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# Development and implementation of an advanced, design-sensitive method for wing weight estimation

A. Elham*, G. La Rocca, M.J.L. van Tooren

*Delft University of Technology, Kluyverweg 1, 2629HS Delft, The Netherlands*

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ABSTRACT

This paper presents the development of an advanced, quasi-analytical method for aircraft wing weight estimation and its detailed technical implementation. Similar to other quasi-analytical methods, it makes use of elementary wing box sizing techniques to compute the amount and distribution of material required to resist the applied loads, in combination with empirical methods to estimate all the other weight contributions. However, a new analytical derivation of the so-called airfoil effective distance parameter and a new advanced load estimation approach have been developed, which allow achieving a higher level of accuracy and design sensitivity than any other similar method found in literature. The proposed wing weight prediction method has been validated using data of various airplanes of different size, category and manufacturer. The computational time is dramatically lower than any finite element based sizing tool, while the achieved level of accuracy is comparable or even higher. Each weight prediction takes few seconds on a standard PC, while the average error on the total wing weight is consistently lower than 2%. The high level of design sensitivity allows the designers to assess the effect of design choices such as different airfoils and planform shapes, different structural layouts and materials, including both metal and composites, etc. The resulting combination of speed, accuracy and high level of design sensitivity makes of the proposed tool also a suitable asset for multidisciplinary design optimization studies.



## 1. Weight estimation methods in the aircraft development process. An introduction

The importance of weight for aircraft performance makes *weight prediction and control* an essential part of the aircraft design process. From the start till the end of an aircraft development project, each design decision is evaluated for its impact on the weight of the aircraft [9]. Given the fact that a different type and amount of data and information about the given aircraft become available during the design process, different weight estimations (or prediction) methods have been developed to meet the needs of the various design stages. The weight estimation methods currently used in aircraft design are generally categorized in different classes. Low class methods, generally addressed as *class I* and *class II* methods, are mostly based on statistics, while higher class methods feature a predominant use of physics based calculations.

*Class I* methods are used at the very beginning of the conceptual design process to estimate the aircraft Maximum Take-off Weight (MTOW) and its three main components, namely the Operational Empty Weight (OEW), the Payload Weight (PLW) and the Fuel Weight (FW). At this stage of the design process, apart from some top level requirements, such as speed and range, designers have very little information at hand, therefore *class I* methods are mostly based on the use of statistical data and basic performance equations. Examples of class I methods are presented by Roskam [27], Raymer [26] and Torenbeek [34].

*Class II* methods are used when, besides the initial estimation of MTOW, FW and OEW, the baseline geometry of the aircraft is available. At this stage, various sets of semi-empirical relations are used to estimate the weight of the main aircraft components (e.g., fuselage, landing gear, wing and systems), on the basis of load factors and geometrical parameters, such as wing span, taper, sweep, fuselage diameter, length, etc. Similar to *class I* methods, these methods are based on statistical data. However, they enable designers to assess the effect on weight caused by their design choices, e.g., due to the selected wing planform layout. Still, the effect of certain structural design solution, such as the number and position of spars and ribs in the wing, cannot be evaluated. Also, the specific weight contribution of the aforementioned structural components is outside the reach of these prediction methods. Examples of *class II* methods are presented by Torenbeek [34], Raymer [26] and Howe [13].

*Class III* methods uses physics based analysis instead of statistical data. Generally, they are based on the use of Finite Element Analysis (FEA) to size the various components of the primary

* Corresponding author. Tel.: +31 15 2781601; fax: +31 15 2789564.
*E-mail address:* a.elham@tudelft.nl (A. Elham).

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**Table 1**
Comparative study of some *class II & 1/2* weight estimation methods. Wing weight estimation error computed as a percentage of the actual wing weight.

| Aircraft | Error of wing weight estimation (%) | | | | | |
|---|---|---|---|---|---|---|
| | AdAstra [32] | Torenbeek [35] | van Dijk [32] | WP15 [32] | Macci [20] | PDCYL [1] |
| A300-600R | 4.7 | – | 4.6 | −0.2 | 12 | – |
| A310-300 | −7.2 | – | −4.0 | −0.3 | – | – |
| A320-100 | −7.0 | – | −6.1 | −8.3 | −4.6 | – |
| A330-300 | 0.1 | – | −12.9 | −12.7 | – | – |
| A340-300 | −1.4 | 1.2 | −2.4 | −5.9 | −2.8 | – |
| A380-800 | 22.6 | – | 8.5 | 0.4 | – | – |
| B737-200 | – | – | – | – | −17.5 | −7.6 |
| B747-100 | – | 1.9 | – | – | −3.5 | 4.1 |
| B747-200 | 29.4 | – | 15.6 | 22.0 | −1.8 | – |
| B747-400 | 55.5 | – | 38.9 | 42.4 | – | – |
| B777-200 | 16.2 | – | 10.8 | 16.1 | – | – |
| DC-8 | – | – | – | – | – | 7.9 |
| MD-11 | – | – | – | – | – | −7.9 |
| MD-83 | – | – | – | – | – | −31.1 |
| L-1011 | – | – | – | – | – | −6.3 |
| Fokker 100 | – | −3.8 | – | – | – | – |
| Cessna Citation II | – | 3.8 | – | – | – | – |

structure and compute their weight using volume and material density data. Examples of this kind of methods are presented by Droegkamp [8], Bindolino [2], Laban [19], Hurlimann [15] and Sensmeier [29]. Although these methods are more accurate than those mentioned above and allow designers to assess specific structural design solution, in general, their implementation requires a large amount of detailed geometry information, which is not usually available in the early design stage. Besides, the time required to prepare a FE model and perform the analysis can be quite large, certainly, orders of magnitude larger than computing the simple parametric relations of any *class II* method. This is an issue that generally makes *class III* methods unsuitable in the early phase of the design process and very challenging when the weight estimation is needed within an optimization process, where hundreds of function evaluations are required.

Also *class IV* weight estimation methods exist, but they are used outside the boundary of conceptual and preliminary design. They are typical for the detail and preproduction phase and combine the results of more detailed FEM models than used in class III, with weight calculations performed on production CAD models and actual component weights from catalogs and suppliers.

The aforementioned weight estimation methods provide tailored solutions to the various stage of the design process. However, none of them *individually* is able to fulfill all the needs of the designer, who would like to have one weight prediction tool that is:

- Design sensitive
- Very fast (computational time in order of seconds/minutes)
- Very accurate (error less than 5%)
- Largely based on physics, rather than statistics, such that also innovative design solutions can be addressed
- Suitable to support MDO studies
- Flexible enough, to account for the inevitable difference in type and amount of data available at different design stages

A hybrid class of tools exists, which possibly represent the best attempt to match the wish list given above. These are the so-called *class II & 1/2* methods, whose strength stays in the ability to combine the low computational time of *class II* methods, with accuracy and design sensitivity levels that are comparable to *class III* methods. An analysis of the state of the art of these hybrid methods for wing weight estimation is provided in the next section.

## 2. Class II & 1/2 methods. Strength and limitations of current applications for wing weight estimation

*Class II & 1/2* wing weight estimation methods are *quasi-analytical* methods that make use of elementary strength/stiffness analysis of wing box structures, in combination with empirical factors and statistics based methods. In particular, the amount of required material to resist the loads applying on the wing is estimated using elementary, physics based structural analysis, while statistics and semi-empirical rules are used to estimate the weight contribution of attachments, joints and other secondary structure elements. The first class II & 1/2 wing weight prediction methods were introduced by Burt and Shanley [3,30] at the beginning of the 1950ies. More recent examples of these methods for wing weight estimation are presented by Torenbeek [35], Macci [20], Kelm [17] and van Dijk [36]. Software implementations exist, such as PDCYL [1], WP15 [32] and AdAstra [32]. The validation results of some of those class II & 1/2 wing weight prediction methods are summarized in Table 1.

