University of Liège
Aerospace & Mechanical Engineering

Aircraft Design
Introduction to Aircraft Structures

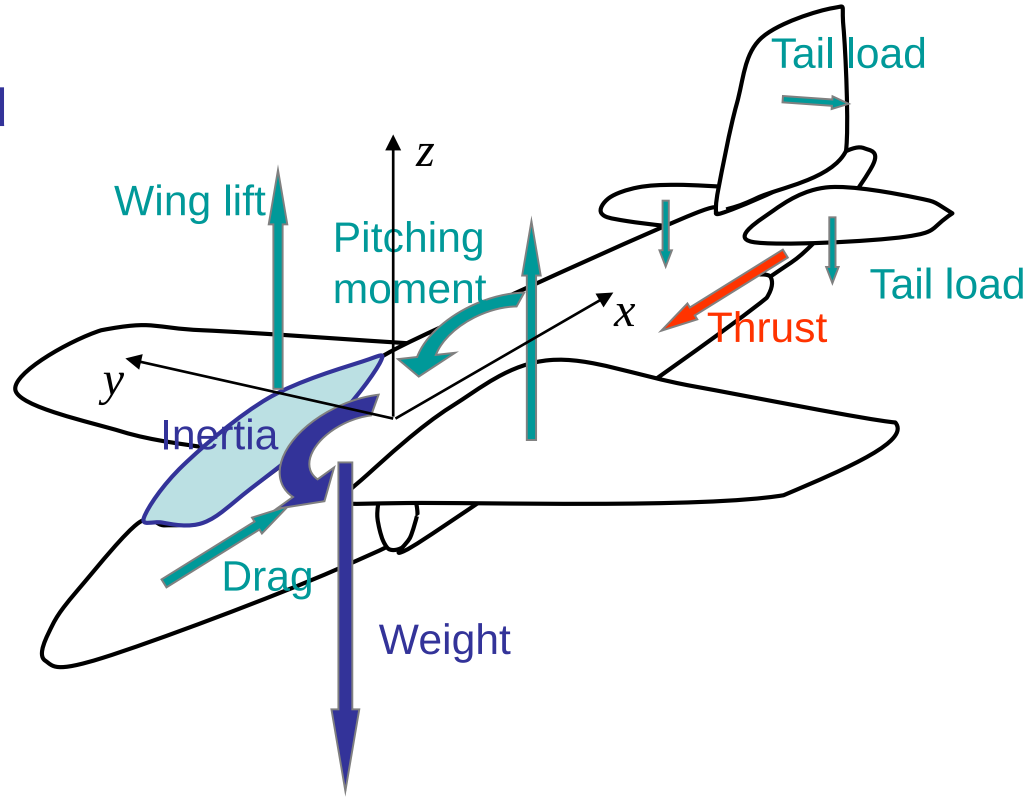
Ludovic Noels

Computational & Multiscale Mechanics of Materials – CM3
http://www.ltas-cm3.ulg.ac.be/
Chemin des Chevreuils 1, B4000 Liège
L.Noels@ulg.ac.be

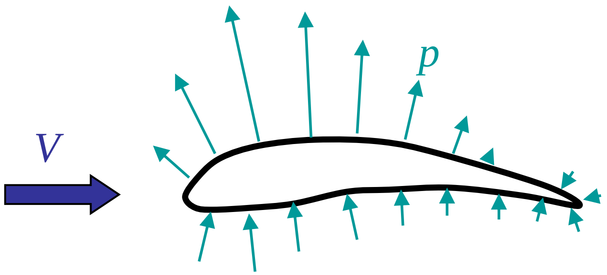
Loading

- Primary purpose of the structure
  - To transmit and resist the applied loads
  - To provide an aerodynamic shape
  - To protect passengers, payload, systems

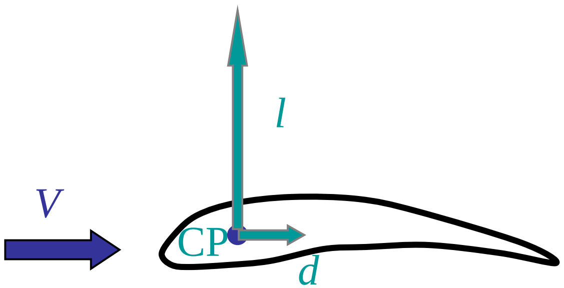
- The structure has to withstand
  - Aerodynamic loadings
  - Thrust
  - Weight and inertial loadings
  - Pressurization cycle
  - Shocks at landing, …



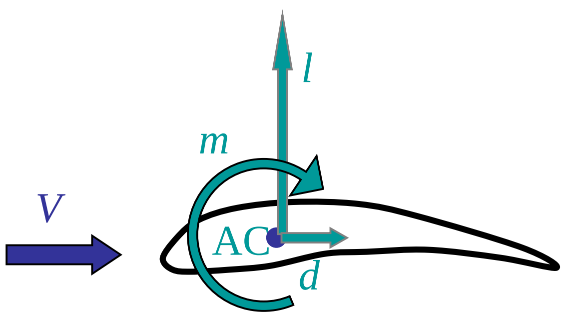
- Example: wing loading
  - Pressure distribution on an airfoil
    - Results from angle of attack and/or camber



  - This distribution can be modeled by
    - A lift (per unit length)
    - A drag (per unit length)
    - Applied at the Center of Pressure (CP)

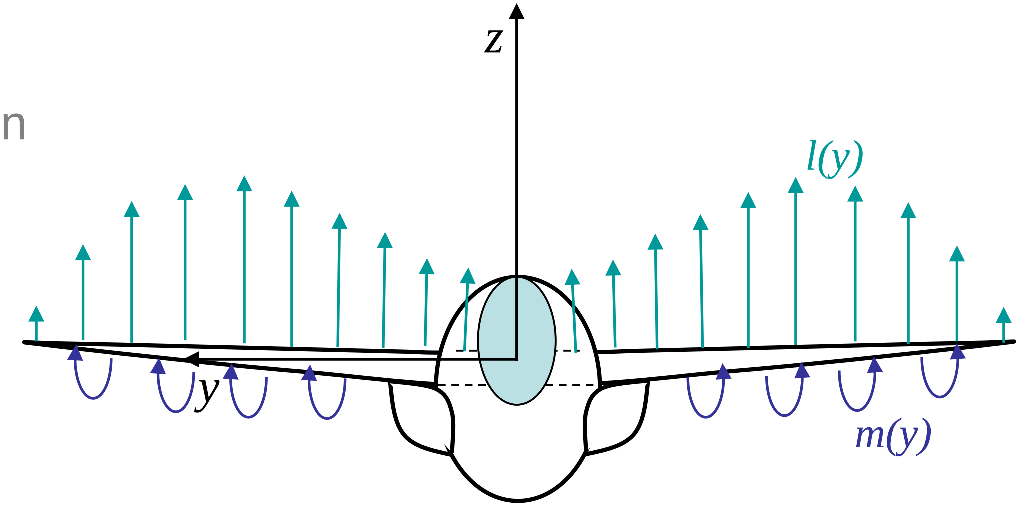


  - As the CP moves with the angle of attack, this is more conveniently modeled by
    - Lift and drag
    - A constant moment
    - Applied at the fixed Aerodynamic Center (AC)
      - Can actually move due to compression effects

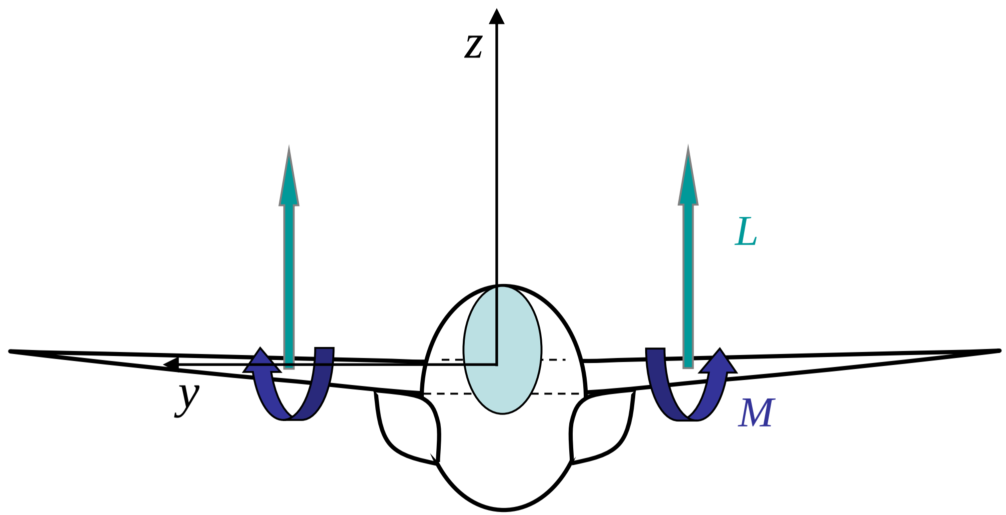


  - As the structural axis is not always at the CP
    - There is a torsion of the wing (particularly when ailerons are actuated)
  - There is always flexion

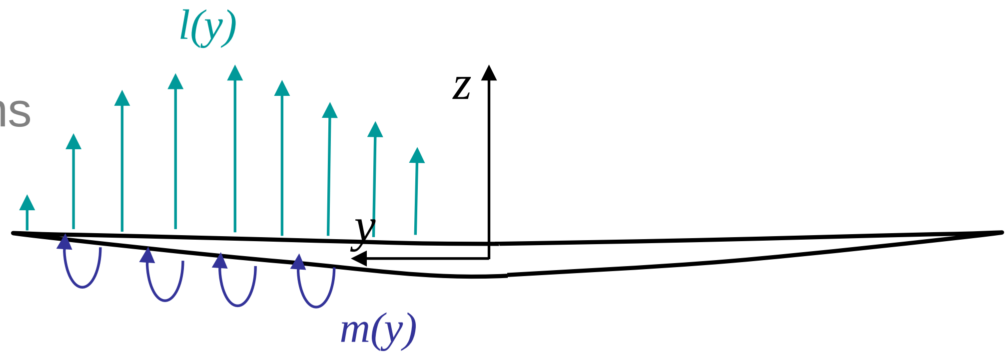
- Example: wing loading (2)
  - The lift distribution depends on
    - Sweep angle
    - Taper ratio
    - …

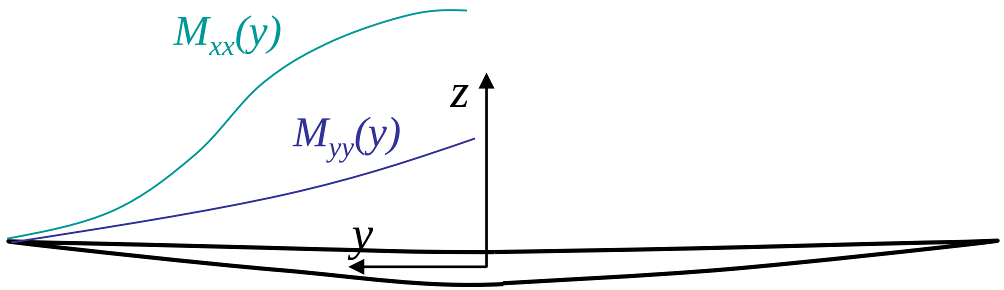


  - Load can be modeled by
    - Lift and moment
    - Applied on the aerodynamic center

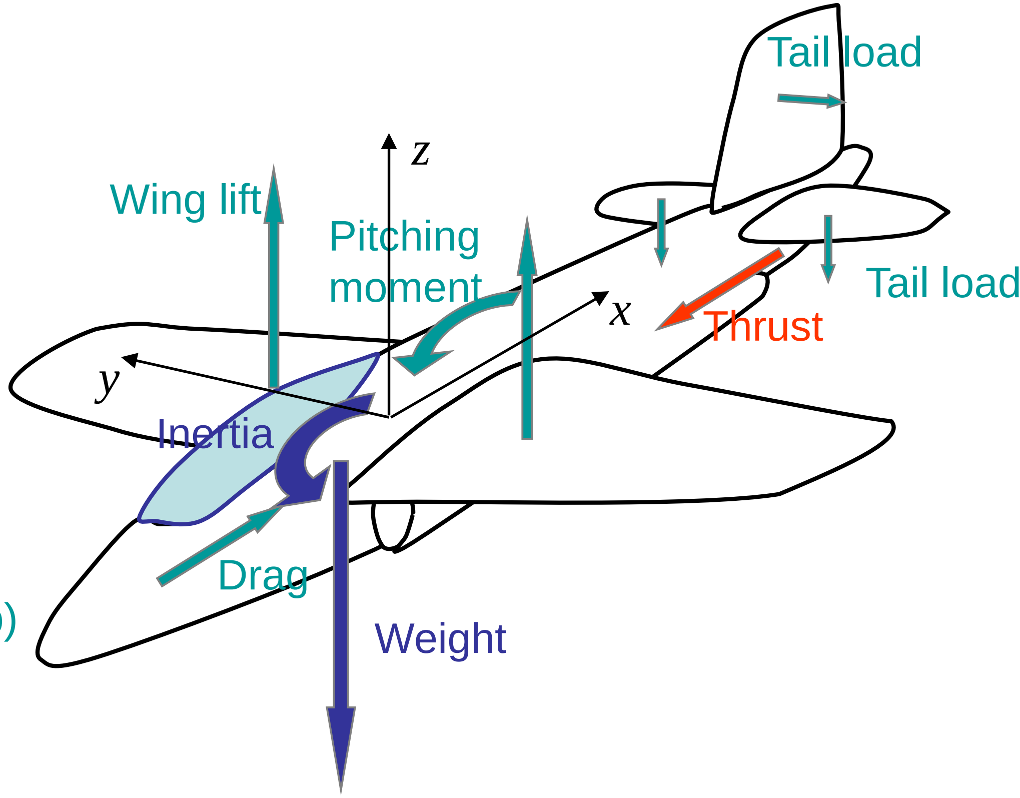


- Example: wing loading (3)
  - The lift and moment distributions result into
    - A bending moment
      - Due to *l(y)*
    - A torsion
      - Due to *m(y)*
      - Due to the fact that *l(y)* is not applied on the structural axis
    - Which depend on
      - Velocity
      - Altitude
      - Maneuver
      - Surface control actuation
      - Configuration (flaps down or up)
      - Gust
      - Take off weight

