Loads

Lift and Inertia

Estimation of loads for the UAV main structural components under assumed critical load cases.

Contents

Aircraft

Fuselage

…

Contents ctd.

Wing

Joints

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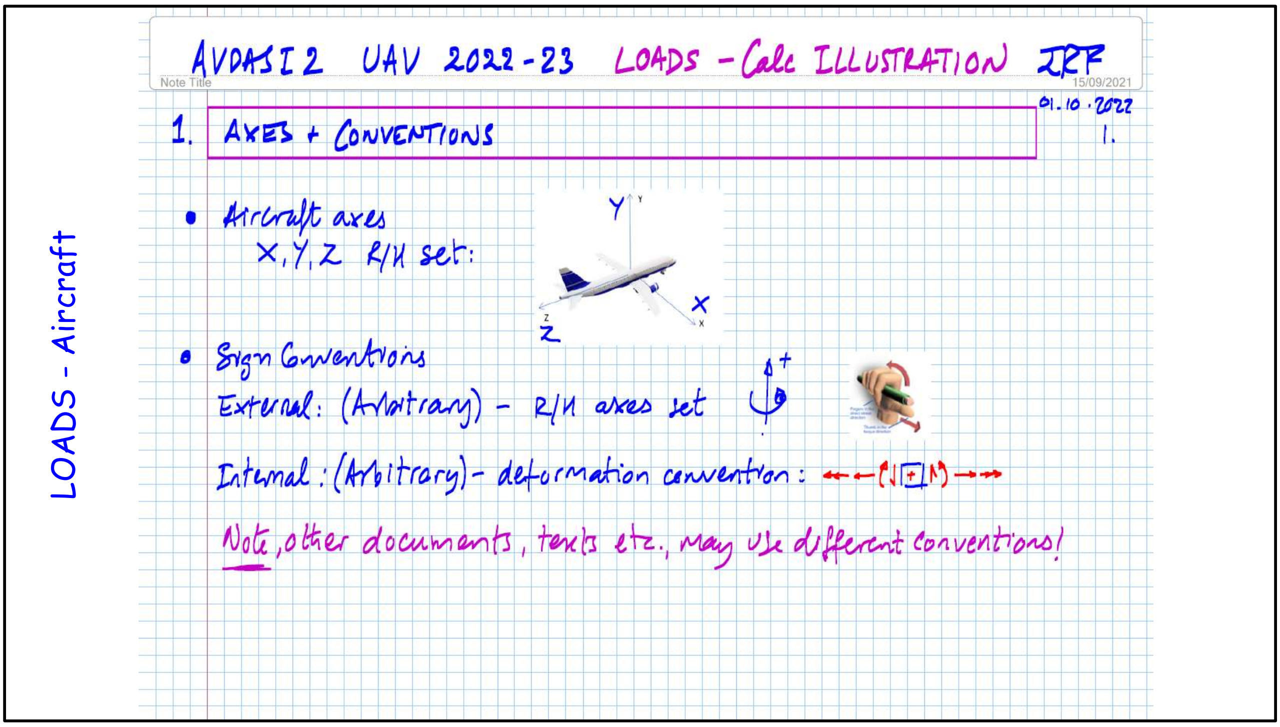
LOADS - Aircraft



Aircraft axes are external/global axes. In my notes, I have used *XYZ* as a right-handed set where:

X is the axis of the starboard wing from root to tip,

Y is the upward vertical axis through the depth of the aircraft, and

Z is the fuselage central axis from nose to tail.

Here, the positive translational directions are as described above and the positive rotational direction is according to the right-hand rule, i.e., the direction of the curled fingers of your right hand when your thumb points in the positive translational axis direction.

Note, there is also an internal "deformation" sign convention defined by our definition of positive deformations. Considering a sub-element, e.g., a section of a wing or fuselage spar as a beam, here we will assume:

Tension is positive

Shear is positive anti-clockwise

Bending moment is positive as sagging

Torsion is positive according to the right-hand rule

These axes and conventions are arbitrary and other documents, texts, analysis packages etc. may use different axes and conventions, e.g., bending positive in a hogging sense. Even so, once chosen you must adhere to the conventions to ensure correct interpretation for your calculations.

Always indicate your chosen sign convention on a sub-element as done on this slide. For consistency in this exercise, you must use the axes and conventions as shown on the slide.

2.

- Loads

Limit "LIM" Load cases, e.g. flight or ground loading

1) 6.0g limit sym' vertical/pitching manoeuvre/gust case 6.0g ↑
2) 6.0g limit lateral yaw/roll manoeuvre/gust case 6.0g → #
3) Empennage limit high speed manoeuvre case

Load case "acceleration factor", n

For each load case, for each structural item - calculate the Shear force, Bending Moment + Torsional loading components "S, M, T"

Initially for 1.0g @ limit then apply acc^n and ULT factors.

i.e.: $1.0g \times n \times \underbrace{k_{USF}}_{1.5}$

Loads are usually defined as limit load cases where "limit" represents the largest load that is expected in service and "load case" represents a particular condition which produces a particular combination of Axial, Shear, Moment and Torsion loading "A,S,M,T ".

There are many types of loads, e.g., flight loads such as gust and manoeuvre loads, and ground loads such as taxi, take-off and landing loads.

There are hundreds of possible load cases which could be considered, e.g. from different phases of flight and different aircraft configurations. For this exercise I have defined just three flight load cases, which should be enough to give us a good initial design for our UAV airframe structure:

Load Case 1) a 6.0g limit symmetric vertical load, representing a pitching manoeuvre or gust.

Load Case 2) a 6.0g limit lateral load, representing a roll or yaw loading from manoeuvre or gust, simplified here to just a lateral acceleration.

Load Case 3) a stall recovery, where the aircraft is spinning vertically at high speed with full opposite rudder kicked in for recovery. (Note, here I have defined the empennage loading as aerodynamic loading only, neglecting the mass of the VTP for our lightweight UAV structure).

Note 6.0g implies an acceleration factor of 6, which is reasonable for the UAV being considered.

For each applied load case, we need to calculate the internal S,M,T loading (neglecting axial loading as secondary here).

You are advised to carry out the initial calculations for 1.0g loading, then apply the acceleration factor to arrive at the load case limit, followed by the Ultimate Safety factor to arrive at the Ultimate Design loading. See appendix X1.1 for definitions.

2. AICRAFT MASSES

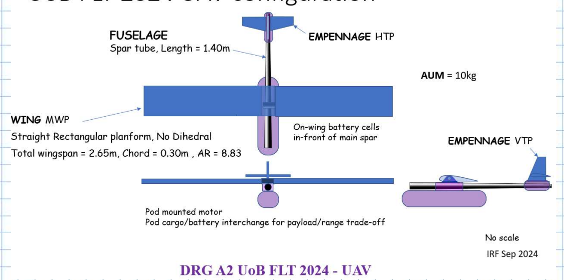
3.

Initial trial values:

	UAV		m	z
			kg	mm
on Fus	Motor		1.000	50
	Batteries		1.000	100
	Payload*		***2.500***	***250***
	Fus' tube		0.200	500
	Pod	+sys	0.500	*350*
	Fus-wing joint		0.050	*400*
	VTP	+sys	0.150	*1250*
	Fus-emp joint		0.300	*1275*
	HTP	+sys	0.300	*1300*
on Wing	Port wing	+sys	1.000	*465*
	Batteries		1.000	*365*
	Star wing	+sys	1.000	*465*
	Batteries		1.000	*365*
	Total		10.000	
	on Fus		*6.000*	
	on wing		*4.000*	

... To be replaced with your masses and locations for your design.

Wing mass distribution considered on p23



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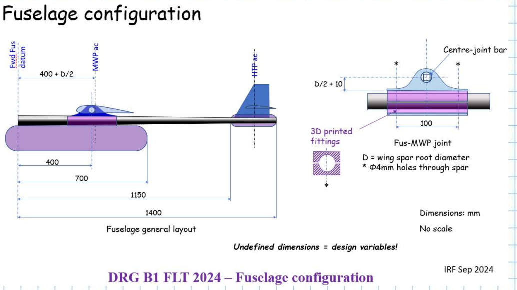

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To calculate the loads, we must first estimate the masses of the major components of the aircraft. I have estimated some rough initial trial values here to get you started. The table here includes the fuselage "Fus" and main wing plane "MWP" masses which sum to the total "all up mass" (AUM) of the aircraft. These masses are only very rough estimates and can be recalculated with more accurate estimates for your designed items after your first design iteration. A breakdown of the main wing plane masses can be found later on in these notes when considering wing loading.

Some of the largest masses include a fuselage mounted motor, on-fuselage payload, on-fuselage batteries and on-wing batteries. These are notional values which you can simply accept as representative for your design.

Note, in this exercise I have specified the nose of the aircraft as the forward (fwd) tip of the fuselage spar tube, even though your final design may result the nose cone projecting slightly further forward than this position.

Note, I have designed for the wing aerodynamic centre (ac) to be tangentially behind the main wing spar to promote nose-down twist as a safe "lift dumping" action.

Note, further acronyms on this slide: EMP for empennage, HTP for horizontal tailplane and VTP for vertical tailplane. A compiled list of acronyms for terminology and notation can also be found in the Contents file in STATICS 1. Aerospace Structural Design.

3. **AIRCRAFT CG** 4.

The CG of the entire aircraft must be forward of the MWP ac for +ve stability … but note: too far forward will make the aircraft too stable and difficult to achieve sufficient nose-up attitude for take-off and landing, whereas too far back will make the aircraft unstable

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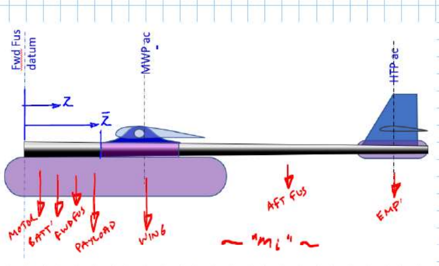


MWP = Main Wing Plane
HTP = Horizontal Tail Plane
VTP = Vertical Tail Plane
AUM = "All up mass" = $\Sigma(m_i)$

z from fwd fus' datum

Considering 1st moments of mass about fwd fus' datum: $\Sigma(m_i).\bar{z} = \Sigma(m_i.z_i)$

$$\bar{z} = \frac{\Sigma(m_i.z_i)}{\Sigma(m_i)} = \frac{\Sigma(m_i.z_i)}{AUM}$$

! Check $\bar{z}$ within specified c.g. limit for the aircraft (See Req'ts)

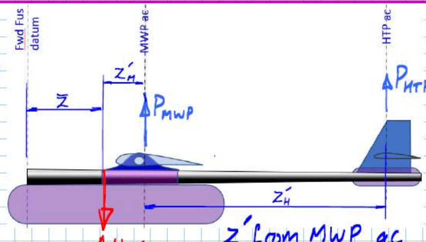
We must know the position of the centre of gravity (CG) to be able to balance the aircraft. In a conventional tail-plane configuration the aircraft CG must be forward of the aerodynamic centre of the main wing (MWP) so that the horizontal tailplane (HTP) provides a balancing down-lift for positive stability . However, too far forward (fwd) will inhibit necessary nose up attitude needed for take-off and landing and too far aft will make the aircraft unstable. Accordingly, there will be a safe range within which the CG must be located for the pitching response and stability to be acceptable.

