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ON THE DEVELOPMENT OF FUSELAGE WEIGHT ESTIMATION TOOL INTO AIRCRAFT MULTIDISCIPLINARY DESIGN ENVIRONMENT

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Abstract

Fuselage weight accounts for 20-30% of the total manufacturing empty weight. Traditional fuselage weight estimation adopts semi-empirical equations based on statistical data. These semi-empirical equations are insufficient to capture effect of materials or structural layouts in the overall aircraft conceptual design phase. An estimation tool based on structural sizing is present in this work. Fuselage is discretized into multiple sections composed of several panels and frames. Each section is sized to satisfy both strength requirements and stiffness requirements. A study case is given to estimate the weight of a glider fuselage. Results conclude that the combination of computational speed, accuracy and sensitivity of structural layouts and materials makes this tool suitable for the overall aircraft conceptual design.

Keywords: weight estimation, fuselage, structural sizing

1. Background

Weight estimation is essential for every phase of aircraft design process. Especially in the early design phase, designs of different structural concepts and materials have to be analyzed and the most promising one has to be selected. Therefore, the methods for weight estimation must be sufficiently accurate, design sensitive and fast enough to make right decisions in a first place. In general, the weight estimation can be divided into three classes: Class I, II, and III method. These methods are used in different phases according to the availability of data and information in the design process.

The Class I method is normally used in the very early design stage, and it is based on statistical data and simple performance equations. Examples of the Class I method are presented by Torenbeek , Roskam and Raymer[1-3]. When more data and information become available after the conceptual design phase, such as the baseline geometry of the aircraft, the Class II method (the component weight estimation method) based on semi-empirical equations is employed to evaluate the effects of planform parameters on component weights. The effects of the structural concept and material are not considered. Both the Class I and Class II methods do not perform physics-based structural sizing.

The Class III method performs physics-based structural sizing based on FEA for various components of primary structures. Elham et al.[4] developed tools that employed an analytical structural sizing method for calculating the mass of primary structures. These tools have been validated with various considered aircraft. Dorbath et.al.[5] and Wenzel[6] et.al. developed a more sophisticated tool that incorporates FEA to calculate the mass of primary and secondary wing structures. Although the weights estimated by the Class III method are sensitive to the structural concept and materials used, this method usually requires more detailed geometrical information which is not available in the early design stage. In addition, the time required for pre/post-processing and analysis of a FE model is much more than the Class I and Class II methods.

2. Proposed Physics-based Structural Sizing Method

An estimation tool based on physics-based structural sizing method has been development in this work. Fuselage is discretized into multiple sections composed of several panels and frames. Each section is sized to satisfy both strength requirements and stiffness requirements. An optimizer is adopted to coverage the estimated weight while meeting the strength and stiffness constraints. The tool was developed for aircraft overall conceptual design to trade-off structural concepts, position of structural elements and material selection.

2.1 Load Cases Considered

Both vertical and lateral loads are chosen for the structural sizing, including flight maneuvering cases, fatigue load cases, landing impact and ground maneuvering cases. The aerodynamic load calculation adopted the freeware vortex lattice code AVL. However, the details on load calculation are out of scope here.

2.2 Shear and Bending Moment Calculation

An example of forces acting on the fuselage is shown in Figure 1.