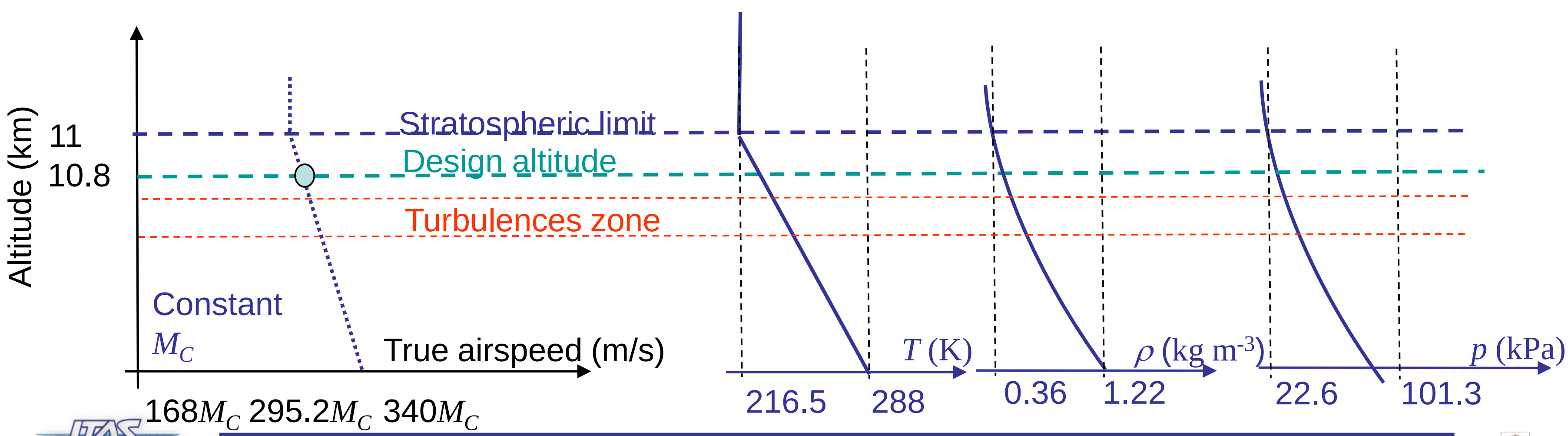




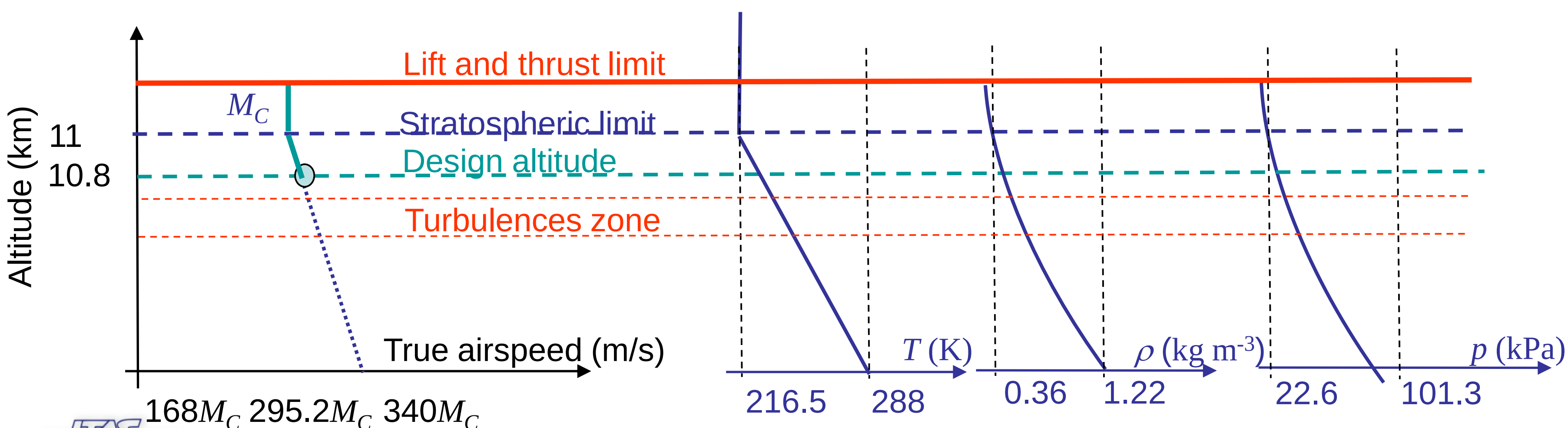
- Load intensity
  - Global loading can be represented by the load factor n (in g-unit)
    - n corresponds to the ratio between
      - The resulting aerodynamic loads perpendicular to the aircraft x-axis
      - The weight
    - When flying: $n \sim L / W$
    - Steady flight: $n = 1$
    - Pullout: n > 1
  - Loading factor depends on
    - Velocity
    - Altitude
    - Maneuver
    - Surface control actuation
    - Configuration (flaps down or up)
    - Gust
    - Take off weight



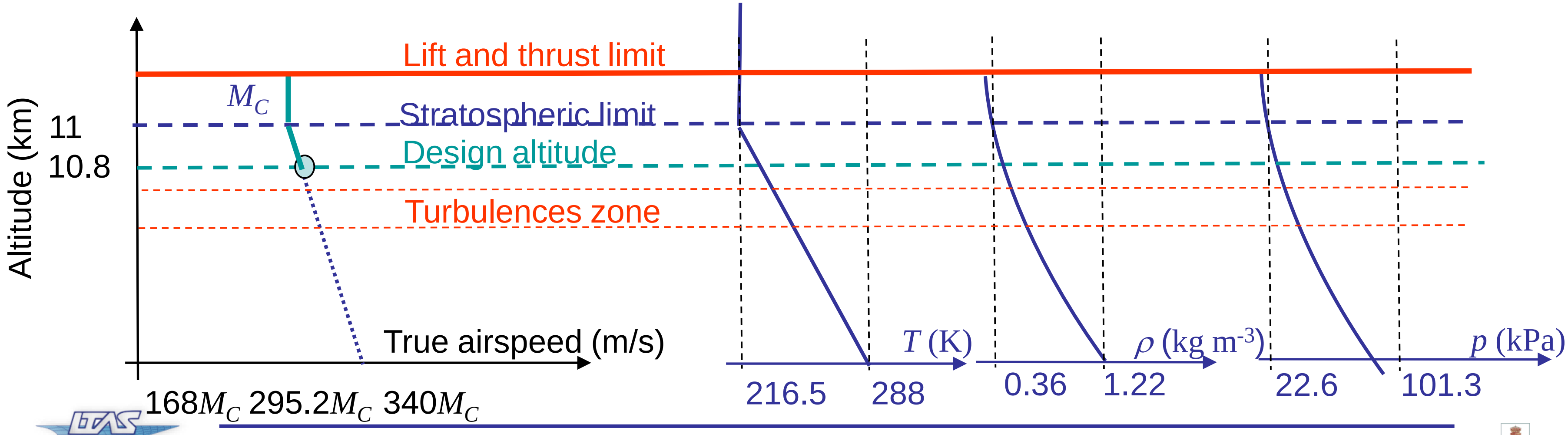
- Placard diagram (Altitude-Velocity dependency)
  - Design altitude
    - High enough to reduce drag (as density decreases with the altitude)
    - Above turbulence zone
  - Design cruise Mach (M_C)
    - Usually maximum operating Mach:
      Mach obtained at maximum engine thrust $\Longrightarrow M_C = M_{mo} \sim 1.06\, M_{cruise}$
    - Temperature evolves linearly with altitude until the stratosphere



- Placard diagram (2)
  - Above the design altitude
    - Although density is reduced, the compressibility effects do not allow flying at higher Mach
    - The plane will fly at the same M_C number
  - Ceiling
    - At high altitude the density is too small
      - The wing cannot produce the required lift
      - The engines cannot produce the required thrust

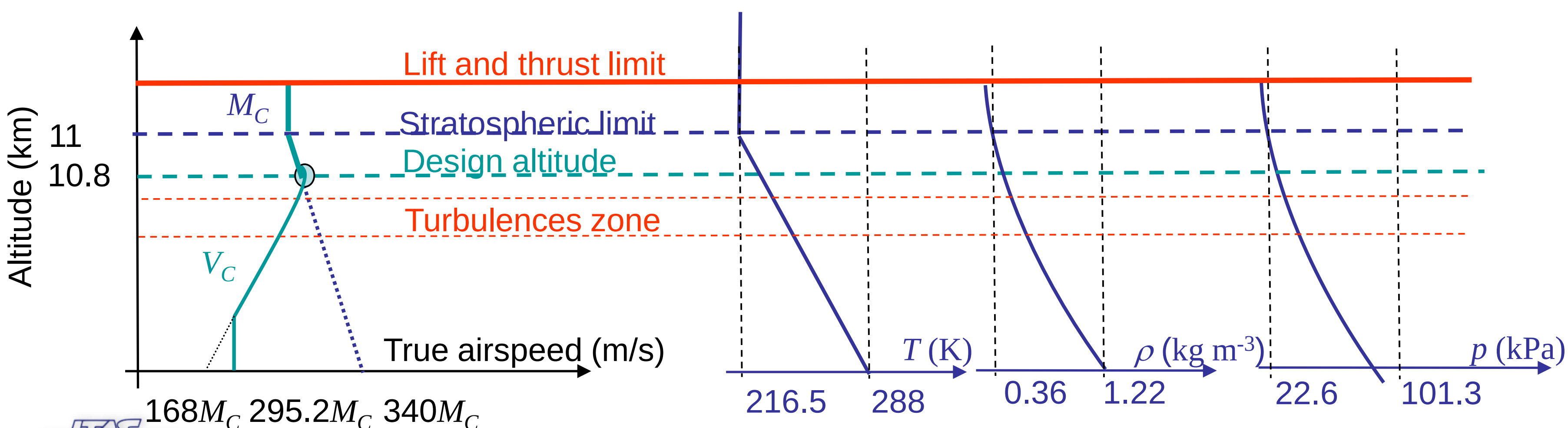


- Placard diagram (3)
  - 1957, Lockheed U2
    - Ceiling 21 km (70000 ft)
    - Only one engine
    - AR ~ 10
    - Stall speed close to maximum speed

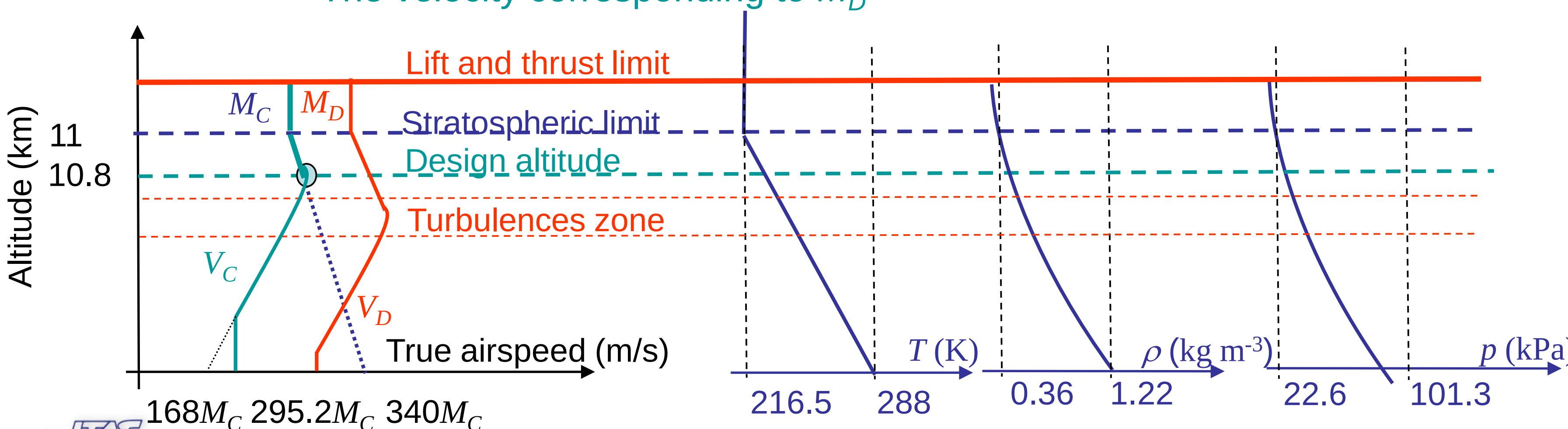


- Placard diagram (4)
  - Below design altitude, when getting closer to the sea level
    - Density increases
      - Engines cannot deliver enough thrust to maintain M_C (drag increases with ρ)
      - Drag has to be kept constant

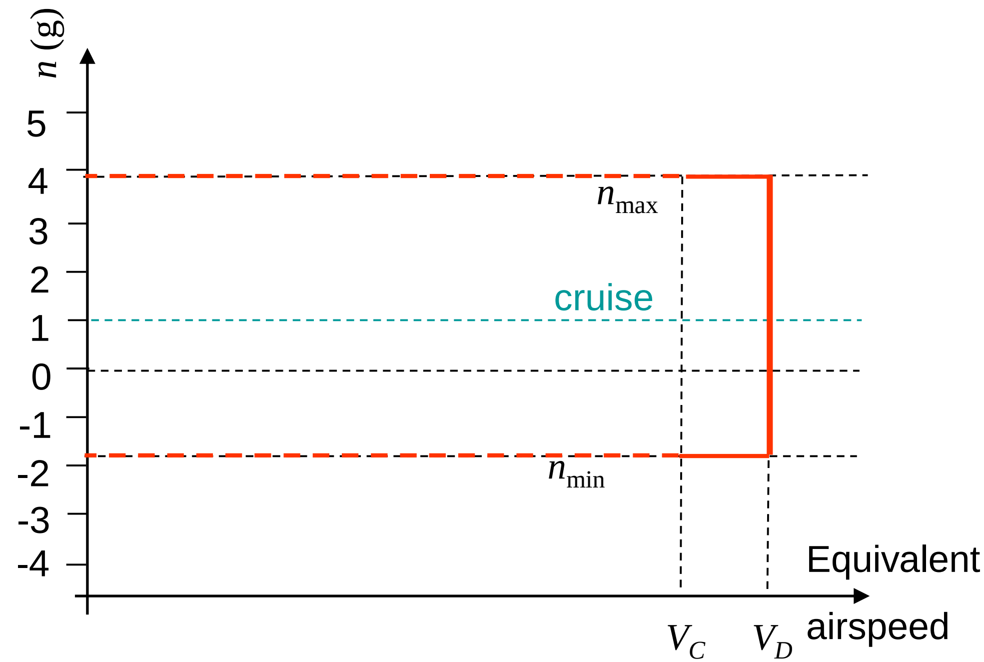
        ⟹ $\rho V_{\text{True}}{}^2/2$ constant (V_{True} is the true airspeed)
      - From the dynamical pressure $\rho V_{\text{True}}{}^2/2$, the equivalent velocity at sea level can be deduced: $V_e = V_{\text{True}} (\rho/\rho_0)^{1/2}$ (ρ_0 = density at sea level)
    - Equivalent velocity is constant ⟹ true airspeed is decreasing
      - There can be an operational limit as take off speed



