INTRODUCTION TO AIRCRAFT STRUCTURES

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INTRODUCTION TO AIRCRAFT STRUCTURES : FUSELAGE STRUCTURES - LOADING

Or ‘Where you put the payload in a typical modern airframe’

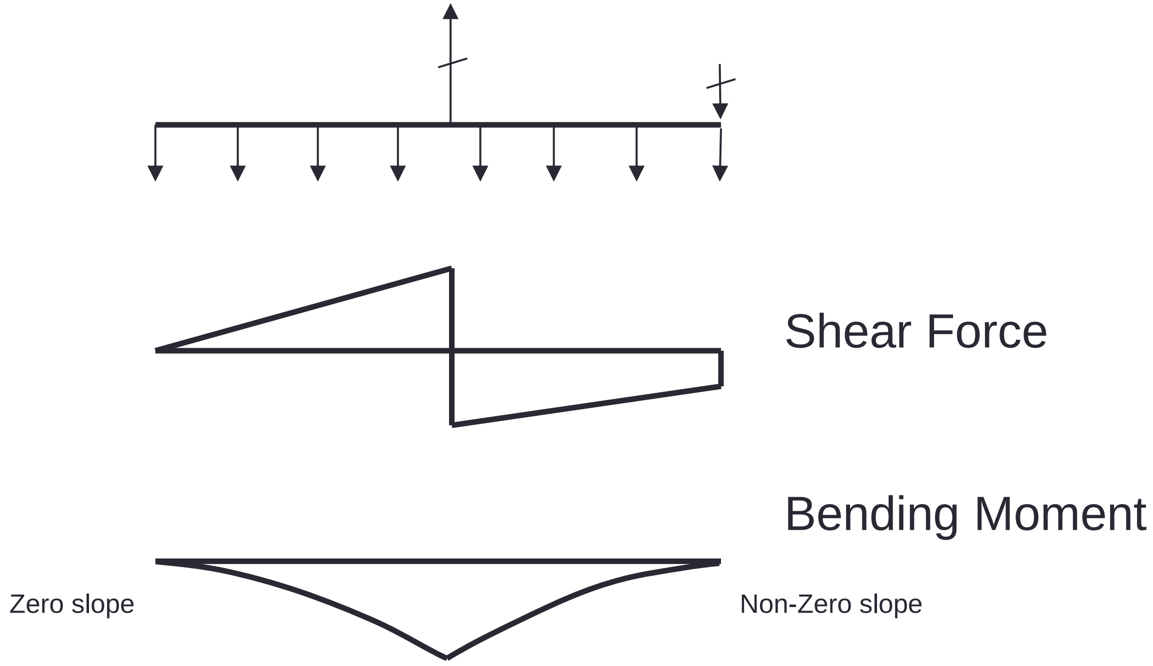
FUSELAGE STRUCTURES

- **Overall Loading on Fuselage**

- LOADING IN LEVEL FLIGHT

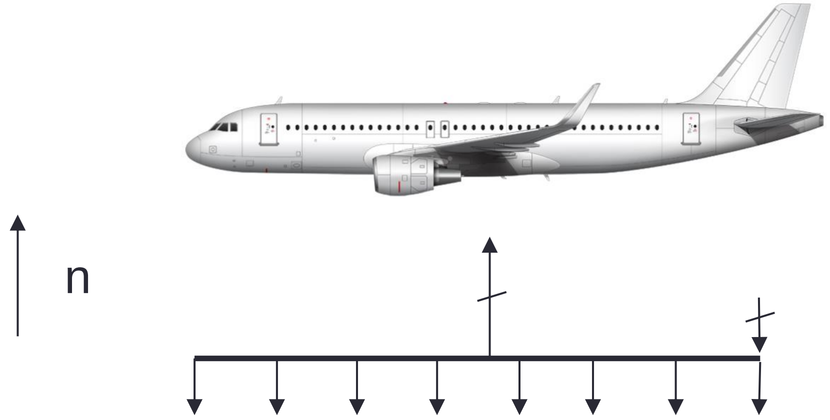
- For simplified illustration assume:

  - uniform distribution of mass along length
  - wing and tail loads act at single points
  - thrust and drag ignored



FUSELAGE STRUCTURES

- LOADING IN SYMMETRIC MANOEUVRE



- A simple manoeuvre such as steady pull-out from a dive gives the same basic type of loading. However the aerodynamic forces are higher, causing an upward acceleration. For steady state conditions this can be treated statically by applying inertia loads in the opposite direction. These are the normal loads due to the weight of the aircraft, but multiplied by a load factor n. The load factor represents the ratio between the inertia loads the aircraft experiences in a particular flight condition and in the 1-g level flight case.

FUSELAGE STRUCTURES

- Typical values of Flight ‘n’ are:
- + 2.5 (transport aircraft)
- + 4.5 (light aircraft)
- + 8.0 to - 6.0 (aerobatic aircraft)
- + 9.0 to – 3.5 (fighter)

FUSELAGE STRUCTURES

- Numerical Example

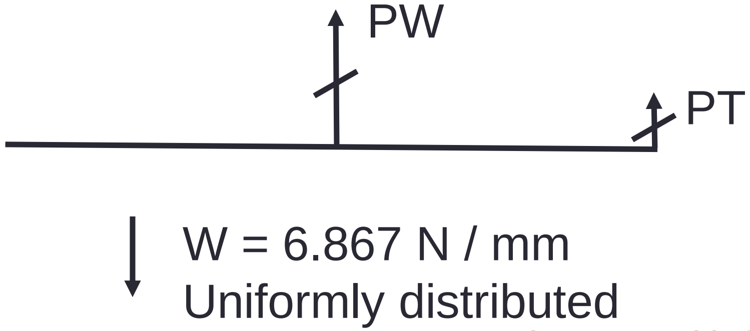
- Consider the simplified case of the fuselage of a business aircraft of length 10m, and diameter 2m.

- The all-up mass of the fuselage is 2000 kg, which is assumed to be uniformly distributed along its length.

- For a symmetric manoeuvre a load factor of 3.5 applies. Wing loading acts through a point 5.5m from the nose, and balance is maintained by a vertical tail load acting at the rear end of the fuselage.