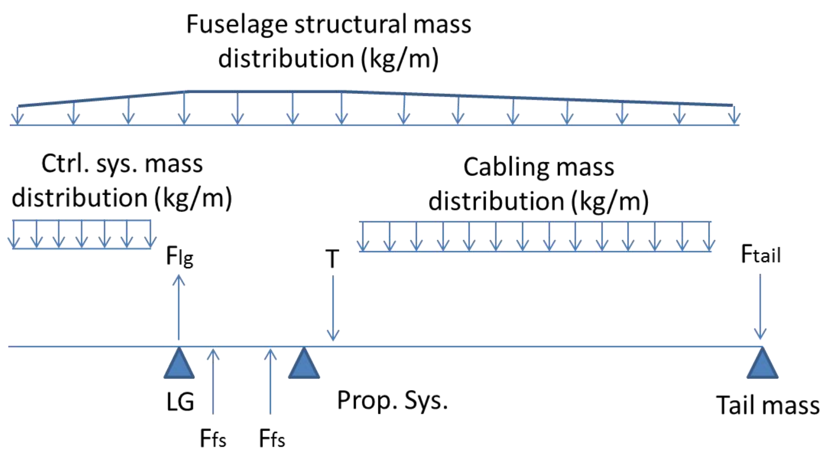


Figure 1: Forces acting on the fuselage.

The fuselage system can be seen to achieve equilibrium by satisfied the following two equations:

$$F_{\mathrm{fs}} + F_{\mathrm{rs}} \ = \ \sum_{i=0}^{Npf} F_{pf}{}^{i} + nMass_{ac}g \tag{1}$$

$$M_{rs} = \ F_{\mathrm{fs}} * d + \ \sum_{i=0}^{Npf} M_{pf}{}^{i} + \ \int_0^L (x - x_{rs})mgdx = 0 \tag{2}$$

Where Ffs and Frs are reacting forces on the front spar and the rear spar of the wing, Fpfi are point forces, i.e. landing gear reacting force, tail trim load and tether tension, n is the load factor, which can be vertical or lateral, d is the distance between the front spar and rear spar, m is the mass distribution along the fuselage (unit is kg/m), Mrs is the bending moment of the fuselage system with respect to the rear spar. From the Eq. 2, the reaction force at rear spar can be calculated:

$$F_{\mathrm{fs}} = -\frac{\sum_{i=0}^{Npf} M_{pf}{}^{i} + \int_0^L (x - x_{rs})mgdx}{d} \tag{3}$$

The reaction force at the front spar can be given by substitute Eq.3 to Eq.1:

$$F_{rs} = \sum_{i=0}^{Npf} F_{pf}{}^{i} + nMass_{ac}g + \frac{\sum_{i=0}^{Npf} M_{pf}{}^{i} + \ \int_0^L (x - x_{rs})mgdx}{d} \tag{4}$$

Since two unknown reacting forces Ffs and Frs are solved, the shear distribution can be obtained from the fuselage nose to the fuselage end by satisfying the vertical equilibrium of forces and the moments at a specified longitudinal location.

2.3 Stress of Load Carrying Elements and Deformation Calculation

The compression/tension stresses of the longerons and the shear stress of the skin are calculated using the procedures suggested by Bruhn[7] :

- Cross-section properties calculation: C.G. of the fuselage cross section, Iy, Iz and Iyz
- Longeron axial stress calculation
- Skin shear flow calculation: delta P method

An assumption was made that longerons carries all the bending moment My and Mz. Therefore, Iy and Iz was calculated by multiplying the longerons area and the square of the distance between longerons and shear center. The contribution of the effective skin thickness was neglected from the calculation. Iyz was kept 0 for all the fuselage cross sections because of the symmetry.

The longeron axial stresses due to bending moments My and Mz are calculated:

$$\begin{gathered}\sigma_{long} = -\left(K_3 M_z - K_1 M_y\right)y - \left(K_2 M_y - K_1 M_z\right)z \\ K_3 = I_{yz}/(I_y\, I_z - {I_{yz}}^2) \\ K_2 = I_z/(I_y\, I_z - {I_{yz}}^2) \\ K_1 = I_y/(I_y\, I_z - {I_{yz}}^2)\end{gathered} \tag{5}$$

The deflection/twist/rotation were calculated based on the Dummy-Unit loads (Bruhn, A7):

$$\begin{gathered}\delta_{end} = \int \frac{Mmdx}{EI} + \iint \frac{q\bar{q}}{Gt} dxdz \\ \theta_{end} = \int \frac{Ttdx}{GJ} \\ \varphi_{end} = \int \frac{Mdx}{EI}\end{gathered} \tag{6}$$

Where m, q and t are bending moment, shear and torque due to virtual unit load. Note: the integration was calculated from the front spar of the wing.