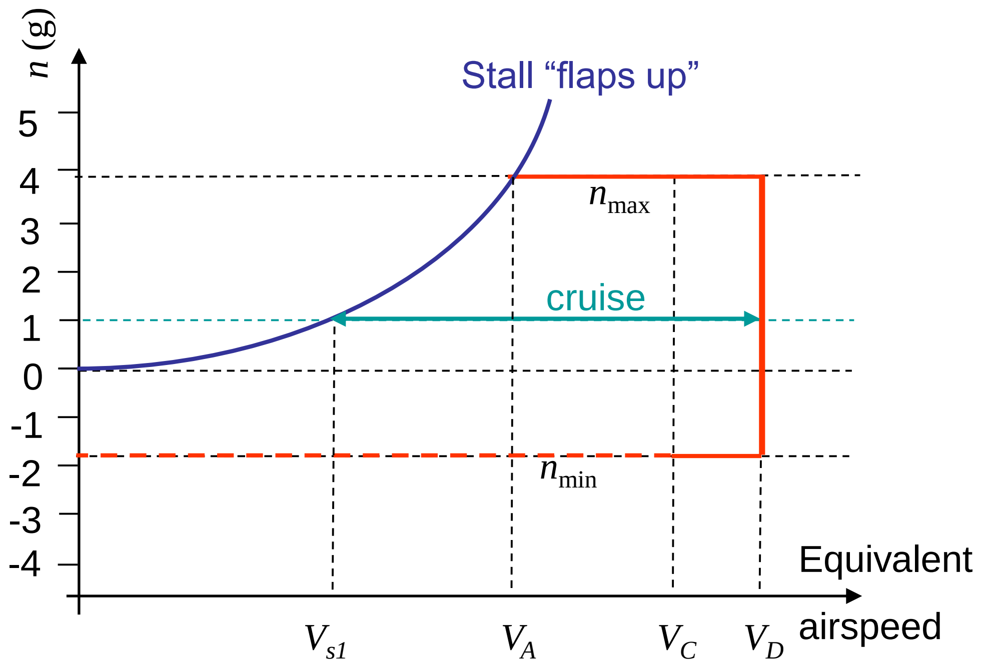
- Placard diagram (5)
  - Maximum velocity?
  - During a dive the plane can go faster than the design mach cruise
    - Design dive Mach (FAR) is defined as the minimum between
      - 1.25 M_C
      - Mach actually obtained after a 20-second dive at 7.5° followed by a 1.5-g pullout ⟹ M_D ~1.07 M_C
    - Above design altitude the maximum velocity is limited by M_D constant
    - Below design altitude the maximum dive velocity V_D is the minimum of
      - 1.25 V_C
      - The dive velocity (20-second dive at …) ~ 1.15 V_C
      - The velocity corresponding to M_D



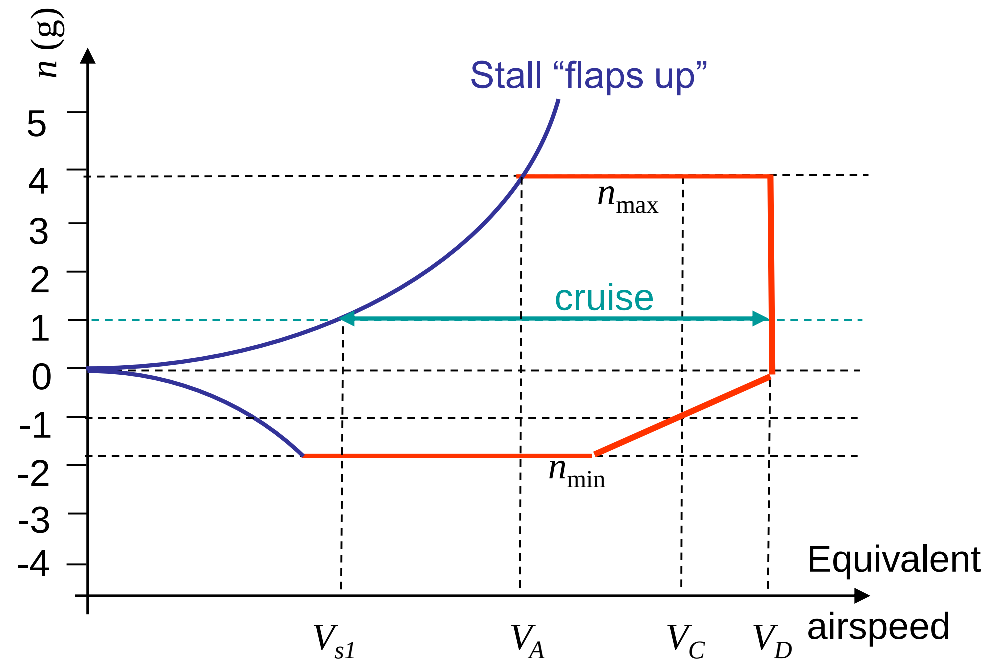
- Maneuver envelope (Velocity-load factor dependency)
  - Extreme load factors
    - Light airplanes (W < 50000 lb)
      - From -1.8 to minimum of
        - » 2.1 + 24000 lb/(W [lb] + 10000 lb)
        - » 3.8
    - Airliners (W > 50000 lb)
      - From -1 to 2.5
    - Acrobatic airplanes
      - From -3 to 6
  - Two design velocities
    - These are equivalent velocities
    - Design dive velocity V_D
      - The plane cannot fly faster
    - Design cruise velocity V_C
  - Are these load limits relevant if the plane fly slower than V_C ?



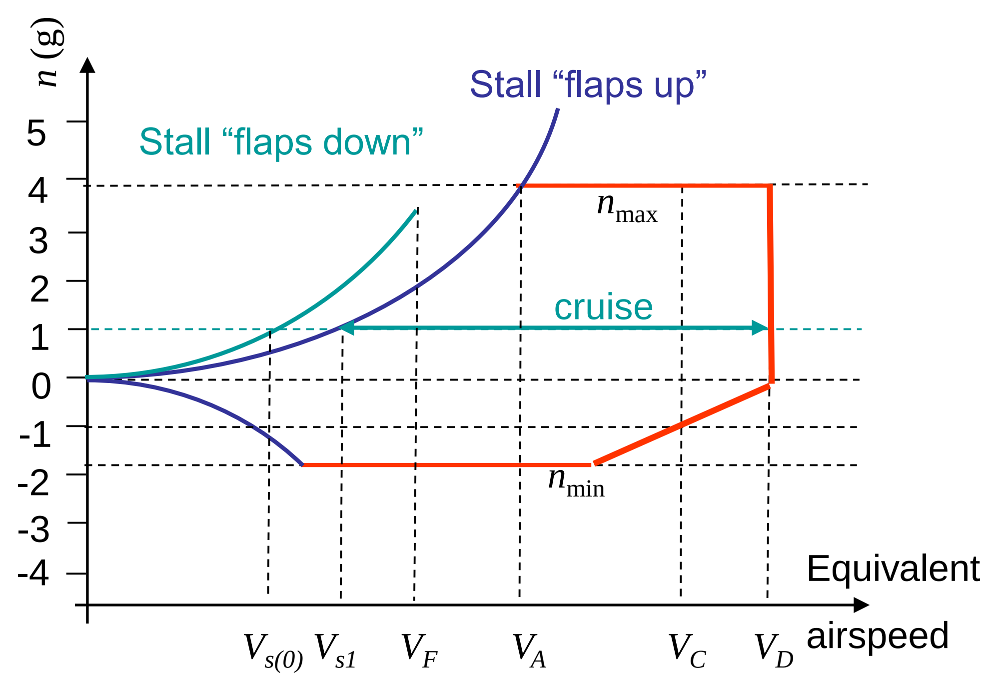
- Maneuver envelope (2)
  - At velocity lower than design cruise V_C
    - A pullout is limited by the maximum lift the plane can withstand before stalling
      - In terms of equivalent velocity and maximum lift coefficient flaps up, the maximum load factor becomes: $n = \frac{L}{W} = \frac{\rho_0 V_e^2 S C_{L\,\max,1}}{2W}$
      - V_A: Intersection between stall line and $n_{\max}$
        - » This is the maximum velocity at which maximum deflection of controls is authorized
      - V_{s1}: Intersection between stall line and $n = 1$
        - » This is the stall velocity in cruise (flaps up)
      - FAR requirement
        - » $V_A > V_{s1}\, n^{1/2}$ but
        - » V_A can be limited to V_C



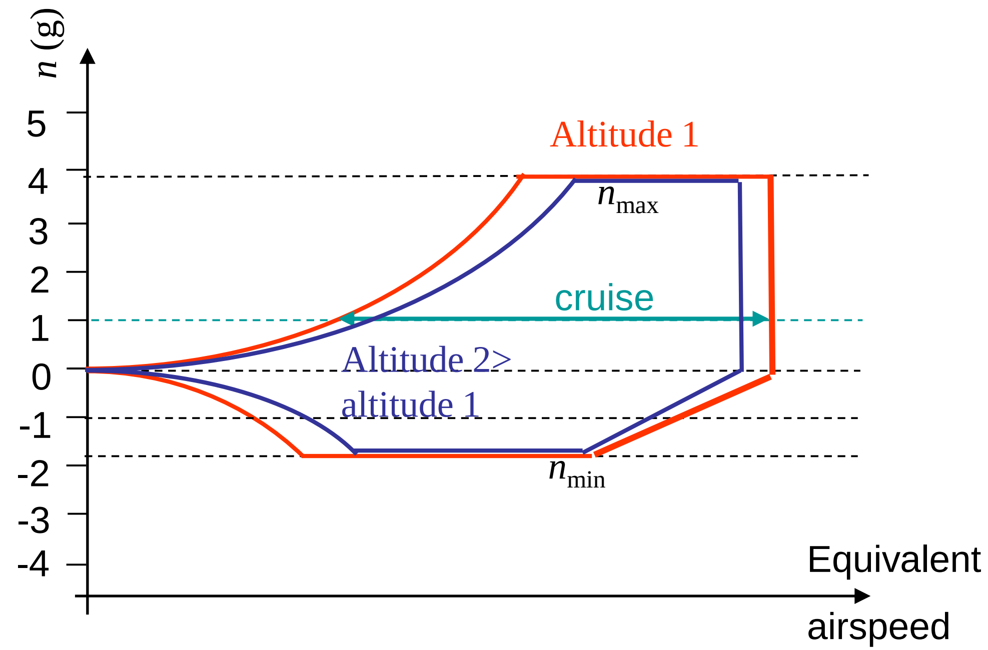
- Maneuver envelope (3)
  - Negative load factor
    - At low velocities
      - Same thing than for pullout: stall limits the load factor
    - At high velocities
      - When diving only a pullout is meaningful
      - Linear interpolation between
        - » $V_e=V_D$ & n=0
        - » $V_e=V_C$ & n=-1



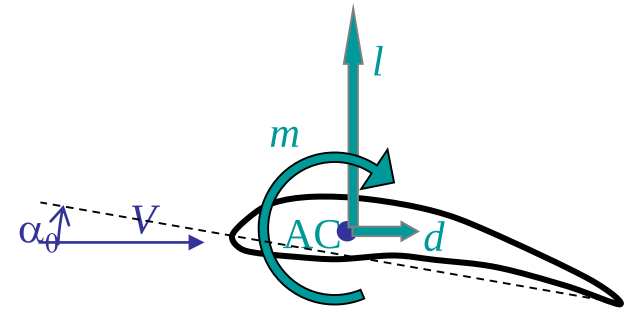
- Maneuver envelope (4)
  - Configuration flaps down
    - The maximum lift coefficient changes, so the load factor
      - Landing configuration $n = \frac{L}{W} = \frac{\rho_0 V_e^2 S C_{L\max,0}}{2W}$
      - Takeoff configuration $n = \frac{L}{W} = \frac{\rho_0 V_e^2 S C_{L\max}}{2W}$
    - Stall velocities
      - V_s: take off
      - V_{s0}: landing
      - V_{s1}: flaps up
    - V_F: velocity below which the flaps can be down (structural limit)
    - FAR requirements
      - $V_F > 1.6\ V_{s1}$ in take off configuration (MTOW)
      - $V_F > 1.8\ V_{s1}$ in approach configuration (weight)
      - $V_F > 1.8\ V_{s0}$ at landing configuration (weight)



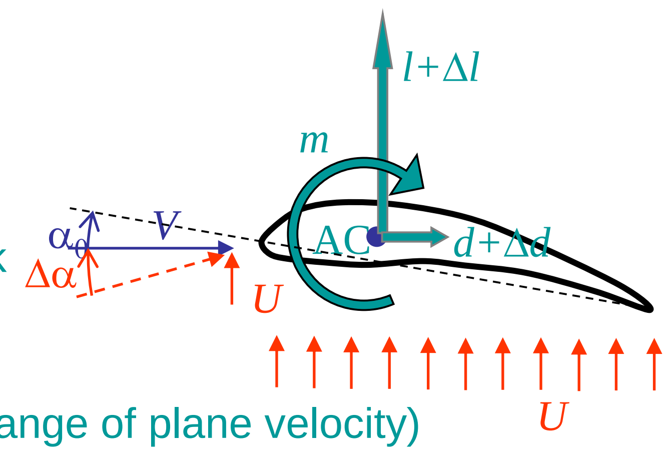
- Maneuver envelope (5)
  - Altitude dependency
    - Use of equivalent velocity reduces the effect of altitude
    - But the envelope still depends on the altitude
      - With the altitude the speed of sounds decreases and density is reduced
        - » For a given equivalent velocity the compressibility effects are higher (higher Mach number) and the maximum lift coefficient decreases
      - The computed V_D will be lower as limited by M_D constant
    - One flight envelope is therefore valid for an altitude range
    - Another factor which is altitude-dependant, and that should also be considered, is the gust factor



- Gust effect
  - Airfoil in still air
    - Airplane velocity V
    - Attack angle α_0



  - Sudden vertical gust U
    - The plane keeps temporarily the same
      - Velocity V
      - Attitude α_0
    - Due to the vertical velocity the angle of attack becomes $\alpha = \alpha_0 + \Delta\alpha \simeq \alpha_0 + \frac{U}{V}$