From Table 1 one can observe that the quality of the wing weight predictions can vary considerably, ranging from very accurate predictions in the order of 1% to quite less accurate ones, often exceeding 25%. Also, each one of the considered methods appears to perform quite differently for aircraft of different type and manufacturer. A critical analysis of the class II & 1/2 methods presented in literature has been performed by the authors and the following three causes have been identified as the possible main sources of weight prediction inaccuracy:

1. Lack of (adequate) geometry model: A large majority of the current class II & 1/2 methods uses an oversimplified geometry model of the wing, which does not (even) include the aerodynamic surface. In some cases, the wing box section is assumed to be simply rectangular, ignoring the actual shape of the upper and lower wing covers. As a matter of fact, most of the weight estimation methods do not have a detailed geometry generator at all. Instead they use a few geometrical parameters defined by users as inputs to built a simplified geometry model. These simplifications may be adequate for early design stages, where exploratory studies do not require high accuracy and the availability of geometry data is very low. However, having a proper geometry model for weight estimation is important for the following reasons:
   - The structural sizing of a wing box, hence its weight, strongly depends on the applied aerodynamic loads. Thereby, the availability of a proper model of the wing outer shape is necessary to obtain a reliable aerodynamic load distribution.
   - The location of the spars has a strong effect on the wing box weight, particularly when the airfoil shape is also modeled. The availability of a proper wing geometry model, including both the outer surface and the inner structure, allows measuring the height of each spar at any spanwise section, whatever the shape of the wing sections. This has a

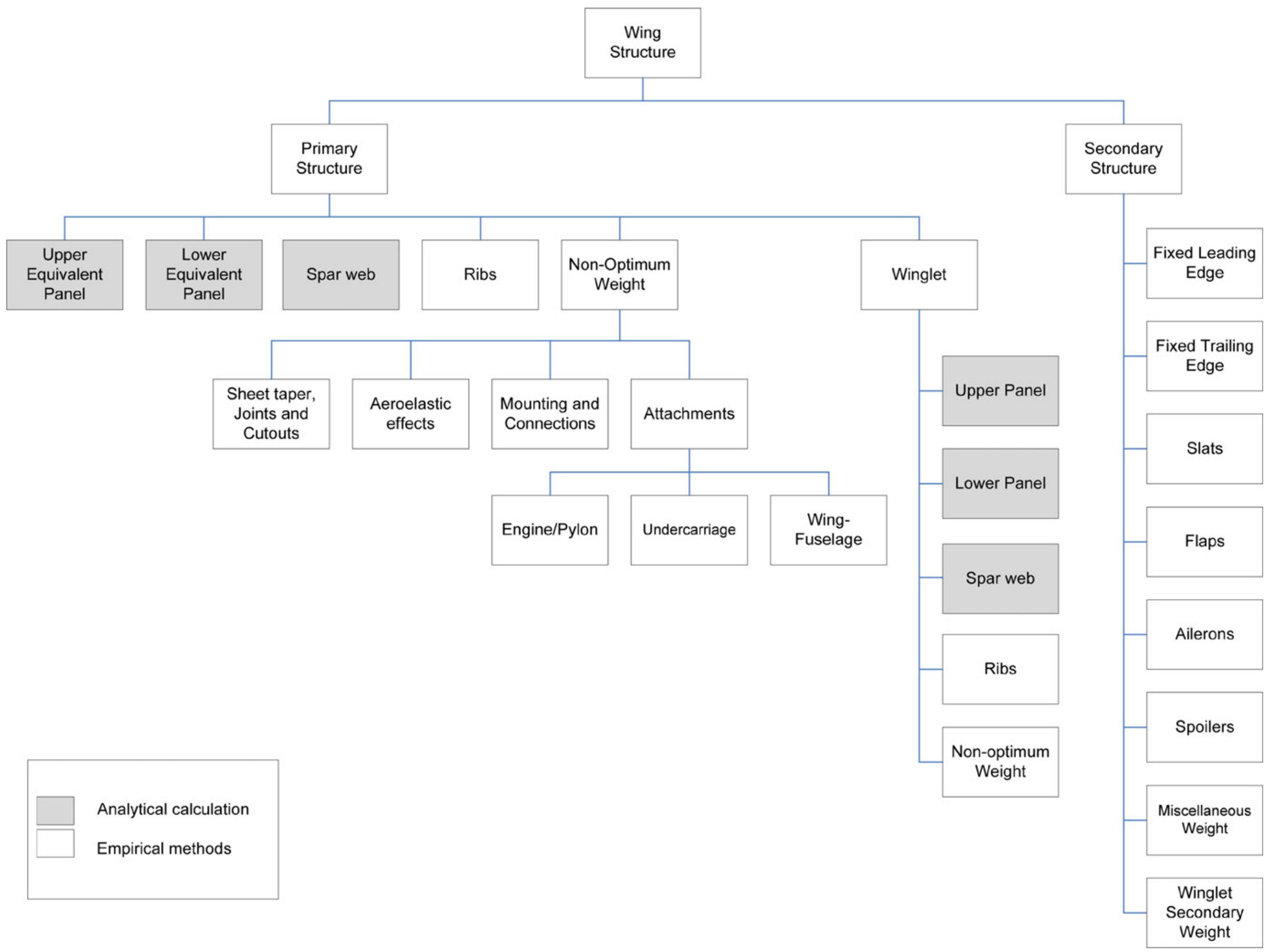



**Fig. 1.** Wing structural weight decomposition.

significant effect on the accuracy of the moment of inertia estimation for each section, which is required to compute normal stresses.
- Most of the empirical equations for secondary weight estimation require the specification of geometrical parameters such as the area of high lift devices or wing thickness at a specific location of the wing span. A good estimation of these and similar parameters is not achievable without an adequately detailed geometry model.

2. Oversimplified/inaccurate loads estimation: Also due to the lack of an adequate geometry model, in most of the examined class II & 1/2 methods, the aerodynamic loads are calculated using empirical equations or by assuming an elliptical lift distribution. Besides, the effects of winglets and tailplanes on the actual lift distribution of the wing are generally not accounted. These are difficult to capture using simplified models that are based on semi-empirical relationships, and would demand additional modeling and simulation efforts. In addition, not all of the load relief factors are (fully) taken into account, such as the presence of fuel tanks, engines and landing gear attachments at various span locations.
3. Strong reliance on empirical coefficients tuning: These methods rely on several empirical coefficients, especially for the weight estimation of other components than skin panels and spars. Since these empirical coefficients are determined and tuned using statistical data, they are dependent on the used database of aircraft. Besides, the fine tuning process applied to these coefficient to better fit a specific class/type of aircraft, most of the time, has a negative impact on the estimation accuracy for others. That might explain the varying performance showed in Table 1, when applying the same method to Airbus or Boeing aircraft.

In order to address these three causes of inaccuracy, while maintaining the high level of computational efficiency and design sensitivity required in preliminary design, a new class II & 1/2 wing weight estimation method has been developed and is presented in this paper. The basic approach of the proposed method is described in Section 3. The analytical sizing method for the wing box structure, which represents the core component of the wing estimation method, is described in Section 4. The overall software implementation of the method, its modular architecture and main components are presented in Section 5. The new method has been validated on a number of aircraft of different size, type and manufacturer. The results are presented and discussed in Section 6, where the advantages of the key improvements featured by the proposed method with respect to other class II & 1/2 methods are demonstrated. Conclusions are provided in Section 7.

## 3. Development of a new *class II & 1/2* wing weight estimation method. Description of the basic approach

Similarly to other class II & 1/2 methods, the estimation of the total weight of an aircraft wing is based on a preliminary decomposition in primary and secondary structure weight contributions, as illustrated in Fig. 1. The primary structure group is further de-

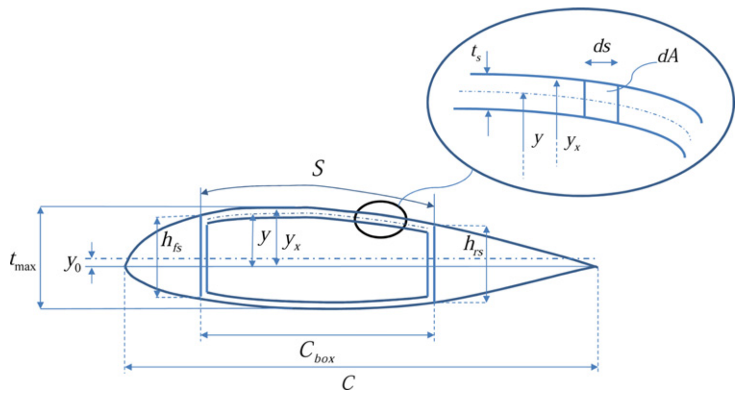



**Fig. 2.** Model of the wing box structure with relative nomenclature.

composed in so-called *optimum* and *non-optimum* structure weight contributions. The former contributions (see gray filled boxes in Fig. 1) include the spar webs and the upper and lower equivalent panels of the wing box, whose weight is determined *analytically*, based on optimum structure sizing (i.e., the amount of material is just that required to carry the critical loads). The latter contributions include weight penalty items such as joints, attachments, cutouts, etc. These, together with the secondary weight contributions, such as fixed leading and trailing edges, high lift devices and movables, are determined by means of empirical methods.

The novel elements of the class II & 1/2 method proposed in this paper are mainly related to the new analytical method to size the wing box structure and the advanced implementation of the modeling and analysis system to support loads estimation. The following two elements characterize the new analytical method to size the wing box:

- A mathematical expression for the so-called *airfoil effective distance* has been derived, which relates the weight of the wing box structure to the geometry of the various wing sections. This novel relation makes the proposed weight estimation method sensitive to the wing outer aerodynamic shape, and allows sizing the upper and lower equivalent panels independently, according to their specific critical load case. Details are provided in Section 4.2.
- The analytical structural sizing method can be applied to both metal and composite structures. The derivation of the allowables for carbon fiber reinforced polymers (CFRP) structural components can be found in Section 4.4.2.