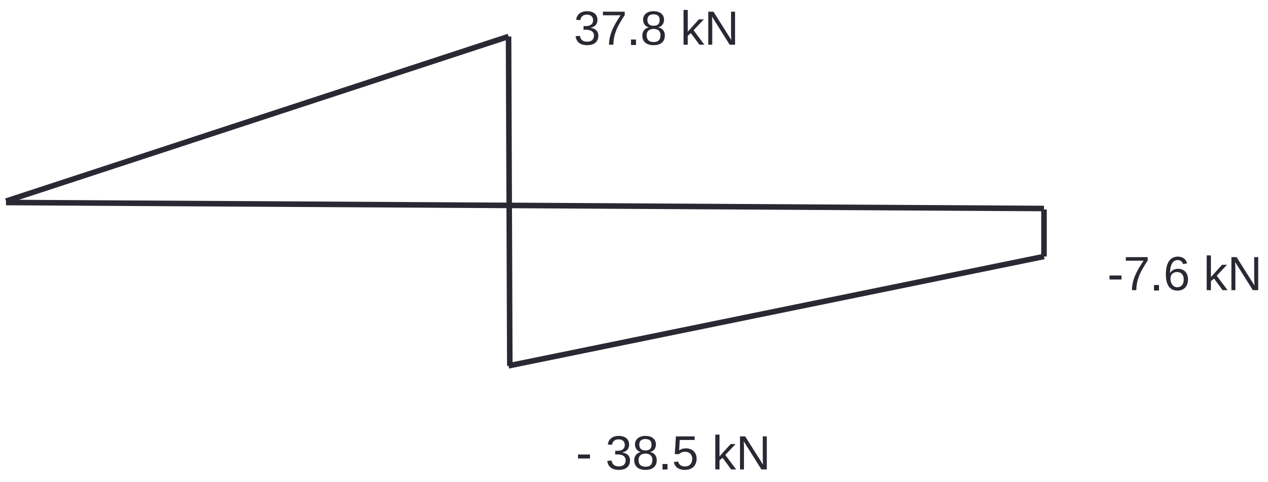
FUSELAGE STRUCTURES

- Inertia load acting downwards per unit length (W) is given by :
- W = (2000 * 9.81 * 3.5) / 10000 = 6.87 N / mm
- Calculate PW and PT by taking moments about the tail :
- PW * 4500 = 6.87 * 10000 * 5000
- PW = 76.3 kN
- PT + PW = 6.87 * 10000
- PT = - 7.6 kN



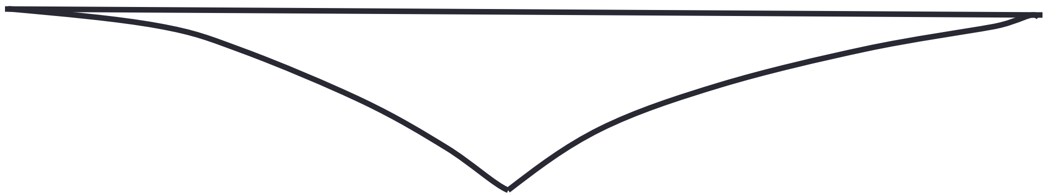
FUSELAGE STRUCTURES

- Shear force diagram with principal values:



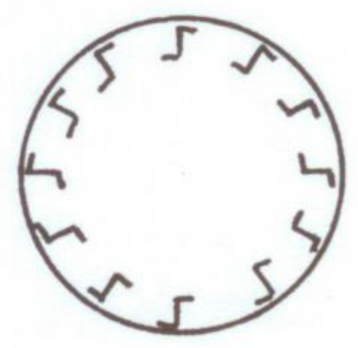
FUSELAGE STRUCTURES

- Bending moment diagram with principal value:



- 103.9 kNm

FUSELAGE STRUCTURES



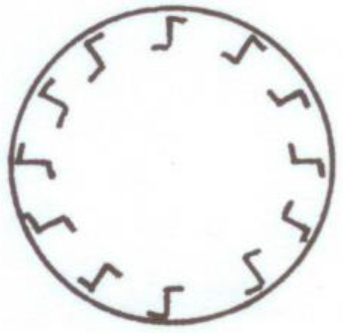
- Calculation of fuselage stresses and deflections

- The fuselage is circular, with a skin of thickness 1 mm.

- It is stiffened by 12 equally spaced Z section stringers with height 50 mm, flange width 25 mm, and thickness 1 mm.

- The material is aluminium alloy, for which Young’s modulus is 70 GPa. Initially assume that no buckling occurs.

FUSELAGE STRUCTURES

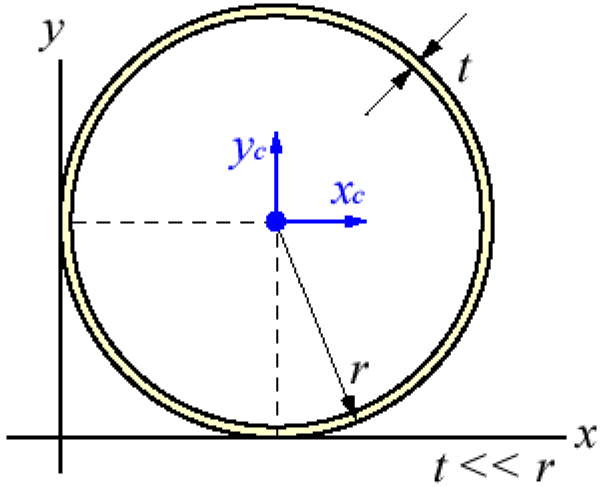


- Estimation of second moment of area I:

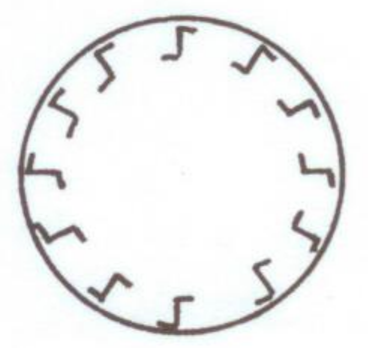
- Equivalent skin thickness : te / t = At / As = Total Area / Skin Area
- Where skin t = 1.0 mm, fuse dia = 2000 mm, 12 stingers of total flange length 100mm and thickness of 1.00 mm

- te =1 x [(2000 x π x 1) + (12 x 100 x 1)] / (2000 x π x 1)
  = 1.19 mm
- For thin walled tubes, Ix = Iy = π * r ^ 3 * te

- I = 3.142 * (1000^3) * 1.19 = 3.74 x 10^9 mm4

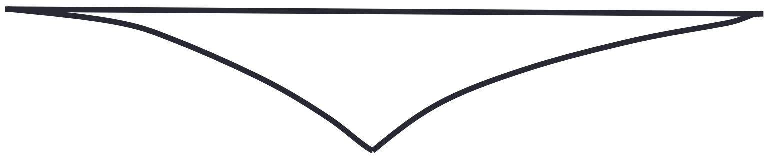


FUSELAGE STRUCTURES



- Stress $\sigma = My/I$

- BM Max = 103.9 kNm
- $I = 3.74 * 10^9$ mm^4
- Max $\sigma = 103.9 \times 10^6 \times \pm 1000 / 3.74 \times 10^9$



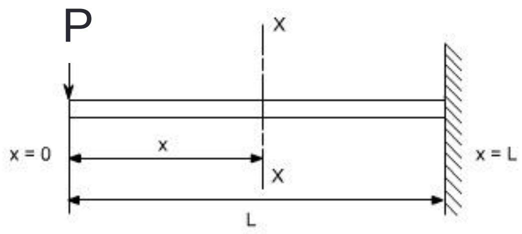
- 103.9 kNm

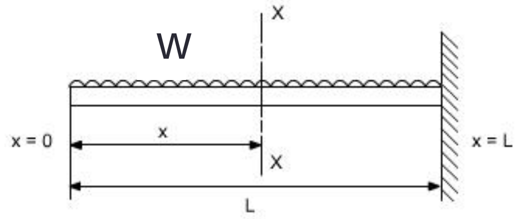
- Max $\sigma = \pm$ 28 MPa
- This is the max bending stress in the fuselage skin

FUSELAGE STRUCTURES

Assumed point of <u>Zero</u> vertical displacement

- Calculation of vertical deflections:
- Where should displacement be assumed to be zero?
- Assume zero displacement and rotation at Wing connection
- Equivalent to a fully constrained end condition
- For a cantilever with end load P, end deflection is $PL^3/3EI$
- For a cantilever with distributed load w, end deflection is $wL^4/8EI$





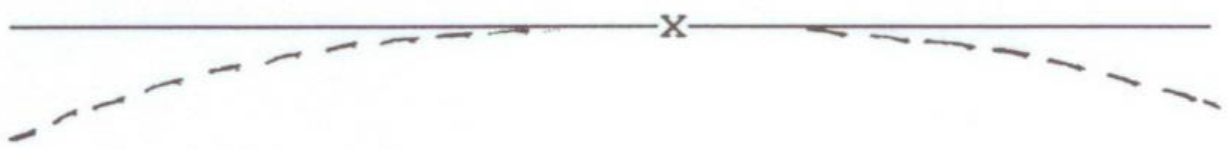
FUSELAGE STRUCTURES

- At nose, δ = 6.87 x 5500 ^ 4 / (8 x 70000 x 3.74 x 10^9) = 3.00 mm

- At tail, by superposition δ = 6.87 x 4500 ^ 4 / (8 x 70000 x 3.74 x 10^9)
  + 7630 x 4500 * 3 / (3 x 70000 x 3.74 x 109)
  = 1.345 + 0.885 = 2.23 mm