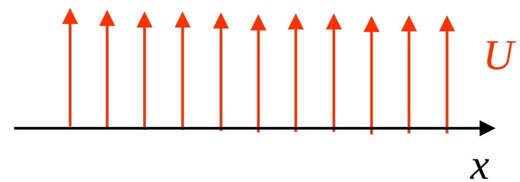


    - Resulting increase of plane lift (neglecting change of plane velocity)

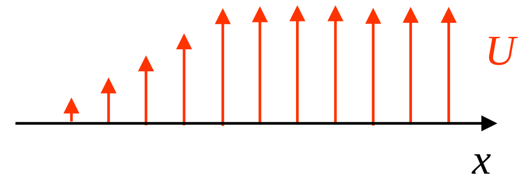
$$\Delta L \simeq \frac{\rho V^2 S \partial_\alpha C_{L\text{plane}} \Delta\alpha}{2} \simeq \frac{\rho V S C_{L\alpha\text{plane}} U}{2}$$

  - Increase in load factor
    - As $\rho U V = \rho_0 U_e V_e$ $\Longrightarrow$ $\Delta n \simeq \frac{\rho_0 V_e S C_{L\alpha\text{plane}} U_e}{2W}$

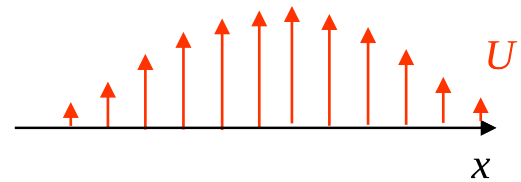
- Gust effect (2)
  - Realistic vertical gust
    - The plane do not really see a sudden vertical gust



    - A real vertical gust can be modeled as graded
      - Ramp



      - Cosine



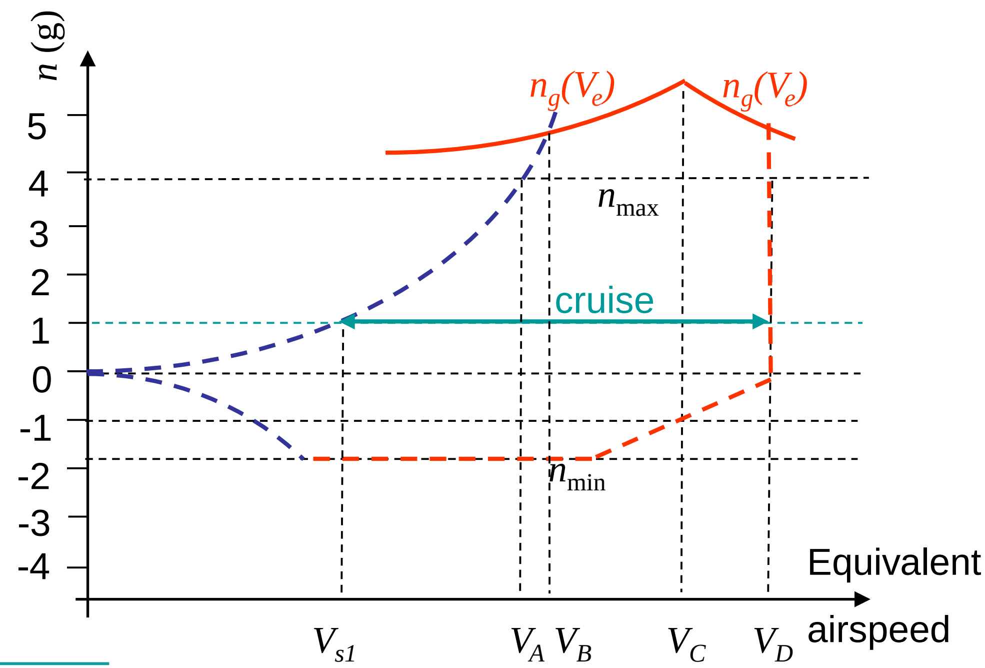
    - Modern methods consider power spectrum analysis
  - Gust alleviation factor: Before gust has reached its maximum value
    - The aircraft has developed a vertical velocity ⟹ reduces the severity
    - The aircraft might be pitching ⟹ effect on the loading (increase of decrease)
    - Elastic deformations of the structure ⟹ might increase the severity
  - So $\Delta n \simeq \frac{\rho_0 V_e S C_{L\alpha\text{plane}} U_e}{2W}$ becomes $\Delta n \simeq \frac{\rho_0 V_e S F C_{L\alpha\text{plane}} U_e}{2W}$
    - F is the gust alleviation factor (<1)

- Gust alleviation factor
  - Expression $\Delta n \simeq \frac{\rho_0 V_e S F C_{L\alpha \text{plane}} U_e}{2W}$ is difficult to be evaluated
  - FAR simple rule $n_g = 1 + \frac{F C_{L\alpha \text{plane}} U_e V_e S}{498 W}$
    - W plane weight in lb
    - V_e equivalent plane velocity in knots (1 knots = 1.852 km /h)
    - Gust alleviation factor $F = \frac{0.88\mu}{5.3 + \mu}$
    - Airplane weigh ratio $\mu = \frac{2W}{\rho C_{L\alpha \text{plane}} c g S}$
    - c mean aerodynamic chord
    - U_e equivalent gust velocity in ft/s
      - Is interpolated from statistical values at different altitudes and for different planes velocities
      - V_B: Velocity when maximum load factor is governed by gust (see next slide)

U_e in ft/s	$V_e = V_B$	$V_e = V_C$	$V_e = V_D$
Sea level	± 56	± 56	± 28
15000 ft	± 44	± 44	± 22
60000 ft	±20.86	±20.86	±10.43

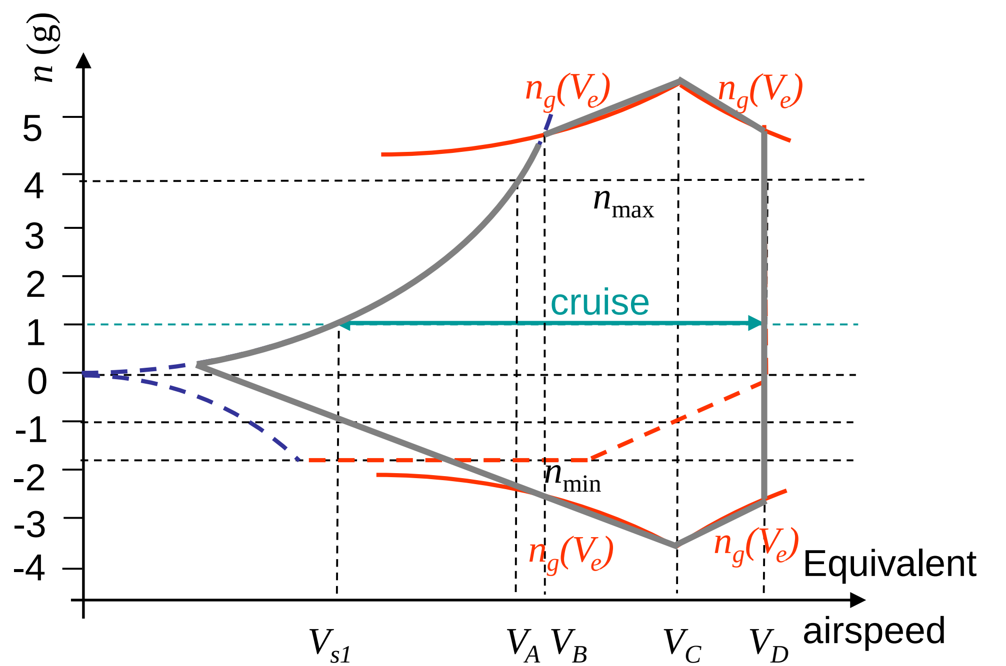
- Gust envelope
  - Gust load factor
    - $n_g = 1 + \frac{F C_{L\alpha\,\text{plane}} U_e V_e S}{498 W}$
    - This gives two branches for $n_g(V_e)$ for $U_e > 0$
    - V_B is the intersection between
      - The stall curve
      - $n_g(V_e)$
    - This means that if
      - $V_e < V_B$ the plane might stall in case of gust
      - So V_B is minimum speed to enter a gust region
    - FAR requirements
      - V_B can be $< V_{s1} [n_g(V_C)]^{1/2}$
      - $V_C > V_B + 1.32 U_e$
      - $V_B > V_{s_1} \sqrt{1 + \frac{\rho_0 V_C S F C_{L_{\alpha_{plane}}}}{2W} U_e}$

U_e in ft/s	$V_e = V_B$	$V_e = V_C$	$V_e = V_D$
Sea level	± 56	± 56	± 28
15000 ft	± 44	± 44	± 22
60000 ft	±20.86	±20.86	±10.43

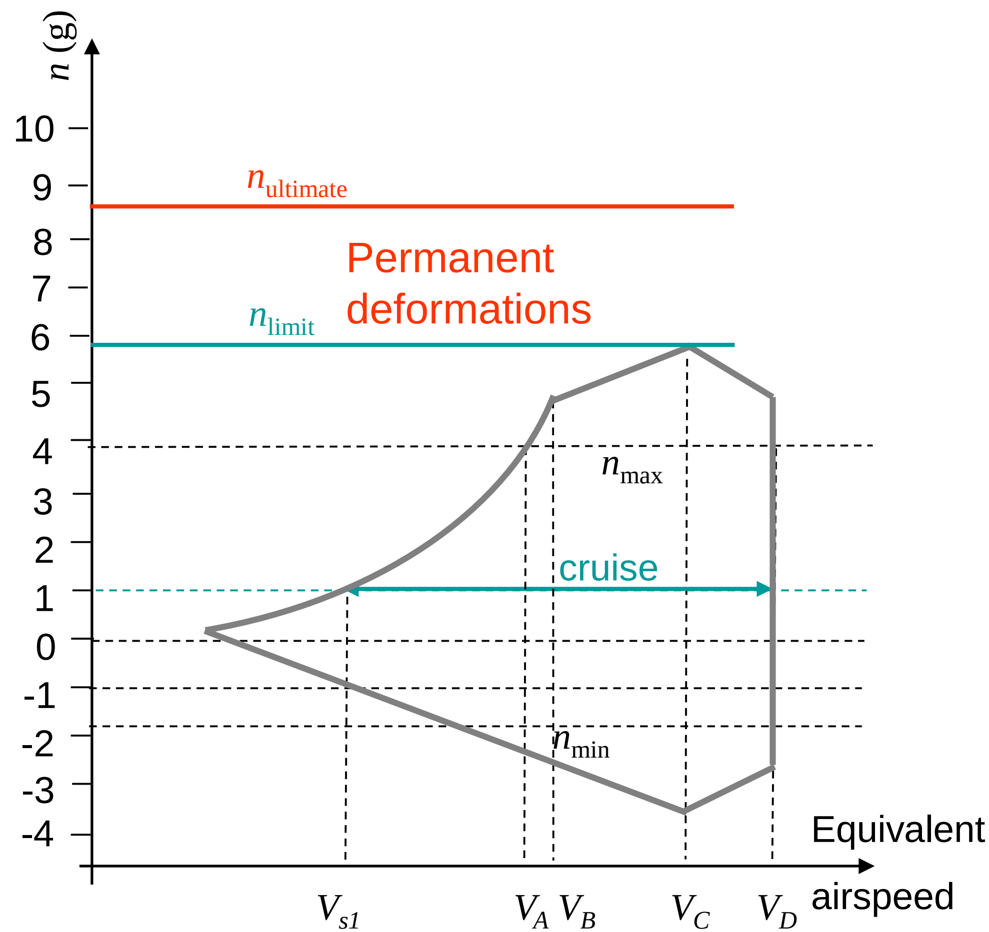


- Gust envelope (2)
  - Gust load factor
    - $n_g = 1 + \frac{F C_{L\alpha\,\text{plane}} U_e V_e S}{498 W}$
    - This gives two branches for $n_g(V_e)$ for $U_e < 0$
  - Gust envelope is the linear interpolation between
    - Positive stall
    - $n_g(V_B)$
    - $n_g(V_C)$
    - $n_g(V_D)$

U_e in ft/s	$V_e = V_B$	$V_e = V_C$	$V_e = V_D$
Sea level	± 56	± 56	± 28
15000 ft	± 44	± 44	± 22
60000 ft	±20.86	±20.86	±10.43



- Design load factors
  - Limit load factor n_{limit}
    - Maximum expected load during service (from gust envelope)
    - The plane cannot experience permanent deformations
  - Ultimate load factor n_{ultimate}
    - Limit load times a security factor (1.5)
    - The plane can experience permanent deformations
    - The structure must be able to withstand the ultimate load for 3 seconds without failure



- First structure designs
  - A wood internal structure smoothed by fabrics
  - A plywood structure was also used for the fuselage

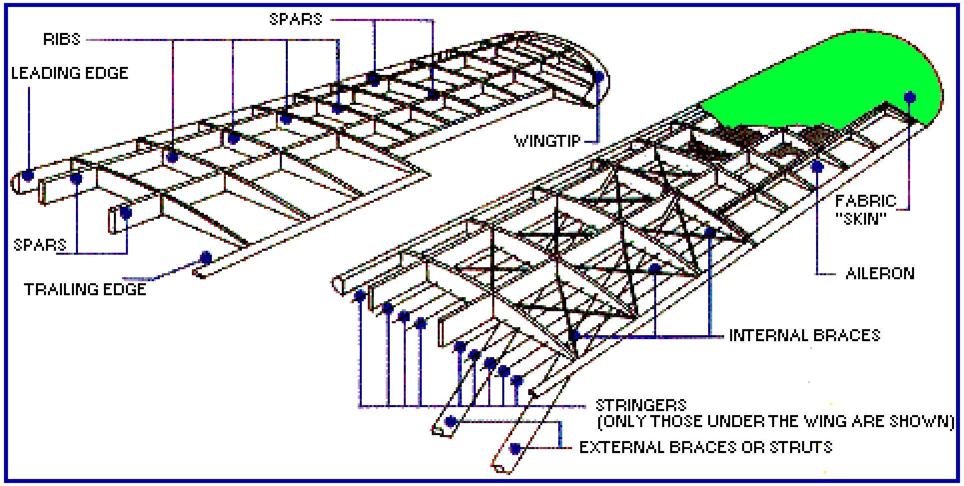


Figure 1-5 Wood-and-fabric-type wing structure

- Was wood a good choice?
  - Specific mechanical properties of wood are favorable to aluminum alloy