In order to generate a realistic load distribution that is sensitive to the actual design of the wing (in terms of planform and wing section shape), a quasi-3D aerodynamic analysis tool has been developed on purpose. This tool is able to estimate the aerodynamic load distribution on the wing, accounting also for the effect of tailplane and winglets. The strength of the tool consists in the ability to integrate a Vortex Lattice Method (VLM), for the estimation of the spanwise lift distribution, with two 2D solvers (one for subsonic and one for transonic regimes) for the estimation of the pitching moment at various wing sections. The computed aerodynamics loads are then fed to the analytical wing box sizing module, together with the relief loads due to the mass of the wing itself, of the fuel and when applicable the engines. The relief loads can be calculated for any other mass such as tip tanks in case of existence. Details on the functionality of the quasi-3D tool and the overall load estimation method are presented in Section 5.3.

The wing weight estimation process is supported by a dedicated parametric geometry generator, which represents a key factor for the accuracy and design-sensitivity of the method proposed in this paper. The geometry generator enables the user to define any type of wing geometry (including outer surface and internal structural layout), from which data are automatically extracted to feed the quasi-3D solver, the analytical wing box sizing module, and, finally, the empirical equations used to estimate the secondary and non-optimum weights. Details on the functionality of the model generator are provided in Section 5.2.

The empirical methods used to estimate the non-optimum weights (including the ribs weight) and the secondary weight contributions are those presented by Torenbeek in [35], hence they will not be discussed in this paper any further.

## 4. Structural sizing method

This section describes the analytical method employed for the preliminary sizing of the wing box structure main elements. Once the amount of material required to carry the applied loads is known, the weight contribution of the wing box can be easily computed. To enable an efficient and effective sizing process, the actual wing box structure is modeled as shown in Fig. 2. The skin, the stringers and the spars caps are modeled using one upper and one lower equivalent panel. The web of the front and back spars are modeled separately using simple vertical panels. The upper and lower panels are sized to withstand the bending moments, whereas the spar webs are sized based on shear loads. There is no ribs sizing performed in this method, although rib pitch is a parameter used to size the upper equivalent panels. As illustrated in Fig. 1, the ribs weight contribution is estimated by means of an empirical method.

### 4.1. Sizing method for the upper and lower equivalent panels

In a wing box with homogeneous material in each panel, the axial stress in each panel element $dA$ (where $dA = t_s\,ds$, being $t_s$ the thickness of the given equivalent panel, see Fig. 2) is determined as follows:

$$\sigma = \frac{M(y - y_0)}{I} \tag{1}$$

where $M$ is the applied bending moment, $I$ is the section second moment of area, $y$ is the distance of the element from the chord line, and $y_0$ is the distance between the neutral axis and the chord line (see Fig. 2). The vertical location of the neutral axis is defined as:

$$y_0 = \frac{\oint y t_s\,ds}{\oint t_s\,ds} \tag{2}$$

The distance of each element of the upper and lower panels ($y_u$ and $y_l$) from the chord line can be expressed in terms of the outer surface coordinates $y_x$ (which are known) and the thickness of the

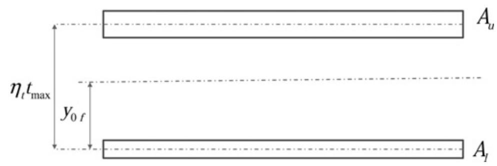



**Fig. 3.** Equivalent flat panels located at effective distance.

given panel $t_s$ (which have to be computed), as follows:

$$y_u = y_{x_u} - t_{s_u}/2$$
$$y_l = y_{x_l} + t_{s_l}/2 \tag{3}$$

Assuming constant chordwise thicknesses $t_{s_u}$ and $t_{s_l}$ for the upper and lower equivalent panels respectively, from Eqs. (1) and (3), it can be found that the axial stress is maximum at the element with maximum $y_x$. Therefore, the stress distribution in the panels can be presented as a function of the airfoil geometry, as follows:

$$\sigma_u = \frac{(y_u - y_0)}{y_{max_u}}\sigma_{max_u}$$
$$\sigma_l = \frac{(y_l - y_0)}{y_{max_l}}\sigma_{max_l} \tag{4}$$

where $y_{max}$ is the maximum of $y - y_0$. In order to further simplify the sizing method, the upper and lower panels are replaced with two flat panels located at a certain distance called the *effective distance* (Fig. 3). This allows sizing the panels assuming a chordwise uniform stress distribution. The effective distance ($\eta_t t_{max}$) is a fraction of the airfoil maximum thickness ($t_{max}$). The parameter $\eta_t$ should be calculated such that the total bending moment in the curved and flat equivalent panels is equal (see Section 4.2).

Using Eq. (2) the vertical location of the neutral axis of the flat panels shown in Fig. 3 is calculated:

$$y_{0_f} = \frac{A_u}{A_u + A_l}\eta_t t_{max} \tag{5}$$

where $A_u$ and $A_l$ are the area of the equivalent upper and lower panels respectively. Using Eq. (5) the proper values of $y - y_0$ in Eq. (1) for the upper and lower panels are calculated:

$$y_{u_f} - y_{0_f} = \eta_t t_{max}\frac{A_l}{A_u + A_l}$$
$$y_{l_f} - y_{0_f} = -\eta_t t_{max}\frac{A_u}{A_u + A_l} \tag{6}$$

Using Eq. (6), the second moment of area ($I$) of the section is derived as follows:

$$I = A_u\left(\eta_t t_{max}\frac{A_l}{A_u + A_l}\right)^2 + A_l\left(-\eta_t t_{max}\frac{A_u}{A_u + A_l}\right)^2$$
$$= \eta_t^2 t_{max}^2\left(\frac{A_u A_l}{A_u + A_l}\right) \tag{7}$$

Substituting Eqs. (6) and (7) in Eq. (1), the minimum required area of the upper and lower panels are determined:

$$A_u = \frac{M/\sigma_{max_u}}{\eta_t t_{max}} \tag{8}$$

$$A_l = \frac{M/\sigma_{max_l}}{\eta_t t_{max}} \tag{9}$$

where $\sigma_{max_u}$ and $\sigma_{max_l}$ are, respectively, the upper and lower panels maximum allowable stresses, which will be defined in Section 4.4. The thicknesses of the curved panels are determined using $t_s = A/S$, where $S$ is the length of the curved panels, measured along the outer wing surface (see Fig. 2):

$$t_{s_u} = \frac{M/\sigma_{max_u}}{\eta_t t_{max} S_u} \tag{10}$$

$$t_{s_l} = \frac{M/\sigma_{max_l}}{\eta_t t_{max} S_l} \tag{11}$$

### 4.2. Calculation method for the effective distance

The wing box effective distance is an important parameter, which relates the structural weight of the wing to the wing outer shape. Shanley [30] derived an equation for the airfoil effective distance based on the rationale that the total bending moment in the curved equivalent panels (Fig. 2) should be equal to the total bending moment in the flat equivalent plates (Fig. 3). Shanley proposes the following method to calculate $\eta_t$ as a function of the airfoil geometry:

$$\eta_t = \frac{1}{N t_{max}(z_{max} - \frac{t_s}{2})}\sum_{n=1}^{N}\left[z_{u_n}^2 + z_{l_n}^2 - t_s(z_{u_n} - zl_n) + \frac{t_s}{2}\right] \tag{12}$$

where $z$ is the airfoil ordinate measured from the bending moment axis (defined as $y_x - y_0$ in Fig. 2), $N$ is the number of elements used to discretize the wing box panels, and $t_{max}$ is the airfoil maximum thickness. It should be noted that, in Shanley's method, the upper and the lower panels are assumed to have the same thickness, indicated as $t_s$ in the equation above.

Torenbeek [35] proposes another empirical relation to estimate the value of $\eta_t$ as a function of the airfoil maximum thickness and the heights of the front ($h_{fs}$) and rear spar ($h_{rs}$):

$$\eta_t \approx \frac{1}{3}\left(1 + (h_{fs}/t_{max})^2 + (h_{rs}/t_{max})^2\right) - 0.025 \tag{13}$$

In this equation, it is assumed that the thicknesses of the upper and lower panels are the same (as in Shanley's) and equal to 2.5% of the maximum thickness of the airfoil $t_{max}$. In case there are not sufficient wing geometry data available to use Eq. (13), Torenbeek suggests to assume $\eta_t$ equal to 0.8.

