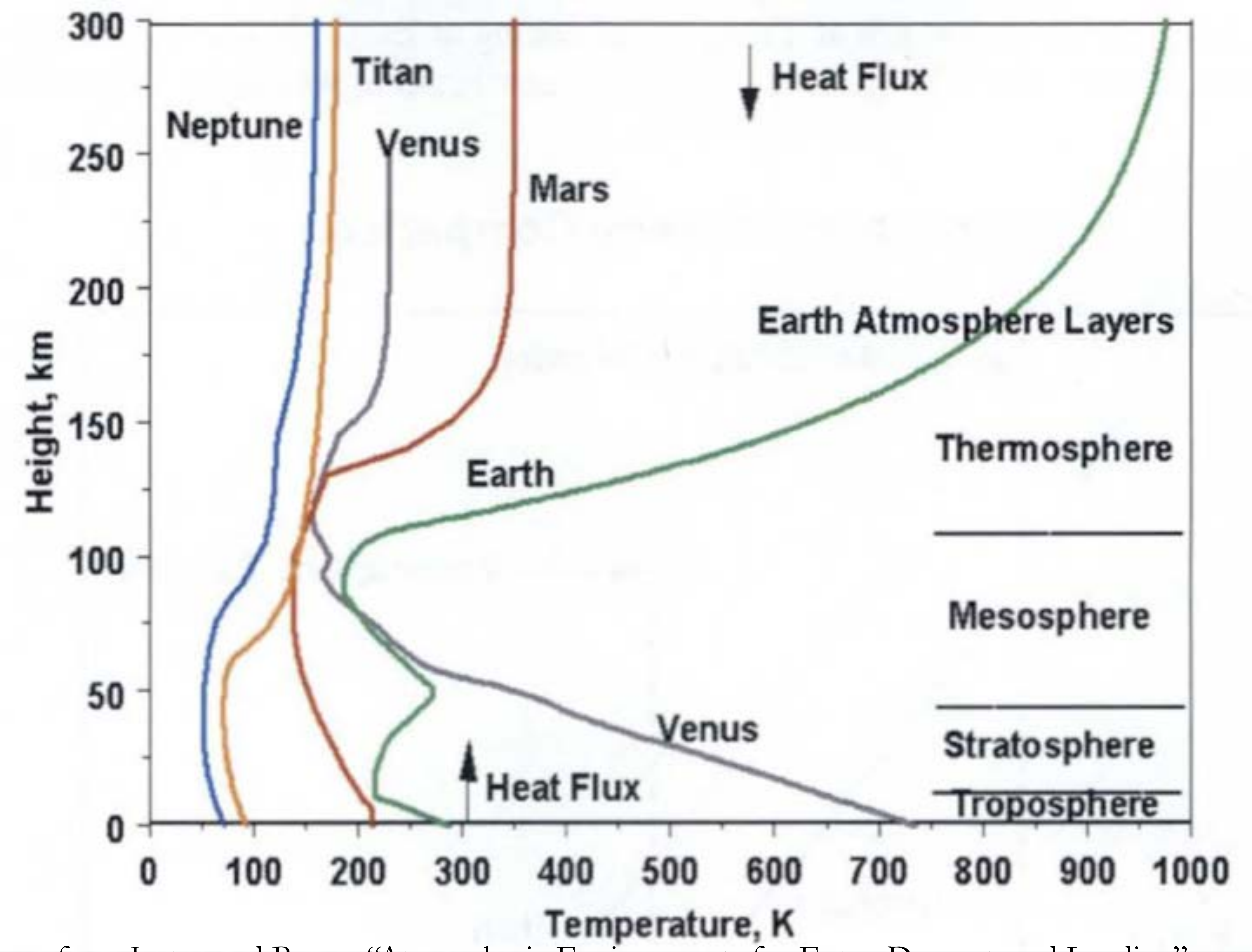
Exponential Atmospheres

$$\rho = \rho_o e^{-h/h_s}$$

$$\rho_o = \text{Reference density}$$

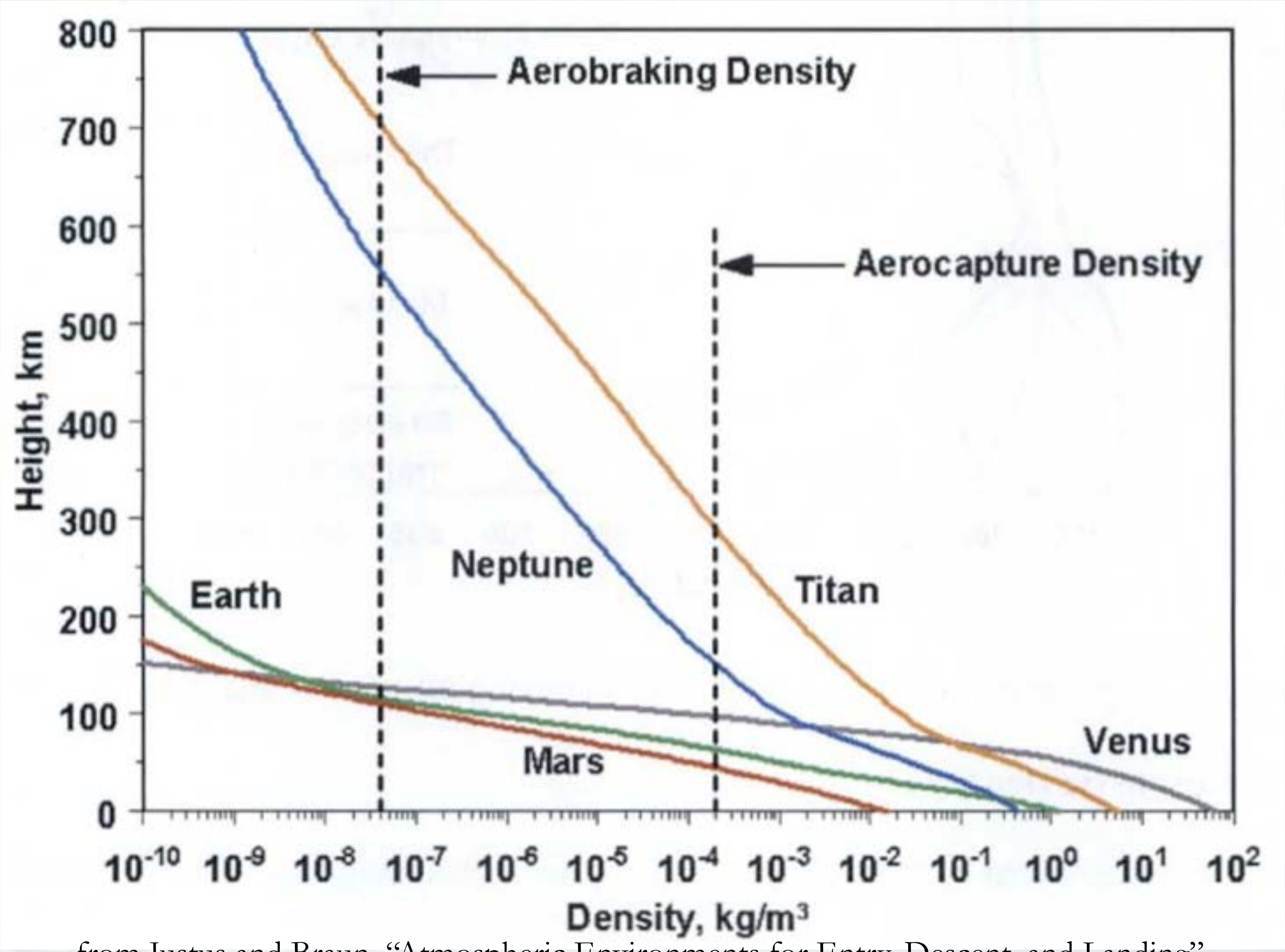
$$h_s = \text{Scale height}$$

Atmospheric Thermal Profiles



from Justus and Braun, "Atmospheric Environments for Entry, Descent, and Landing", 5th International Planetary Probes Workshop, August 2006

Planetary Atmospheric Density

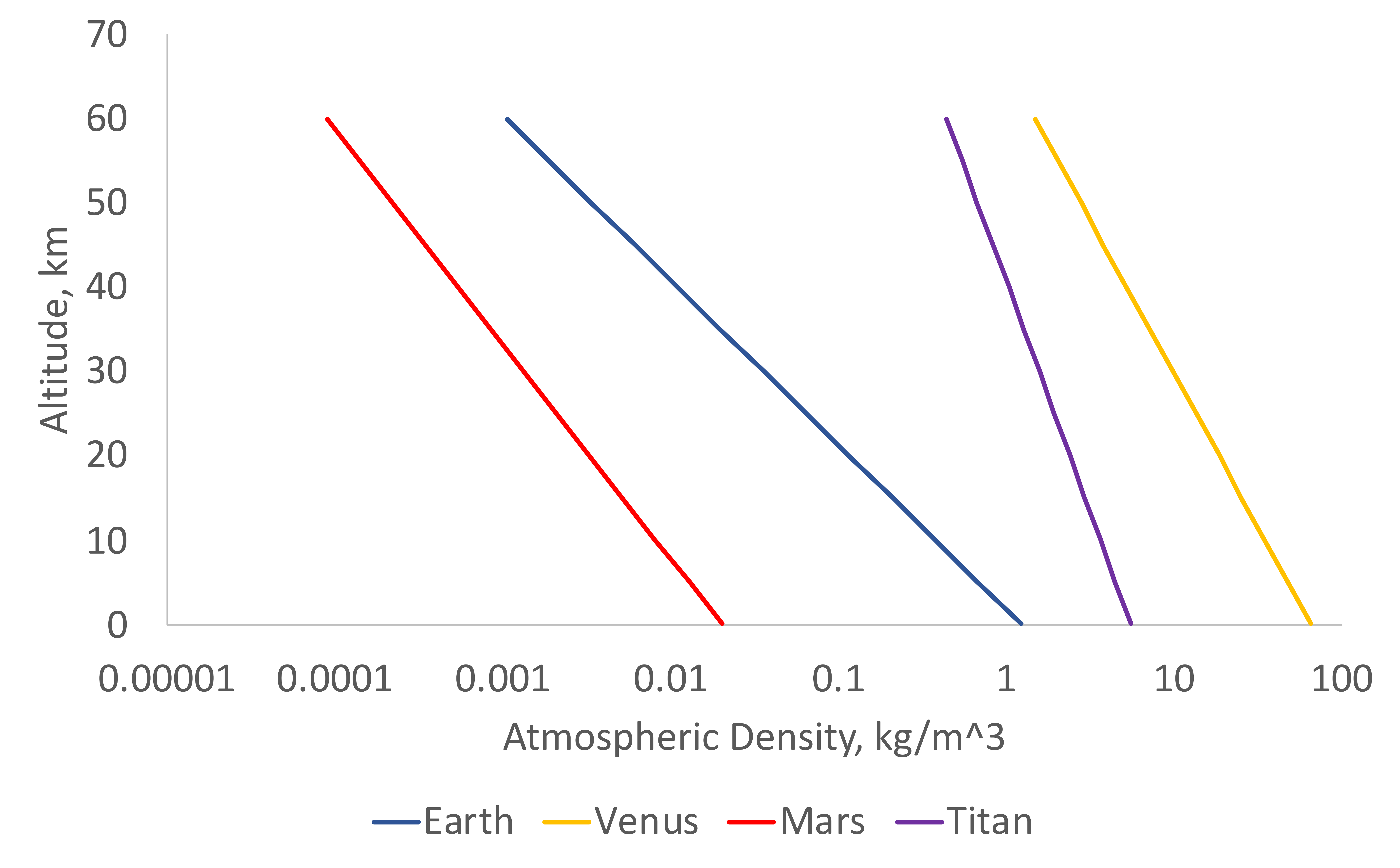


from Justus and Braun, “Atmospheric Environments for Entry, Descent, and Landing”, 5th International Planetary Probes Workshop, August 2006

Planetary Entry - Physical Data

	Radius (km)	μ (km³/sec²)	ρ_o (kg/m³)	h_s (km)	v_{esc} (km/sec)
Earth	6378	398,604	1.217	8.5	11.18
Mars	3393	42,828	0.020	11.1	5.025
Venus	6052	325,600	64.79	15.9	10.37
Titan	2575	8969	5.474	23.93	2.639