- Note: superposition can be used for any linear analysis

- Deflected shape:



δ = 3.00 mm

δ = 2.23 mm

FUSELAGE STRUCTURES

- LOADING IN GUSTS

- Symmetric gusts can be treated in a similar way using an appropriate load factor. Both positive (up) and negative (down) gusts have to be considered. The positive gust is generally more critical because it adds to the 1g level flight loading.

1g cruise inertia ↓

1g cruise inertia ↓ + ↓ ‘n’ g gust inertia

FUSELAGE STRUCTURES

- LOADING DURING LANDING

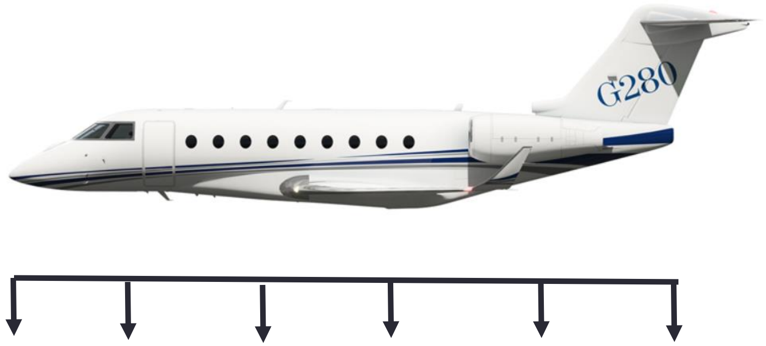
- Various different cases have to be considered e.g. for an aircraft with nose and two main undercarriages:
  - Three point landing
  - Tail down landing
  - Touchdown on one wheel / bogie

- Significant fore-aft loading arises due to
  - Braking
  - Attitude of aircraft in tail down landing

FUSELAGE STRUCTURES

- UNBALANCED PITCHING MOMENTS

- For cases where there are net pitching moments on the aircraft, the inertia loading also includes a rotational acceleration about the c.g. In such cases the load factor n is no longer constant. It varies linearly with distance from the c.g.