To find the CG position, we consider the first moments of mass of the entire aircraft. We can then check the CG position against the acceptable CG range for the aircraft as specified in the Requirements document.

4. AIRCRAFT MWP and HTP LIFT

5.

Consider 1.0g "steady level flight."



z' from MWP ac

ac = aerodynamic centre = $\frac{1}{4}c$

c = chord

Don't forget kg×g → N @ 1.0g

Calculate @ 1.0g first

then factor up × n for load case

and " up × k_{USF} for ULT

Full FBD:

Taking moments about MWP ac

$\sum \overset{+}{\circlearrowleft}_{ac} = 0: \quad AUM \times g \times z'_M + P_{HTP} \times z'_H = 0$

$\hookrightarrow P_{HTP} = -\dfrac{AUM \times g \times z'_M}{z'_H}$ Note -ve ie downward!

$\sum \uparrow^{+} = 0: \quad -AUM \times g + P_{MWP} + P_{HTP} = 0$

$\hookrightarrow P_{MWP} = AUM \times g - P_{HTP}$

But since value of P_{HTP} is -ve it adds to the load carried by the wing!

Total wing load carried by both wings i.e. $P_{MWP} = AUM \times g - (-\#)$ @ 1.0g Limit.

Knowing the CG position, we can now evaluate the required lift at the respective aerodynamic centres of the main wing plane (MWP) and horizontal tail plane (HTP) for equilibrium. To do this we consider 1.0g steady level flight and take moments about the MWP aerodynamic centre. Here, I have defined distances from the MWP aerodynamic centre (ac) as z' with appropriate suffixes.

For a conventional tail-plane design, we find that the HTP lift acts downwards in normal flight. The resultant lift on MWP must then carry the downward HTP lift as well as the AUM.

Remember, the forces here represent the total acting on port <u>and</u> starboard wings. I.e., half the calculated load acts on the main wing on each side.

Note, my symbolic statements on the left represent equilibrium statements for moments and forces. I.e.:
"The sum of the moments about the MWP ac equals zero", taking anticlockwise as my chosen arbitrary positive rotational direction.
And:
"The sum of the forces vertically equals zero", taking upwards as my chosen arbitrary positive translational direction.

Note, to avoid ambiguity when drawing a free body diagram (FBD) or when writing an equilibrium equation, please proceed as follows: 1) On the FBD, indicate the sense of a load by the direction of the arrow and omit any signs. 2) In the equilibrium equation, start with all the terms on the left-hand side of the equation and equate to zero, indicating the sense by the sign of the terms according to your chosen positive direction. You do not have to second-guess the resulting direction. If a result is negative, it simply means that it is in the opposite direction to your drawn arrow on your FBD.

Oh, and don't forget to multiply your masses by *g* to get forces!

6.

From the lift load calculations above, it should become evident that the HTP will need to be sized and configured to provide a balancing lift to maintain equilibrium for the aircraft CG position.

[Note, the HTP should be set at an incidence that provides the balancing lift with no trim (i.e. zero elevator deployment) for the major phase of the flight mission, e.g. the cruise or loiter phase, to avoid unecessary drag. Refer to the Aerodynamics content for further advice on tail incidence setting]

Similarly, the VTP will need to be sized to give adequate lateral stability.

For initial VTP and HTP planform sizing at the conceptual design stage, we shall use "**tail coefficients**" which relate tail size to the main wing and fuselage dimensions for given aircraft types. For our UAV, we shall use reduced sailplane coefficients.

Text as written on slide.

7.

VTP sizing:

$$S_{VTP} = C_{VTP} \times \frac{b_w S_w}{L_{VTP}}$$

HTP sizing:

$$S_{HTP} = C_{HTP} \times \frac{\bar{c}_w S_w}{L_{HTP}}$$

where: S_{VTP} = plan area of VTP <u>above</u> fuselage

S_{HTP} = " " " HTP <u>total</u> span – including through fuselage.

C_{VTP} = VTP tail coefficient , C_{HTP} = HTP tail coefficient

b_w = total MWP span

S_w = area of total MWP span – in both expressions

L_{VTP} = distance between the aerodynamic centres (¼ chord locations) of the mean aerodynamic chords (m.a.c) of the MWP and VTP

L_{HTP} = " ——————————— " MWP and HTP

$\bar{c}_w$ = m.a.c of the MWP, ie simply the chord of the main wing for our UAV rectangular planform wing.

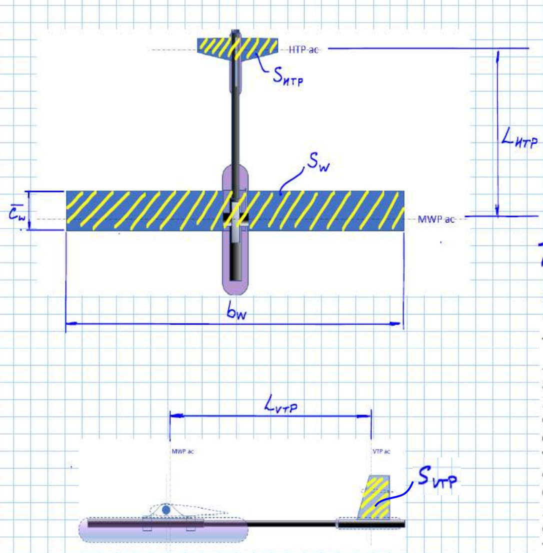
m.a.c = mean aerodynamic chord

Typical VTP and HTP planform sizing equations are given here, based on “tail coefficients”, MWP area, and the distances between the MWP ac and VTP and HTP ac’s.

S_{VTP} = Plan area of VTP above the fuselage
S_{HTP} = Plan area of HTP including through the fuselage
C_{VTP} = VTP tail coefficient
C_{HTP} = HTP tail coefficient
b_w = Total tip to tip main wing plane span
S_w = Total area of main wing plane span
L_{VTP} = Distance between ac of MWP and VTP
L_{HTP} = Distance between ac of MWP and HTP
$\hat{c}_w$ = mean aerodynamic chord of the MWP

These sizings will give a reasonable starting point for tailplane design based on coefficients to suit particular aircraft types, as illustrated on the next slide.

8.



Tail coefficients for typical aircraft types

Vertical and aft-horizontal tail coefficients.

	C_{VT}	C_{HT}	
Sail Plane	0.02	0.50	× 0.75
Homebuilt	0.04	0.50	
General Aviation (single engine)	0.04	0.70	
General Aviation (twin engine)	0.07	0.80	
Twin Turboprop	0.08	0.90	
Combat Jet Trainer	0.06	0.70	
Combat Jet Fighter	0.07	0.40	
Military Transport/Bomber	0.08	1.00	
Commercial Jet Transport	0.09	1.00	
UoB FLT 2024-25:	0.015	0.375	

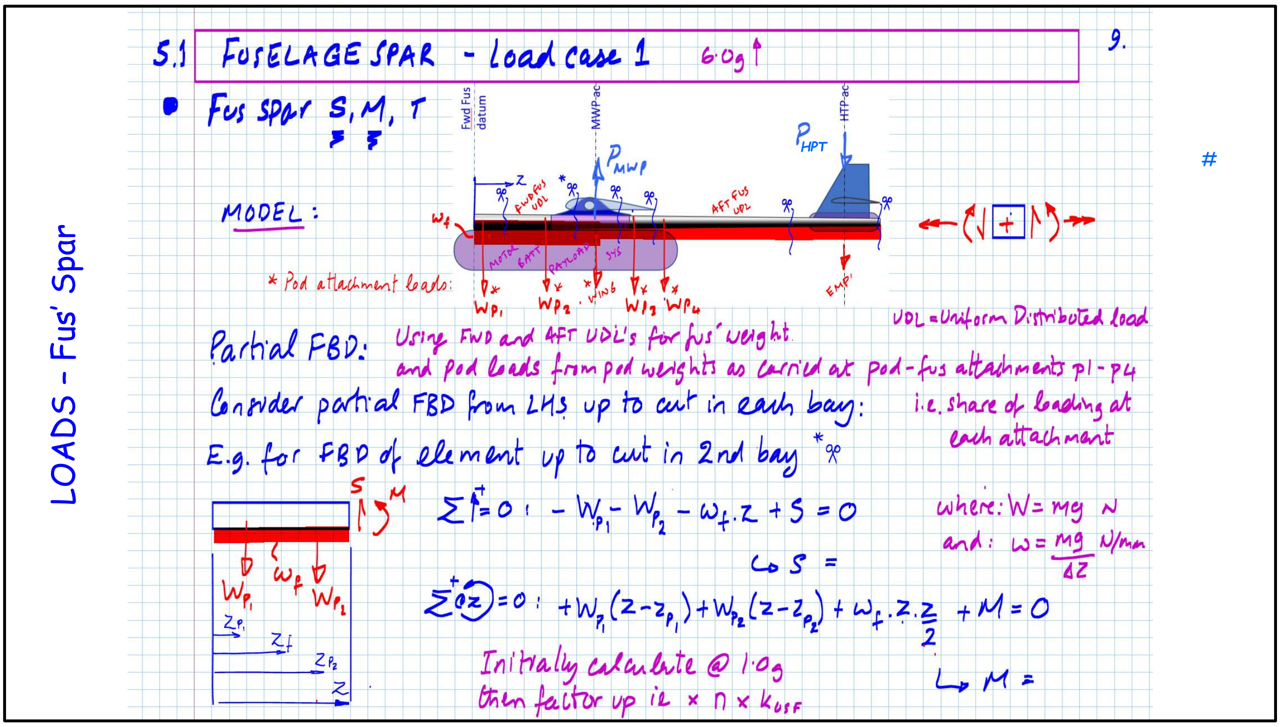
Watch out for the "cross-over" terminology in design!
e.g. S = span, area, shear force !... but should be obvious!?

The dimensions used in the tailplane sizing equations on the previous slide are illustrated here, along with a table of typical coefficients for different aircraft types.

We will use the coefficients for a sailplane with some reduction to suit our UAV.

Please be aware of cross-over of terminology where the same term may be used to represent different items within aero or structures calculations. E.g., *S* = span or area or shear force etc., although it should become reasonably obvious as your familiarity with the equations grows.