3. Estimation Tool Implementation

The tool was implemented in the MATLAB R2012b. The sizing problem was formulated as follows:

Given: load cases, fuselage width and height dimension along the fuselage, longerons position, the longitudinal positions where the areas of longerons and skin thickness will be varied in the optimization process.

Minimize: the skin weight + longeron weight

Design variables: the areas of longerons and skin thickness at the specified longitudinal positions.
Constraints: failure criteria and deflection constraints. Meanwhile the minimum facing thickness constraint was included, considering the manufacturability of the fuselage structural components.

The function breakdown of the tool is shown in Figure 2.

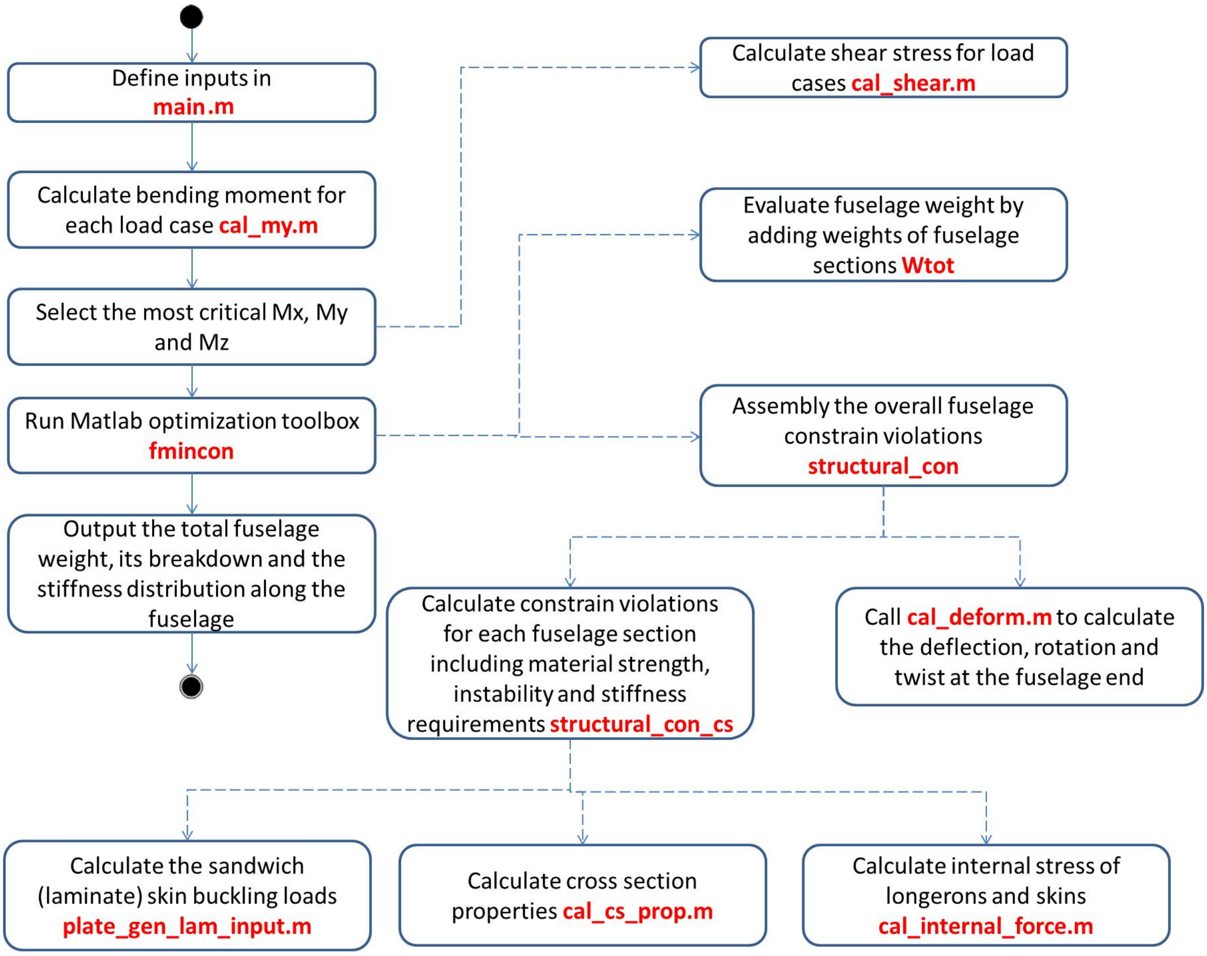


Figure 2: Overview of the weight estimation tool

4. Study Case of a Glider Fuselage

4.1 Glider Fuselage Structural Model and Initial Estimation Results

The initial structural concept of a glider fuselage is unidirectional(UD) tapes and sandwich skin concept. Unidirectional tapes were used as longerons. Details, such as cut-outs and wing-fuselage intersection, are not included since the main purpose of the structural model is used for estimating the fuselage weight.