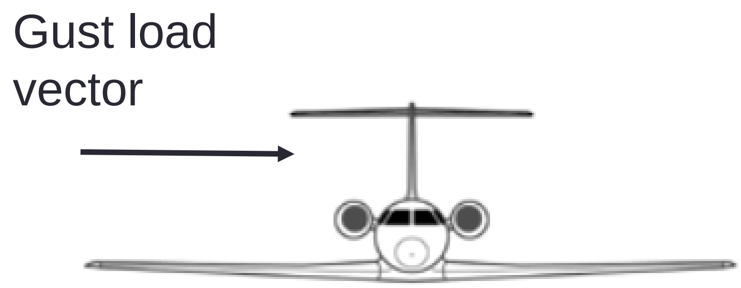
Constant ‘n’ load factor

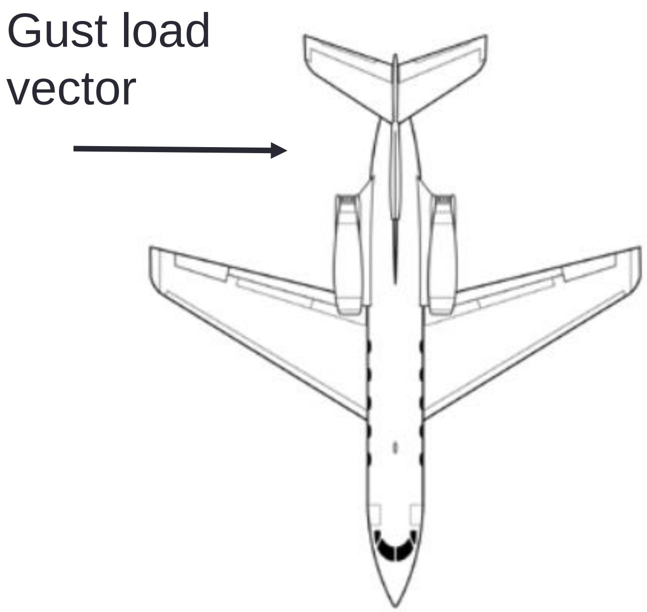
‘n’ load factor varies along fuselage length

FUSELAGE STRUCTURES

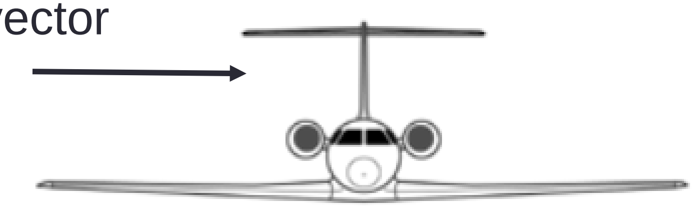
- LOADING IN ASYMMETRIC CASES

- Asymmetric manoeuvre with side load from tail
- Lateral gust





FUSELAGE STRUCTURES



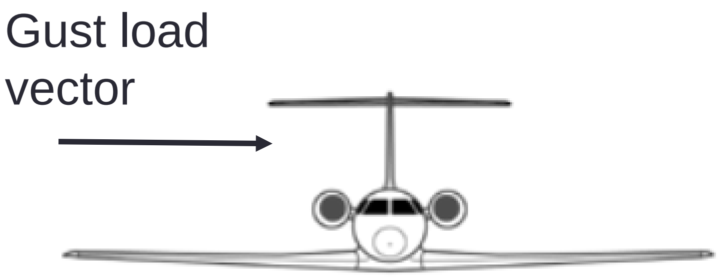
- Numerical Example

- The same business aircraft from the fuselage bending analysis example has a vertical tail fin extending 2m above the top of the fuselage.

- In a lateral gust a side load of 10 kN arises acting on the tailplane at a point 1m above the top of the fuselage.

- Calculate the stresses in the fuselage and the lateral deflection at the top of the tail fin due to torsion in the fuselage assuming all the torque is reacted at the wings.

FUSELAGE STRUCTURES



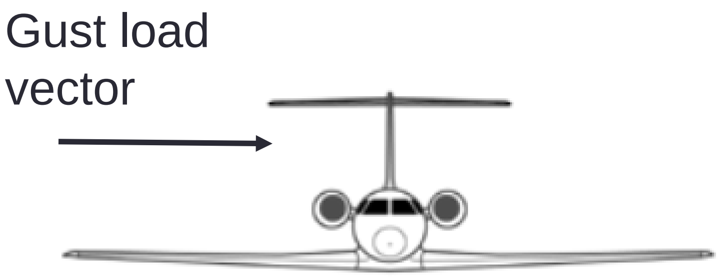
- Numerical Example

- Torque T = 10 * (1.0 +1.0) = 20 kNm

- Polar second moment of area for a thin walled cylinder $J = 2\pi r^3 t$
- J = 2 * π * 1000 ^ 3 * 1.0 (note, fuselage skin thickness t is used, not te)
- $J = 6.28 * 10^9$ mm^4

- T/J = τ/r : where τ is the shear stress in the fuselage skin
- Hence $\tau = (20 * 10^6 * 1000) / 6.28 * 10^9 =$ 3.2 mpa