LOADS – Fus’ Spar

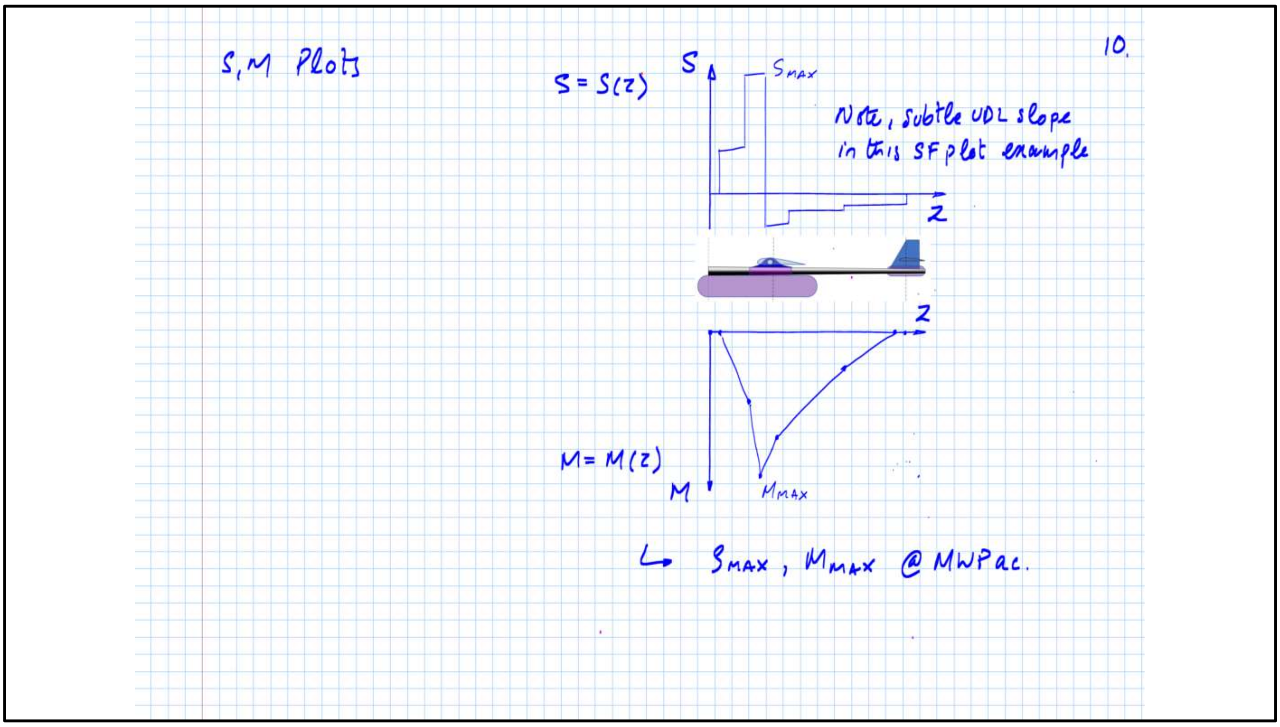


Moving on to the actual loading calculations, and starting with the fuselage spar, we will evaluate the internal shear, bending moment, and torsion (S,M,T) loading along the spar for load case 1) as the most critical case. Remember, only use arrows to indicate sense on the free body diagram (FBD). I.e., do not include signs with the arrows to avoid ambiguity.

For the S,M loading, consider the fuselage spar as a beam and draw the FBD, accounting for point and smeared loads from the masses and lift. Next, consider sections at bays along the beam where we make a notional cut to reveal the internal loading. Create a partial FBD of the section to the left or right of the cut, with the internal loading in the positive direction on the exposed edge according to the internal sign convention. Now write out the equilibrium equation and transpose to calculate the revealed internal S,M loading. Remember to initially write the equation with all terms on the left-hand side (LHS) and equate to zero, interpreting the signs from the directions of arrows on the diagram.

By proceeding with this method for each bay along the beam we can build up a set of equations to illustrate the shear and moment distributions along on the spar.

I advise initially performing your calculations for the 1.0g loading at limit then simply factoring up by the acceleration factor and Safety factor to arrive at the ultimate design load for the load case.

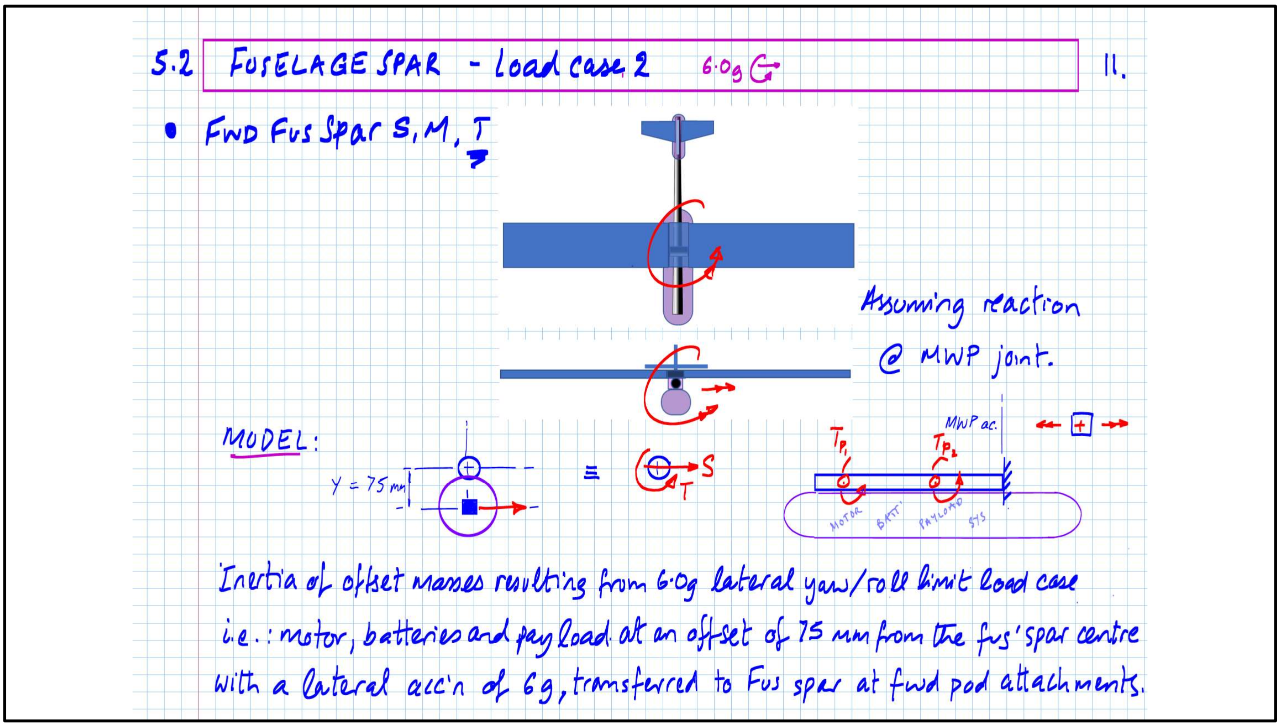


Having obtained a set of equations for internal loading along the beam you should plot the shear force and bending moment diagrams.

For the combined point and smeared loadings, you will see a stepped shear force diagram where the steps in a smeared load region will be sloped, and a faceted bending moment diagram where the facets in a smeared loading region will be curved.

We will of course be most interested in the peak shear and peak bending moment values to check our spar design. If the position of the maximum internal loading is obvious, we could of course just consider a section and FBD up to that position. Even so, it is always advisable to plot the distributions to understand the loading along the complete beam.

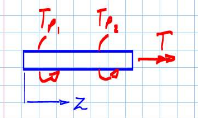
We could simplify further than the illustration given here by adding up and smearing more of the point loads as uniform distributed loads by taking the sum of the loads and dividing by the length over which they apply. Starting with a simplified model is advisable because you can arrive at your estimate quickly with less chance of making an error and acquire more feel for the values. You can then increase the fidelity of the model in stages to improve the detail of the representation and check against the results from the simpler models to provide confidence.



For a simple estimate of internal torsion loading, we shall consider the forward (fwd) and aft portions as separate shafts which are effectively reacted at the MWP joint.

For the fwd fuselage, load case 2) can be considered as the most critical for torsional loading. I.e., the 6.0g lateral yaw/roll case which is simplified as a sideways acceleration here. Considering the inertial loading from the main pod masses (motor, batteries and payload, systems) at their respective positions and the lateral acceleration we can evaluate the accumulated torque along the fwd fuselage up to the MWP joint where the fus' spar is assumed to be fixed. The torque resulting from the offset loading in the pod is transferred to the fus' spar tube at the pod-fus' attachment points. For conservatism and simplification, we can assume that only the two forward attachments are active (the other two being loose or failed).

12.



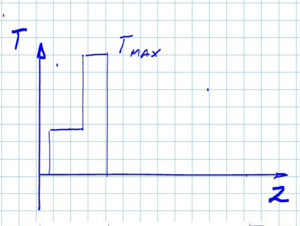
$\sum_{@z} \overset{+}{\twoheadrightarrow} = 0: \quad -T_{P_1} - T_{P_2} + T = 0$ i.e superposition of torques

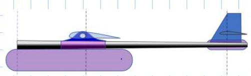
↳ T @ MWP joint

where T_{P_i} = share of $\sum m_i \cdot g \cdot Y_i$ at each pod-fus attachment.

Calc for 1.0g @ LIM then factor up
for load case @ ULT
i.e. $\times n \times k_{USF}$

$T = T(z)$

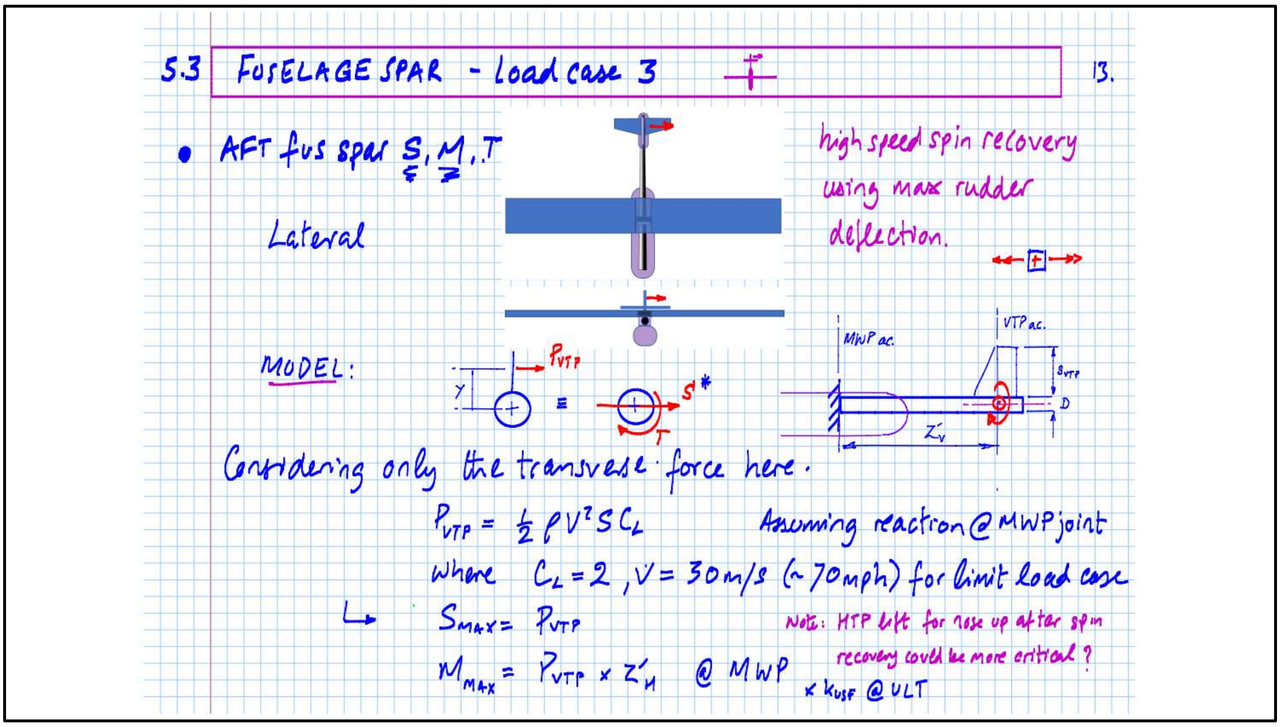




↳ T_{MAX}, @ MWPac.