Exponential Atmospheric Density Models



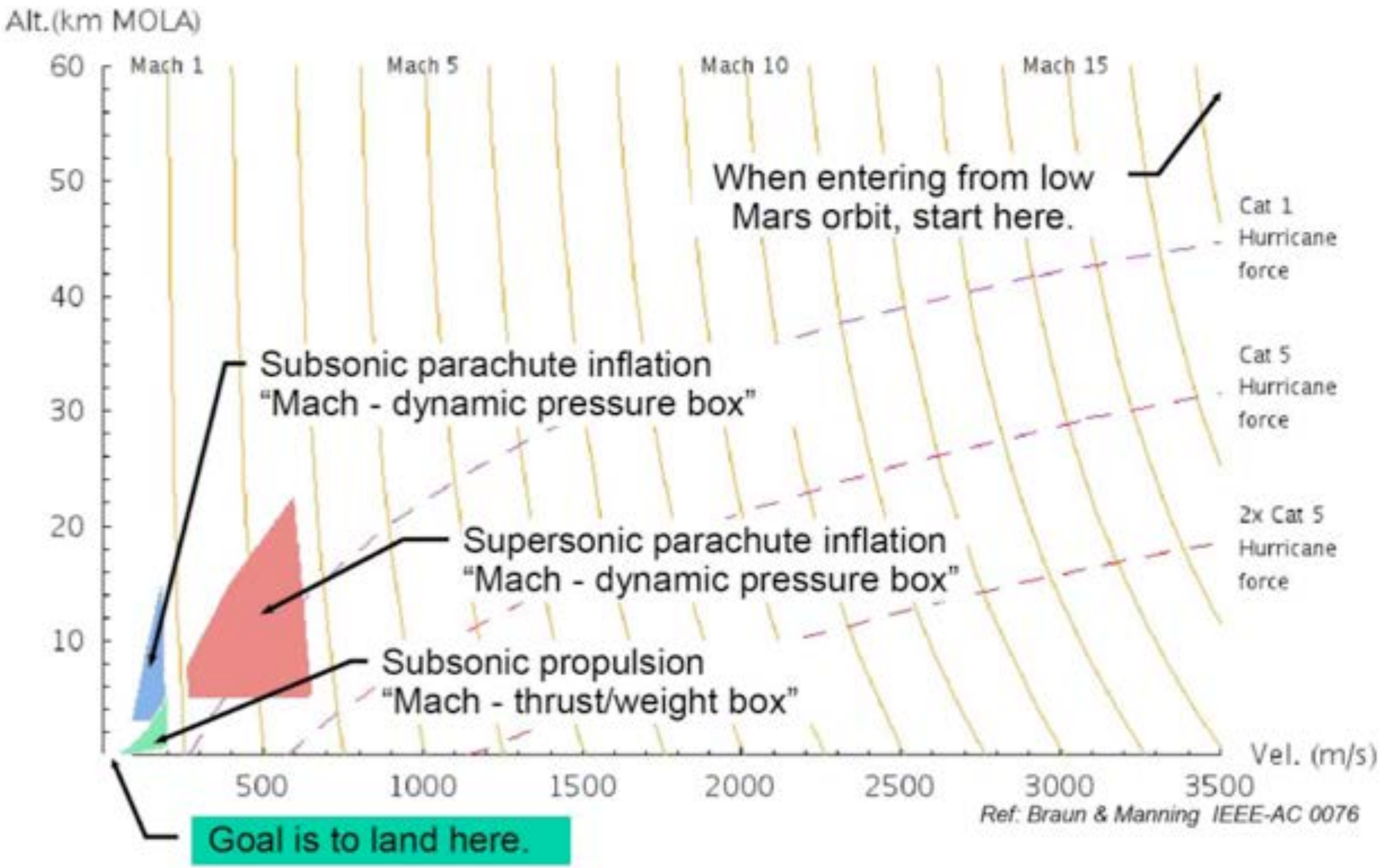
Atmospheric Entry, Descent, and Landing

- Savings in propellant by dissipating entry energy in atmosphere
  - Δv for lunar landing ~2200 m/sec
  - Δv for Mars landing ~500 m/sec
- Requires heat shield/aeroshell, aerodynamic decelerators, etc.
- Terminology
  - **Entry** covers atmospheric interface through peak heating and deceleration
  - **Descent** covers atmospheric deceleration to subsonic velocity and ground proximity
  - **Landing** covers deceleration to touchdown velocity and stable orientation on surface

Orbital Entry - The Physics

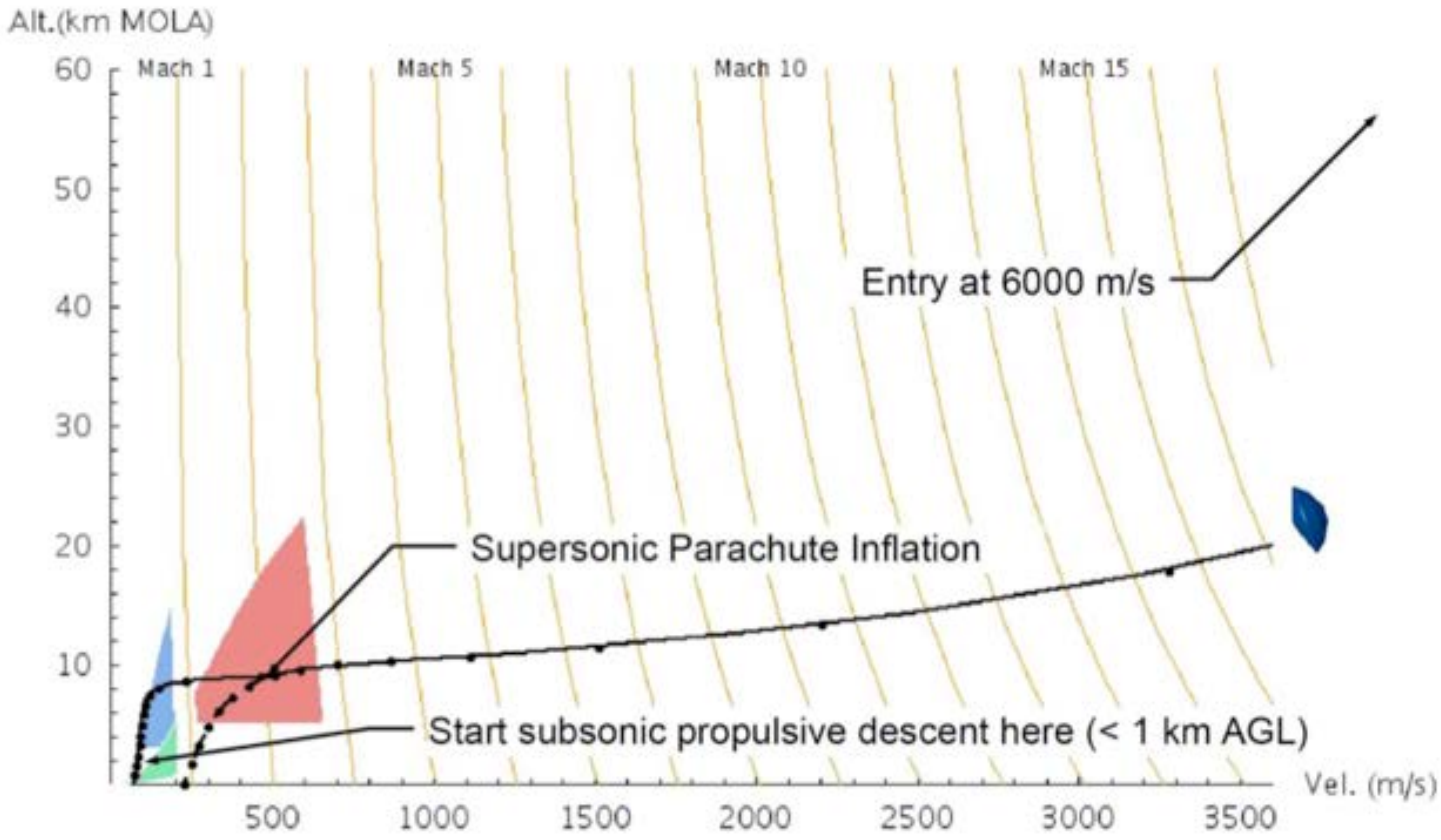
- 32 MJ/kg dissipated by friction with atmosphere over ~8 min = 66kW/kg
- Pure graphite (carbon) high-temperature material: c_p=709 J/kg°K
- Orbital energy would cause temperature gain of 45,000°K!
- Survival depends on two factors
  - Dumping 99.9% of heat to atmosphere as the entry vehicle passes through mitigates stagnation point heating to ~3000°K
  - Heat shield to protect payload from residual entry heat

EDL Phase Plot – A Handy Way to Visualize EDL

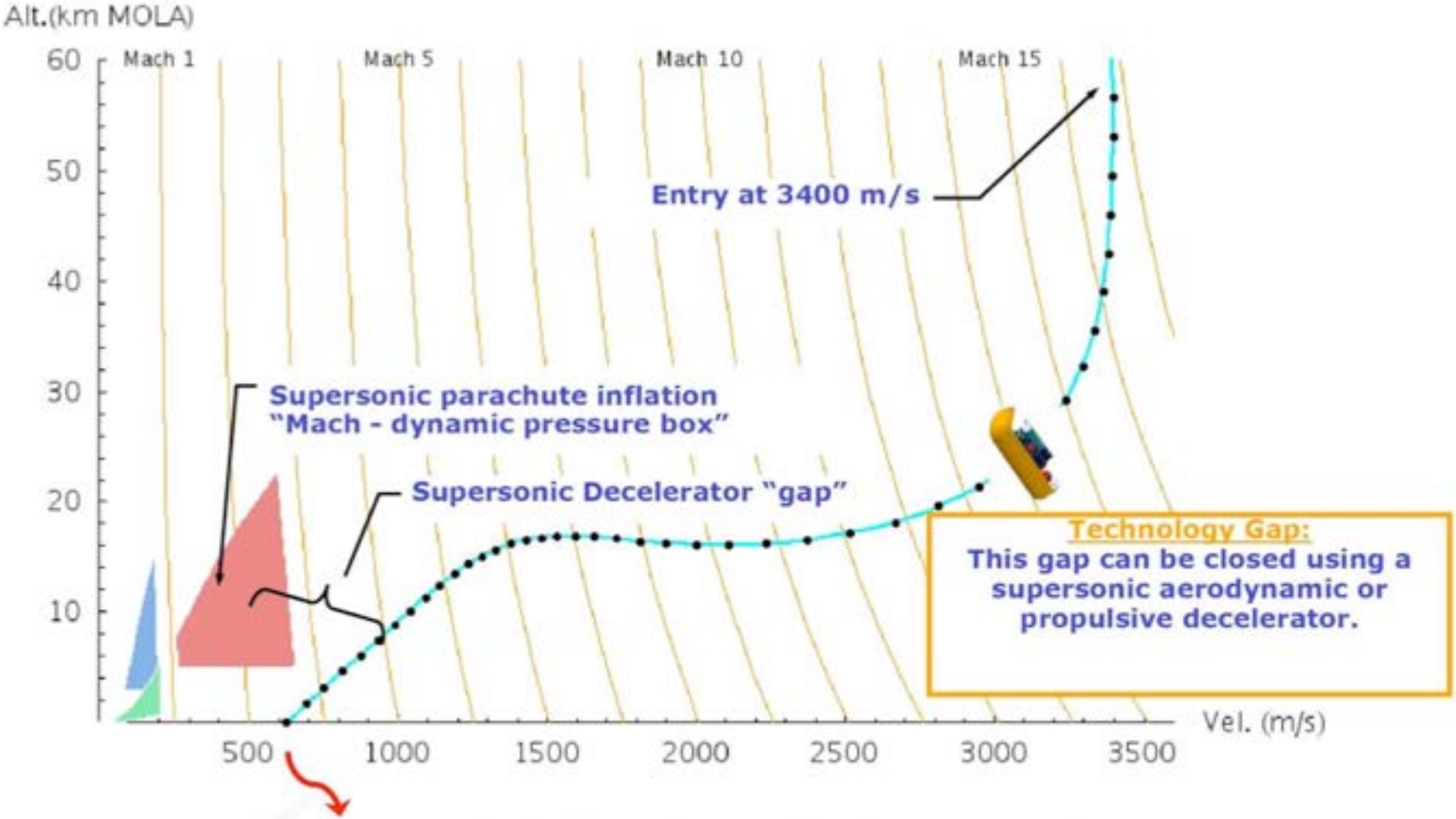


Ref: Braun & Manning IEEE-AC 0076

Robotic program: No gap so far ….



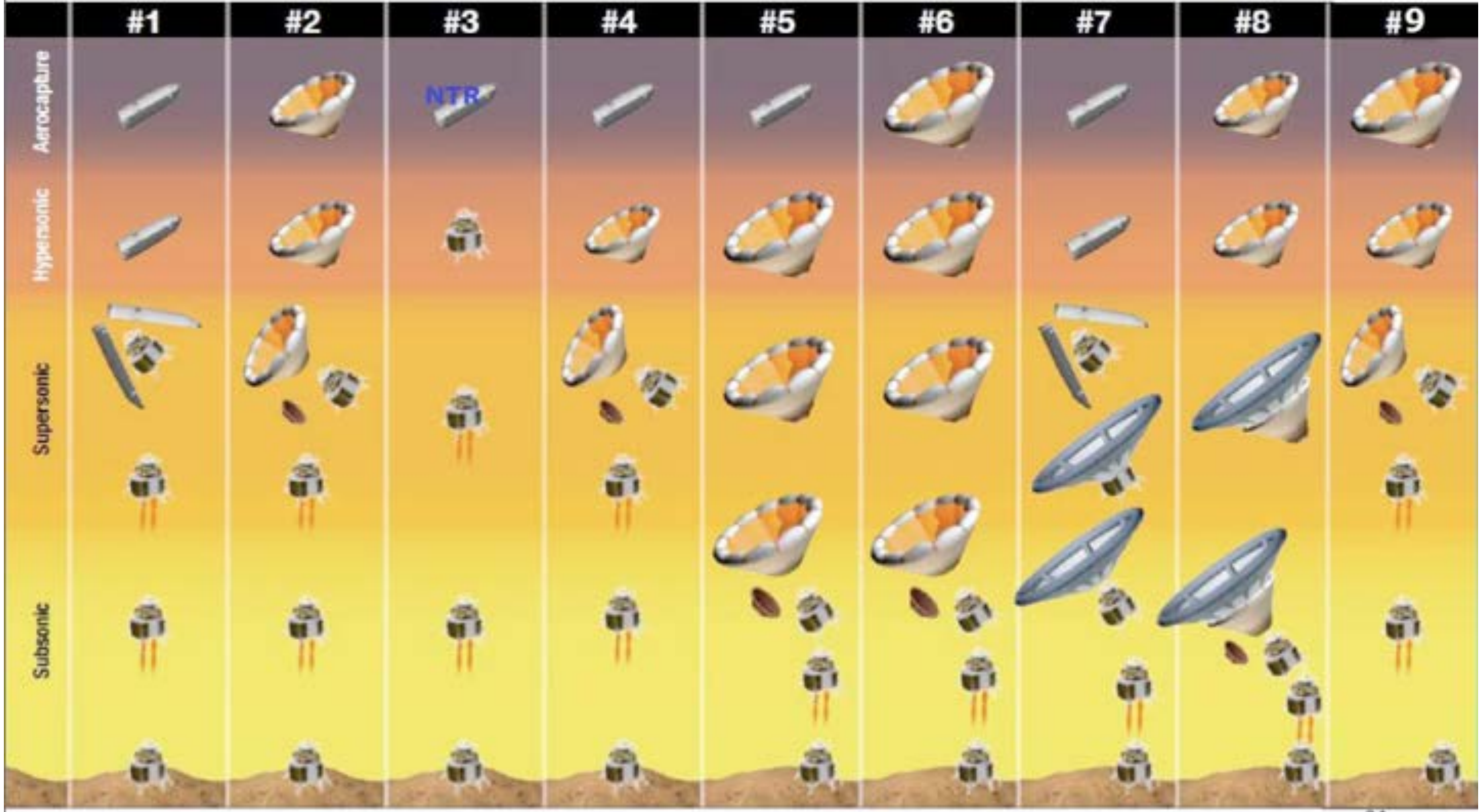
How would Humans Land?