Elham et al. [11] investigated the effect of different values of $\eta_t$ on the wing weight estimation of various transport aircraft. They concluded that the influence of $\eta_t$ can be so large (e.g., increasing $\eta_t$ from 0.8 to 1 causes a 12% error in the weight estimation of the Boeing 747 wing), that a precise estimation of this parameter is a great necessity. To this purpose, a new method to calculate the effective distance parameter $\eta_t$ has been developed and is described here below. The proposed method is developed based on the same rationale of the Shanley's method (equal total bending moment in the curved equivalent panels and the flat equivalent plates), however the thickness of the upper and lower panels are allowed to be different (but both constant over the whole chord) and those panels are allowed to be made of different materials (with various allowables). Having different thicknesses in the panels also affects the location of the bending moment axis, which cannot be calculated accurately by assuming the same thicknesses in both upper and lower panels.

The total bending moment in the box with curved panels is equal to:

$$M_c = \oint \sigma\, y\, dA$$
$$= \sum_{n=1}^{N}\sigma_{u_n}(y_{u_n} - y_0)t_{s_u}\,ds_{u_n} + \sum_{n=1}^{N}\sigma_{l_n}(y_{l_n} - y_0)t_{s_l}\,ds_{l_n} \tag{14}$$

Substituting Eq. (4) in (14), the total bending moment in the curved panels is calculated:

$$M_c = \frac{\sigma_{max_u}}{y_{max_u}} \sum_{n=1}^{N} (y_{u_n} - y_0)^2 t_{s_u}\, ds_{u_n} + \frac{\sigma_{max_l}}{y_{max_l}} \sum_{n=1}^{N} (y_{l_n} - y_0)^2 t_{s_l}\, ds_{l_n} \tag{15}$$

The bending moment in the flat plates is presented as following:

$$M_f = \sigma_{u_{max}} A_u (\eta_t t_{max} - y_{0_f}) + \sigma_{l_{max}} A_l\, y_{0_f} \tag{16}$$

Assuming that $t_{s_u}$ and $t_{s_l}$ are constant along the chord, $A_u$ and $A_l$ can be rewritten as $t_{s_u} S_u$ and $t_{s_l} S_l$, respectively, where $S_u$ and $S_l$ are the curvilinear length of the upper and lower equivalent panels, measured along the outer wing surface (see Fig. 2). Substituting Eq. (5) in Eq. (16), the equation of the total bending moment in the flat plates is reduced to:

$$M_f = \eta_t t_{max} \left( \frac{t_{su} S_u t_{sl} S_l}{t_{su} S_u + t_{sl} S_l} \right) (\sigma_{max_u} + \sigma_{max_l}) \tag{17}$$

Having set $M_c = M_f$, after some reductions, the following new $\eta_t$ relation can be obtained:

$$\eta_t = \frac{1}{t_{max}} \left( \frac{1}{S_u y_{max_u}} \sum_{n=1}^{N} (y_{u_n} - y_0)^2\, ds_{u_n} + \frac{1}{S_l y_{max_l}} \sum_{n=1}^{N} (y_{l_n} - y_0)^2\, ds_{l_n} \right) \tag{18}$$

Fig. 4 shows the value of $\eta_t$ for the NACA 23012 airfoil, calculated using two different methods, namely the new proposed method (indicated as Elham's method in the figure) and Shanley's method. The wing box is assumed to be between 20% to 60% of the airfoil chord. In this figure, $\eta_t$ is plotted as a function of the thickness of the equivalent panels $t_s$, which has been assumed to be the same for the upper and lower equivalent panels. This because of the limitation of Shanley's method, whilst the new method proposed in this thesis would allow to consider different thickness values. The value of $\eta_t$ suggested by Torenbeek ($\eta_t = 0.8$ for $t_s/t_{max} = 0.025$) is also shown in that figure. The value of $\eta_t$ suggested by Torenbeek is only for one particular value of $t_s/t_{max}$, which is assumed by Torenbeek to be a typical value for passenger aircraft. However, with the other two methods, $\eta_t$ varies linearly with $t_s$. Indeed, for a given airfoil shape, the larger the thickness of the panels, the lower their effective distance. It can be noted that the gradients of the curves ($\frac{d\eta_t}{dt_s}$) computed using Shanley's and Elham's methods are very close to each other and equal to $-0.985$ and $-0.975$, respectively.

The effect of the different methods to compute $\eta_t$ on the wing weight estimation of a few aircraft is reported in Section 6.

The effect of the airfoil shape on the value of $\eta_t$ is shown in Fig. 5, where four different airfoils with the same maximum thickness to chord ratio equal to 0.12 are tested. From Fig. 5(a) one can observe that the value of $\eta_t$ for the Withcomb airfoil is about 5.5% higher than for the NACA 63412 airfoil. Considering Eqs. (8) and (9), such difference in the value of $\eta_t$ can cause more than 5% difference in the weight of wing box skins, stringers and spars caps. This example demonstrates the importance to derive a $\eta_t$ value that is specific to the actual geometry of the wing, whereas assuming a fixed $\eta_t$ value (such as 1 or 0.8) would make the weight estimation both inaccurate and not design sensitive.

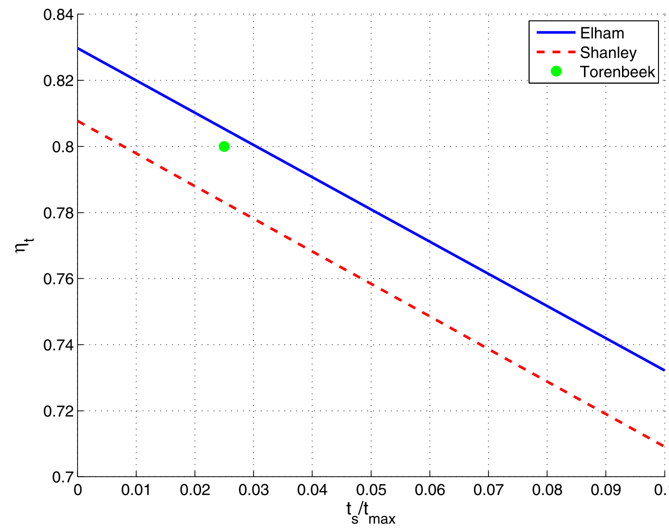



**Fig. 4.** $\eta_t$ for the NACA 23012 airfoil calculated using three different methods.

### 4.3. Sizing method for spars web

As anticipated at the beginning of Section 4, the webs of the spars are sized to withstand only shear stresses. The shear flow in the spar webs is the sum of the shear flow due to the vertical load and the shear flow due to the torsional moment.

The shear flows due to the vertical load in the front and rear spars are estimated implementing the method presented by Howe [14]:

$$q_{v_{fs}} = \frac{h_{fs}}{(h_{fs}^2 + h_{rs}^2)} V \tag{19}$$

$$q_{v_{rs}} = \frac{h_{rs}}{(h_{fs}^2 + h_{rs}^2)} V \tag{20}$$

where $h_{fs}$ and $h_{rs}$ are the heights of the front and rear spar respectively and $V$ is the total vertical force.

The shear flow due to torsional moment is calculated as follows:

$$q_t = \frac{T}{2A_{box}} \tag{21}$$

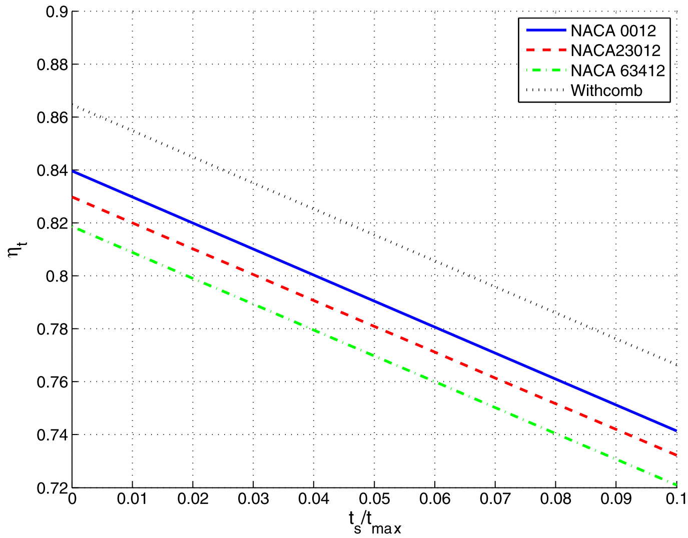
where $T$ is the torsional moment and $A_{box}$ is the enclosed area of the wing box. The torsional moment should be calculated about the elastic axis (shear center for a 2D section). The location of the shear center is calculated using the method presented by Megson [21]. In this method the location of the shear center is determined using the condition that a shear load acting through the shear center of a section produces zero twist:

$$\frac{d\theta}{dz} = \frac{1}{2A_{box}} \oint \frac{q}{Gt}\, ds = 0 \quad \Rightarrow \quad \oint \frac{q}{Gt}\, ds = 0 \tag{22}$$

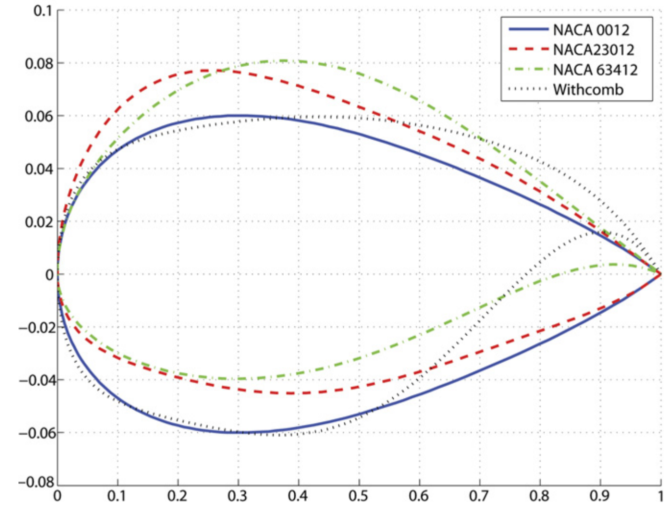
Using Eq. (22) the chordwise location of the shear center is found by applying a vertical shear force, $V$, through the $y$ axis:

$$x_{sc} = -\frac{1}{V} \oint \frac{q}{Gt}\, ds \tag{23}$$

From Eqs. (22) and (23) one can find that the location of the shear center is a function of the panels thickness ($t$). So it should be determined in an iterative process (see Section 5.4).