I.e. Loading at pod attachment frames
with share of loading at each attachment.

Considering a free body diagram for the fwd spar as a shaft in torsion, we can write out the torsion equilibrium equation. Again, consider a set of notional cuts in each bay between the changes of loading to reveal the internal torque loading at any position. The maximum torque is of course at the MWP joint position where the spar is assumed to be fixed. We could of course proceed directly to this position to just determine the maximum value. Even so, you are advised to consider each bay to draw a torque diagram illustrating the internal torque distribution along the fwd spar. Again, starting from a simplified model, we could refine to increase fidelity and accuracy.



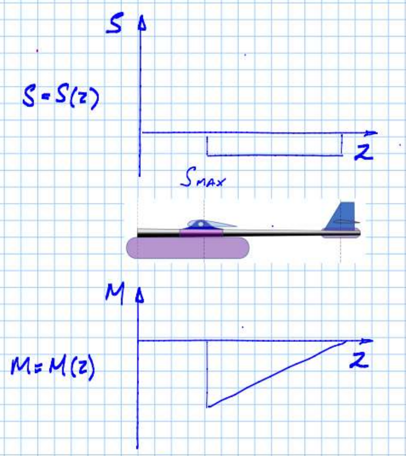
For the aft spar, load case 3) presents the critical case. I.e., stall recovery with maximum rudder deflection at high speed. Here we are considering the VTP lift with full rudder deflection at 30 m/s to represent the limit loading (rather than an inertial loading with an acceleration factor). Here I have defined the empennage loading as aerodynamic loading only, neglecting the mass of the VTP for our lightweight UAV structure.

To simplify the problem, we consider the loading from the tail separately, either 1) as a transverse load without offset, to obtain internal shear and moments in the spar as a beam under transverse loading, or 2) as a torque due to the offset load, to obtain internal torque in the spar as a shaft.

Initially considering the aft spar as a beam and the VTP lift as a transverse loading without offset so the shaft is essentially a cantilever fixed at the MWP joint with a point load at the VTP aerodynamic centreline. FBD and equilibrium analysis then reveals the internal shear force and bending moment distribution, with maximum bending moment obviously at the fixed end.

S,M Plots

14.



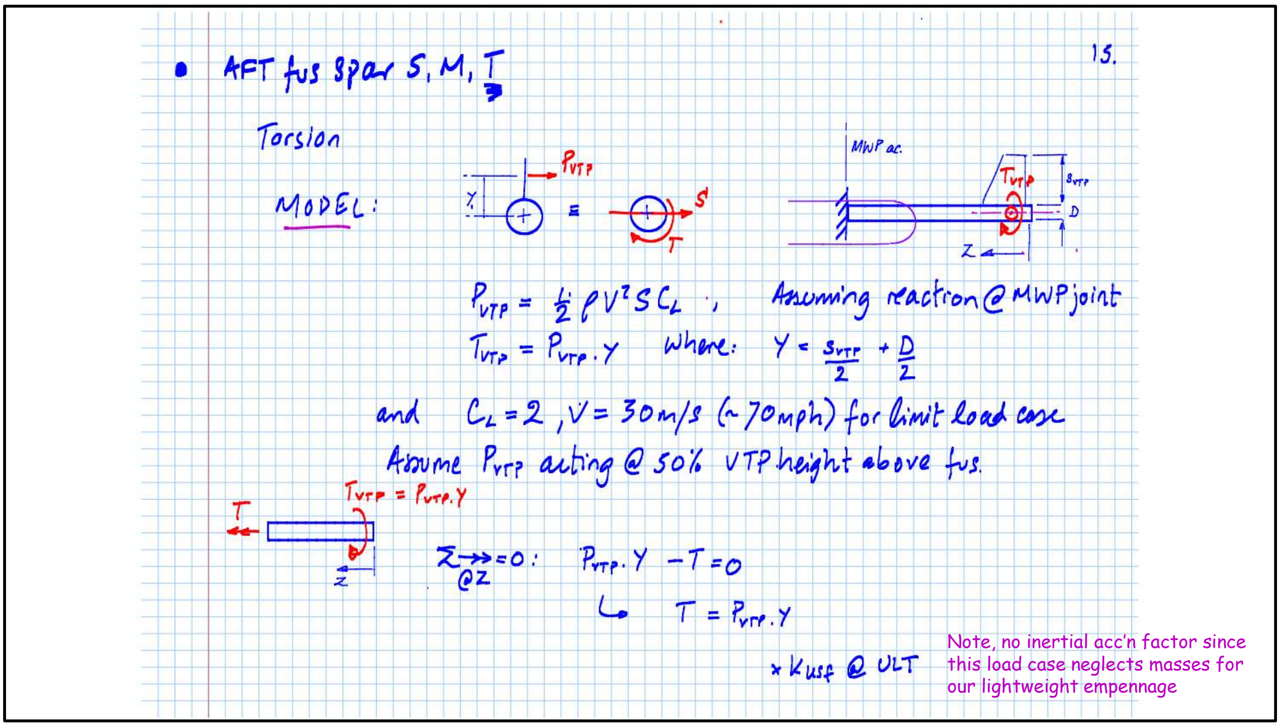
S_{MAX} constant for tip load

M_{max} @ MWP ac.

Plot for 1.0g and factor up for load case @ ULT i.e. $\times \cancel{n} \times k_{USF}$

Note, no inertial acc'n factor since this load case neglects masses for our lightweight empennage

I.e., internal S,M distributions for a simple point loaded cantilever.

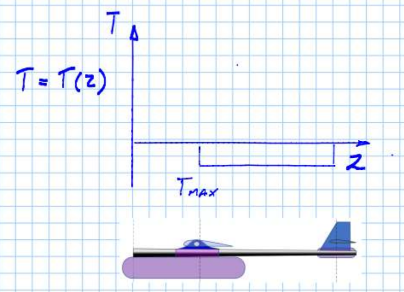


Next, considering the torque due to the offset load on the spar as a shaft.

The applied torque is calculated as the resultant VTP lift load times its offset to the centre of the spar tube, as applied at the line of the VTP aerodynamic centre. As a point torque, FBD and equilibrium analysis reveals a constant internal torque along the spar as illustrated on the next slide.

T Plot

16.

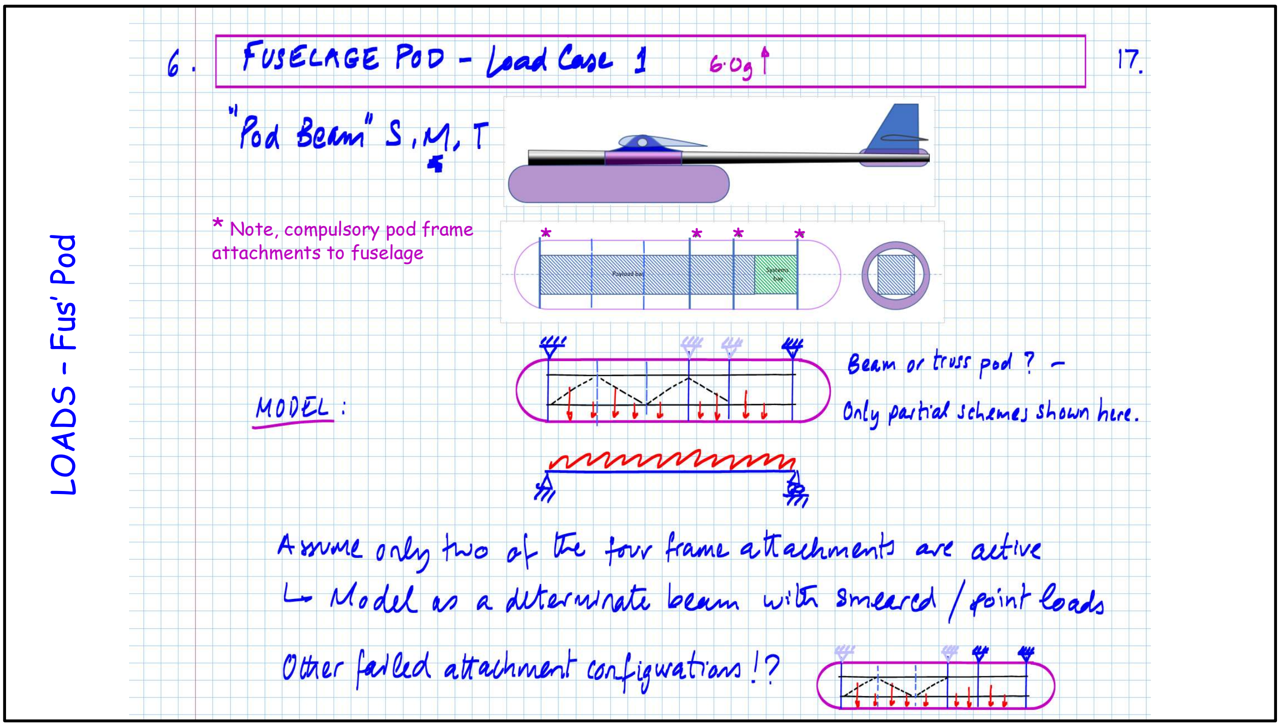


Plot for 1·0g and factor up for load case @ ULT i.e. $\times n \times k_{USF}$

↳ T_{MAX}, @ MWPac.

…

LOADS – Fus' Pod

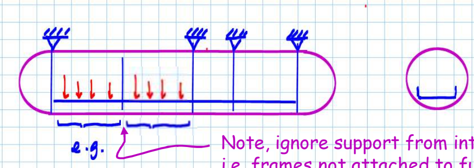


Moving on to the fuselage pod, we can consider it as a beam with smeared UDL inertial loading from the pod deck where the beam is supported at the frame attachments to the fuselage. Here we will consider load case 1) but the other load cases may also be severe!?

The four compulsory frame attachments to the fus spar makes the pod beam statically indeterminate but for conservatism and simplification, we can assume that only two attachments are active (the other two being loose or failed). The problem then reduces to a standard deterministic beam with UDL. If we assume that two attachments are inactive at the middle of the pod, we end up with a simply supported beam. Alternatively, if we assume that two attachments have failed at one end of the pod then we end up with a propped cantilever beam, which may be more severe!?

18.

- Pod Deck S, M, T



Note, ignore support from intermediate frames,
i.e. frames not attached to fuselage

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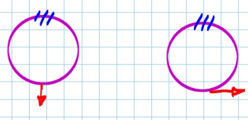
MODEL: ss beam between 'frames'.