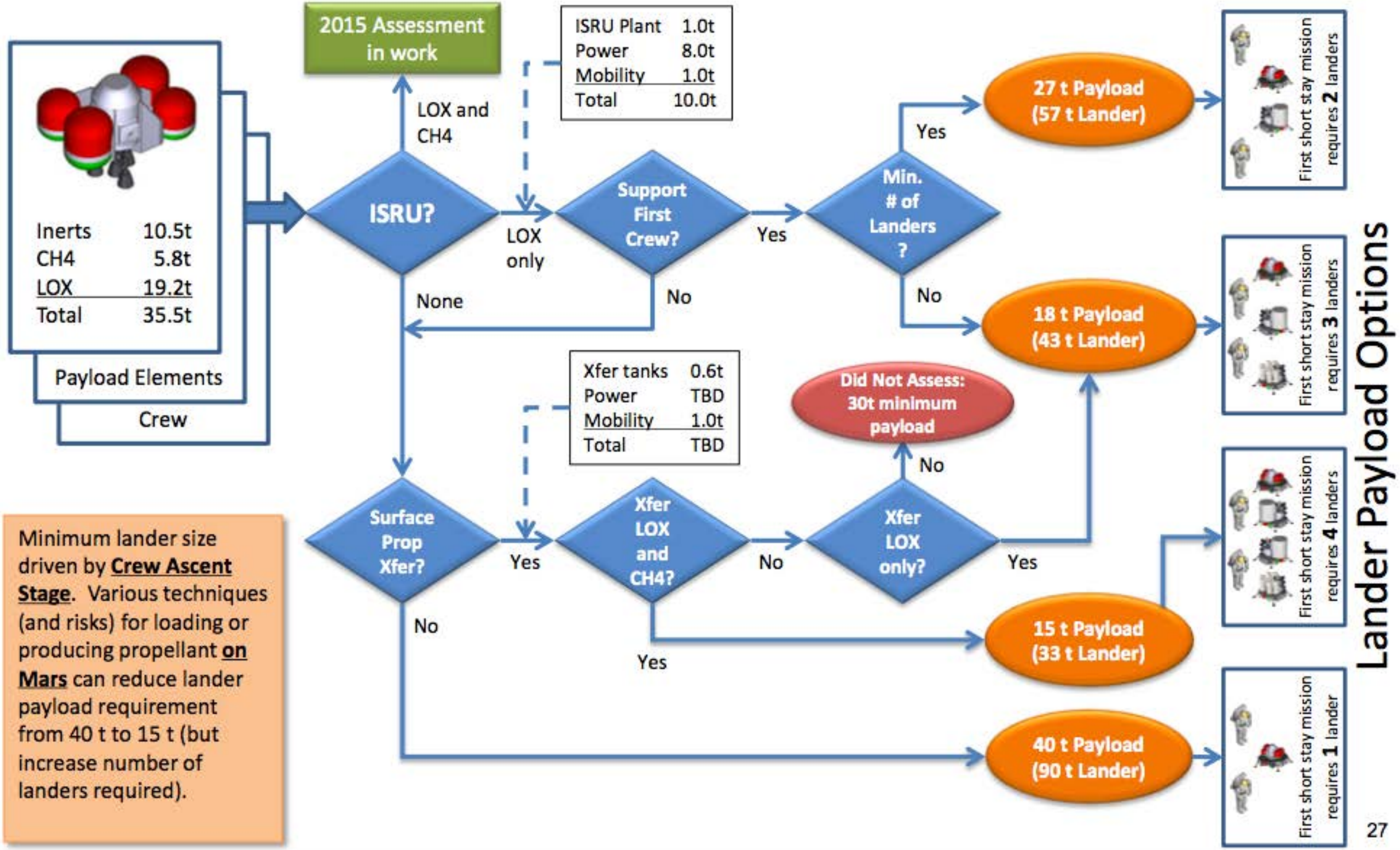
Without new technologies we have surface impact at Mach 2.5

Potential Exploration Architectures

Some possible combinations…



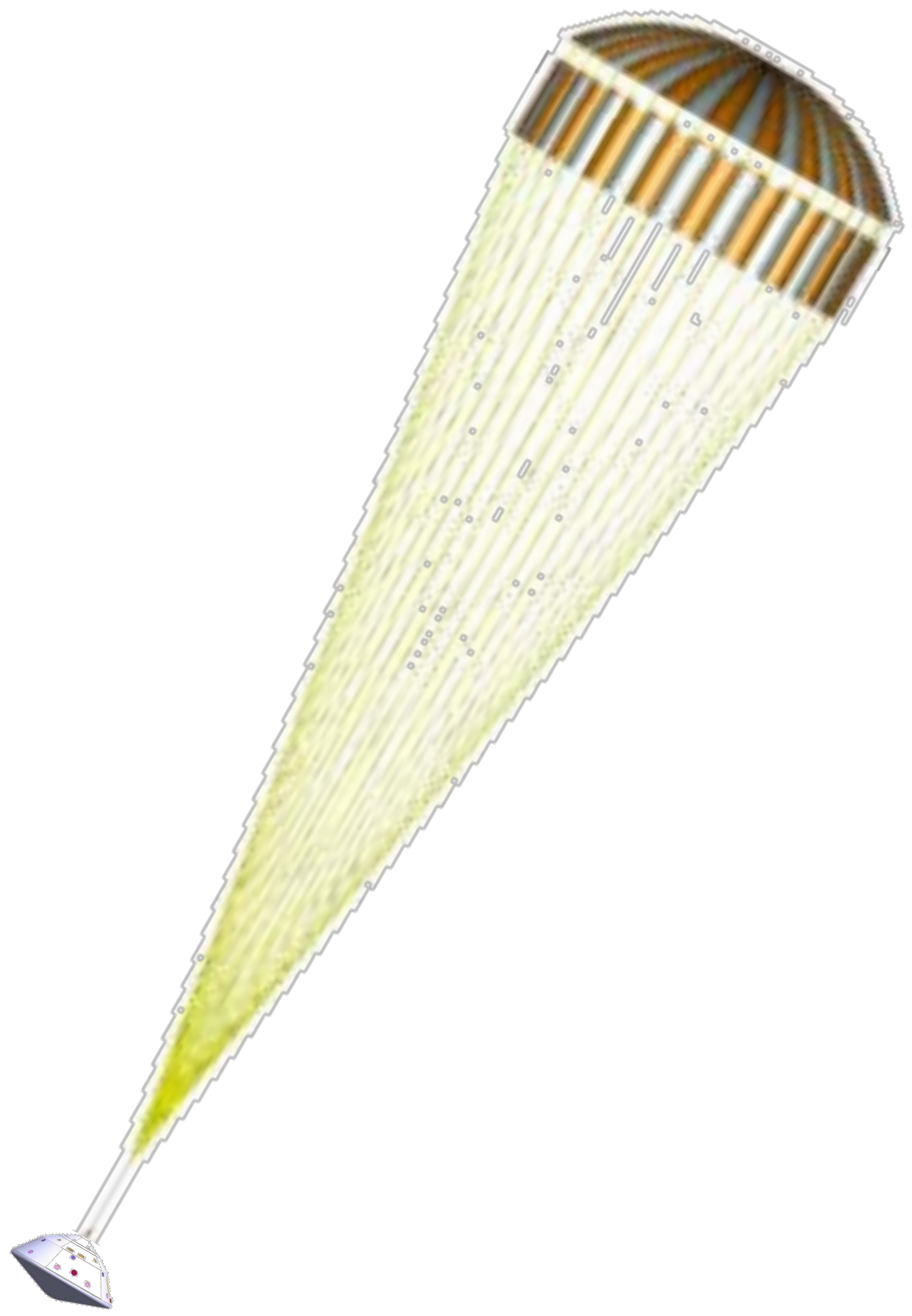
Largest Indivisible Payload Element and Options for Size of the Lander



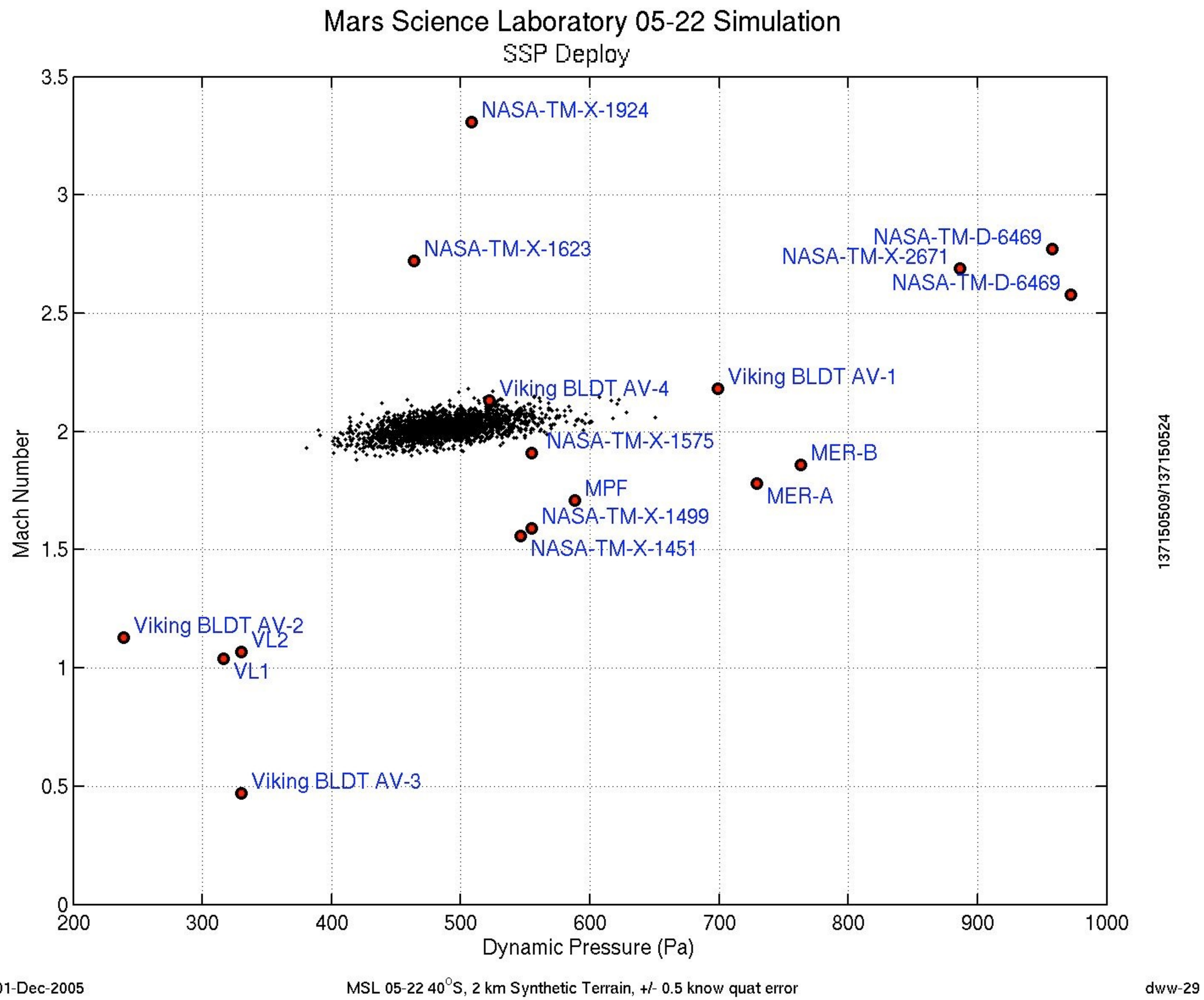
Minimum lander size driven by **Crew Ascent Stage**. Various techniques (and risks) for loading or producing propellant **on Mars** can reduce lander payload requirement from 40 t to 15 t (but increase number of landers required).