(a) $\eta_t$ as a function of panel thickness.




(b) Airfoils geometry.

**Fig. 5.** $\eta_t$ for various airfoils with maximum thickness to chord ratio equal to 0.12.

Finally, the thickness of each spar web is calculated using the total shear flow ($q = q_v + q_t$) and the maximum allowable shear stress ($\tau_{max}$):

$$t_w = \frac{q}{\tau_{max}} \tag{24}$$

### 4.4. Determination of the allowables

In order to compute the thickness of the equivalent panels and the spar webs by means of Eqs. (10) and (11) and Eq. (24), respectively, it is necessary to determine first the values of the allowables $\sigma_{max_u}$, $\sigma_{max_l}$ and $\tau_{max}$. The methods used to estimate these allowables for metal structures and composites are presented in the following subsections.

#### 4.4.1. Metal alloys

The lower panel is assumed to be under tension when the wing is subjected to positive load factors. For this reason it seems reasonable to assume the maximum allowable stress in that panel equal to the maximum tensile stress of the given metal alloy. In reality, the actual value of the allowable stress could be limited by fatigue and associated crack initiation and propagation issues. However, during the preliminary design stage the knowledge about the design is generally not sufficient to consider those issues. Therefore, in the proposed method, the lower panel is sized using the maximum tensile yield stress ($F_{t_y}$) of the used material. The validation results (see Section 6) confirms that this is a correct assumption for preliminary weight estimation.

For the upper panel, which is under compression for positive load factors, the maximum allowable stress is selected as the minimum of the compressive yield stress ($F_{c_y}$) and the buckling stress. The buckling stress of a stiffened panel under compression can be calculated using the stiffened panel efficiency method [24]:

$$\sigma_{max_b} = F\left(\frac{PE}{L}\right)^{1/2} \tag{25}$$

where $F$ is the stiffened panel efficiency factor, $P$ is the load intensity, $E$ is the Young's modulus and $L$ is the rib pitch. The load intensity in the upper panel is equal to:

$$P = \frac{\sigma A_u}{C_{box}} \tag{26}$$

where $C_{box}$ is the distance between the front and the rear spar (i.e., the wing box chord), defined in Fig. 2. Substituting Eq. (8) in (26), the load intensity is represented in such a way that $P = \frac{M}{\eta_t t_{max} C_{box}}$. Using this value for $P$, the maximum allowable stress is determined as follow:

$$\sigma_{max_u} = \min\left(F_{c_y}, F\sqrt{\frac{ME}{\eta_t t_{max} C_{box} L}}\right) \tag{27}$$

Concerning the sizing of the spar webs, the maximum allowable shear stress can be estimated using the method presented in [22]:

$$\tau_{max} = \frac{F_{t_y}(L) + F_{t_y}(LT) + F_{c_y}(L) + F_{c_y}(LT)}{4} \times \frac{2F_{s_u}}{F_{t_u}(L) + F_{t_u}(LT)} \tag{28}$$

where subscripts $t$, $c$ and $s$ denote tension, compression and shear, respectively; $y$ and $u$ denote yield and ultimate stresses; $L$ and $LT$ denote the longitudinal and long transverse direction of the material.

#### 4.4.2. Composite materials

The strength of a composite structure takes a range of values as a result of material variability, environmental effects, and sensitivity to damage. In preliminary design, the so-called cutoff strain can be used to size composite structures [14,16]. The cutoff strains are determined based on the *worst of all situations* [16]. The allowables for composite materials are calculated using these cutoff strains in combination with the so-called *10% rule* [5]. The 10% rule allows to estimate the mechanical properties of a laminate from the ply properties and the percentages of plies in each direction. Implementing this approach, the maximum allowable stress in the lower panel is calculated using Eq. (29):

$$\sigma_{maxl} = E_x \epsilon_t \tag{29}$$

where $E_x$ is the laminate stiffness in $x$ direction and $\epsilon_t$ is the cutoff tensile strain.

For the upper panel, the maximum allowable compressive stress is calculated using the same type of equation as (29), where a cutoff compressive strain ($\epsilon_c$) is used instead of $\epsilon_t$. The buckling stress is calculated using the method presented by Tetlow [33]:

**Table 2**
Suggested values for cut off strains.

| Tensile | Compressive | Shear |
|---|---|---|
| 0.005 | 0.0045 | 0.006 |

$$\sigma_{maxb} = 0.725F\left({E_{x0}}^2\ Z\right)^{1/4}\sqrt{\frac{P}{L}} \tag{30}$$

where $Z$ is calculated using the following equations:

$$Z = \frac{E_x K}{E_{x0}}$$
$$K = \frac{\pi^2}{6(1-\nu_{xy}\nu_{yx})E_{x0}} \times \left((E_x E_y)^{1/2} + \nu_{xy}\frac{E_y}{2} + \nu_{yx}\frac{E_x}{2} + 2(1-\nu_{xy}\nu_{yx})G_{xy}\right) \tag{31}$$

Concerning the spars webs, the maximum allowable shear stress is calculated as follows:

$$\tau_{max} = G_{xy}\epsilon_s \tag{32}$$

where $G_{xy}$ is the laminate shear stiffness and $\epsilon_s$ is the cutoff shear strain.

The required laminate properties in Eqs. (29), (31) and (32) are calculated using the 10% rule [5] as follows:

$$E_x = \left(0.1 + 0.9\frac{m}{100}\right)E_{x0}$$
$$E_y = \left(0.1 + 0.9\frac{100-m-n}{100}\right)E_{x0}$$
$$G_{xy} = \left(0.028 + 0.234\frac{n}{100}\right)E_{x0}$$
$$\nu_{xy} = \nu_{yx} = \frac{1}{1+4(\frac{100-m-n}{n})} \tag{33}$$

where $m$ is the percentage of plies at 0 degree, $n$ is the percentage of plies at ±45 degree and $E_{x0}$ is the stiffness of the unidirectional lamina along the principal direction.

The cutoff strains can be determined using Angle Minus Longitudinal (AML) plots method [23]. The AML is the percentage of plies in ±45° minus the percentage of plies in 0° $(n-m)$. In order to create an AML plot, laminates of different layups, which provide a range of AML, are drilled or impacted and tested to failure in various situations: under room temperature, hot/wet, and cold/dry conditions. The allowable strains are derived from test data are plotted as a function of AML. These plots can be used to determine the cut off strains as a function of the composite layup. Example of AML plot is presented in Ref. [23]. If such plots are not available, the cut off strains provided in Table 2 can be used.

## 5. Software implementation of the proposed method

A software implementation of the method illustrated above has been developed using Matlab and is named EMWET (Elham Modified Weight Estimation Technique). The tool has been developed as a standalone application that reads an input file and creates an output report with all the results. The main structure of the tool is shown in Fig. 6. EMWET consists of three main modules, namely the *geometry generator*, the *load calculator* and the *weight estimator*, plus a data handling module that coordinates the activities of the main modules and enables their data exchange.

The geometry generator uses state-of-the-art parametric modeling techniques to generate 3D models of wings and tailplanes. The generated geometry data are sent to the data handling module and, from here, they are fed to the load calculator and the weight estimator to support the actual computations.

The load calculator module contains a quasi-3D aerodynamic solver that computes the aerodynamic loads (lift and moment) on the wing and a load integration module that combines the aerodynamic loads with the inertia relief contributions. The final load distributions are used by the weight estimator module to size the wing box.

Also the weight estimator module consists of more sub modules: the wing box weight estimator, where the analytical wing box sizing method discussed in the previous sections is implemented; the non-optimum and secondary weight estimator, where the empirical methods to compute the other weight contributions are applied; and the total weight calculator, which simply adds the two contributions to compute the total wing weight.