- Frame S, M, T



MODEL: Isolated ring beam – see Refs,
e.g. Roark "Formulas for Stress + Strain"

Considering the deck as a beam carrying load between supports at adjacent frames, we can initially model a portion of the deck as a simply supported beam carrying UDL between the frame supports.

The frames can be modelled as standard ring-beams, assuming each frame carries a share of the inertial loading on the deck. See the Aeronautical Design Handbook in the appendix X2.2 and Ref "Formulae for Stress and Strain" by Roark.

LOADS – Emp' Spar

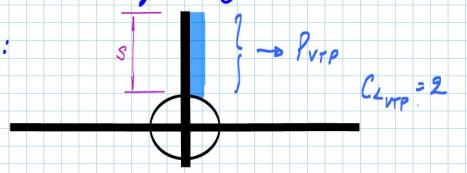
LOADS - Emp' Spars

7. **EMPENNAGE SPARS – Load Case 3** 19.

- VTP spar $\underset{F}{S}, \underset{S}{M}, T$ High speed/high C_L
  – full rudder spin recovery limit case.

MODEL: Cantilever with UDL lift, neglecting inertia relief

VTP spar loading:



where $P_{VTP} = \frac{1}{2}\rho V^2 S C_L$ $\quad S$ = plan area, s = single span.

and $C_L = 2$, $V = 30\,m/s$ (~70 mph) for limit load case

$S_{max} = P_{VTP}$

$M_{max} = P_{VTP} \cdot \frac{s}{2}$

$\times k_{ULF}$ @ ult

Assuming UDL lift and neglecting inertia relief of empennage.

Moving on to the empennage spars we will consider spin recovery load case 3) as most critical for both the VTP and the HTP main spars. Assuming a conservative uniform distributed wing lift distribution, we can easily arrive at the maximum internal shear and bending moment at the root of the main spars.

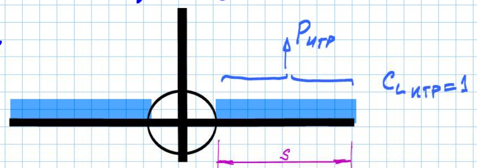
So, for the VTP as a cantilever beam under uniform distributed loading (UDL) calculated for lift at 30 m/s and a full rudder deflection with a lift coefficient C_L =2, the maximum shear and bending moment values can easily be found. Note, here I have defined the empennage loading as aerodynamic loading only, neglecting the mass of the VTP for our lightweight UAV structure.

20.

- HTP spar S_F, M_S, T - Easing the nose up after spin recovery limit case

MODEL: Cantilever with UDL lift, neglecting inertia relief

HTP spar loading:



where $P_{HTP} = \frac{1}{2}\rho V^2 S C_L$ $\quad S$ = plan area, s = single span.

and $C_L = 1$, $V = 30\,m/s$ (~70 mph) for limit load case

$S_{max} = P_{HTP}$

$M_{max} = P_{HTP} \cdot \frac{s}{2}$

$\times k_{USF}$ @ ult

Assuming UDL lift and neglecting inertia relief of empennage.

Plot!

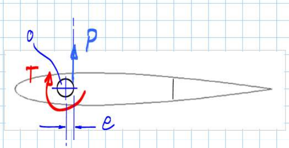
Similarly, for the HTP, assuming that full elevator is applied to return to straight and level after the spin is corrected, we perform a similar calculation for the HTP as we did for the VTP to obtain the maximum internal shear and bending moment. Again, the empennage loading is taken as aerodynamic loading only here, neglecting the mass of the HTP for our lightweight UAV structure.

As well as obtaining the maximum internal loadings we can also plot the shear force and bending moment diagrams for the standard UDL cantilever beam models. Etc.

21.

- VTP, HTP spar S, M, T

MODEL:



$\sum \overset{+}{\circlearrowleft}_{O} = 0$: $T = P.e$

Neglecting VTP or HTP weight

Where $P = P_{VTP}$ or P_{HTP}

$e = D/2$

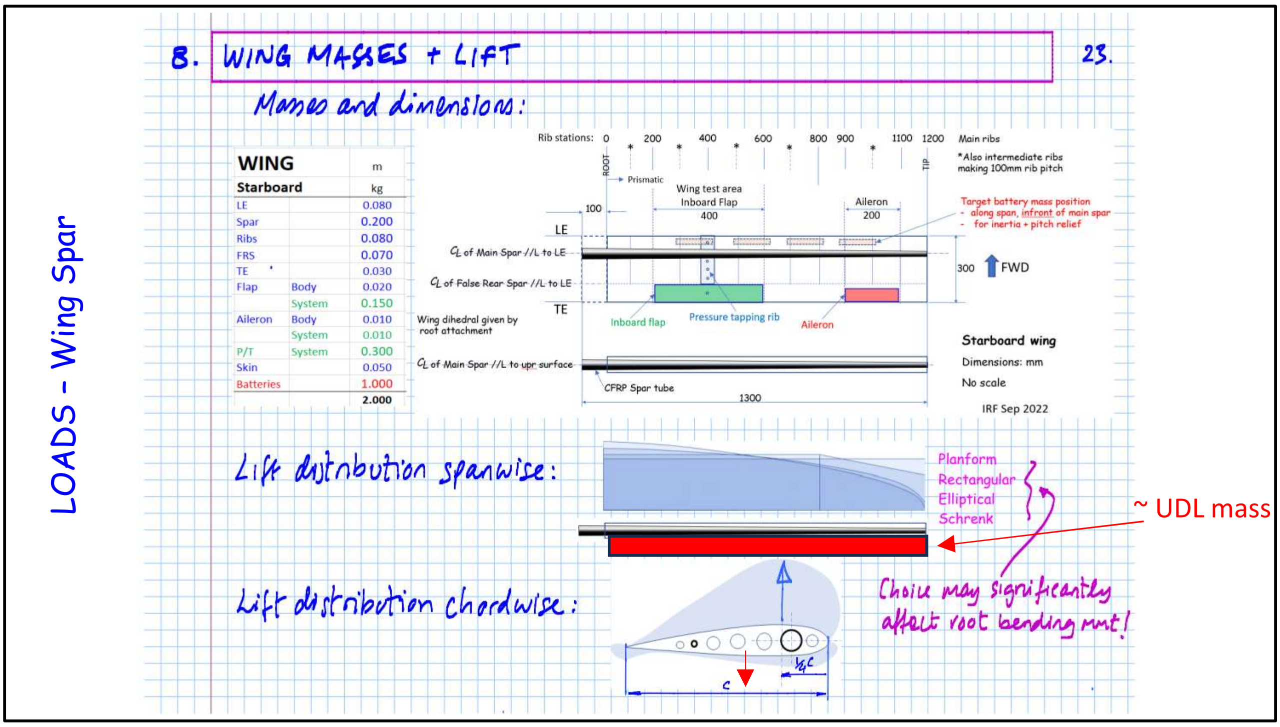
$D = \phi_{SPAR}$

For torsion, referring to the tailplane cross-sections and the advised design configuration for this UAV, where the main spars are to be located tangentially in front of the aerofoil ac, we can calculate the torsion due to lift according to the offset from the centre of the spar tube.

ac = aerodynamic centre

22.

...



An initial estimate of the MWP wing masses is given in the table on this slide, summing to a total of 2kg for each wing, i.e., a total of 4kg for both wings. Note, the majority of the masses are assumed to be smeared along the span of each wing, including the on-wing batteries, where the design intent is for them to be carried in the nose section ahead of the main wing spar to promote nose-down inertial torque. The flap and aileron and respective systems are also assumed to be smeared initially for simplicity.

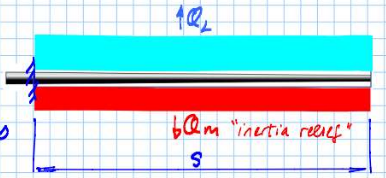
The spanwise wing lift distribution is commonly represented as a quarter ellipse, reducing towards the tip. A more accurate distribution would be a Schenk distribution which also accounts for the wing planform. For a conservative initial estimate, and since the wing planform is rectangular for our design, we shall use a rectangular UDL lift distribution. Note the spanwise position for resultant lift will be at 50% span for a UDL distribution, compared to ~40% span for an elliptical distribution.

The chordwise lift distribution is sketched at the bottom of the slide and the resultant will typically be at the quarter chord position for most aerofoils.

8.1 **WING SPAR - Load case 1** $6.0g\uparrow$ 24.

- Wing $\underline{S}$, $\underline{M}$, T.

<u>MODEL</u>: UDL Cantilever:

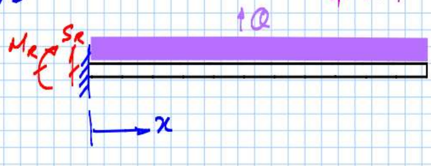


assuming UDL lift and mass distributions
conservative!

Where $Q_L = \dfrac{P_{MWP}}{2} / s$, $Q_M = M_W \cdot g / s$ N/mm i.e for one wing

Looking at starboard wing spar tube from TE, i.e. z: ⊙

Note P_{MWP} accounts for -ve P_{HTP}!



$\Sigma\uparrow$: Resultant $Q = Q_L - Q_M$

Taking x from root:

First consider full FBD to evaluate reactions @ root, i.e. @ $x=0$

$\Sigma\uparrow^{+}=0: \quad S_R + Q.s = 0 \rightarrow S_R = -Q.s$

$\Sigma \circlearrowleft_{x=0} = 0: -M_R + Q.s.\frac{s}{2} = 0 \rightarrow M_R = Q\frac{s^2}{2}$

i.e. $S_R = P$, $M_R = P.\frac{s}{2}$ where P = resultant loading for UDL mass + lift distributions.