Parachute Descent

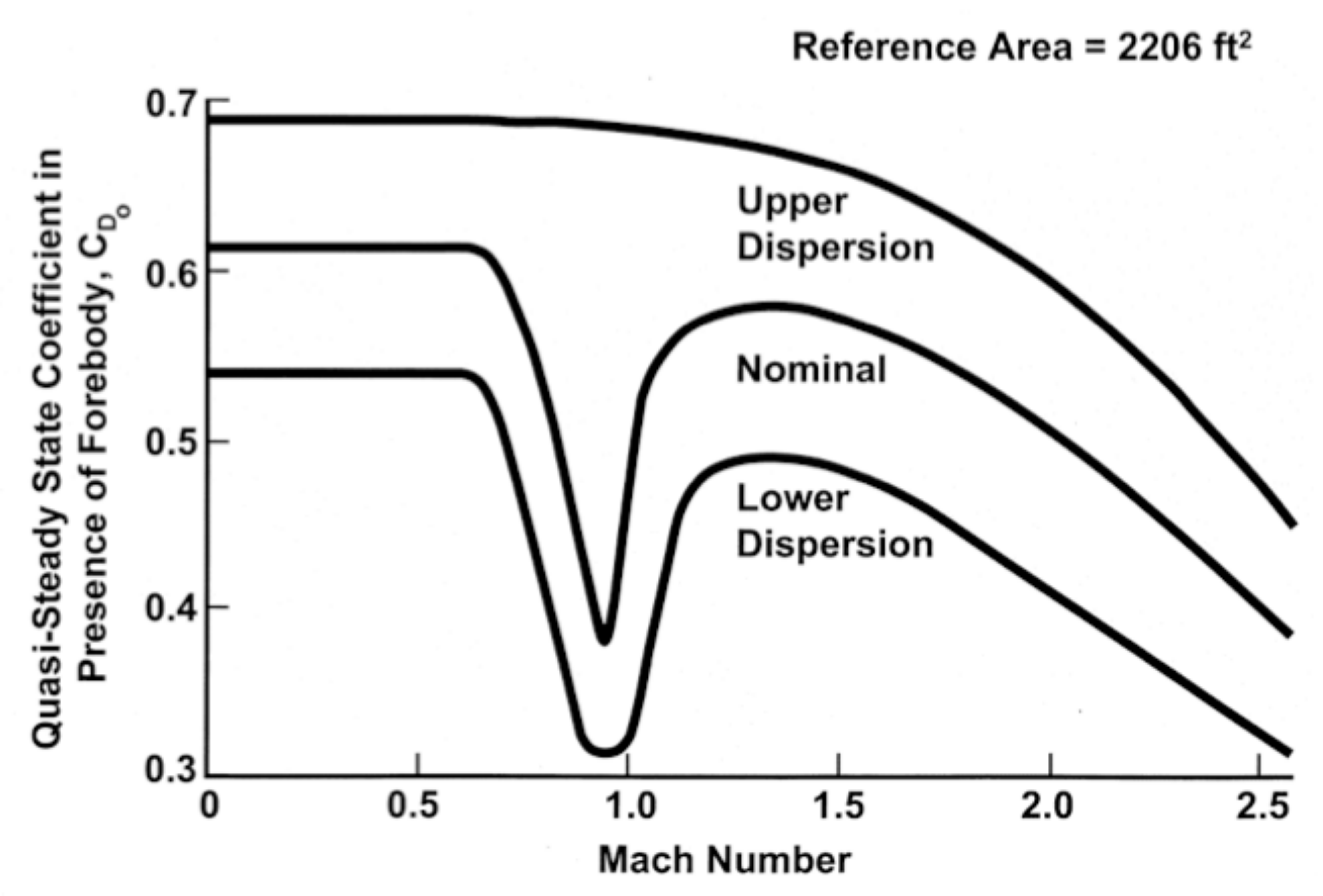
- Secondary decelerator is Parachute drag
  - Approximately 95% of remaining Kinetic energy is dissipated to the atmosphere
- Viking configuration parachute
  - Larger diameter (19.7 m vs 16.1 m)
  - Modern materials (kevlar vs. polyester)
- Deployment conditions
  - Mach number < 2.15 (Viking)
  - Dynamic Pressure < 850 Pa (MER)
  - Deployment AoA @ deploy < 15 deg. (Viking)
- Parachute scaled to closely match Viking test post deployment flight conditions
  - Area ratios
  - On chute ballistic coefficient
  - Area oscillations matched



Parachute Deployment



Viking Parachute Drag Coefficient Model



from Cruz and Lingard, "Aerodynamic Decelerators for Planetary Exploration: Past, Present, and Future", AIAA 2006-6792, AIAA Guidance, Navigation, and Control Conference, August 2006

Terminal Velocity

Full form of ODE -

$$\frac{d\left(v^2\right)}{d\rho} - \frac{h_s}{\beta \sin\gamma} v^2 = \frac{2gh_s}{\rho}$$

At terminal velocity, $v = \text{constant} \equiv v_T$

$$-\frac{h_s}{\beta \sin\gamma} v_T^2 = \frac{2gh_s}{\rho}$$

$$v_T = \sqrt{-\frac{2g\beta \sin\gamma}{\rho}}$$

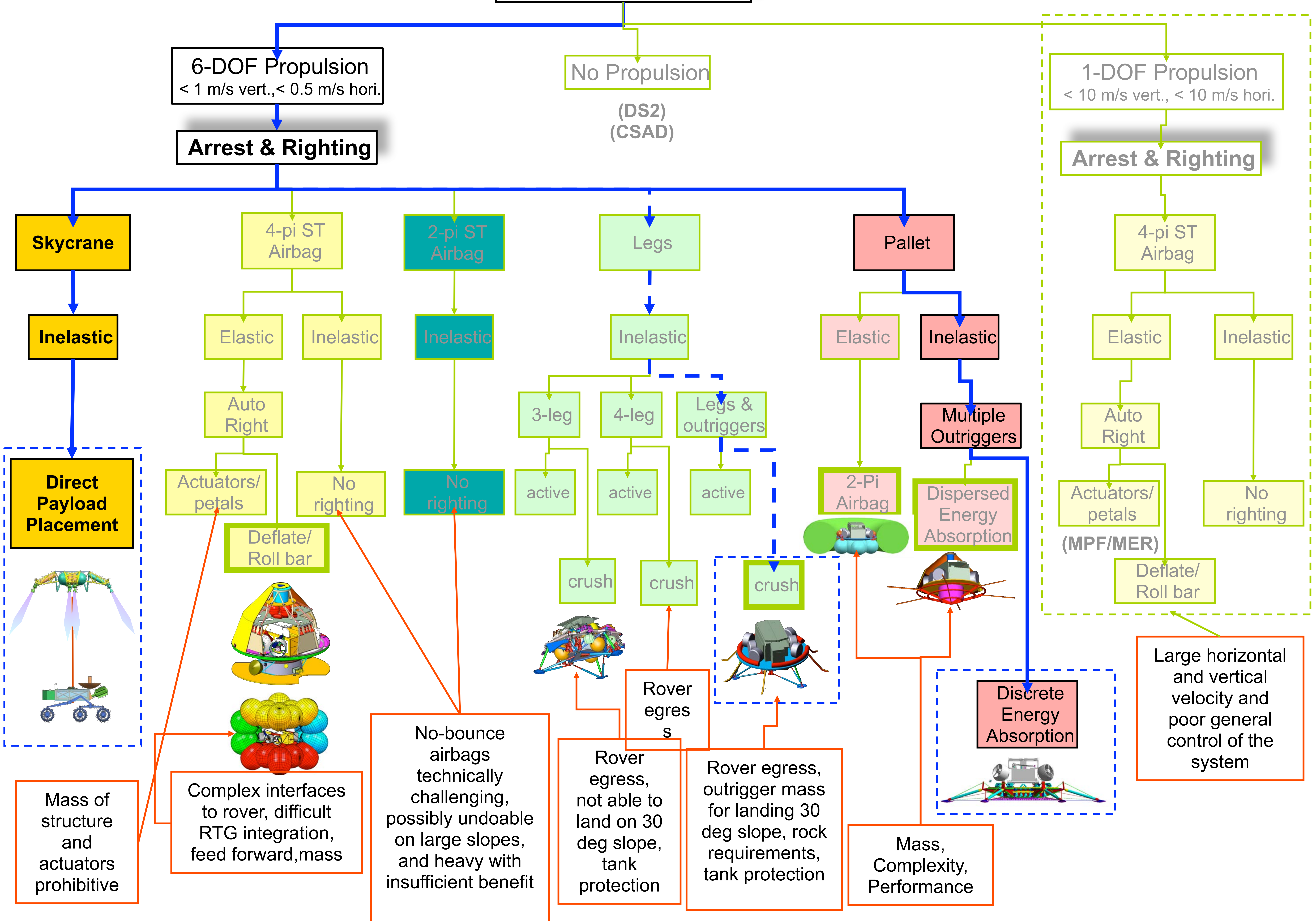
Viking Terminal Velocity Under Chute

$$\beta = \frac{m}{c_D A} = \frac{930\ kg}{0.62\left(\frac{\pi}{4}\right)(16.15\ m)^2} = 7.322\ \frac{kg}{m^2}$$

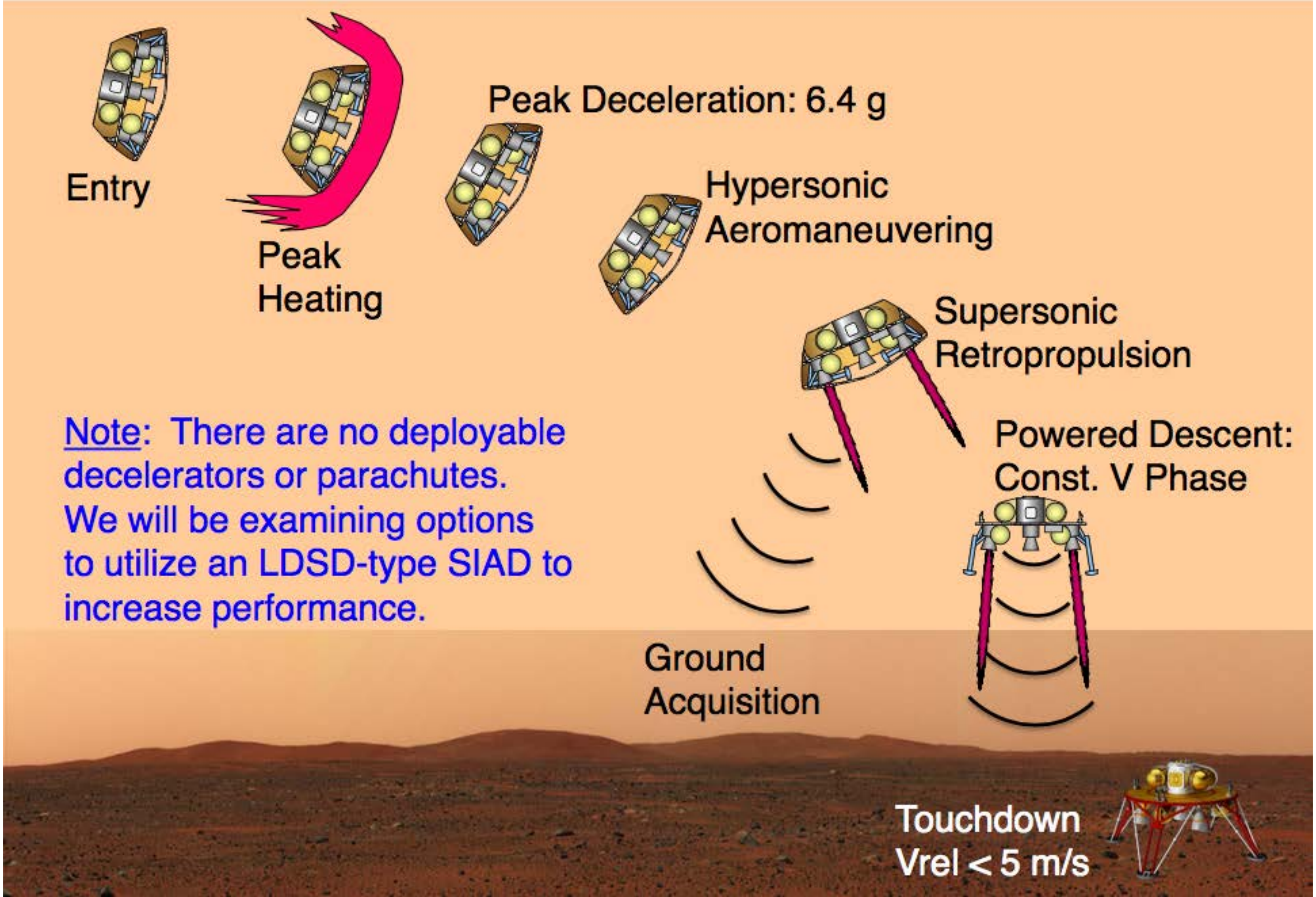
$$v_T = \sqrt{-\frac{2g\beta\sin\gamma}{\rho}} = \sqrt{-\frac{2(3.711\ m/s^2)(7.322\ kg/m^2)\sin(-30^o)}{0.02\ kg/m^3}} = 36.9\ \frac{m}{sec}$$

$$\beta_{crit} = -\frac{\rho_o h_s}{\sin\gamma} = -\frac{0.02\ kg/m^3(10,800\ m)}{\sin(-30^o)} = 432\ \frac{kg}{m^2}$$

Trade Coverage Example: Terminal Descent



EDL Concept for Blunt Body Mars Lander



Atmospheric Neutral Buoyancy

- Given an enclosed volume V of gas with density ρ
- Lift force is $V(\rho_{atm}-\rho)$ - must be $\geq mg$
  - on Earth ~1 kg lift/cubic meter of He
  - on Mars ~10 gms lift/cubic meter of He
- Horizontal velocity at equilibrium is identical to wind speed
- Interior pressure generally identical to ambient (except for superpressure balloons)
- Can generate low density through choice of gas, heating

Buoyancy by Light Gases

- Ideal gas law $$PV = nRT$$
- Given same volume and temperature, gas densities scale proportionally to molecular weight *n*
- Mars' atmosphere is essentially CO_2 – $n = 44$
  - He: $n = 4; \quad \Delta\rho = 90.3 \ gm/m^3$
  - H_2: $n = 2; \quad \Delta\rho = 94.8 \ gm/m^3$
- *Hindenburg* airship would have a total lift capacity of 49,900 kg in Mars atmosphere and gravity (Earth lift capacity 232,000 kg - factor of 4.6)