The functionalities of the aforementioned modules are discussed in more detail in the following sections. For a complete overview of the interaction and data exchange between the various modules, the reader can refer to the N2 chart provided in Fig. 7.

### 5.1. Input parameters

The input parameters required to operate EMWET are illustrated in Table 3. These include the parameters required to define the geometry of wing (both aerodynamic surface and internal structural layout) and tail (only aerodynamic surface), the parameters required to estimate the relief loads (i.e., number and position of engines, size and position of fuel tanks), the material specification parameters (e.g., density and allowables), and, finally a set of performance parameters, such as flight altitude, maximum operative speed, etc. (mainly used for the aerodynamic loads calculation), and maximum takeoff weight, maximum zero fuel weight, etc. (mainly used for the weight estimation formulas). Fig. 7, illustrates the specific data flow from the input file to the various computational modules of EMWET. All these data are contained inside a single input file, with exception of the airfoils and material data, which are stored in two separate and extensible databases (see Fig. 6). In this case, the input file contains just the name of the given airfoil and material, while EMWET takes care of accessing the right files in the databases and extract the required data.

### 5.2. Geometry generator

The geometry generator module is able to generate any sort of wing geometry, just based on the data provided by the user via input file. A minimum of two airfoils, one at the root and one at the tip must be provided to build a simple, single tapered wing. However, any type of swept and twisted wing can be defined, including multiple airfoils and featuring multiple kinks and dihedral angles. Winglets can be defined as well.

The airfoils contained in the library can be either defined as sets of 2D Cartesian points or as lists of coefficients to support three alternative parametric airfoil representations, namely: the CST parameterization method, which is based on Bernstein polynomials [18]; the CSRT method, which is based on the extension of the CST method using B-Splines [31]; and a third parameterization based on Chebychev polynomials [37]. As demonstrated in Ref. [10], the possibility to use these parameterization methods makes EMWET usable within an optimization framework, where different aerodynamic shapes can be evaluated, while keeping the number or related design variables low.

The possibility to define the internal structure is at the moment limited to the classical spar and ribs layout. The user can specify the position of one front and one rear spar as percentages of the

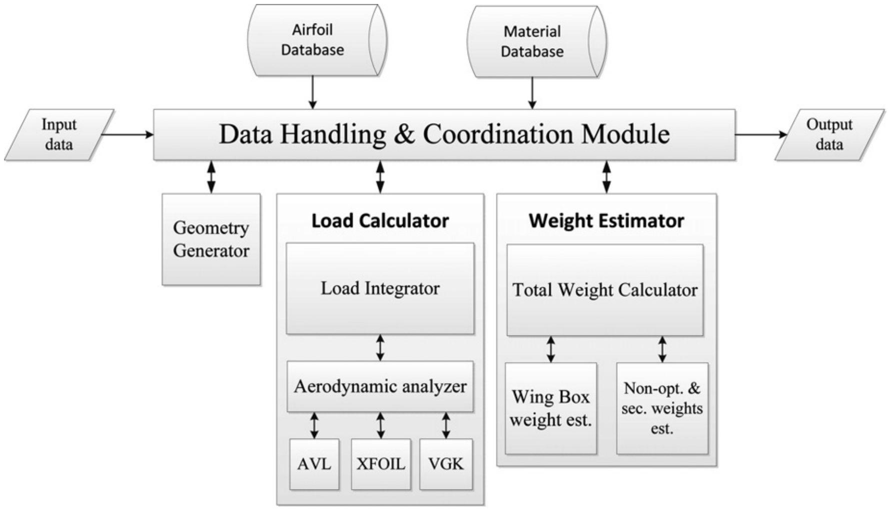



**Fig. 6.** EMWET main structure.

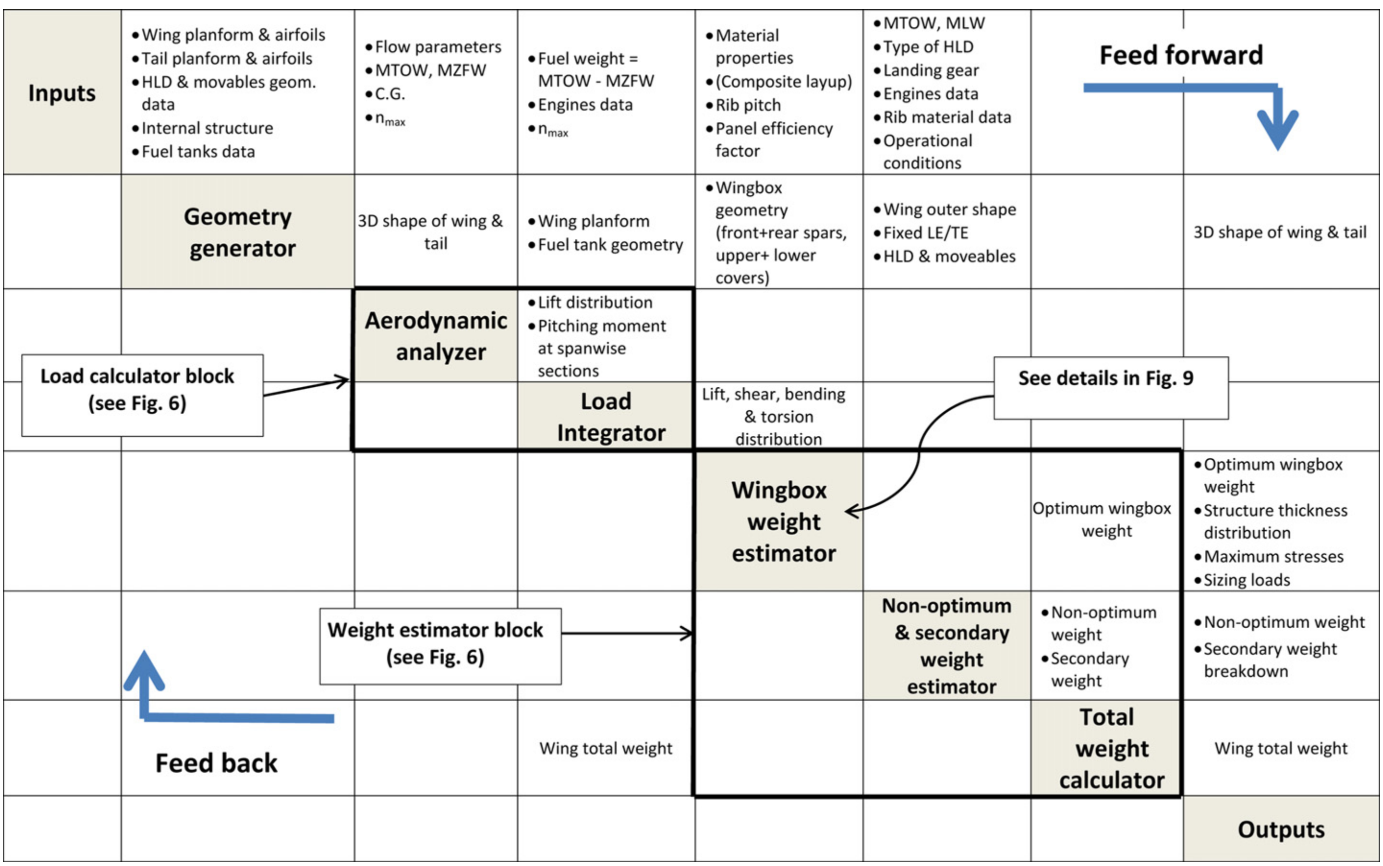



**Fig. 7.** EMWET N2 chart.

chord length at various (spanwise) wing stations, so that it is possible to model both straight and cranked spars.

Based on the same approach used to define the wing, also the geometry of various tail configurations can be defined. Concerning the wing weight estimation method discussed in this paper, only the aerodynamic surface of the horizontal tail is used, to account for its effect on the wing aerodynamic load distribution. However, a simplified version of EMWET has been successfully deployed also for the estimation of the tail group weight [11]. In that case, the geometry generator was used to define the inner and outer geometry of both wing and tail empennages. Eventually, the geometrical data necessary for the load calculator module (including the layout of the fuel tanks for the inertia relief), for the wing box sizing tool (including the detailed shape of the wing panels) and for the non-optimum and secondary weight estimator (including the areas of the various movable surfaces and high lift devices) are all consistently generated by the geometry generator. More details on the data exchange between the model generator and the other EMWET modules can be found in Fig. 7.

Apart from supporting the analysis process, the generator can be used also as simple visualization tool, to check the quality of the geometry model defined via input file. Examples of wing and tail configurations created by the geometry generator are shown in Fig. 8.