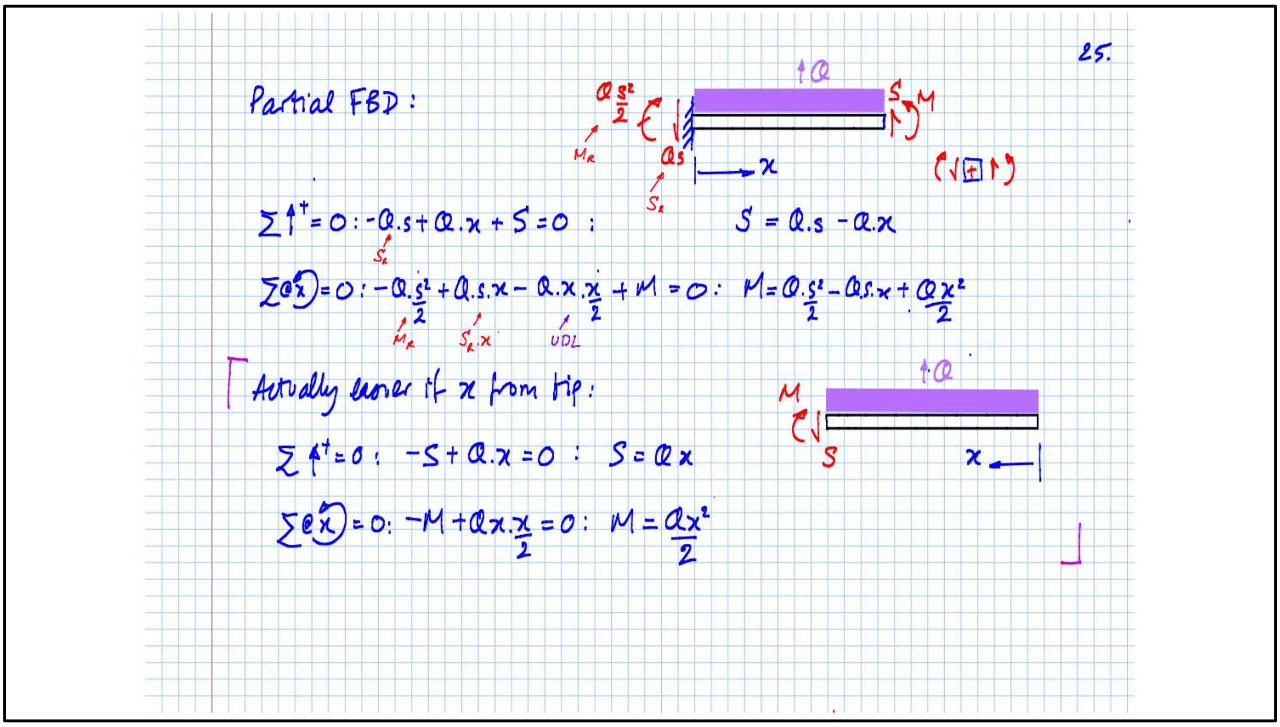
I.e. further simplification just to obtain root loading

Assuming both lift and inertia loadings are UDL the resultant UDL becomes a simple algebraic addition where the wing mass loading intensity* subtracts from the wing lift loading intensity* as "inertia relief".

Load case 1) presents the most critical case for the wing. Modelling the wing as a cantilever beam subjected to the resultant UDL, we can apply FBD's and equilibrium equations to quickly arrive at the internal shear force and bending moment anywhere along the wing.

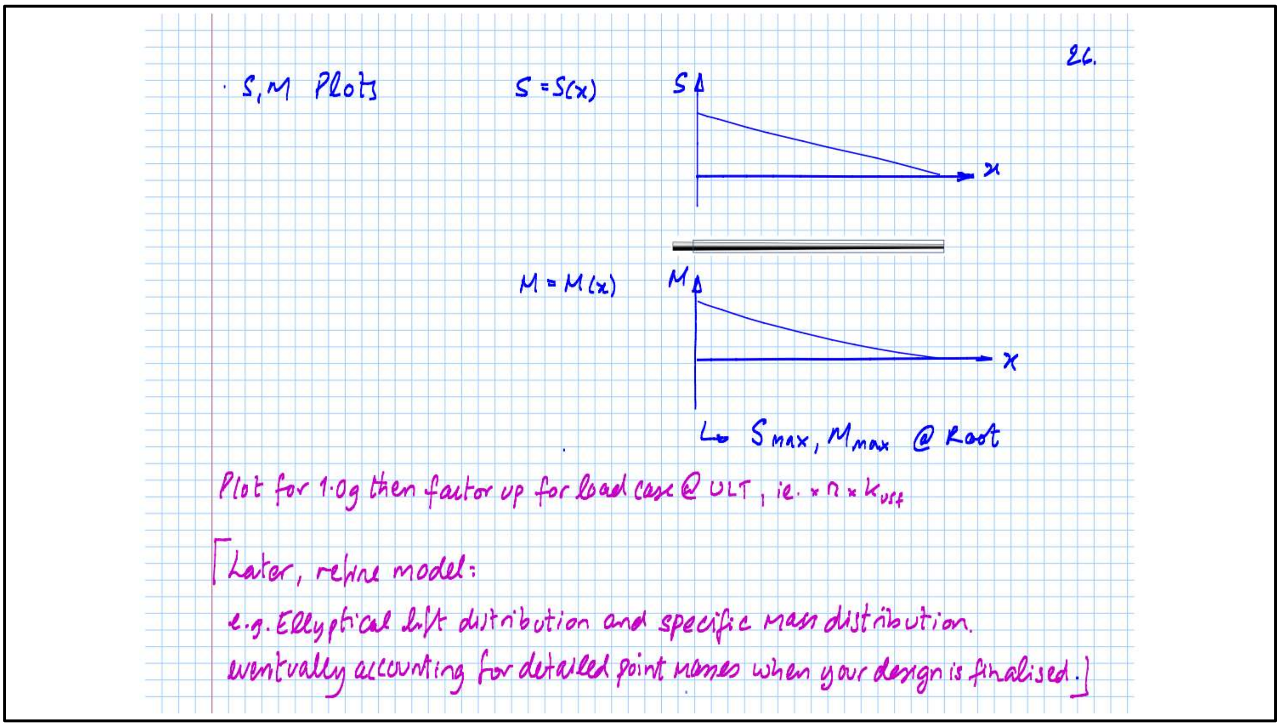
Starting from the root we first need to evaluate the root reactions.

*Loading intensity is simply a smeared load, i.e. UDL, where the total load is divided by the length over which it acts to provide a load per unit length.



Considering a FBD and equilibrium of a section along the beam provides a general expression for internal shear and bending moment anywhere along the beam.

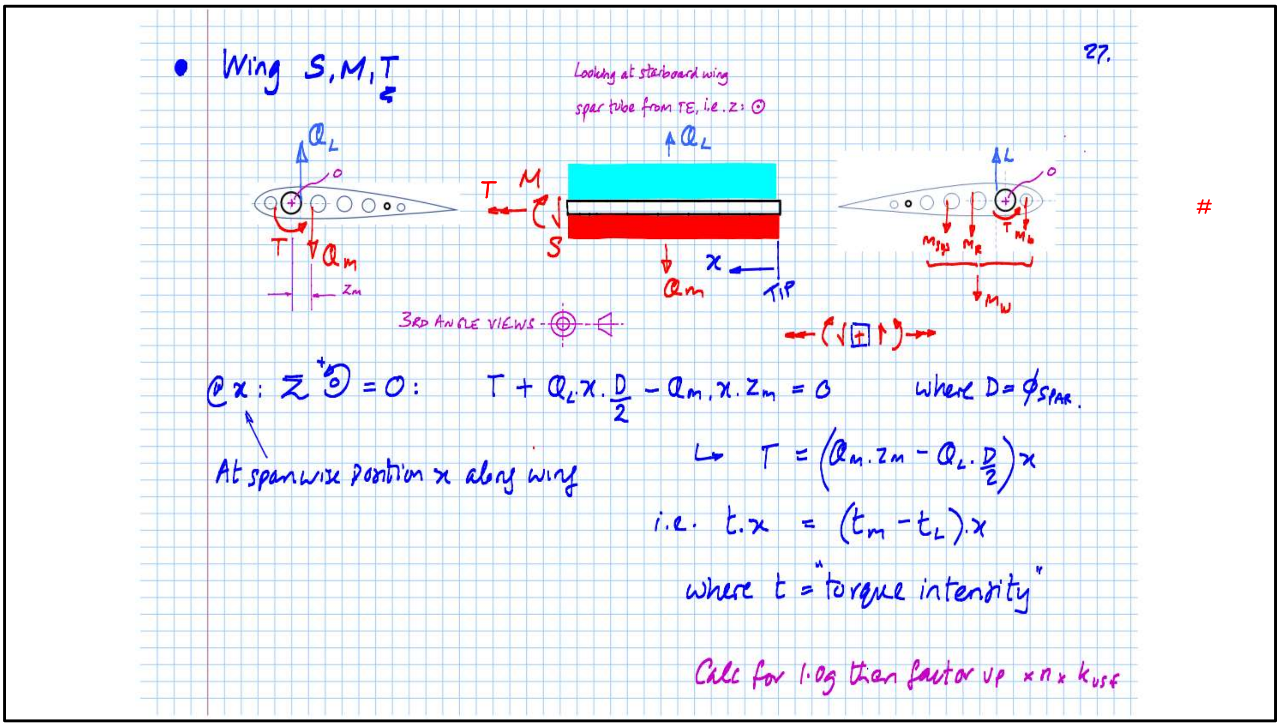
Actually, the process is slightly easier if you work from the tip, since the reactions at the root are then not needed to solve for values along the beam.



The internal shear force and bending moment diagrams illustrate the loading distribution which of course reduces from a maximum at the root to zero at the tip.

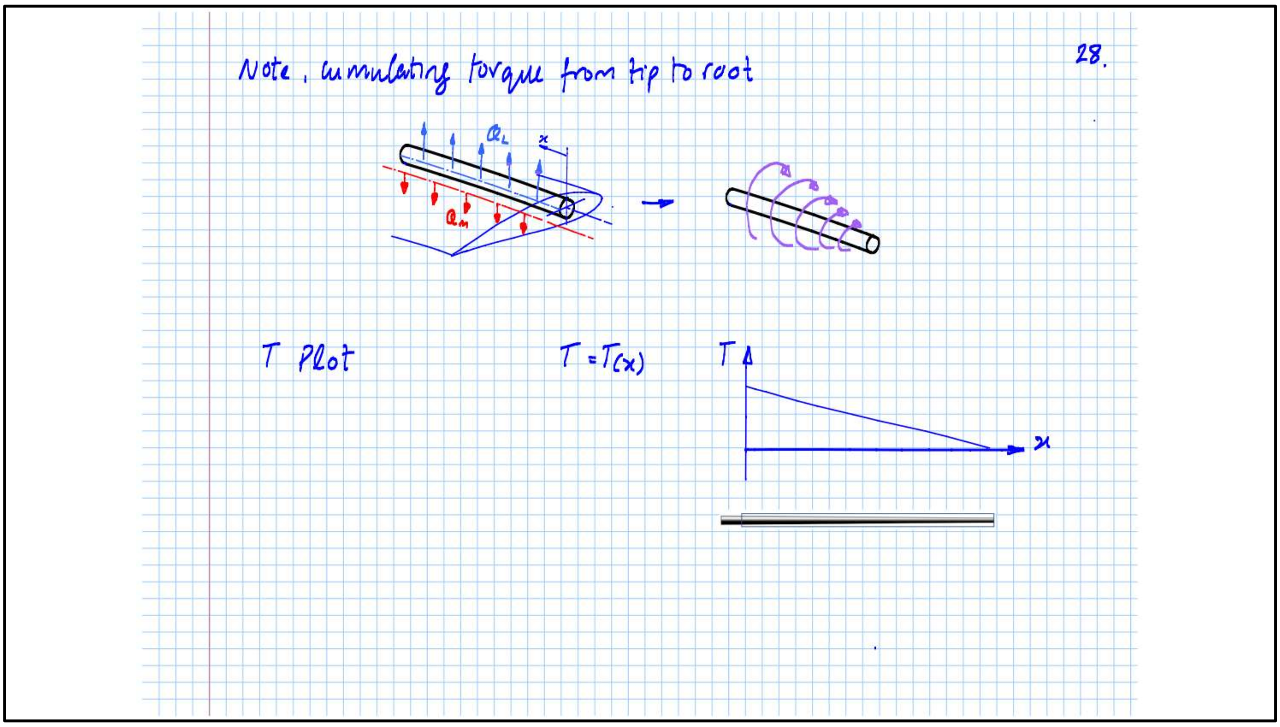
As advised previously, calculate and plot for 1.0g then factor up to account for the load case acceleration factor and ultimate safety factor.