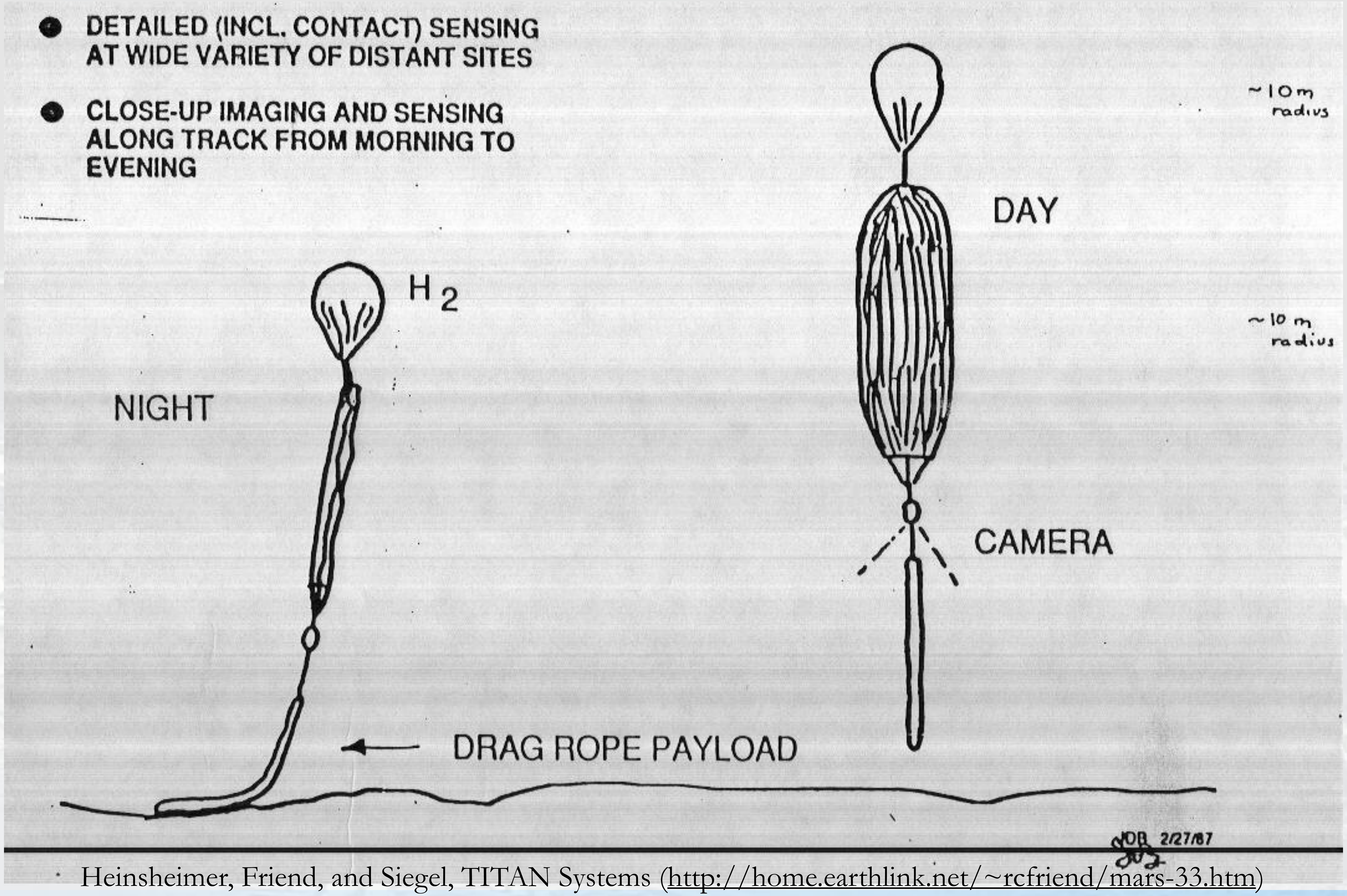
Goodyear Blimp

- Volume 5380 m^3
- Empty mass 4252 kg
- Gross mass 5824 kg
- Mars lift 1278 kg

Thermal Balloons ("Montgolfieres")

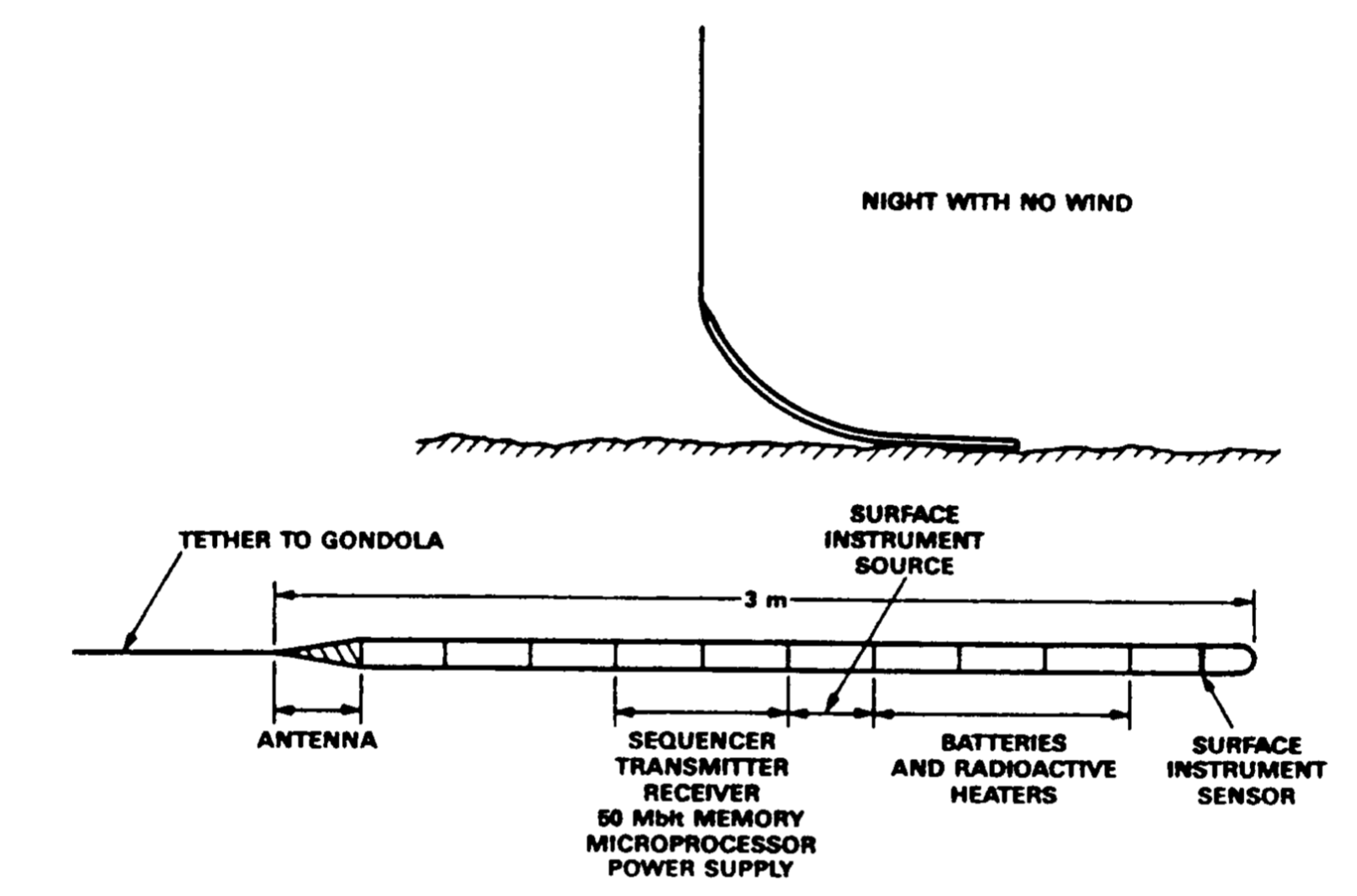
- Use ambient gases and thermal difference to create lift
- Ideal gas – gas density inversely proportional to temperature
- Ambient atmospheric temperature on Mars ~200K
- Heat gases to 300K: lift force 33 gm/m^3 (about 1/3 of He or H_2 balloon)

Dual-Lift Mars Balloon Concept



Heinsheimer, Friend, and Siegel, TITAN Systems (http://home.earthlink.net/~rcfriend/mars-33.htm)

Data Collection by Dragging



Heinsheimer, Friend, and Siegel, "Concepts for Autonomous Flight Control for a Balloon on Mars" NASA 89N15600

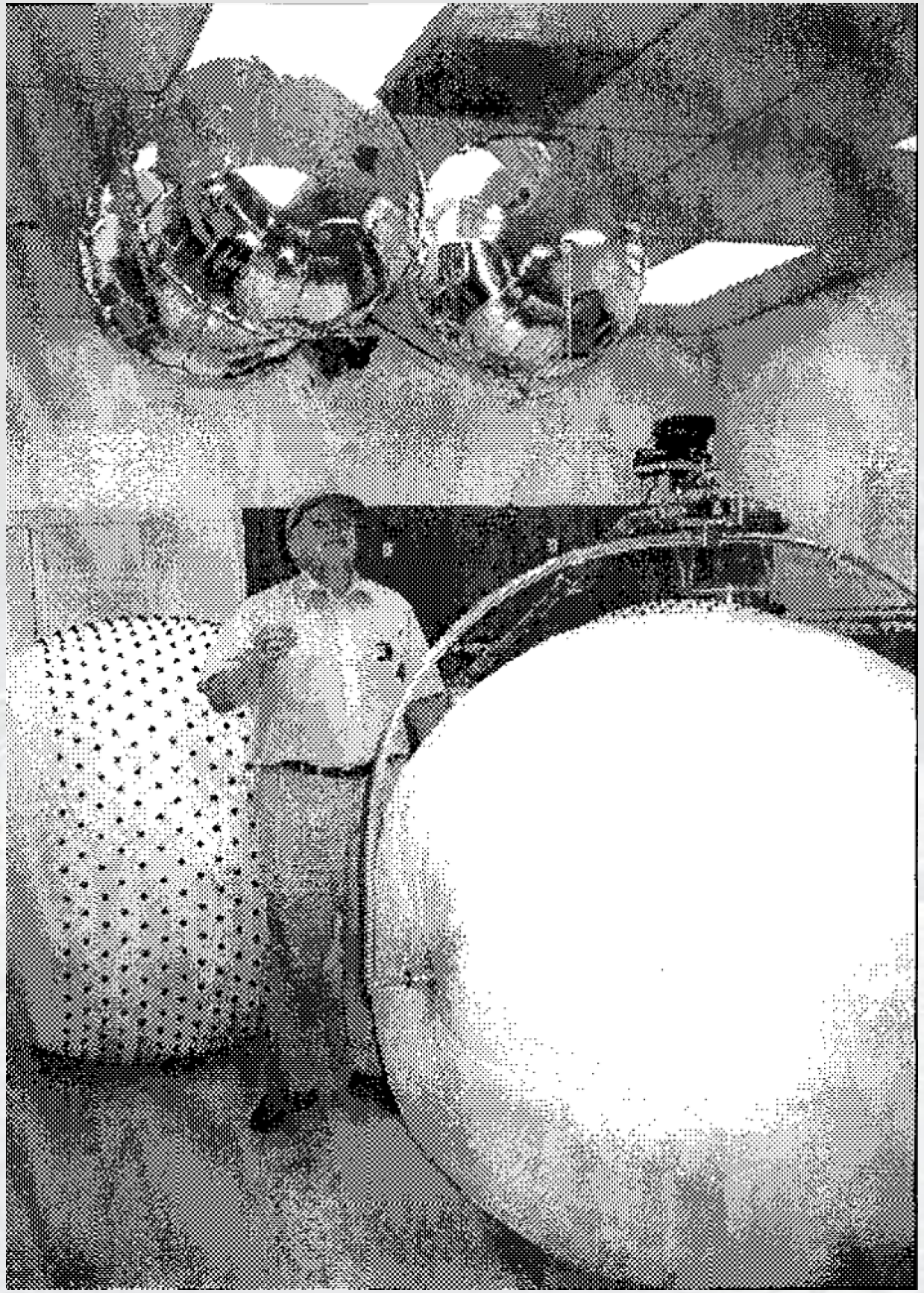
Superpressure Balloons

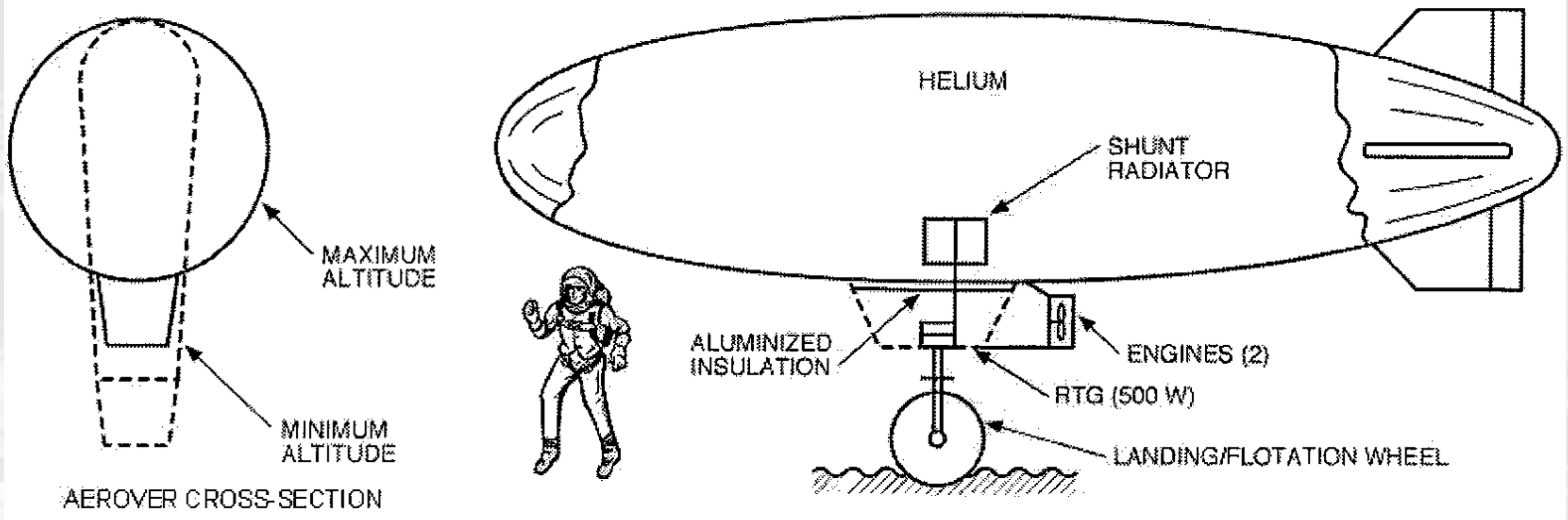
- Interior pressure greater than external ambient
- Envelope is relatively insensitive (in terms of volume) to interior pressure changes
- Diurnal temperature changes have minimal effect on lift
- Provides stable long-term platform for extended flights
- Envelope must be significantly stronger (and therefore heavier) than ambient-pressure balloons