	Yield or tensile strength* [MPa]	Young [MPa]	Density $[kg \cdot m^{-3}]$	Ratio Young-Density	Ratio Strength-Density
Wood	100*	14000	640	21.9	0.156
Structural steel	200	210000	7800	26.9	0.025
Aluminum	75	70000	2700	8.9	0.027
High strength steel alloy A514	690	210000	7800	26.9	0.088
Aluminum alloy 2014	400	73000	2700	9.3	0.148
Titanium alloy 6Al-4V	830	118000	4510	26.17	0.184
Carbon fiber reinforced plastic	1400* (theoretical)	130000	1800	72.2	0.777

- Was wood a good choice (2)?
  - Drawbacks of wood
    - Moisture absorption changed shape and dimensions
    - Glued structures affected by humidity
    - Strongly anisotropic
    - Oversee import
    - Not suited to stress concentration
  - Wood-fabric structures
    - Were not always waterproof
      - Picture Fokker Dr.I
    - Did not allow to build high-aspect ratio wing
      - Most of the planes were biplanes or triplanes with lower lift/drag ratio

Photo Courtesy Hans Franke

- Was wood a good choice (3)?
  - Nowadays, only light aircrafts are built using this concept (ex: Mudry)

  - In 1915, Junkers constructed a steel plane
    - Cantilevered wing
    - Steel is too heavy (specific tensile strength too low)

- Duralumin
  - 1909, Alfred Wilm, Germany
    - An aluminum alloy containing
      - 3.5 per cent copper
      - 0.5 per cent magnesium
      - Silicon and iron as impurities

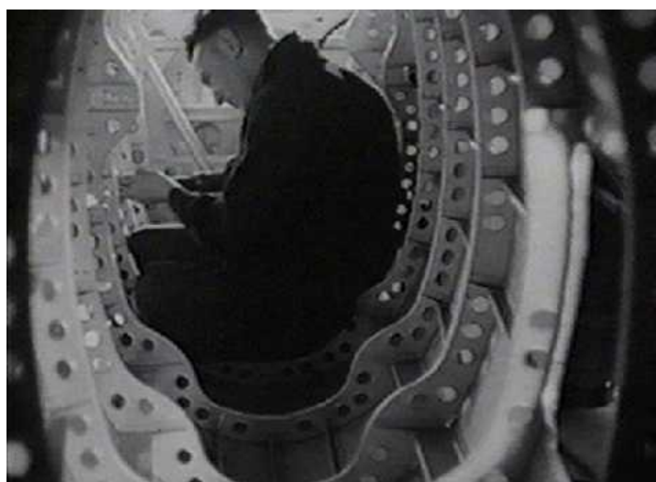
    spontaneously hardened after quenching from about 480°C.
  - This alloy had interesting specific mechanical properties
    - Yield 230 Mpa but
    - Density only 2700 $\mathrm{kg \cdot m^{-3}}$
  - The question was
    - How to efficiently use this duralumin?

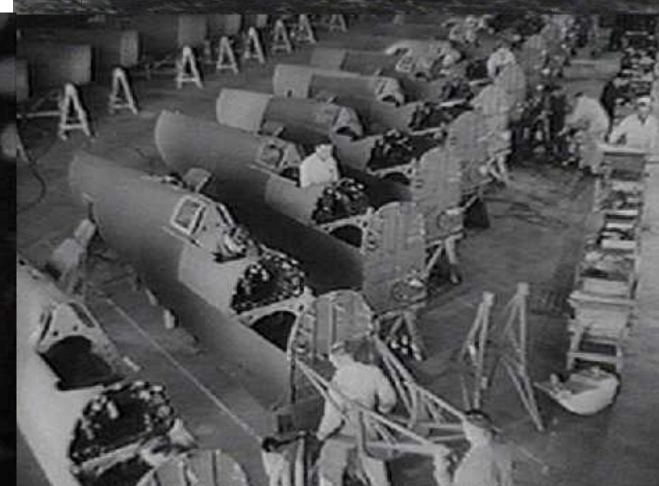
- Monocoque
  - Instead of
    - Using a frame as main structure and
    - Covering it with thin metal sheets
  - The skin of the structure can be such that it resists the load by itself
    - Lighter than framed structures
    - Sport cars (carbon fiber)
    - Soda can (aluminum)
      - As long as it is filled, it is resistant
      - Empty, it is subjected to buckling
  - These structures are subject to buckling and cannot be used for an aircraft



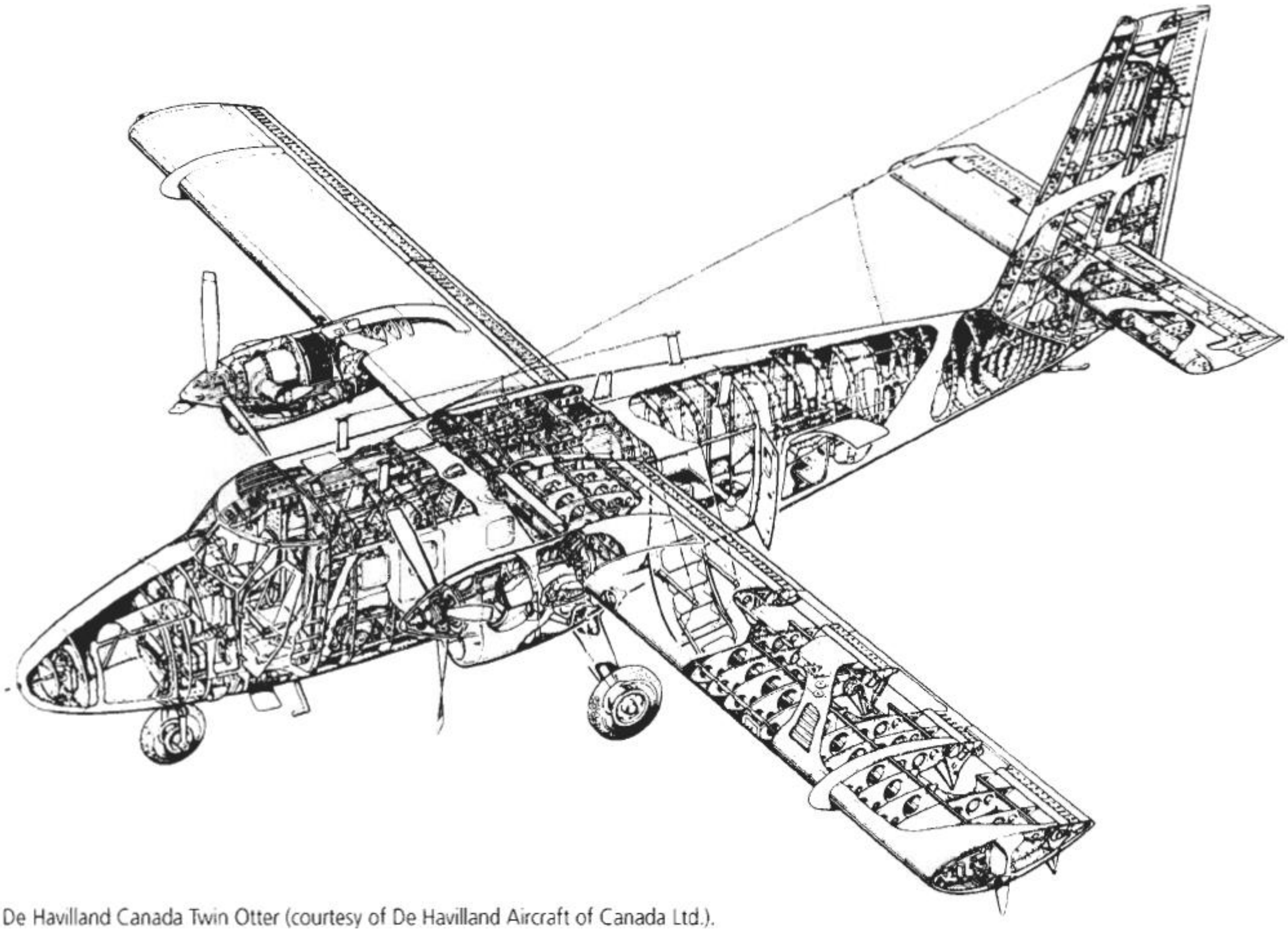
- Semi-monocoque
  - Monocoques are subject to buckling
  - The skin of the shell is usually supported by
    - Longitudinal stiffening members
    - Transverse frames

    to enable it to resist bending, compressive and torsional loads without buckling
  - These stiffeners are fixed to the skin instead of putting a skin on a structural frame
- First semi-monocoque aircrafts were made of duralumin (example: spitfire)





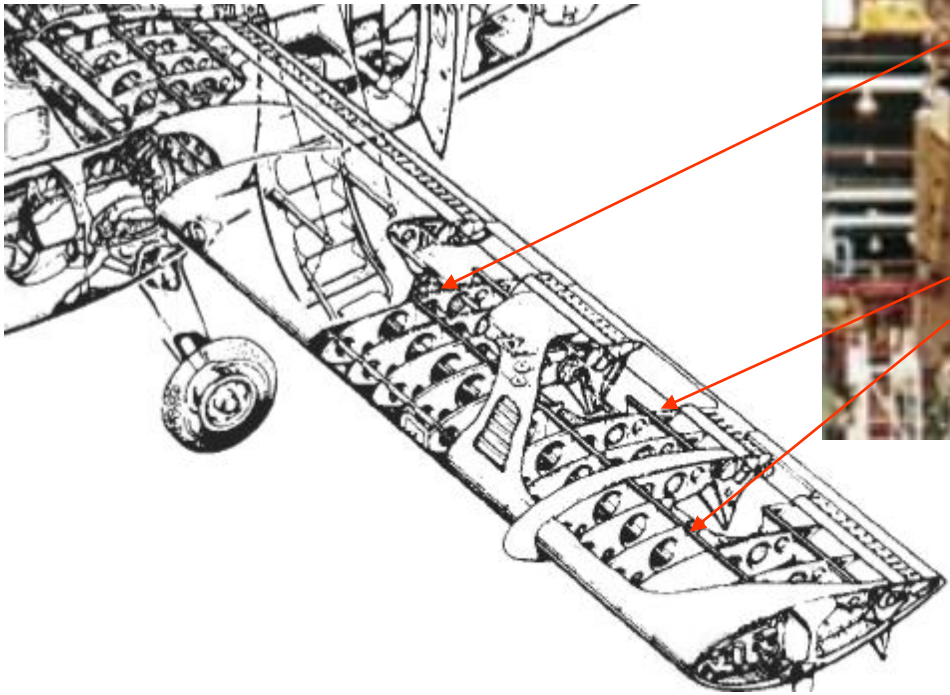
- Global view

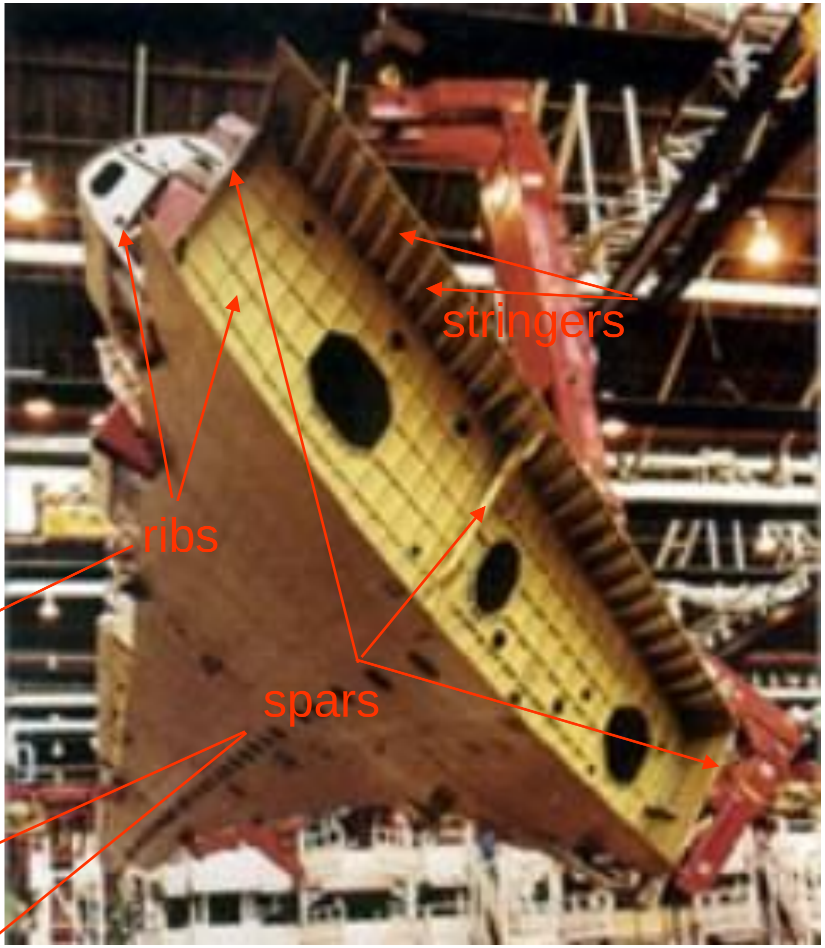


De Havilland Canada Twin Otter (courtesy of De Havilland Aircraft of Canada Ltd.).