FUSELAGE STRUCTURES



- Shear modulus G = E/[2(1+v)]
- G = 70/[2(1+0.3)] = 26.9 GPa

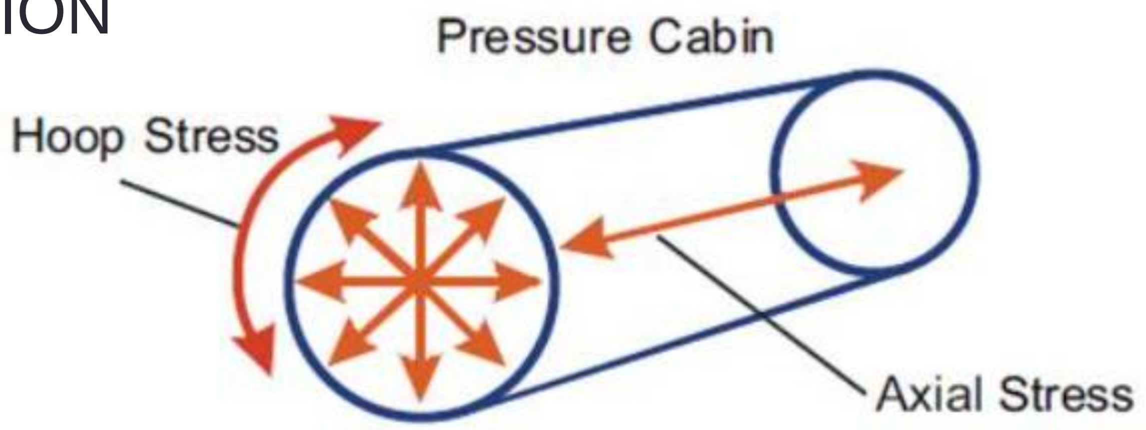
- Length of fuselage under torque L = 4500 mm
- T/J = Gθ/L
- Hence θ = 5.35 * 10^-4 Radians

- Lateral deflection at top of tail = 5.35 * 10^-4 * 3000 = 1.61 mm

- Note: the lateral load will also cause bending of the tail fin and fuselage

FUSELAGE STRUCTURES

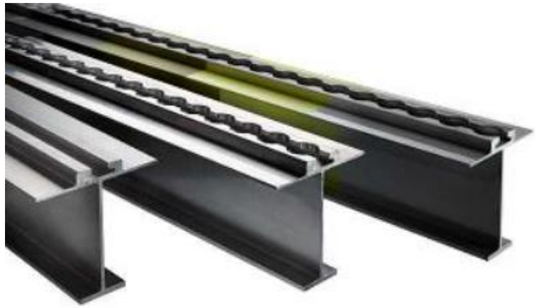
- LOADING DUE TO CABIN PRESSURISATION
- Hoop Stress due to pressure P
- Take P = 0.065 N/mm^2
- σhoop = P r / t
- σhoop =0.065 * 1000 / 1.0 = 65 Mpa

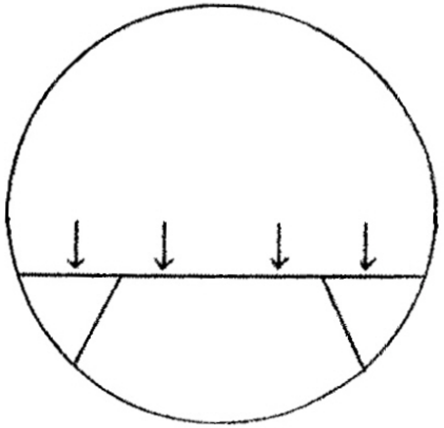


- Axial stress due to reaction of pressure P at fuselage end bulkheads
- (note – equivalent thickness te can be used to account for stringer axial area)
- σaxial may be approximated as = P r / 2 te
- σaxial = 0.065 * 1000 / 2 * 1.19 = 27.3 Mpa

FUSELAGE STRUCTURES

- **Local Loads on Fuselage Structure**





Internal loads are mainly defined by the inertia forces acting on the attached items of mass.

Loads can be considered to be reacted in the fuselage skin in shear via the fuselage frames