Flight Missions with Balloons

- Venus: Vega - Russian Vega missions put two French balloons in Venus atmosphere in 1985
  – One died in 56 minutes
  – One operated for two days (battery limitations)
- Mars: French dual-balloon system (solar thermal balloon tied to He/H2 balloon - gas balloon keeps solar balloon off the ground, thermal balloon lifts payloads when sun warms envelope) -never flew

Future Concepts – Titan Aerover



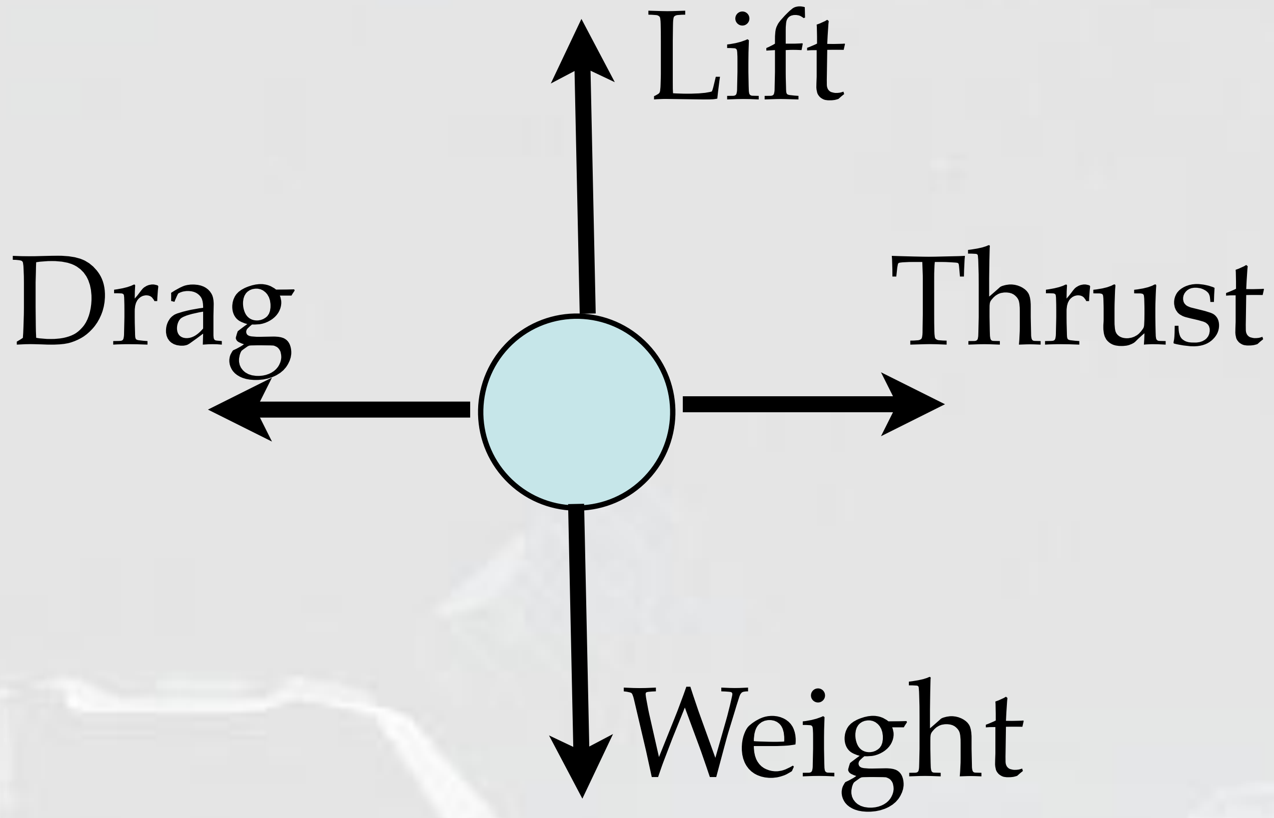


NASA Concept for Venus Habitation

“Heavier than Atmosphere” Approaches

- Fixed wing
  - Gliders
  - Powered
    - Propellers
    - Jet
    - Rocket
- Rotary wing
- Hybrid/Reconfigurable

Dynamic Atmospheric Lift



$$D = \frac{1}{2}\rho v^2 S c_D$$

$$L = \frac{1}{2}\rho v^2 S c_L$$

For steady, level flight: $T = D \qquad L = W = mg$

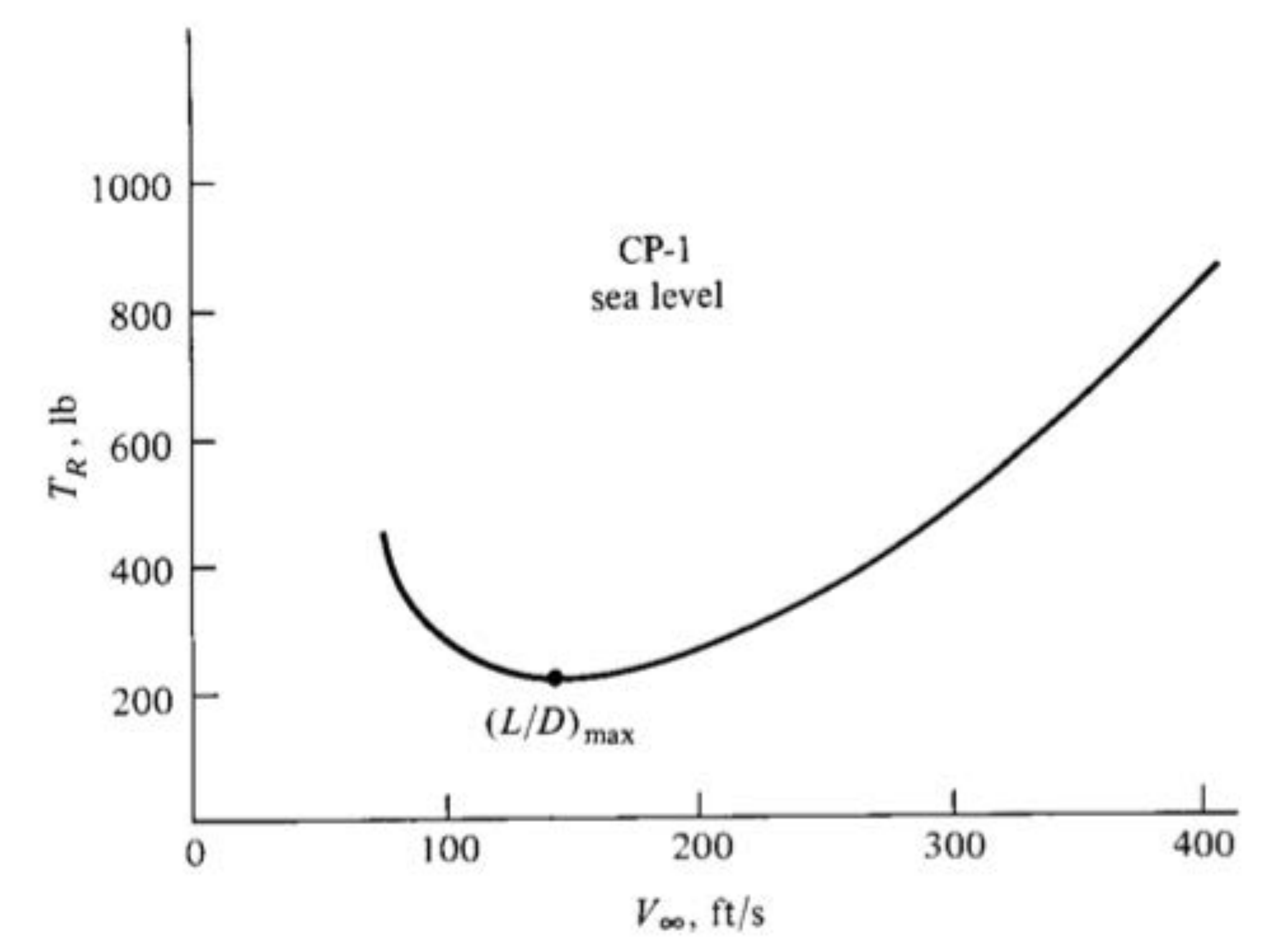
$$W = L = D\frac{L}{D} = T\frac{L}{D} \qquad T = \frac{W}{L/D}$$