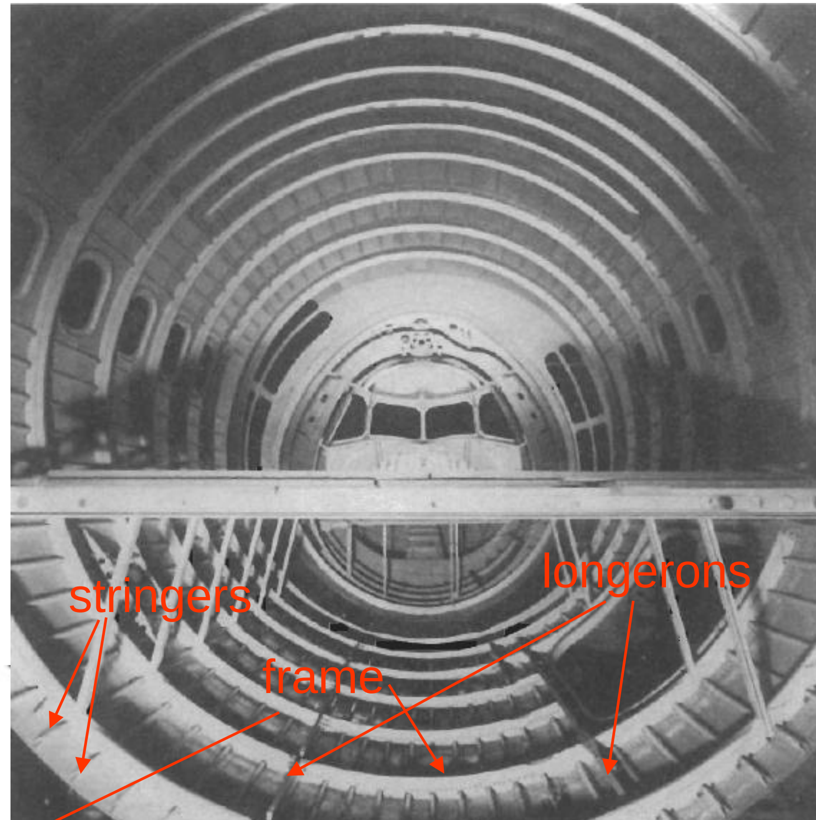
Semi-monocoque structure

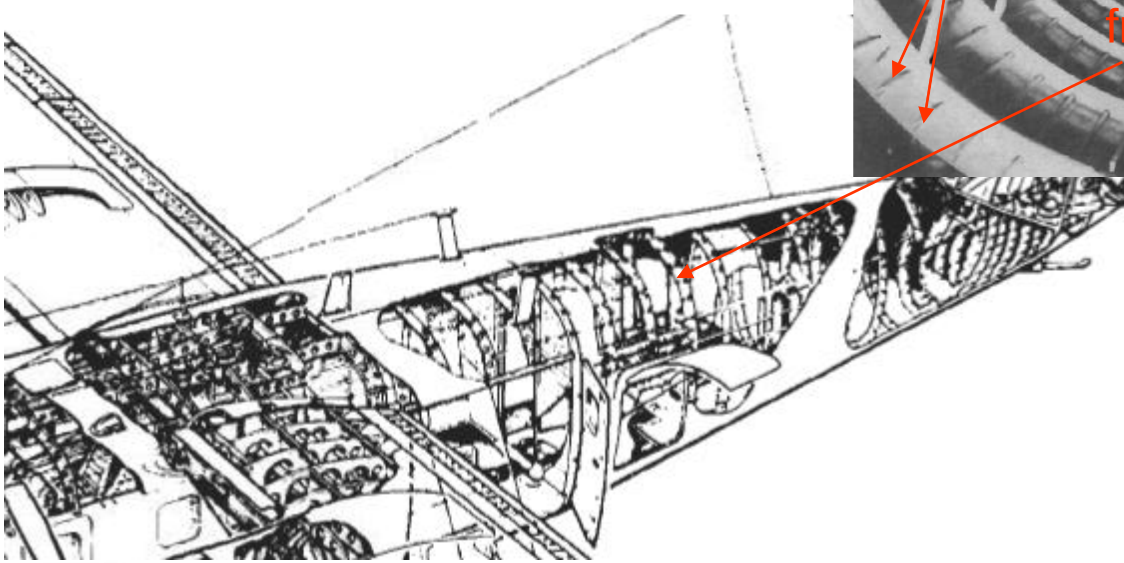
- Wing: Box-beam structure
  - 2 or 3 spars
  - Ribs
  - Stringers fixed to the skin
  - Transport aircrafts
    - Skin >~ 1. mm
    - Ribs >~ 0.5 mm
    - Spars >~ 1. mm



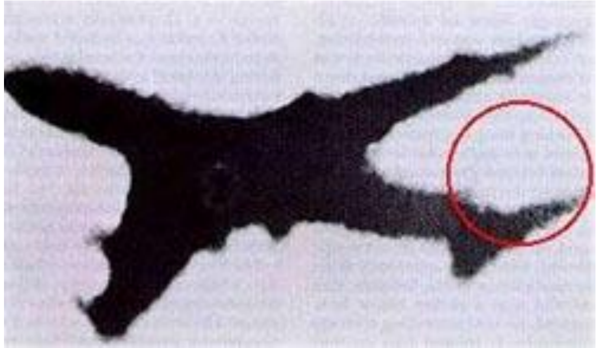


- Fuselage
  - Circular if pressurized
  - Longerons
  - Stringers
  - Frames or formers
  - Bulkheads (see next slide)





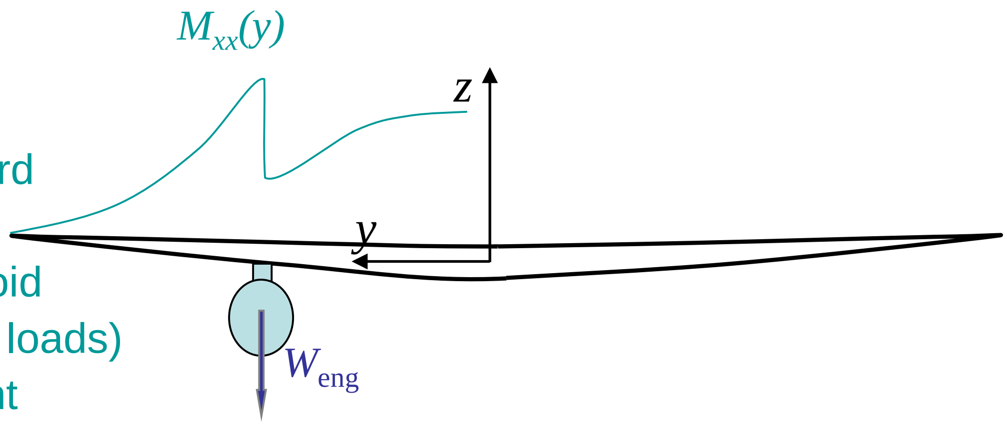
- Fuselage (2)
  - Circular if pressurized
  - Longerons
  - Stringers
  - Frames or formers
  - Bulkheads
    - Reinforcement at
      - Wing root
      - Empennage fixation
      - Engine fixation
      - …
    - Pressurization
      - Between cabin and tailfin
      - B747, Japan Airline 123: bulkhead repaired with a single row of rivets instead of two



Design criteria

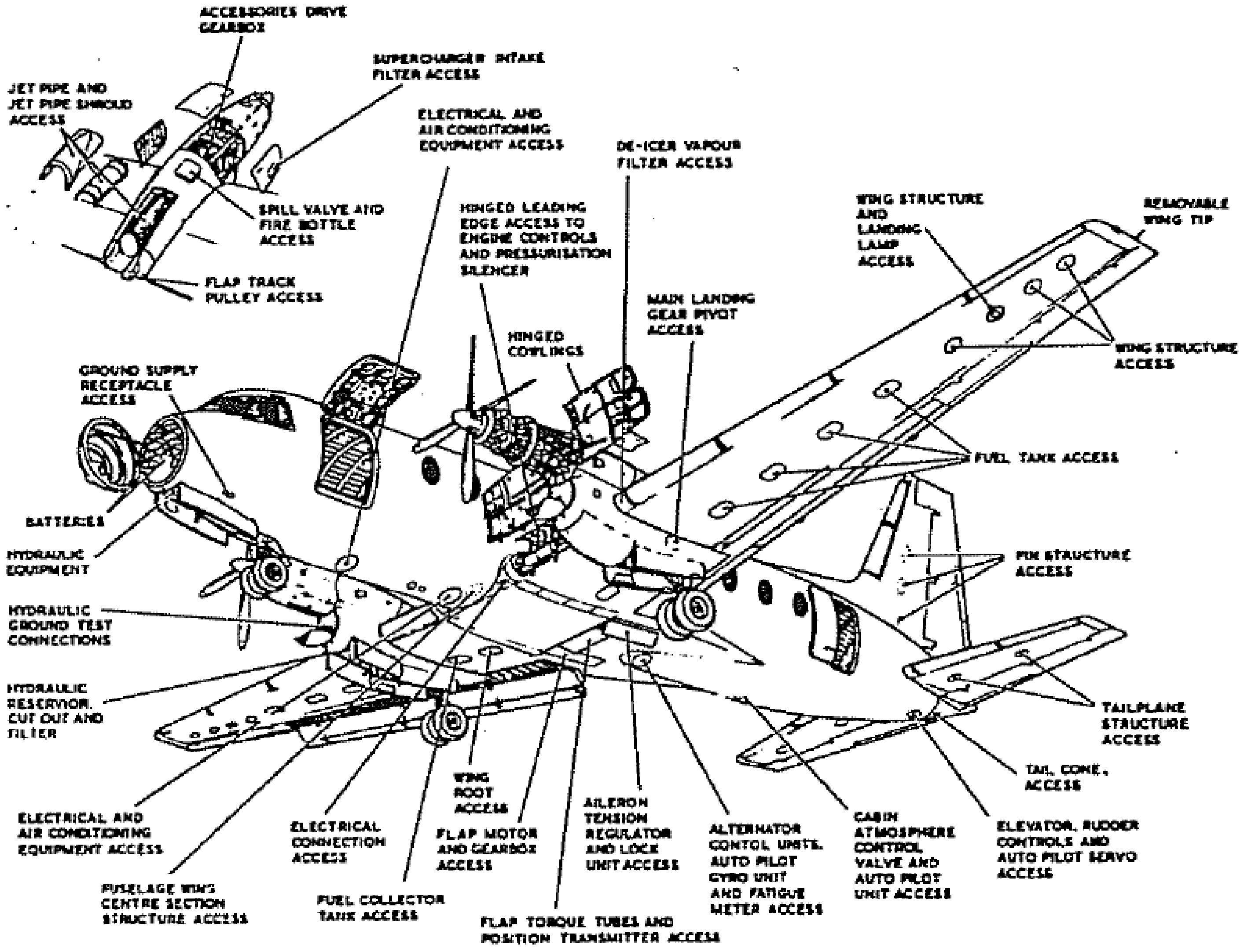
- Structural integrity of the airframe
  - Must be ensured in the event of
    - Failure of a single primary structural element
    - Partial damage occurrence in extensive structures (e.g. skin panels)
    - Crack propagation
      - Adequate residual strength and stiffness
      - Slow rate of crack propagation
  - Design for a specified life in terms of
    - Operational hours
    - Number of flight cycles (ground-air-ground)

- Minimum structural weight
  - Wing
    - Fixed items & fuel tank outboard of wing (reduce wing loading)
    - 1-m free of fuel at wing tip (avoid fire risk in case of electrostatic loads)
    - Heavy mass at the wing in front of the structural axis (reduce aeroelastic issues)
    - Use the same ribs to support landing gear, flaps, engine
    - If possible wing in one part (throughout the fuselage for mid-wing)
  - Landing gear
    - Commonly attached to the wing
    - Should not induce bending nor shearing larger than in flight
      - Close to the root
      - Just forward of flexural axis



- Minimum structural weight (2)
  - Fuselage
    - Heavy masses near the CG (reduce the inertia loads)
    - Limited number of bulkheads
  - Empennages
    - Far from the wing (to reduce the aerodynamic loading)
    - Supported by an existing bulkhead
  - Other
    - Simple structures (avoid rollers, …)

- Ease of maintenance and inspection

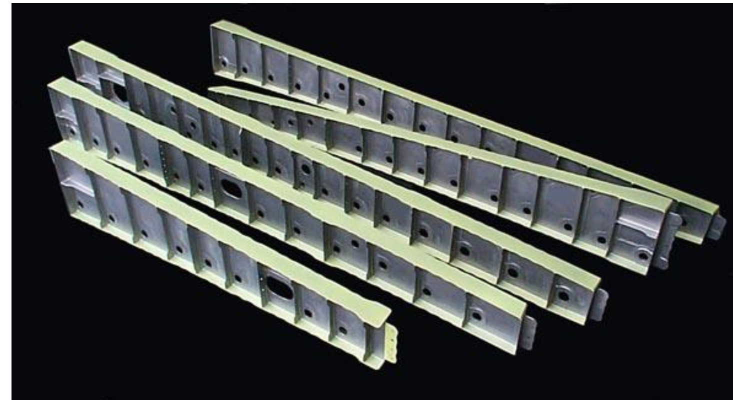


- Aluminum alloys
  - Duralumin (2xxx)
    - 4-7% Cu, 0.5-1.5% Mg, 0.2-2% Mn, 0.3% Si, 0.2-1% Fe
    - Picture: F15 horizontal stabilizer skin
  - Magnesium-Silicon alloy (6xxx)
    - 0.1-0.4% Cu, 0.5-1.5% Mg, 0.1-0.4% Mn, 0.3-2% Si, 0.1-0.7% Fe
  - Aluminum-Zinc-Magnesium alloy (7xxx)
    - 1-2.5% Cu, 1-7% Zn, 1-3% Mg, 0.3% Si
  - Used on fuselage and wing, also for rivets, …

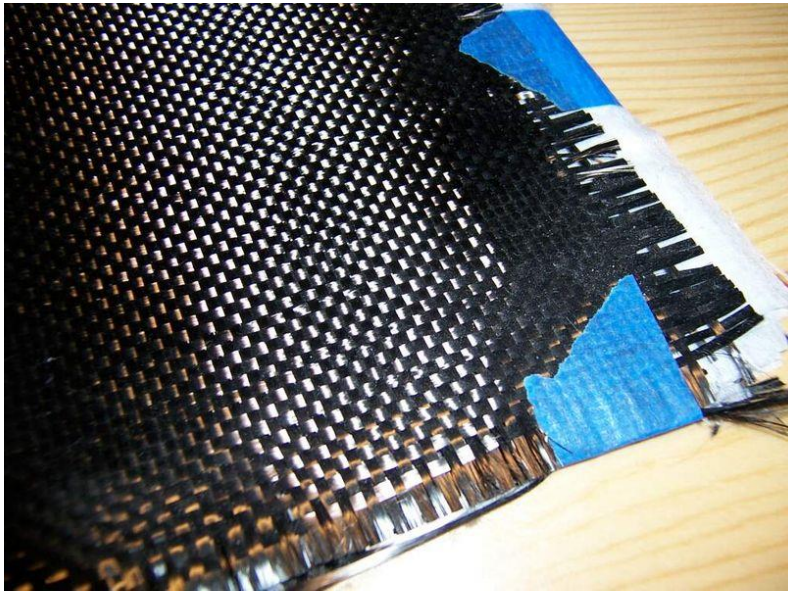
	Yield [MPa]	Weldability	Machinability	Corrosion resistance	Fatigue properties
2024-T351	270	No	Average	Poor	Excellent
6061 T6	240	Excellent	Good	Good	Good
7075 T651	400	No	Average	Average	Good