Later on, you could refine your model to account for the specific mass distribution of your design, accounting for point masses and an elliptic lift distribution to improve the representation of the real configuration.



For the internal torsion loading we need to consider the chordwise position of the lift and wing mass. The lift can be assumed to occur at the ac, i.e., at the 25% "quarter" chord position from the nose of the aerofoil. Also note, by design our UAV wing structure is arranged so that the spars are tangentially infront of the 25% chord position so that lift results in a nose down torque, tending to reduce the angle of attack and resulting lift (a "lift dumping" response). The resultant mass loading acts according to the chordwise distribution of structural mass, system mass, and battery mass. As a first estimate, you can reasonably assume that the wing CG occurs at 40% of the chord from the nose.

ac = aerodynamics centre



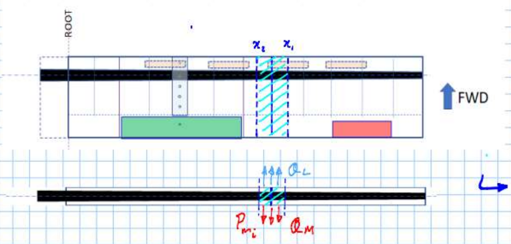
The torque accumulates linearly to a maximum at the root.

LOADS - Wing Ribs

9. RIB LOADING - Load Case 1 6.0g↑

29.

Ribs can be considered as carrying the aero loading and wing mass acting on half of the rib bay on either side and transferring this to the main spar.



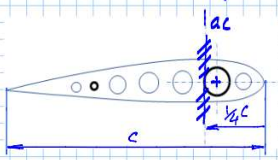
$$\hookrightarrow \quad P_{RIB} = \sum_{x_1}^{x_2} P_{m_i} + \int_{x_1}^{x_2} q\, dx$$

i.e rib loading can be calculated as the sum of point loads and integral of distributed loads on half the rib bay on either side of the rib.

The ribs are generally relatively lightly loaded by the wing lift and inertia loading, although specific ribs might see additional local loading from attached items such as flaps, undercarriage or motors.

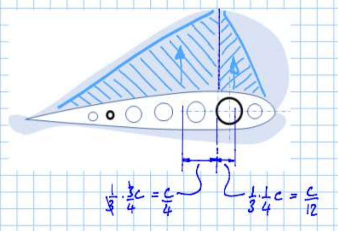
Here we will initially consider just the wing lift and inertia loading on a rib (local loadings can be accounted for when considering your specific design. Each rib effectively carries the lift and mass loading up to half the rib bay on each side of the rib and transfers it to the main spar.

The fwd rib and aft rib elements are effectively cantilevered off the main spar. 30.

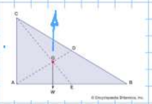


Assuming cantilevered from aerodynamic centre position.

Consider approximate triangular distributions of lift loading.
The effective centre of loading of the Δ distributions can be taken as acting at $\frac{1}{3}$ of their length from the ac.

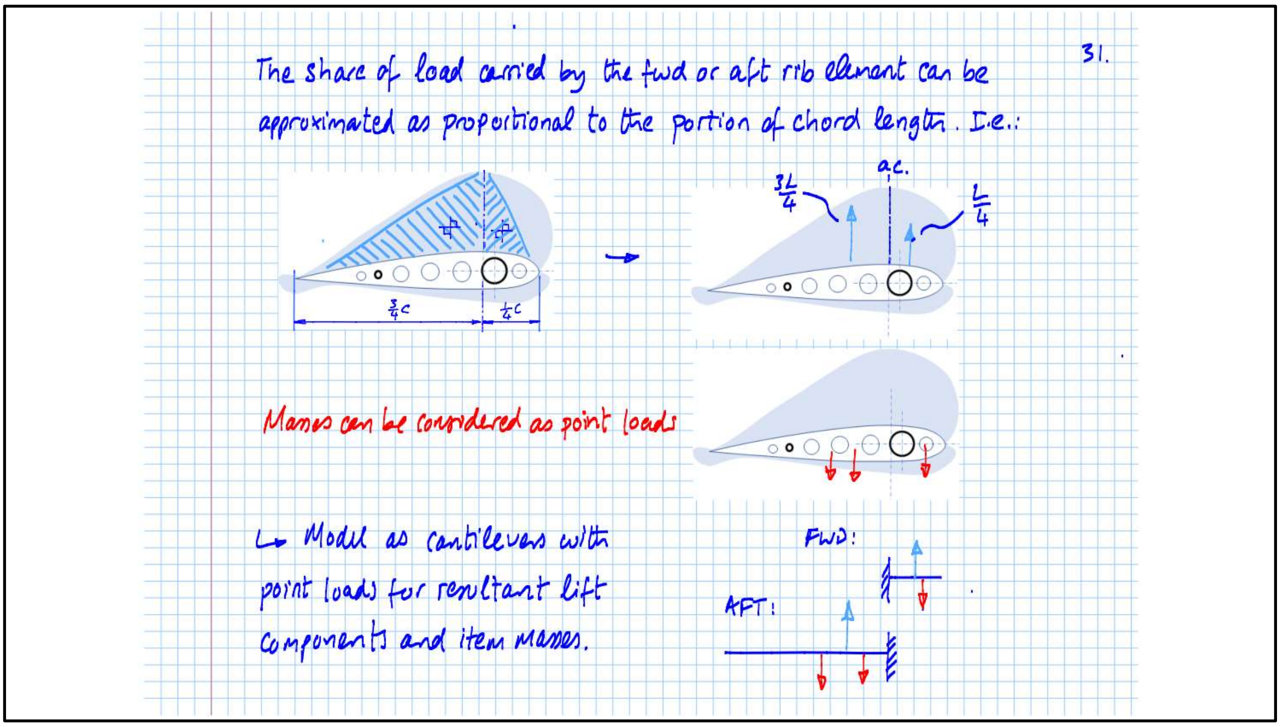


Centres of Δ lift distrib'n @ $\frac{1}{3}$ length from ac.



As a rough representation, we can model the rib as a beam cantilevered off the main spar forward or aft of the ac position, which corresponds to the back of the spar tube by design for our UAV.

Considering the chordwise lift distribution on each side of the ac to be approximately triangular, we can take the resultant lift on each side of the ac to be one third of the rib length on each side. I.e., the fwd rib lift resultant occurring at $^1/_3$ x $^1/_4c = c/_{12}$ in front of the ac and the aft rib lift resultant occurring at $^1/_3$ x $^3/_4c = c/_4$ behind the ac.



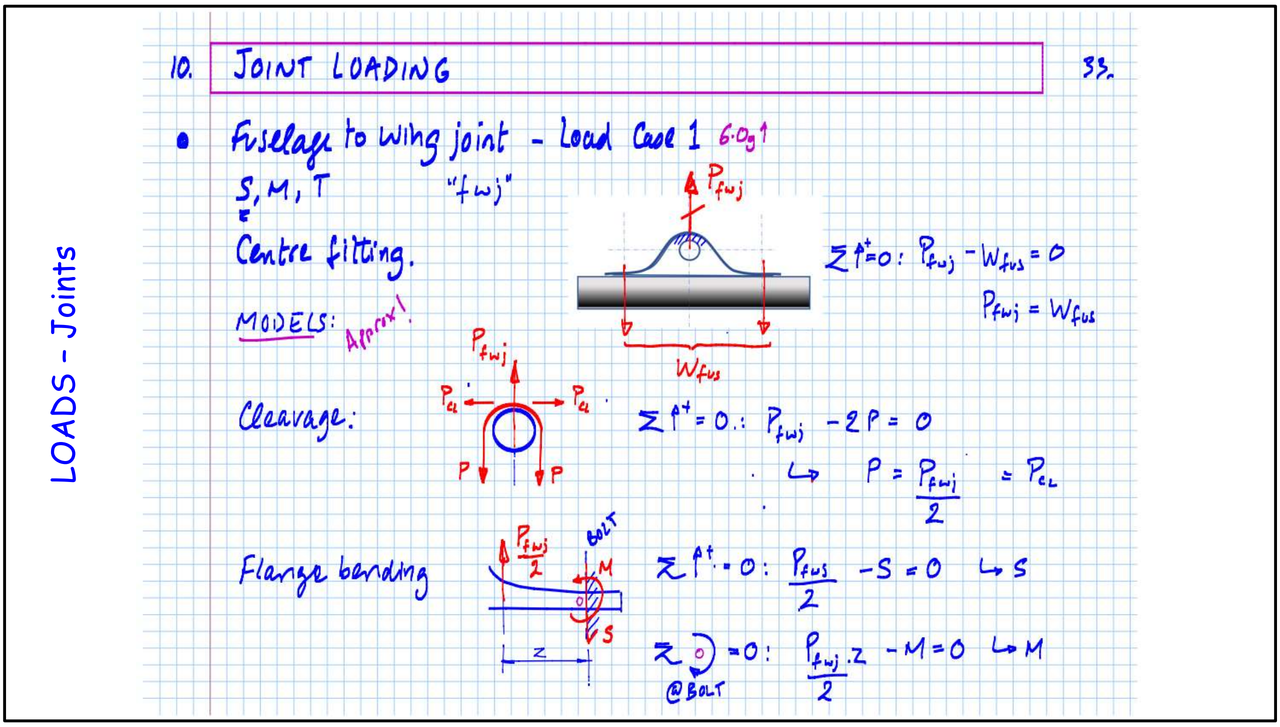
As approximated triangular distributions, the fwd and aft lift resultants can be assumed to be in proportion to the base length of the distribution, i.e., as $^{1}/_{4}$L and $^{3}/_{4}$L respectively, where L is the total lift in the rib region.

Finally, the fwd and aft rib portions can be modelled as stubby cantilever beams on each side of the ac with point loadings from lift and masses.

You can of course enhance the fidelity of these models later on by referring to specific distributions and point mass positions for your actual design.

32.