$$L = \frac{1}{2}\rho v^2 S c_D \frac{L}{D}$$

Atmospheric Flight Performance

$$L = \frac{1}{2}\rho v^2 S c_L$$

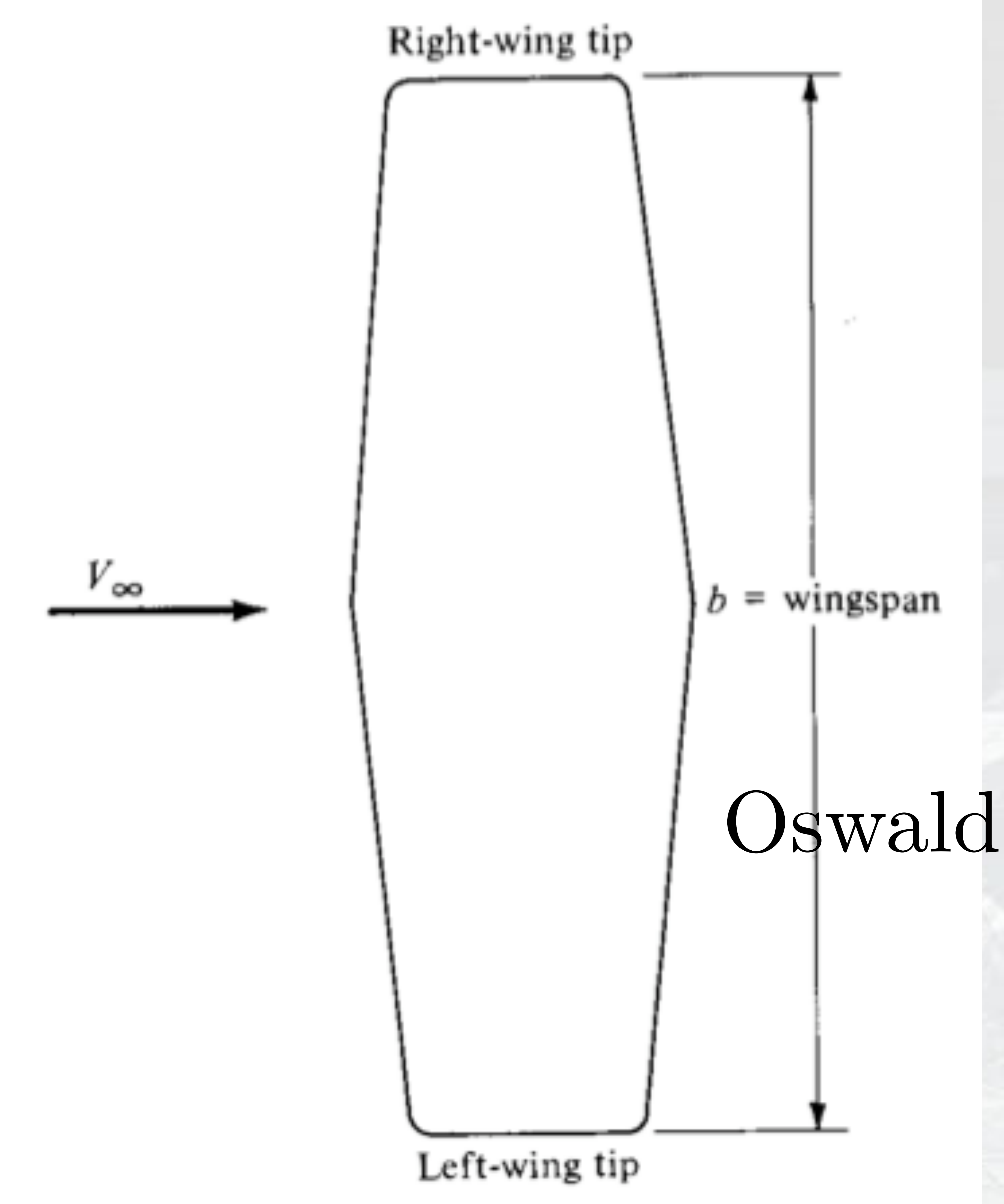
$$D = \frac{1}{2}\rho v^2 S c_D$$



from Anderson, *Introduction to Flight*, Third Edition McGraw Hill, 1989

$$c_D = c_{Do} + c_{Di} = c_{Do} + \frac{c_L^2}{\pi e(AR)}$$

Aspect Ratio

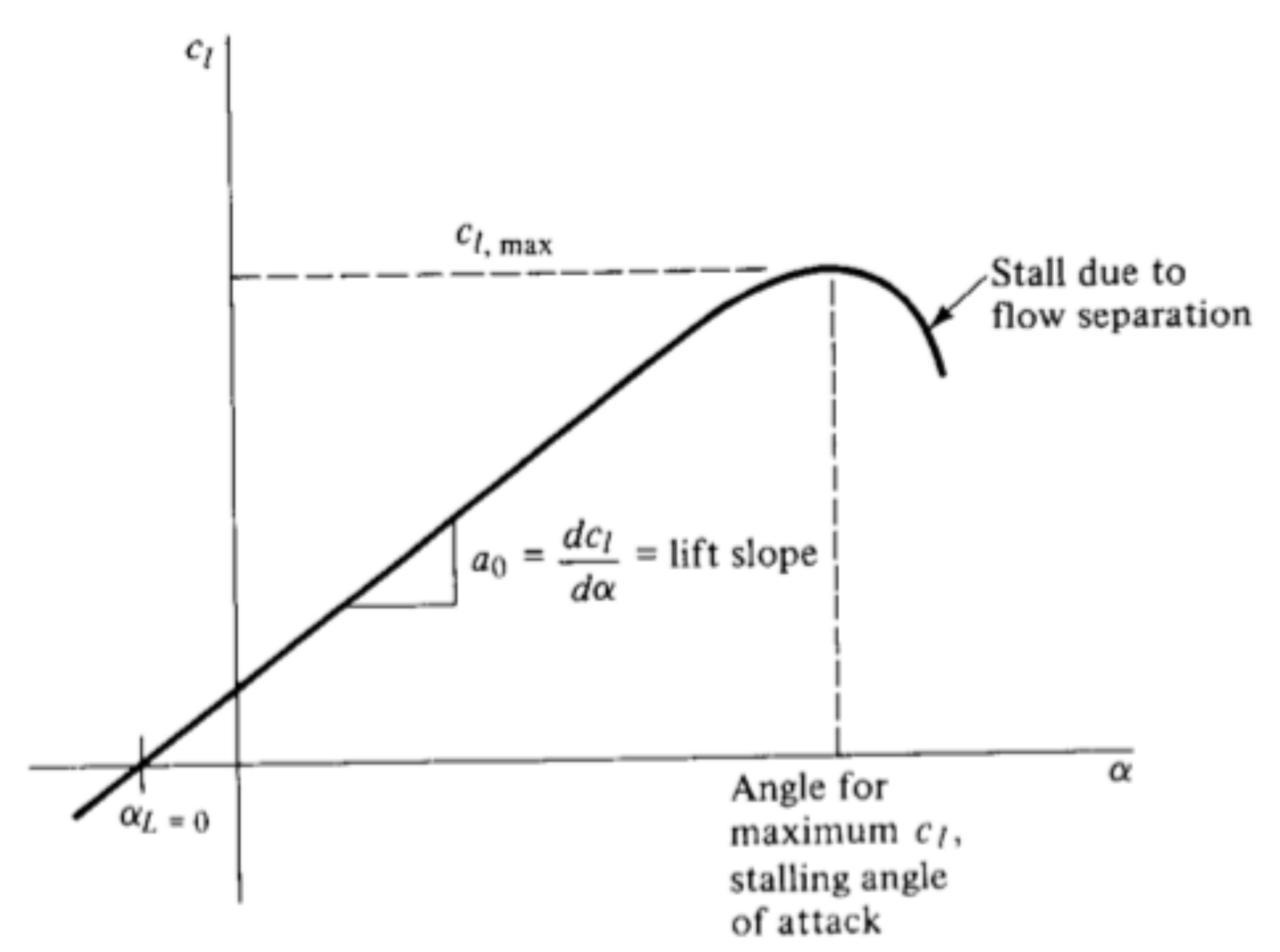


$$\text{Wing area} \equiv S$$

$$\text{Aspect ratio} \equiv AR = \frac{b^2}{S}$$

$$\text{Oswald efficiency factor} \equiv e \approx 0.9$$

Lift Curve



from Anderson, *Introduction to Flight*, Third Edition McGraw Hill, 1989

Mars Atmosphere

$$\rho = 0.020 \ \frac{kg}{m^3}$$

$$T = 210 \ K$$

$$g = 3.71 \ \frac{m}{sec^2}$$

$$R = 188.92 \ \frac{J}{kg \ K}$$

$$\gamma = 1.2941$$

$$\text{Speed of sound } a = \sqrt{\gamma R T} = 226.6 \ \frac{m}{sec}$$

Aircraft Flight Performance

- U-2 high-altitude spy plane
- Cruises at "70,000+ feet"
- m=18,000 kg
- b=32 m
- S~64 m²

$$v_{stall} = \sqrt{\frac{mg}{S}\frac{2}{\rho c_{L(max)}}}$$

$$\text{U-2 } v_{stall(Mars)} = 228.4\ \frac{m}{sec^2}$$

Stable Gliding Flight

$$\text{Flight path angle } \gamma$$

$$D = mg \sin\gamma$$

$$mg = W = L \implies \sin\gamma = \frac{1}{L/D}$$

$$\text{High performance glider } L/D \approx 30$$

$$\text{Deploy at 10 km} \implies Range \approx 300\ km$$

$$V \approx 200\ \frac{m}{sec} \implies \text{Flight time } 25\ min$$

Powered Flight

$$T = \dot{m}(v_e - V)$$

$$v_e = \text{Exhaust velocity; } V = \text{Flight velocity}$$

$$\text{Power into flow } P_f = \frac{\dot{m}}{2}\left(v_e^2 - V^2\right)$$

$$\text{Power into flight } P_v = TV$$

$$\text{Propulsive efficiency } \eta_{prop} = \frac{2}{1 + \frac{v_e}{V}}$$

Power Required

$$\text{Powered required } \equiv P_R = T_R V$$

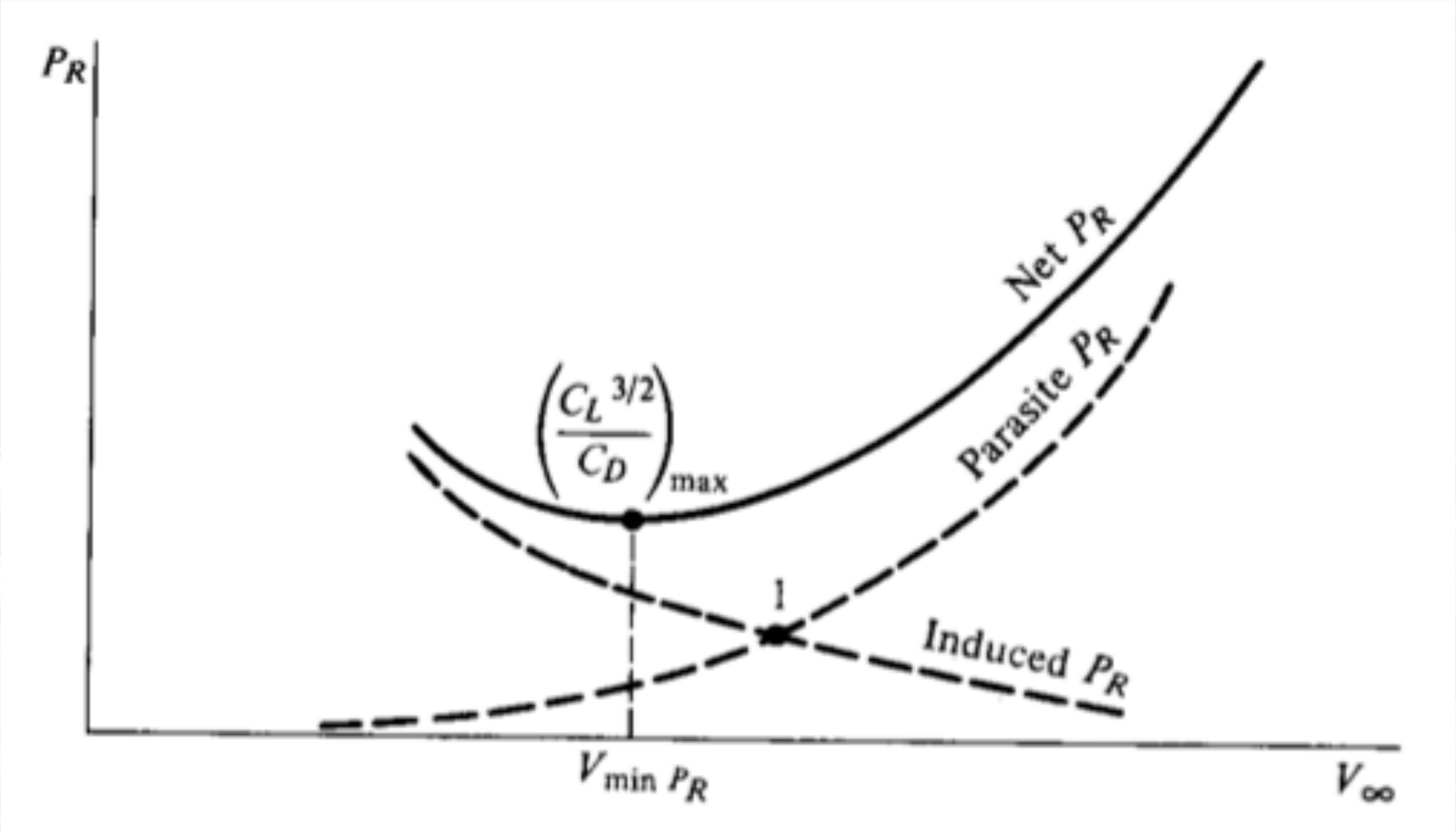
$$\text{Thrust required } \equiv T_R = \frac{W}{c_L/c_D}$$

$$L = W = \frac{1}{2}\rho V^2 S c_L \qquad V = \sqrt{\frac{2W}{\rho S c_L}}$$

$$P_R = \frac{W}{c_L/c_D}\sqrt{\frac{2W}{\rho S c_L}}$$

$$P_R = \sqrt{\frac{2W^3 c_D^2}{\rho S c_L^3}} \propto \frac{1}{c_L^{3/2}/c_D}$$

Power Required with Velocity



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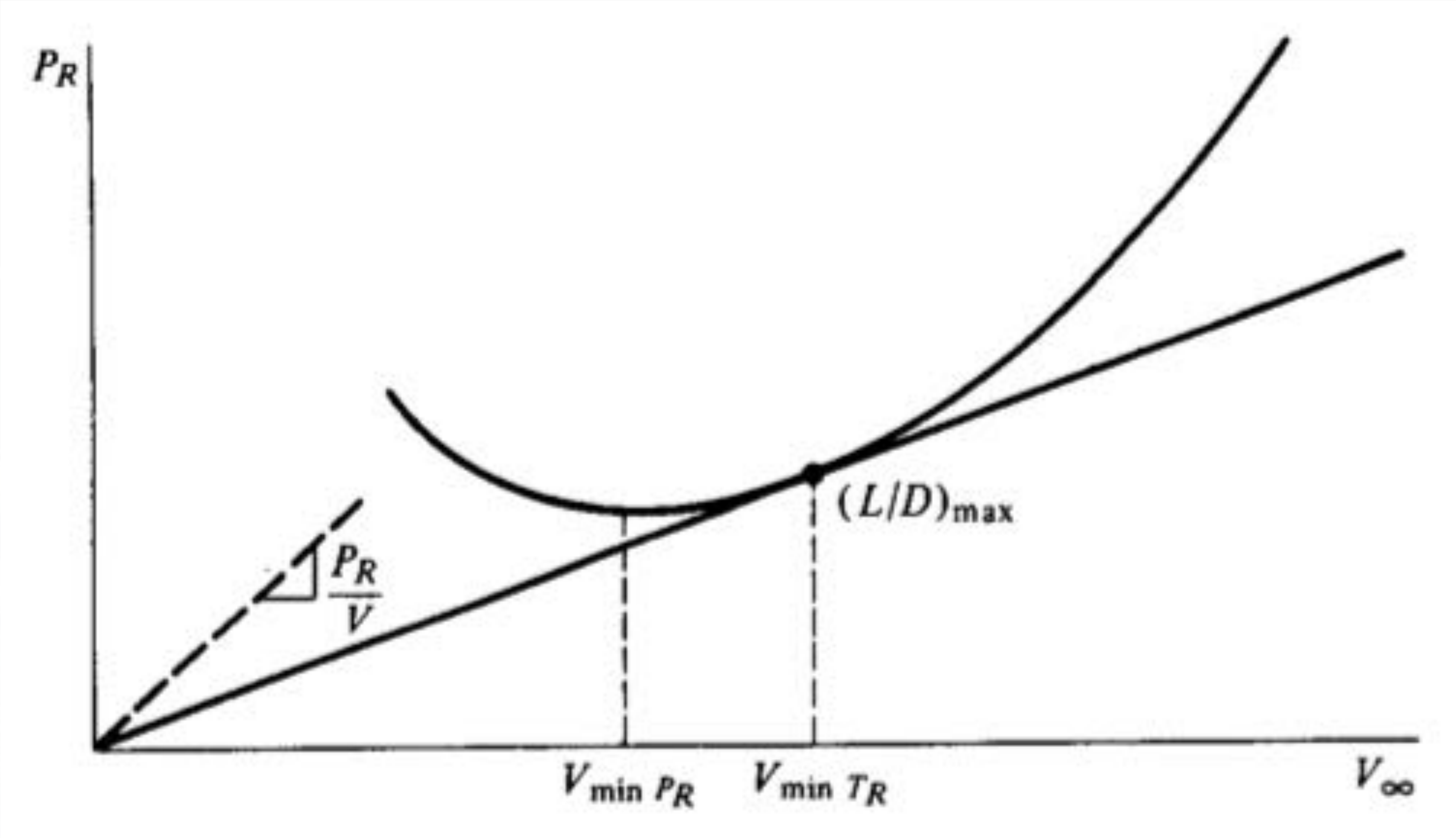
Minimum Power and Thrust