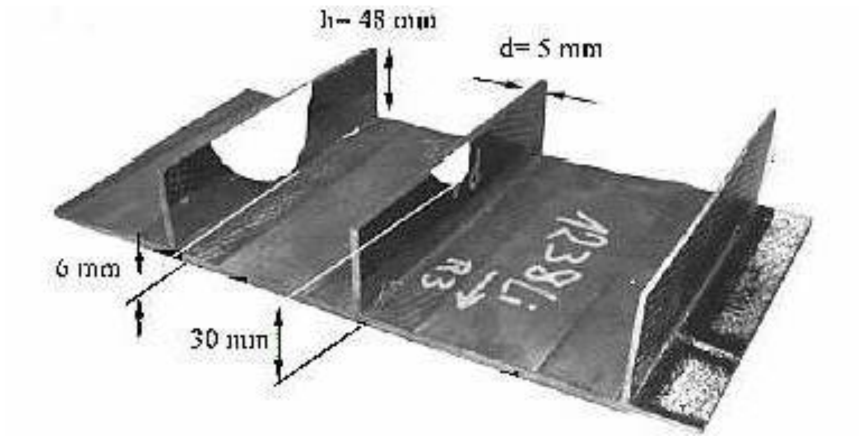
- Steel
  - Iron
    - Specific strength too low
  - Ultra-high-tensile strength carbon alloys
    - Brittleness
    - Not easily machinable, nor to weld
  - Maraging steel
    - Low carbon (<0.03%)
    - 17-19% Ni, 8-9% Co, 3-3.5 Mo, 0.15-0.25% Ti
    - High Yield strength (1400 MPa)
    - Compared to carbon-alloy
      - Higher toughness
      - Easier to machine and to weld
      - Better corrosion resistance
      - 3x more expensive
    - Aircraft arrester hook, undercarriage, …
    - Can be used at elevated temperature (400°C)

- Titanium alloy
  - High specific strength
    - Example Ti 6Al-4V
      - Yield 830 MPa, density 4510 $kg \cdot m^{-3}$
  - Properties
    - High toughness
    - Good fatigue resistance
    - Good corrosion resistance
      - Except at high T° and salt environment
    - Good Machinability and can be welded
    - Retains strength at high T° (500°C)
  - High primary and fabrication cost
    - 7X higher than aluminum alloys
  - Uses
    - Military aircrafts
      - Picture: F22 wing spars (Ti 6Al-4V)
    - Slat and flap tracks
      - Picture: B757 flap track (Ti 10V-2Fe-3Al)
    - Undercarriage



- Composite
  - Fibers in a matrix
    - Fibers: polymers, metals or ceramics
    - Matrix: polymers, metals or ceramics
    - Fibers orientation: unidirectional, woven, random
  - Carbon Fiber Reinforced Plastic
    - Carbon woven fibers in epoxy resin
      - Picture: carbon fibers
    - Tensile strength: 1400 MPa
    - Density: 1800 $kg \cdot m^{-3}$
    - A laminate is a stack of CFRP plies
      - Picture: skin with stringers

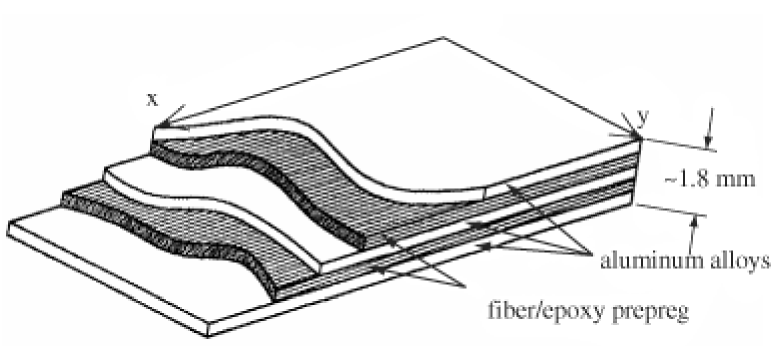


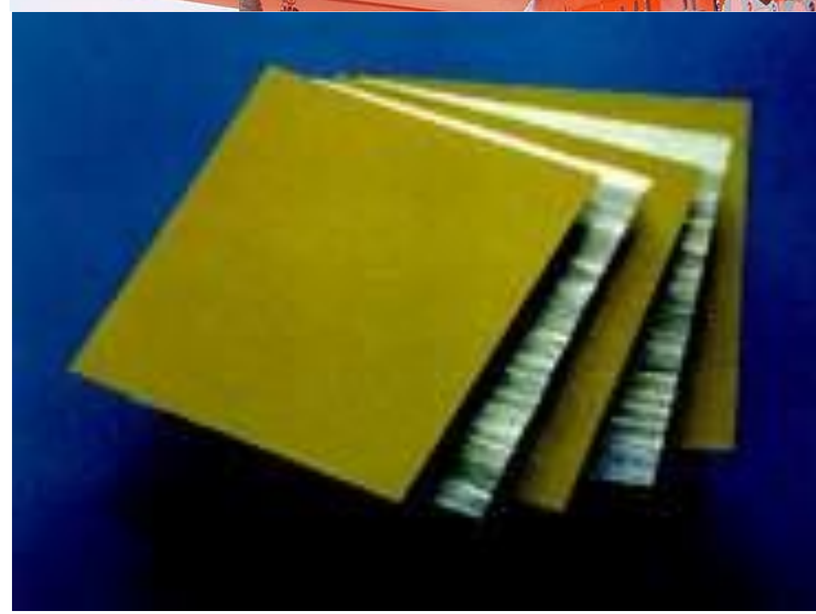


- Composite (2)
  - Wing, fuselage, …
  - Typhoon: CFRP
    - 70% of the skin
    - 40% of total weight
  - B787:
    - Fuselage all in CFRP

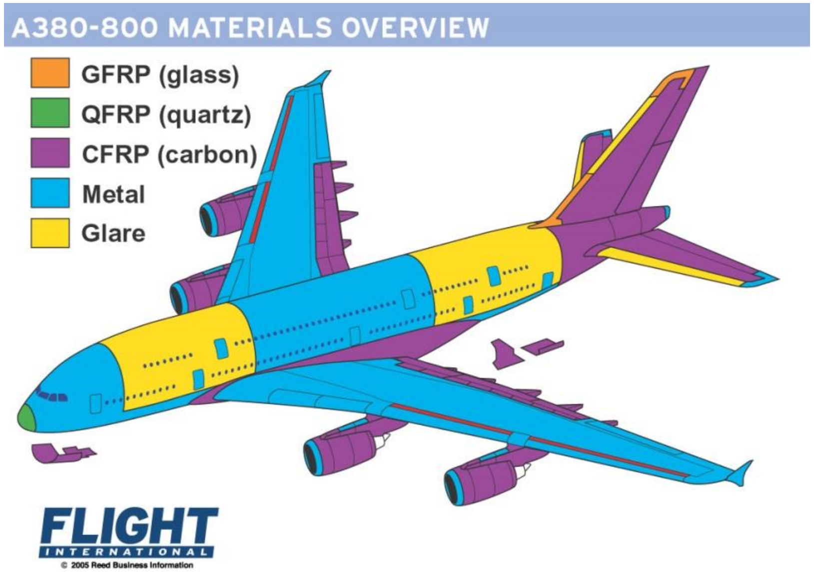
- Composite (3)
  - Drawbacks
    - "Brittle" rupture mode
    - Impact damage
    - Resin can absorb moisture

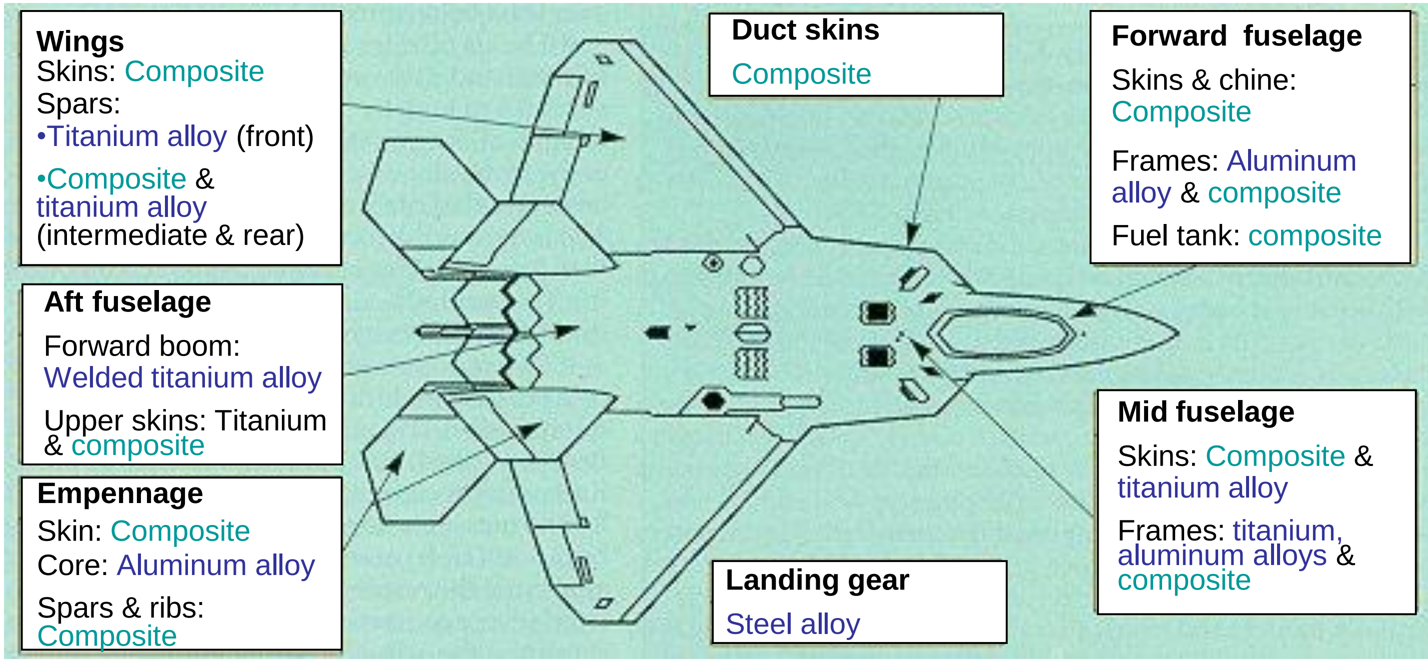
  - Glare
    - Thin layers of aluminum interspersed with Glass Fiber Reinforced Plastic
    - Improves damage resistance



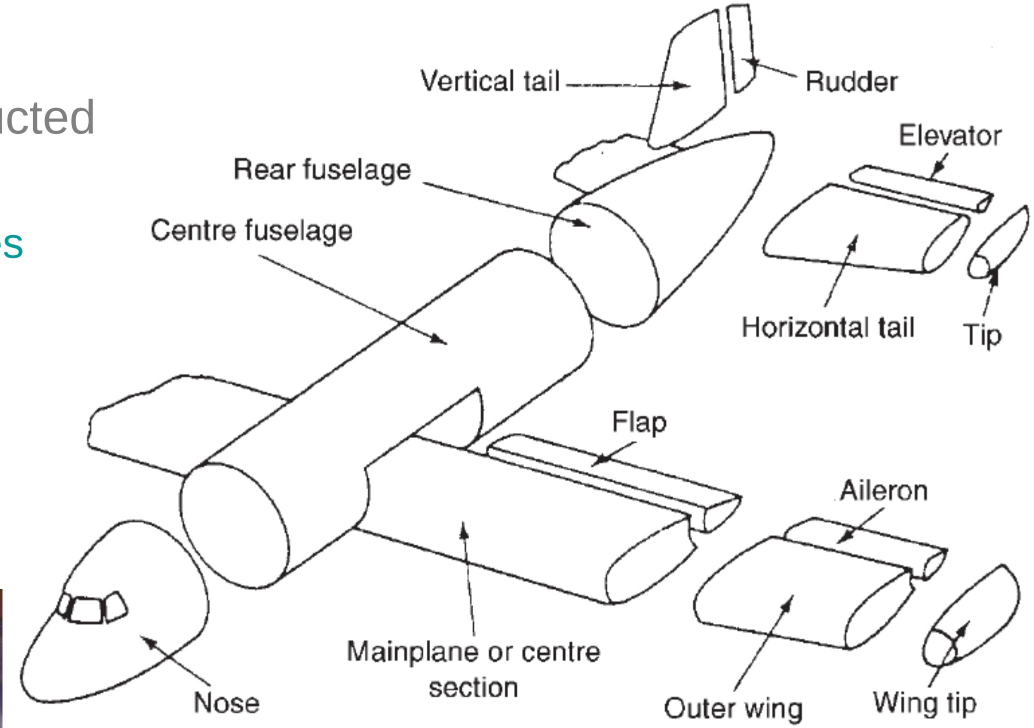


- Materials summary
  - Military aircrafts use more
    - Composite
    - Titanium alloy
  - Civil aircrafts
    - More and more composite

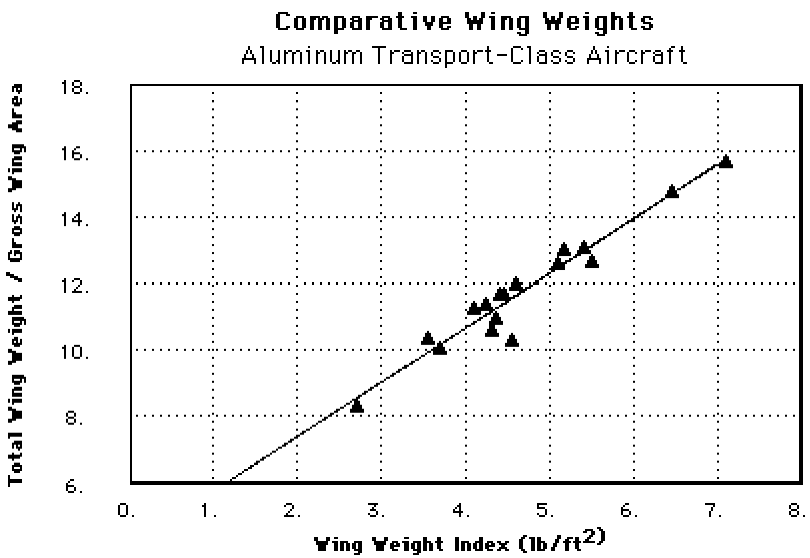




- Sub-assembly
  - Each sub-assembly is constructed
    - In specialized designed jigs
    - In different factories, countries