…



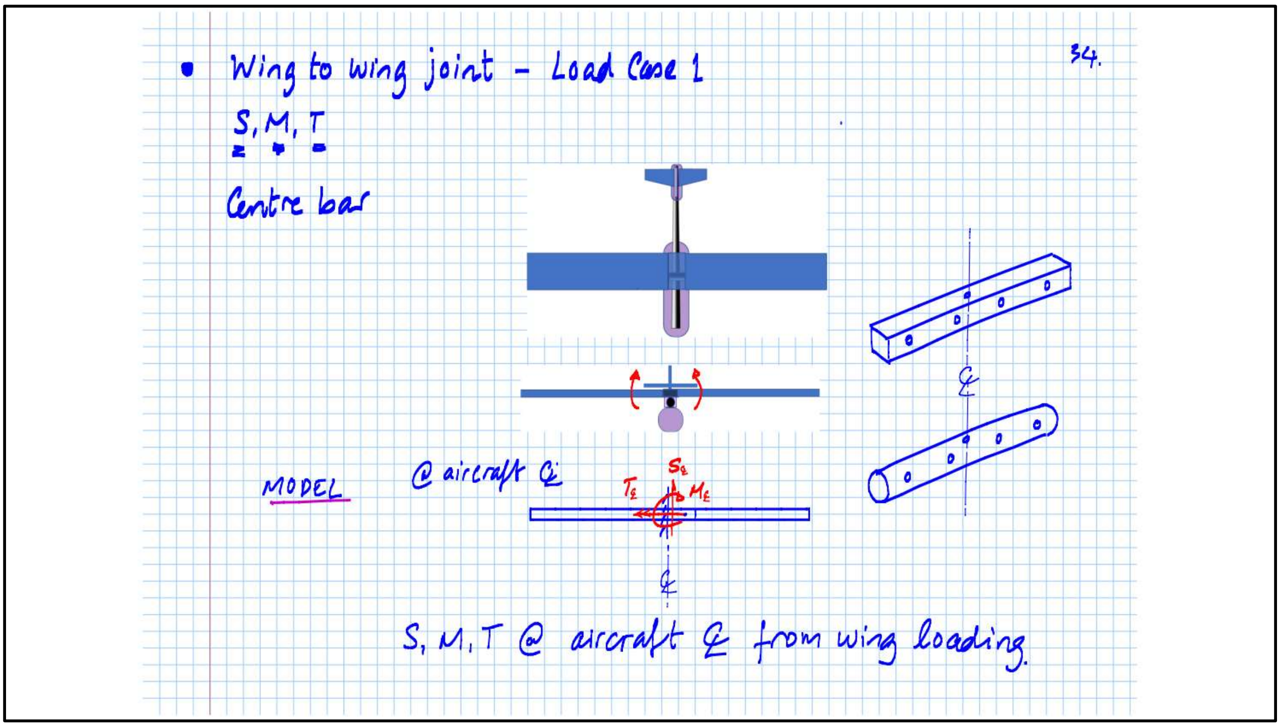
Here, we shall consider the loading carried by the major joints between the main elements of the structure.

The port and starboard MWP wings are joined to the fuselage by a centre bar which passes through a fuselage lug joint fitting inserted into each main wing spar at the wing root.

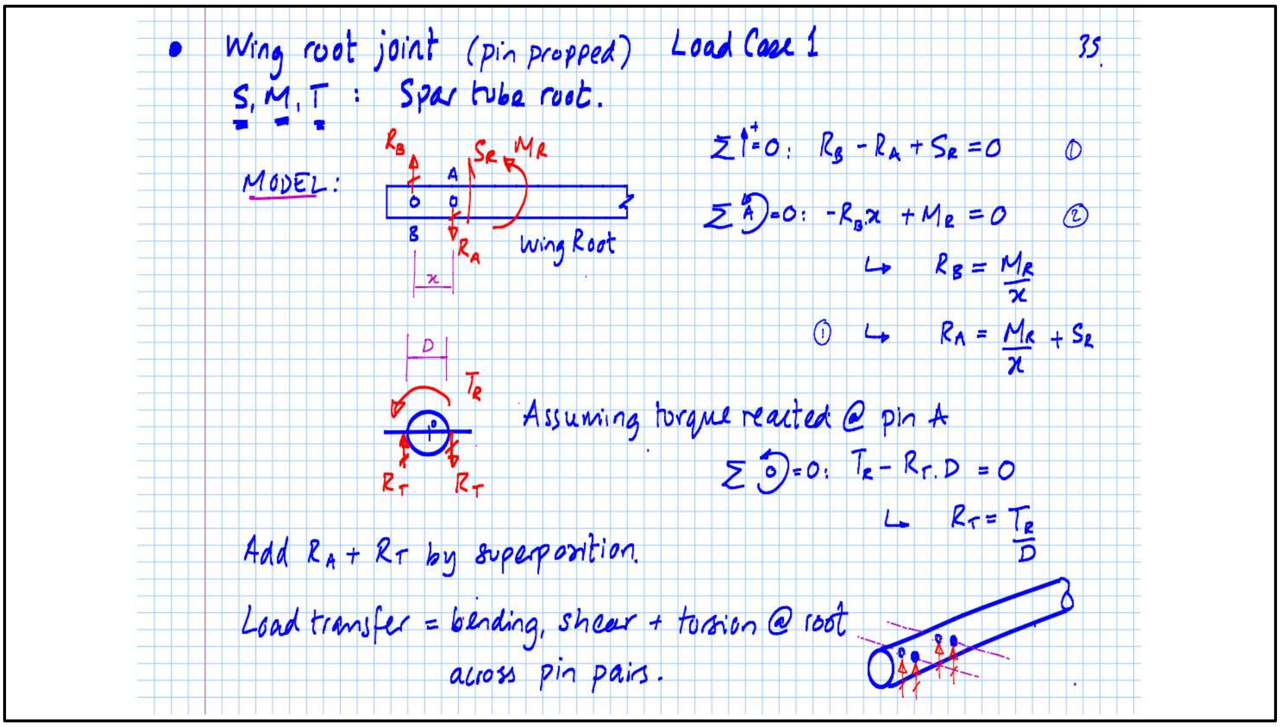
Considering the fuselage-to-wing joint fixing as a lug and considering load case 1) as most critical, the lug load can be estimated as the inertial weight of the fuselage (including pod and empennage) for the load case, i.e., including the acceleration factor.

This load can then be translated as a "cleavage load" on the main lug and a "prying load" on the lug feet where simple FBD's and equilibrium equations reveal the values.

P_{fwj} = Fus-to-wing joint lug load
W_{fus} = Fus inertial weight



Next, considering the centre-bar which passes through the fus-to-wing joint lug and inserts into the port and starboard wing main spar tubes at the wing roots. The centre-bar acts as a beam to react the wing root shear, bending and torsion loading.



Considering the wing main-spar root-joint on the spar tube side, the joint consist of a pair of bolts to transfer the wing root loading as a couple. From the FBD and equilibrium equations we can find the joint transfer loading due to shear, bending and torsion and add these algebraically to find the load on each bolt, on each side of the tube.

Note, to ensure the wing root joint is adequate for the wind tunnel testing we design for the torque to also be reacted by this joint in this slide. In your UAV design the torsion load should actually be reacted by a pin joint between the wing false rear spar (FRS) and the fuselage spar.

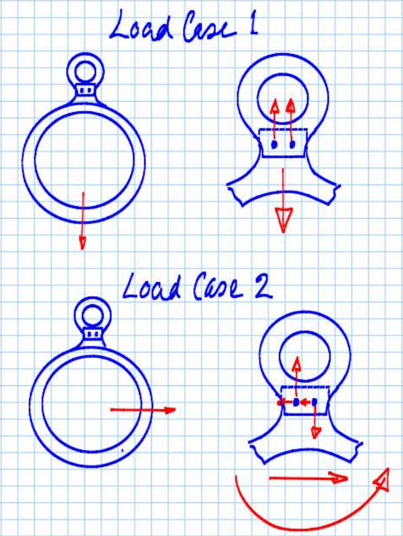
36.

Pod to fuselage spar joint - Load Case 1, 2

S, M, T

Twin bolted lug.

MODEL: Couple load at attachment bolt pair



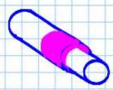
The connection of the pod to the fus spar will be through a twin bolted lug or pairs of lugs to react loading as a couple. It would be prudent to check load case 1) and 2) here.

37.

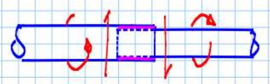
- Spar tube joint - eg. load case 3 for aft fus spar.

$\underline{S}$, M, $\underline{T}$

bonded tubes



MODEL:



Load transfer = shear, ie transverse + torsion across bond.

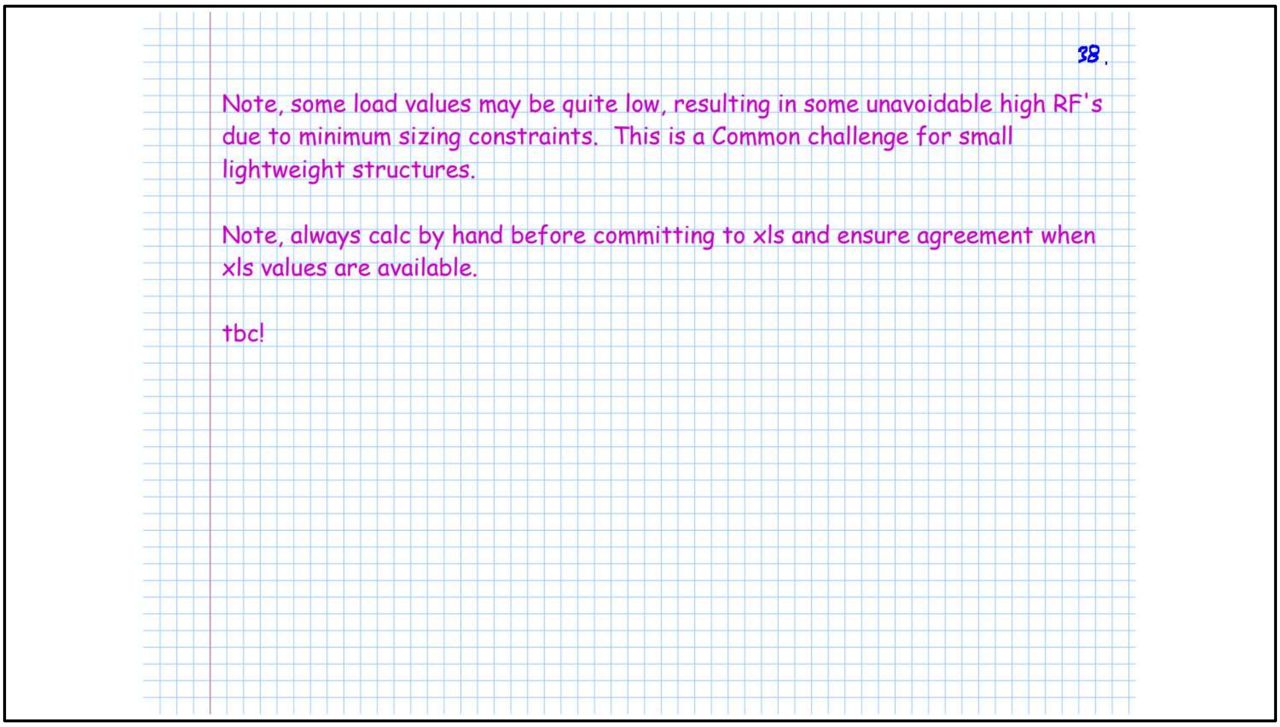
$$S, T = S(x), T(x)$$

@ joint position x along tube.

- Empennage joint ...

etc.

If you have decided to splice tubes to create a spar you will need to bond them together and consider the load transfer. The bonded joint will be responsible for transferring translational and torsional shear loading between the joined tubes, as carried by the adhesive between them.



… See text on slide and also note:

All the methods above are simplified to provide initial estimates with expedience.

You are encouraged to refine the representation within the hand calc methods as you proceed with your design **but do not attempt to apply any unsubstantiated Finite element analysis (FEA). FEA is for the refinement of an existing design and should only be acceptable with substantiated models and properties. FEA does not belong in this preliminary design exercise!**