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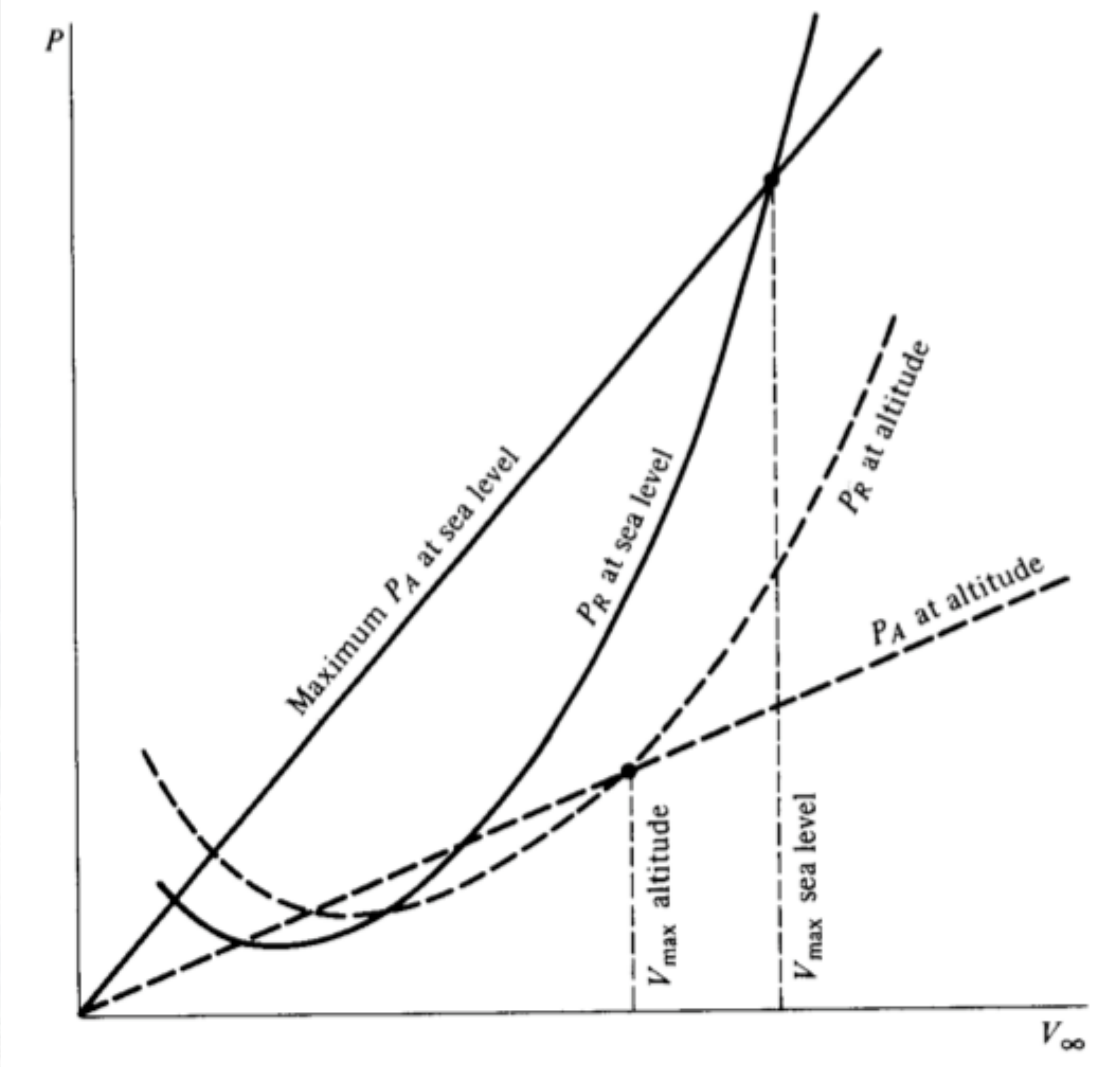
Effect of Altitude on Power



from Anderson, *Introduction to Flight*, Third Edition McGraw Hill, 1989

Actuator Disk Size

Engine intake area A

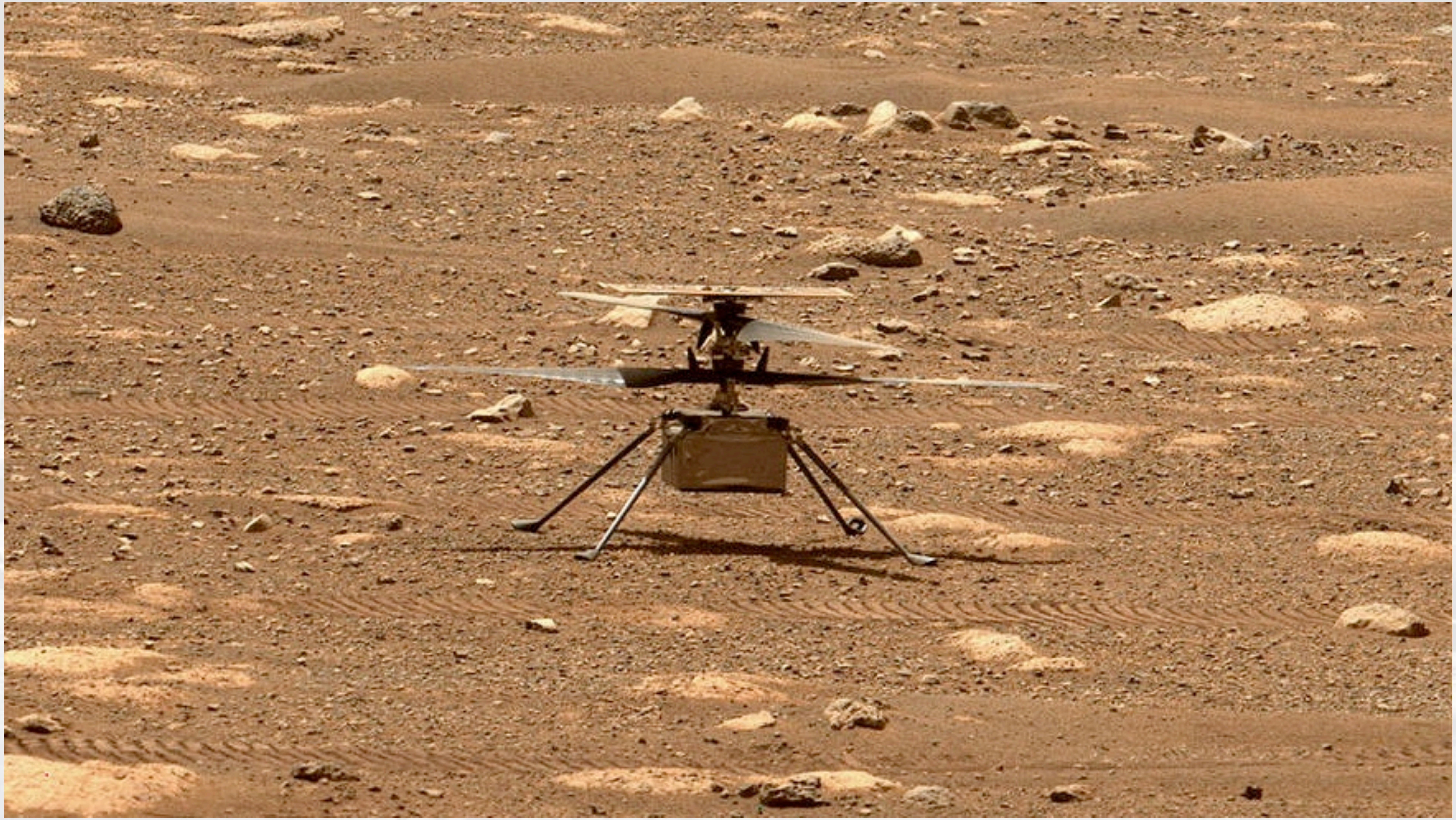
$$\dot{m} = \rho A V \qquad\qquad T = D = \frac{W}{L/D}$$

$$T = \dot{m}V = \rho A V (v_e - V)$$

$$\rho A V (v_e - V) = \frac{W}{L/D}$$

$$A = \frac{W}{(L/D)\rho V (v_e - V)}$$

Ingenuity – Mars Helicopter (2021)



Rotorcraft (Quick and Dirty)

- Thrust is downwards
- Hovering flight T=W
- Power calculations same as before if L/D=1
- Incline lift vector angle β from vertical

$$W = T\cos\beta \implies T = \frac{mg}{cos\beta}$$

$$D = T\sin\beta \implies D = mg\tan\beta$$

$$\frac{1}{2}\rho V^2 S c_D = mg\tan\beta \implies V = \sqrt{\frac{2mg\tan\beta}{\rho S c_D}}$$

(Classic) Helicopter Flight Controls

- Cyclic
  - Varies the angle of attack of the rotor blades as they rotate around the hub
  - Controls horizontal velocity
- Collective
  - Varies the angle of attack of all rotor blades simultaneously
  - Controls climb/descent
- Tail rotor
  - Corrects for the torque required for the rotor blades
  - Controls heading angle
- Throttle – engine speed/torque as required for flight

Alternative Vertical Flight Configurations

- Compound helicopter – stub wings for lift in forward flight (AH-56A Cheyenne)
- Coaxial – Two counter-rotating rotors, one above the other (Ingenuity)
- Tandem – two counter-rotating rotors separated by fuselage (CH-47 Chinook)
- Synchropter – two counter-rotating rotors mounted close together at an angle and synchronized so they rotate through each other (Kaman K-MAX)
- Tiltrotors – rotating engines/rotors to provide combination of lift/forward thrust (V-22 Osprey)
- Multirotors – Three or more rotors (quadcopters)

Differences of Multirotors

- Still utilized counter-rotating rotors to neutralize torque
- Translation accomplished by differential lift rotating thrust vector
- Simpler - no swash plates for collective/cyclic, fixed rotor blades
- Higher disk loading $\Longrightarrow$ lower efficiency
- More motors $\Longrightarrow$ more chance of failure, but increased potential for redundancy to mitigate failure(s)

Looking for Equation for Aircraft Range

$$\text{Efficiency} = \frac{\text{propulsive power}}{\text{fuel power}} = \frac{T v_e}{\dot{m}_f h}$$

$$h \equiv \text{heating value of fuel}$$

$$\eta_{overall} = \frac{T v_e}{\dot{m}_f h} \qquad \frac{dW}{dt} = -\dot{m}_f g = \frac{-W}{\frac{L}{D}\frac{T}{\dot{m}_f g}}$$

$$\frac{dW}{dt} = \frac{-W v_e}{\frac{h}{g}\frac{L}{D}\frac{T v_e}{\dot{m}_f h}} = \frac{-W v_e}{\frac{h}{g}\frac{L}{D}\eta_{overall}}$$

More Aerial Range

Rewrite and integrate

$$\frac{dW}{W} = \frac{-v_e dt}{\frac{h}{g}\frac{L}{D}\eta_{overall}} \Longrightarrow \ln W = C - \frac{-v_e t}{\frac{h}{g}\frac{L}{D}\eta_{overall}}$$

Initial conditions - $\text{at } t = 0 \; W = W_{init} \rightarrow C = \ln W_{init}$

$$\text{Range} = \frac{h}{g}\frac{L}{D}\eta_{overall} \ln \frac{W_{init}}{W_{final}}$$

$$\text{Range} = \frac{V\frac{L}{D}}{g \; SFC} \ln \frac{W_{init}}{W_{final}}$$

-->Breguet Range Equation

Some Notes on Breguet Range Eqn

$$SFC \equiv \text{Specific Fuel Consumption}$$

For propeller-driven aircraft,

$$SFC = \frac{\text{mass of fuel}}{(\text{power})(\text{time})}$$

For jet aircraft,

$$SFC = \frac{\text{mass of fuel}}{(\text{thrust})(\text{time})} = \frac{\dot{m}}{T}$$

So $SFC = \dfrac{1}{v_e}$ (for suitable definitions of v_e)

Specific Fuel Consumption

Engine	SFC $\frac{lb(fuel)}{hr - lb(thrust)}$	SFC $\frac{kg(fuel)}{sec - N(thrust)}$	$v_e(effective)$ m/sec
CF-6 (747)	0.605	17.1×10^{-6}	58,400
J-58 (SR-71)	1.9	54×10^{-6}	19,000
SSME	7.95	225×10^{-6}	4440

Breguet Endurance Equations

For propeller-driven aircraft,

$$E = \frac{\eta}{SFC}\frac{c_L^{3/2}}{c_D}\sqrt{2\rho S}\left(\frac{1}{\sqrt{m_f}} - \frac{1}{\sqrt{m_o}}\right)$$

For jet aircraft,

$$E = \frac{1}{SFC}\frac{c_L}{c_D}\ln\frac{m_o}{m_f}$$