**Table 3**
Input parameters of EMWET.

| | | |
|---|---|---|
| Geometry | Wing<br>• Coordinate of leading edge point<br>• Chord<br>• Twist angle<br>• Incidence angle<br>• Name of the airfoil<br>Winglet<br>• Same data as wing<br>Tail<br>• Same data as wing | These data must be provided at least for two sections (root and tip). The airfoils coordinate must be provided in separate files. |
| Structure | • Chordwise position of front and rear spars<br>• Rib pitch<br>• Efficiency of stiffened panels | Position of spars are defined as percentage of local chord, at least at two sections (root and tip). |
| Material | Upper equivalent panel<br>• Material name<br>Lower equivalent panel<br>• Material name<br>Front spar<br>• Material name<br>Rear spar<br>• Material name | Each material is defined in a separate file including material properties such as density, Young modulus, ultimate and yield stresses (tensile, compressive and shear), etc. |
| Performance | • Flight altitude<br>• Maximum operating speed and Mach number<br>• Dive speed and Mach number<br>• Maximum load factor | The maximum maneuver load factor is given and the gust load factor is calculated by the tool. The critical one is used. |
| Weight | • Maximum take-off weight<br>• Maximum zero fuel weight<br>• Maximum landing weight<br>• C.G. position | |
| Engine | • Number of engines attached to wing<br>• Spanwise position of each engine<br>• Weight of installed engine (per engine) | The weight of installed engine including the weight of nacelle and pylon. |
| L.G. | • Number of landing gear attached to wing | |
| Fuel tank | • Spanwise position of fuel tank start and end | Fuel tanks are assumed to be placed between the front and rear spars. |

### 5.3. Load calculator

The sizing loads considered for the wing are derived from two scenarios, based on the flight maneuver and gust conditions, as defined in CS 25.331 to 23.341 [12]. The critical one is assumed for the weight estimation:

1. Symmetric maneuver load for maximum positive load factor at maximum takeoff weight and maximum operating speed ($V_{MO}$).
2. Gust load for maximum zero-fuel weight at maximum operating speed ($V_{MO}$).

In order to obtain proper load estimations, the aerodynamic loads are calculated on the complete set of lifting surfaces in trimmed condition. The proper elevator deflection angle is calculated in such a way to have zero pitching moment about the aircraft center of gravity. To this purpose, both the position of the aircraft center of gravity (C.G.) and the geometry of wing and horizontal are required, which are provided via the EMWET input file (Table 3) and delivered by the geometry generator, respectively. When computing the pull up maneuver load case, a quasi-equilibrium condition is assumed and the dynamic response of the aircraft to any elevator deflection is ignored. The aircraft weights (maximum take-off weight and zero fuel weight) provided via the input file are used for the aerodynamic calculations. Although the changes in wing weight during the iterations might alter the c.g. position, hence require a new aerodynamic calculation, this effect is ignored because of secondary order. However, an iteration loop is performed to account for the changes in the relief load due to the wing mass: since the loads are functions of the wing mass, the load calculator is called by the structural sizing module in an iteration loop until the solution converges. This feedback loop is clearly visible in Fig. 7 (see the wing total weight iteration).

As anticipated in Section 3, the aerodynamic loads on wing and tail are calculated by means of a quasi-3D aerodynamic solver, which has been developed especially in the context of for EMWET and represents one of the key elements to the accuracy of the proposed weight estimation method. This solver combines the commercial Vortex Lattice Method (VLM) tool AVL [6] with a 2D solver to calculate wing section properties at several spanwise stations. Actually, the 2D solver is one automatically selected by EMWET between two integrated analysis tools, namely XFOIL [7] and VGK [38]. XFOIL is used for subsonic flow conditions ($M < 0.6$), while

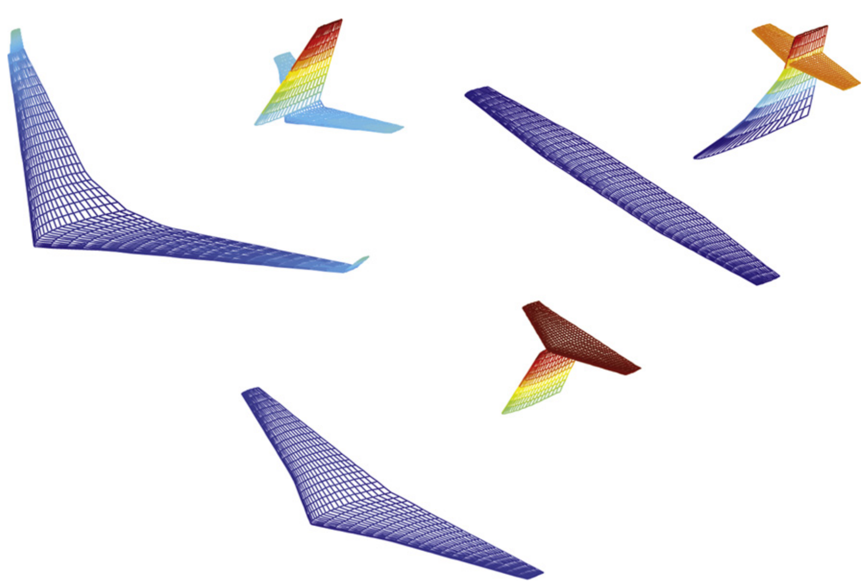


**Fig. 8.** 3D models of different wing and tail configurations.

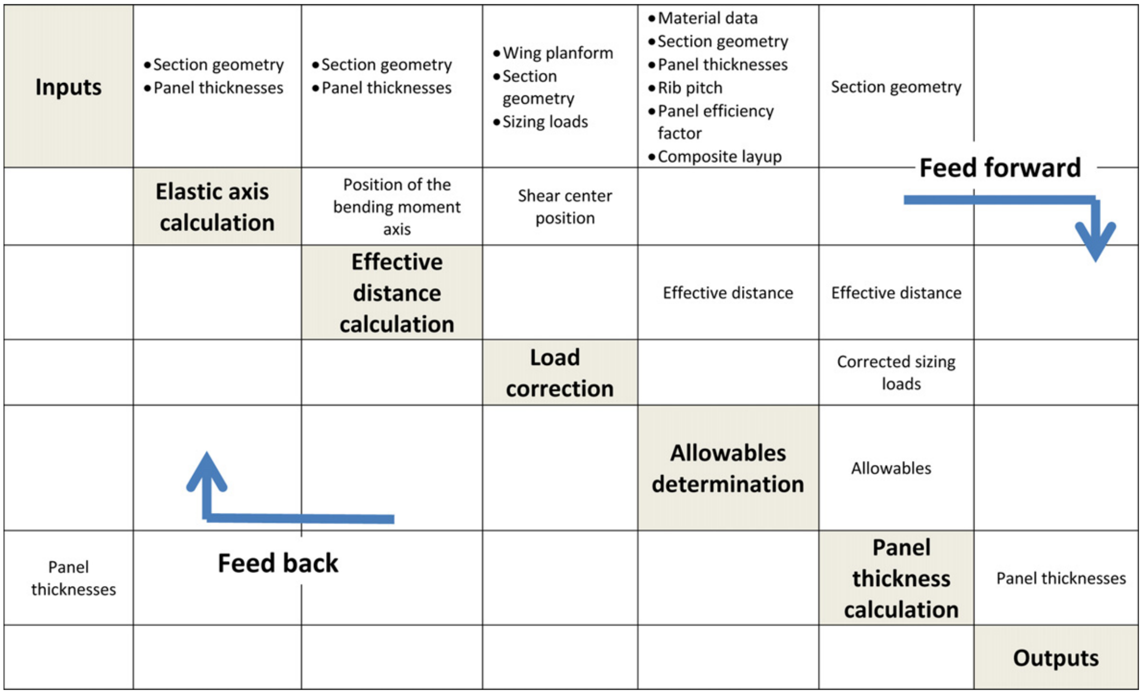



**Fig. 9.** N2 chart of the wing box weight estimation module.

VGK is used for transonic conditions ($M > 0.6$). First, the VLM code is used to calculate the lift distribution on the wing. Then, based on the calculated lift distribution, the local lift coefficients are determined at several spanwise positions. The selected 2D aerodynamic solver is then used to calculate the pitching moment of the wing sections at those spanwise positions, when operating at the determined local lift coefficients.

It should be noticed that the calculation of the aerodynamic pitching moment component is another important feature of the EMWET tool. Most of the weight estimation methods found in literature use only the pitching moment component due to the wing sweep angle, which is computed using the lift distribution and the wing planform geometry. However, also the effect of the airfoils shape on the wing pitching moment must be taken into account for an accurate estimation of the sizing loads, especially when advanced supercritical airfoils with aft-cambering are used, which produce a significant amount of pitching moment.

The inertia relief effect due to the mass of wing, fuel and wing mounted engines is also computed, such that the spanwise distributions of the shear force and the bending moment are calculated using the *net* vertical forces acting on the wing.