The total length of the glider fuselage is 6m. In the fuselage structural model, it is divided into a typical 30 fuselage sections. An assumption is assumed that only UD longerons take all the bending moments and skins carry all the shear loads. Considering manufacturbility, the thickness of sandwich core might be different across frame station while it was kept circumferentially equal.

The material allowable values in Table 1 was used for weight estimation.

Table 1: material mechanical properties.

Material	E1/Gpa	E2/Gpa	G12/Gpa	v12	σ1/Mpa	σ2/Mpa	τ12/Mpa	ρ/kg/m3
Unidirectional fiber	151	12	3.5	0.34	tension: 600 fatigue:300 compression : 800	tension: 50 compression : 50	91	1570
+/-45 fabric facing	13	13	15	0.837	tension: 400 fatigue:200 compression : 300	tension: 400 fatigue:200 compression : 300	350 fatigue:17 5	1470
Nomex core	44	44	22	0.3	-	-	1.2	1000

The convergence history of the optimization is shown in Figure 3. The optimized dimensions are given in Table 2.

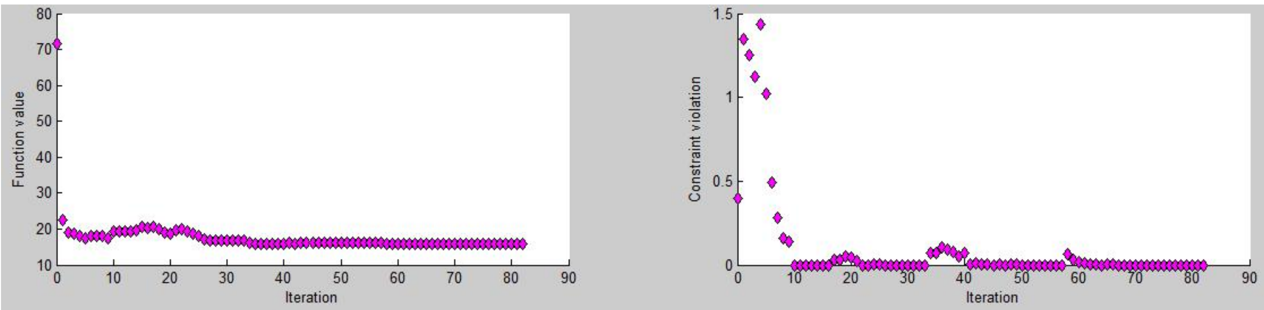


Figure 3: Optimization history, the shown function value is the weight of skin panels and longerons.

Table 2: weight estimation results.