- Component weight can be estimated
  - For conceptual design
  - Based on statistical results of traditional aluminum structures
  - Example: wing



- Structural weight [lbs]
  - Wing with ailerons

$$W_w = 4.22\,S + 1.642\,10^{-6}\,\frac{n_{\mathrm{ultim}} b^3 \sqrt{W_{\mathrm{to}} \mathrm{ZFW}}\,(1+2\lambda)}{\left.\frac{t}{c}\right|_{\mathrm{avg}} \cos^2\Lambda\, S\,(1+\lambda)}$$

S: gross area of the wing [ft^2]

ZFW: zero fuel weight [lb]

Λ: sweep angle of the structural axis

t: airfoil thickness [ft]

W_{to}: take off weight [lb]

b: span [ft]

λ: taper (c_{tip}/c_{root}),

c: chord [ft]

  - Horizontal empennage & elevators

$$W_T = 5.25\,S_{T\,\mathrm{exp}} + 0.8\,10^{-6}\,\frac{n_{\mathrm{ultim}} b_T^3 W_{\mathrm{to}} \bar{\bar{c}} \sqrt{S_{T\,\mathrm{exp}}}}{\left.\frac{t_T}{c_T}\right|_{\mathrm{avg}} \cos^2\Lambda_T\, l_T\, S_T^{\frac{3}{2}}}$$

$S_{T\,exp}$: exposed empennage area [ft^2]

l_T: distance plane CG to empennage CP [ft]

$\bar{\bar{c}}$: average aerodynamic chord of the wing [ft]

S_T: gross empennage area [ft^2]

b_T: empennage span [ft]

t_T: empennage airfoil thickness [ft]

c_T : empennage chord [ft]

Λ_T: sweep angle of empennage structural axis

- Structural weight [lbs] (2)
  - Fin without rudder

$$W_{F'} = 2.62\, S_F + 1.5\, 10^{-5}\, \frac{n_{\text{ultim}} b_F^3 \left(8 + 0.44 \frac{W_{\text{to}}}{S}\right)}{\left.\frac{t_F}{c_F}\right|_{\text{avg}} \cos^2 \Lambda_F}$$

S_F: fin area [ft²]
b_F: fin height [ft]
t_F: fin airfoil thickness [ft]
c_F: fin chord [ft]
Λ_F: sweep angle of fin structural axis
S: gross surface of wing [ft²]

  - Rudder: $W_r / S_r \sim 1.6\, W_{F'} / S_F$
  - Fuselage
    - Pressure index $I_\text{p} = 1.5\, 10^{-3}\, \Delta p_\text{max} \text{width}_\text{fus}$
    - Δp [lb/ft²] (cabin pressure ~2600m)
    - Bending index

$$I_\text{b} = 1.91\, 10^{-4} n_{\text{limit at ZFW}} \left(\text{ZFW} - W_w - W_{\text{wing-mounted engines}}\right) \frac{\text{length}_\text{fus}}{\text{height}_\text{fus}^2}$$

    - Weight depends on wetted area S_wetted [ft²] (area in direct contact with air)

$$W_\text{fus} = \left(1.051 + 0.102\, I_\text{fus}\right) S_\text{fus,wetted}$$

$$I_\text{fus} = \begin{cases} I_\text{p} & \text{if } I_\text{p} > I_\text{b} \\ \frac{\left(I_\text{p}^2 + I_\text{b}^2\right)}{2 I_\text{b}} & \text{if } I_\text{p} < I_\text{b} \end{cases}$$

- Structural weight [lbs] (3)
  - Systems
    - Landing gear $W_{gear} = 0.04\ W_{to}$
    - Hydromechanical system of control surfaces $W_{SC} = I_{SC}\ (S_{Texp}+S_F)$

      I_{sc} [lb/ft^2] : 3.5, 2.5 or 1.7 (fully, partially or not powered)
    - Propulsion $W_{prop} = 1.6 W_{eng} \sim 0.6486\ T_{to}^{0.9255}$

      T_{to} : Static thrust (M 0) at sea level [lbf], *1lbf ~ 4.4 N
    - Equipment
      - APU $W_{APU} = 7\ N_{seats}$
      - Instruments (business, domestic, transatlantic) $W_{inst} = 100, 800, 1200$
      - Hydraulics $W_{hydr} = 0.65\ S$
      - Electrical $W_{elec} \sim 13\ N_{seats}$
      - Electronics (business, domestic, transatlantic) $W_{etronic} = 300, 900, 1500$
      - Furnishing if < 300 seats $W_{furn} \sim (43.7 - 0.037\ N_{seats})\ N_{seats} + 46\ N_{seats}$

        if > 300 seats $W_{furn} \sim (43.7 - 0.037{*}300)\ N_{seats} + 46\ N_{seats}$
      - AC & deicing $W_{AC} = 15\ N_{seats}$
  - Payload ($W_{payload}$)
    - Operating items (class dependant) $W_{oper} = [17 - 40]\ N_{pass}$
    - Flight crew $W_{crew} = (190 + 50)\ N_{crew}$
    - Flight attendant $W_{attend} = (170 + 40)\ N_{atten}$
    - Passengers (people and luggage) $W_{pax} = 225\ N_{pass}$
  - Definitions :
    - ZFW: Sum of these components $ZFW = \Sigma\ W_i$

- Structural weight [lbs] (4)
  - Examples

Aircraft System	CITATION-500	MDAT-30	MDAT-50	F-28	MDAT-70	DC-9-10	BAC-111	DC-9-30	737-200	727-100
Wing System	1,020	3,143	4,360	7,526	5,910	9,366	9,817	11,391	11,164	17,682
Tail System	288	1,010	1,193	1,477	1,505	2,619	2,470	2,790	2,777	4,148
Body System	930	4,276	5,692	6,909	7,118	9,452	11,274	11,118	11,920	17,589
Alighting Gear System	425	1,379	1,874	2,564	2,440	3,640	3,465	4,182	4,038	7,244
Nacelle System	241	948	1,294	866	1,684	1,462	1,191	1,462	1,515	2,226
Propulsion System (less Dry Engine)	340	1,140	1,338	988	1,702	1,478	1,788	2,190	1,721	3,052
Flight Controls System (less Auto Pilot)	196	600	699	1,404	805	1,102	1,655	1,434	2,325	2,836
Auxiliary Power System	0	343	400	320	460	805	719	817	855	0
Instrument System	76	300	300	267	300	490	504	575	518	723
Hydraulic and Pneumatic System	94	257	300	406	345	681	1,391	753	835	1,054
Electrical System	361	617	825	953	1,040	1,631	1,610	1,715	2,156	2,988
Avionics System (incl. Auto Pilot)	321	586	586	923	586	1,039	1,368	1,108	1,100	1,844
Furnishings and Equipment System	794	2,657	3,548	3,535	4,772	6,690	7,771	8,594	9,119	11,962
Air Conditioning System	188	325	435	520	550	1,016	1,062	1,110	1,084	1,526
Anti-icing System	101	384	448	520	511	472	234	474	113	639
Load and Handling System	2	20	20	--	20	19	9	57	--	15
Empty Weight (less Dry Engine)	5,377	17,985	23,312	29,178	29,748	41,962	46,328	49,770	51,240	75,528
Dry Engine Weight	1,002	2,480	3,373	4,327	4,392	6,113	5,434	6,160	6,212	9,322
Empty Weight (M.E.W.)	6,379	20,465	26,685	33,505	34,140	48,075	51,762	55,930	57,452	84,850
Takeoff Gross Weight	11,650	34,480	46,850	62,000	61,000	86,300	99,650	108,000	104,000	161,000

Manufacturer empty weight

- Structural weight [lbs] (5)
  - Examples

Aircraft System	727-200	707-320	DC-8-55	DC-8-62	DC-10-10	L-1011	DC-10-40	747	SCAT-15*
Wing System	18,529	28,647	34,909	36,247	48,990	47,401	57,748	88,741	83,940
Tail System	4,142	6,004	4,952	4,930	13,657	8,570	14,454	11,958	8,590
Body System	22,415	22,299	22,246	23,704	44,790	49,432	46,522	68,452	54,322
Alighting Gear System	7,948	11,216	11,682	11,449	18,581	19,923	25,085	32,220	28,720
Nacelle System	2,225	3,176	4,644	6,648	8,493	8,916	9,328	10,830	15,650
Propulsion System (less Dry Engine)	3,022	5,306	9,410	7,840	7,673	8,279	13,503	9,605	6,310
Flight Controls System (less Auto Pilot)	2,984	2,139	2,035	2,098	5,120	5,068	5,188	6,886	10,777
Auxiliary Power Plant System	849	0	0	0	1,589	1,202	1,592	1,797	--
Instrument System	827	550	1,002	916	1,349	1,016	1,645	1,486	3,400
Hydraulic and Pneumatic Group	1,147	1,557	2,250	1,744	4,150	4,401	4,346	5,067	10,670
Electrical System	2,844	3,944	2,414	2,752	5,366	5,490	5,293	5,305	6,002
Avionics System (incl. Auto Pilot)	1,896	1,815	1,870	2,058	2,827	2,801	3,186	4,134	4,178
Furnishings and Equipment System	14,702	16,875	15,884	15,340	38,072	32,829	33,114	48,007	20,615
Air Conditioning System	1,802	1,602	2,388	2,296	2,386	3,344	2,527	3,634	2,820
Anti-icing System	666	626	794	673	416	296	555	413	210
Load and Handling System	19	--	55	54	62	--	62	228, -896**	--
Empty Weight (less Dry Engine)	86,017	105,756	116,535	118,749	203,521	198,968	224,148	297,867	256,204
Dry Engine Weight	9,678	19,420	16,936	17,316	23,229	30,046	25,587	35,700	45,020
Empty Weight (M.E.W.)	95,695	125,176	133,471	136,065	226,750	229,014	249,735	333,567	301,224
Takeoff Gross Weight	175,000	312,000	325,000	335,000	430,000	430,000	565,000	775,000	631,000

Manufacturer empty weight

- CG locations
  - Wing: 30% chord at wing MAC
  - Horizontal tail: 30% chord at 35% semi-span
  - Fin: 30% chord at 35% of vertical height
  - Surface controls: 40% chord on wing MAC
  - Fuselage: 45% of fuselage length
  - Main Gear: located sufficiently aft of aft c.g. to permit 5% - 8% of load on nose gear
  - Hydraulics: 75% at wing c.g., 25% at tail c.g.
  - AC / deicing: End of fuse nose section
  - Propulsion: 50% of nacelle length for each engine
  - Electrical: 75% at fuselage center, 25% at propulsion c.g.
  - Electronics and Instruments: 40% of nose section
  - APU: Varies
  - Furnishings, passengers, baggage, cargo, operating items, flight attendants: From layout. Near 51% of fuselage length
  - Crew: 45% of nose length
  - Fuel: Compute from tank layout

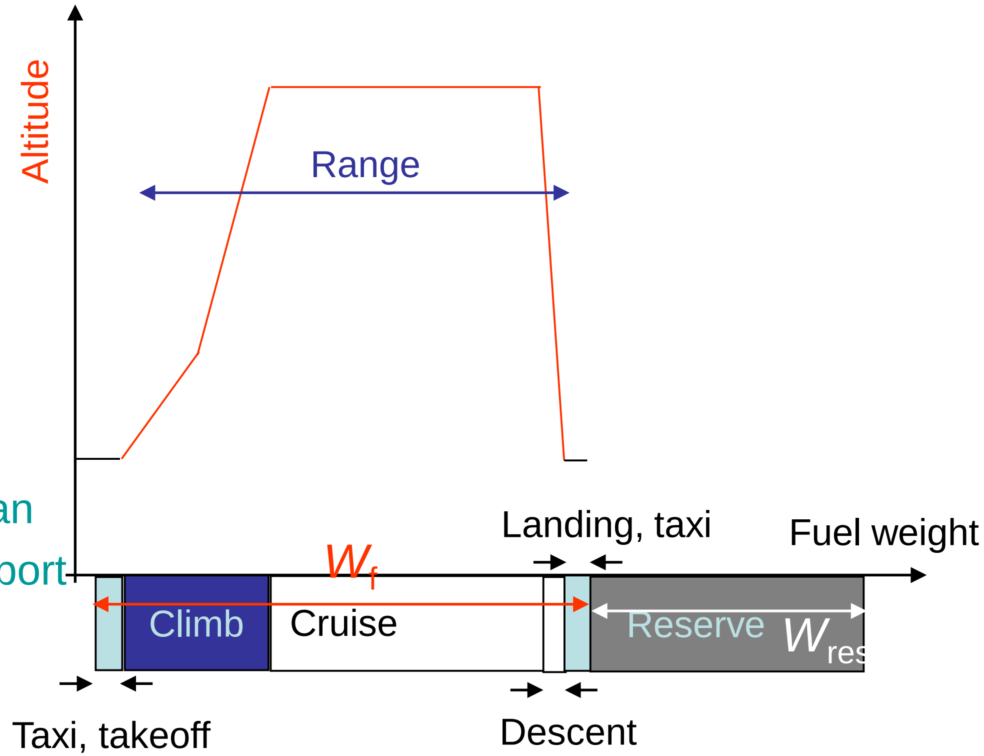
- For a given mission
  - Taxi & takeoff

    $W_{taxi} = 0.0035\ W_{to}$
  - Landing & taxi

    $W_{land} = 0.0035\ W_{to}$
  - Reserve
    - Should allow
      - Deviations from the flight plan
      - Diversion to an alternate airport
    - Airliners
      - $W_{res} \sim 0.08\ \text{ZFW}$
    - Business jet
      - W_{res} fuel consumption for ¾-h cruise
  - Climbing (angle of ~ 10°)

$$\frac{W_{\text{climb}}}{W_{\text{TO}}} \simeq \frac{1}{100}\left[\frac{\text{cruise altitude [ft]}}{31600\ \text{[ft]}} + \frac{1}{2} M_{\text{cruise}}^2\right]$$

  - Descend: ~ same fuel consumption than cruise
  - Take Off Weight (TOW): $W_{to} = \text{ZFW} + W_{res} + W_f$
  - Landing weight: $\text{ZFW} + W_{res} + 0.0035\ W_{to}$



References

- Lecture notes
  - Aircraft Design: Synthesis and Analysis, Ilan Kroo, Stanford University, http://adg.stanford.edu/aa241/AircraftDesign.html
- Other reference
  - Book
    - Aircraft Structures for engineering students, T. H. G. Megson, Butterworth-Heinemann, An imprint of Elsevier Science, 2003, ISBN 0 340 70588 4