Station (m)	UD1 Area /mm^2	UD2 Area /mm^2	UD3 Area /mm^2	UD4 Area /mm^2	Thickness of panel 1 facing /mm	Thickness of panel 2 facing /mm	Thickness of panel 3 facing /mm	Thickness of panel 4 facing /mm	Core thickness /mm
0	1.6	1.3	1.3	1.6	0.4	0.4	0.4	0.4	4
2	320.0	234.9	234.9	320.0	1.0	1.1	0.6	1.0	5
3.5	910.6	431.2	431.2	910.6	0.9	1.1	1.1	0.9	4
6	67.3	26.4	26.4	67.3	2.3	2.5	2.3	2.3	4

The total estimated fuselage weight is 14.3 kg. The weight breakdown is shown in Table 3.

Table 3: weight breakdown.

Wlongeron /kg	Wfacing /kg	Wcore /kg	Wframe/kg
5.6	4.4	3.3	1.1

4.2 Effect of Different Frame Pitch on Weight Results

As frame pitch varied from 100mm to 900mm, the weight estimation process was run for each frame pitch. The calculation results are shown in Table 4. The longeron weight increased gradually because more longeron material are needed to resist Euler buckling. Meantime, smaller frame pitch means more frames and more frame weight. In addition, more frames needs more assembly effort from the perspective of manufacturing. The minimum total weight can be achieved when frame pitch is 500mm.

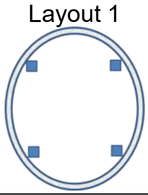
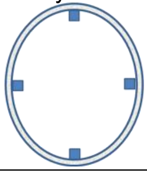
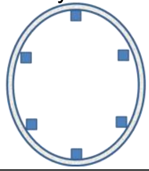
Table 4: weight breakdown vs. frame pitch.

Frame Pitch/mm	100	300	500	700	900
Wlongeron /kg	5.3	5.4	5.6	6.2	7.7
Wfacing /kg	4.4	4.4	4.4	4.4	4.4
Wcore /kg	3.3	3.3	3.3	3.3	3.3
Wframe/kg	5.4	1.8	1.1	0.8	0.6
Total weight/kg	18.4	14.9	14.3	14.6	15.9

4.3 Effect of UD Layout on the Weight Results

Three longeron layouts with frame pitch of 500 mm were investigated in this section. Results in Table 5 show that layout 2 yielded the least fuselage weight. Because layout 3 has most material to carry axial forces, its skin could take less shear force. Therefore, the layout 3 needs less core material than other two layouts.

Table 5: weight breakdown vs. UD layout.

UD layout	Layout 1 	Layout 2 	Layout 3 
Wlongeron /kg	5.6	7.4	7.4
Wfacing /kg	4.4	4.4	4.7
Wcore /kg	3.3	4.4	2.8
Wframe/kg	1.1	1.8	1.0
Total weight/kg	14.3	18.0	15.9

4.4 Effect of Different Material

The weight of glider fuselage with skin panels and stringers made of AL2024 was estimated using this method, while the structural layout was kept the same as the one described in Section 4.1. The total fuselage weight yielded 18.1 kg while the skin and stringers were 17.2 kg. Compared with the metal fuselage made from AL2024, sandwich fuselage gave a 3.8 kg weight saving, 20.9% of the total fuselage weight.

5. Conclusions and Discussions

An estimation tool based on structural sizing has been development in this work. Fuselage is discretized into multiple sections composed of several panels and frames. Each section is sized to satisfy both strength requirements and stiffness requirements. An optimizer is employed to coverage the estimated weight while meeting the strength and stiffness constraints.

A study case is given to estimate the weight of glider fuselage. Results show the tool is able to fast estimate the composite fuselage and quantify the sensitivity of structural layouts and materials. The computational time required for each weight estimation is in the order of 60 seconds, using a computer with a 2.00 Ghz Intel Core2Qurd processor and 4Gb RAM memory, and includes the time taken to generate the fuselage structural model, determine the panel dimensions, and perform the required iterations.

Further study will be on the use of the developed method and tools to estimate the fuselage weight of commercial aircraft and calibration of the method with real weights from the existing aircraft program.

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7. Copyright Statement

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