The torsional moment acting on the wing box structure is computed as the sum of the following three components: the aerodynamic pitching moment of the various wing sections (computed using the 2D solver), the moment due to the wing sweep angle (negative for back swept wings), and the moment due to the misalignment of the aerodynamic center (where the aerodynamic load and moments are placed) and the shear center of the various wing sections. The first two components are calculated by the load calculator about the quarter-chord line and later are corrected by the wing box weight estimation module based on the location of the elastic axis. The last component is actually added by the wing box weight estimation module, where the location of the shear center is determined. More details are provided in the next subsection.

### 5.4. Weight estimator

The weight estimator module contains three modules: 1) the wing box weight estimator, which includes the equivalent panels and spar webs analytical sizing, 2) the non-optimum and secondary weight estimator, and 3) a simple module called total weigh calculator that sums all the contributions.

The wing box weight estimation module contains the implementation of the structural sizing method described in Section 4. Here the geometrical data from the geometry generator, the structural and the material data provided via the input file, and the loads calculated by the load calculator are finally used to size the main wing box structural elements (i.e., upper and lower panel and spar webs).

An additional N2 chart is provided in Fig. 9, which illustrates the specific details of the analytical process used to size the wing box, as embedded in the module *wing box weight estimator*. The loads received from the load calculator need to be adjusted before proceeding with the actual structural sizing. The torsional loads should be transformed from the quarter-chord line to the elastic axis. Then all the loads are corrected in two steps: the loads are first multiplied by a safety factor equal to 1.5, and then aligned with the elastic axis of the wing in the following way:

$$V_\perp = V$$
$$M_\perp = \frac{M}{\cos \Lambda_{ea}}$$
$$T_\perp = \frac{T}{\cos \Lambda_{ea}} \tag{34}$$

where the $V_\perp$, $M_\perp$ and $T_\perp$ are, respectively, the shear force, the bending moment and the torsional moment perpendicular to the wing box elastic axis, and $\Lambda_{ea}$ is the sweep angle of the elastic axis (see Fig. 10).

The location of the elastic axis, which is required to correct the loads by means of Eq. (34), is determined using the thickness of the equivalent panels. However, the sizing loads required for calculating the thickness of the panels are functions of the location of the elastic axis. Beside that, the effective distance parameter at each wing section is a function of the thicknesses of the equivalent panels. On the other hand, the thicknesses of the equivalent panels can be determined once the value of the effective distance is known. In addition to these dependencies, also the allowables

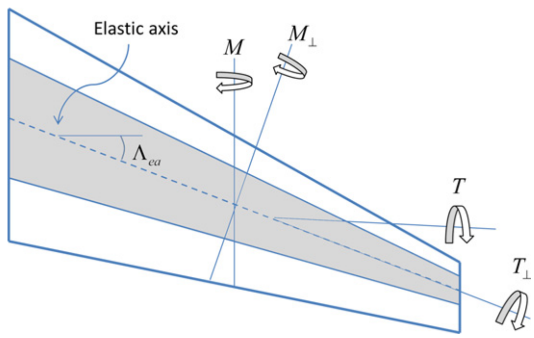



**Fig. 10.** Corrected sizing load based on the location of the elastic axis.

(yield and ultimate stresses, see Section 4.4.1) required to size the equivalent panels, are functions of panel thicknesses. This is true for metal structures, where the allowables for sheets and plates of the same alloy, but rolled or extruded with various thicknesses, are different [22]. All these dependencies are actually solved by a single panel thicknesses iteration loop, as shown in the N2 chart of Fig. 9. An initial value of the panel thicknesses is provided as input for the first calculations. After that, the new values of panel thicknesses are used for the next iterations until the solution converges.

The sizing module guarantees that all the calculated thicknesses are never lower than the minimum gage thickness defined by the user via input file.

Once the thicknesses of the equivalent panels and the spar webs have been computed by the structural sizing module, the weight contribution of the wing box can be easily determined on the basis of the density values of the used materials.

The weight contributions of the ribs and of all the various non-optimum and secondary structure components are computed using the empirical methods proposed by Torenbeek [35].

Finally, the total weight of the wing is determined as the sum of the above mentioned contributions and stored in an output file, together with the detailed breakdown.

## 6. Validation and discussion

The weight prediction method presented in this paper has been validated using the actual weight of various aircraft from different manufacturers, size and category. The required data for the validation, that is the geometrical and structural data of the considered aircraft and their actual weight breakdown, were gathered from literature [24,28,25] and via private communication with manufacturers. The results are presented in Table 4, where the error on the wing weight estimation is calculated as follows:

$$error = \frac{W_{\text{calc}} - W_{\text{actual}}}{W_{\text{calc}}} \tag{35}$$

The average accuracy of the method is about 2%, which is better than the results presented in Table 1 and comparable to the best existing FEM-based weight estimation methods. The computational time required for each weight estimation is in the order of 30 seconds, using a computer with a 2.00 GHz Intel Core2Duo E4400 processor and 2GB RAM memory, and includes the time to generate the wing geometrical model, compute the aerodynamic loads, and perform the required iterations.

It is interesting to observe that the achieved level of accuracy seems independent from aircraft type and manufacturer, although no specific tuning of the semi-empirical coefficients was performed within EMWET. Indeed the semi-empirical methods used in EMWET for the estimation of the non-optimum and secondary weight contributions were the same as proposed by Torenbeek [35]. This seems to suggest that by improving the quality of the wing box sizing process, the overall weight estimation results become less sensitive to the various semi-empirical coefficients.

**Table 4**
Validation results of wing weight estimation method.

| Aircraft | Maximum take-off weight (kg) | Error of wing weight estimation (%) |
|---|---|---|
| Fokker 50 | 20 820 | −0.72 |
| Boeing 737-200 | 52 390 | 0.15 |
| Boeing 727-300 | 95 028 | −2.71 |
| Airbus A300-600R | 170 500 | 1.86 |
| Airbus A330-300 | 217 000 | −2.18 |
| Boeing 777-200 | 242 670 | 2.66 |

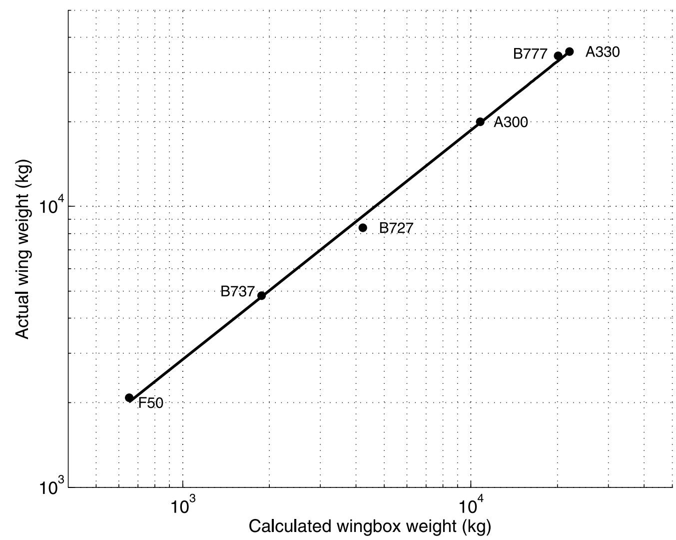



**Fig. 11.** Good correlation between calculated wing box weight and actual total wing weight.

Unfortunately, it was not possible to extend the validation process to a larger number of aircraft, due to the great difficulty in obtaining complete and reliable aircraft data set as required to run EMWET and perform comparisons. In particular, none of the aircraft used for validation has composite wing structure. Although EMWET ability to deal with composite structures has been verified [10], its validation is still an open issue.

A simple regression analysis has been performed to derive a relationship between the wing box weight contribution computed analytically (i.e., the weight of the upper and lower skin panels, stringers, spar caps and webs) and the actual total wing weight. The result of the power regression is given in Eq. (36) and illustrated in Fig. 11.

$$W_{total} = 10.147\, W_{wingbox}^{0.8162}$$
$$R^2 = 0.9982 \tag{36}$$

Given the very good fit, one could estimate the aircraft wing weight just using Eq. (36), without the need of any empirical method for secondary and non-optimum weights. This would reduce the amount of required input data significantly and appears to be a very suitable approach for early design stages, when data about high lift devices and control surfaces, for example, are not available yet.

The validation results presented in this section are for aircraft with conventional configurations. It should be noticed that, although the weight estimation method presented in this paper is developed and validated for aircraft with conventional configurations, the fact that the primary weight is calculated using physics based analysis and fully stressed design principles, makes it largely applicable to other kinds of lifting surface configurations, such as those considered, for example, within the NASA N + 3 program [4]. Strut braced wings, twin boom configurations, and flying wing configurations can be addressed by the method proposed in this paper, without significant modifications. For more radical designs such as box wing and joined wing configurations some equations may require modification or partial re-derivation. For example, in order to apply the proposed method to a joined wing configurations [39], where a large portion of the primary wing structure material is concentrated at the corners of the wing box because of the tilted bending moment axis of the complex wing system, all the equations derived assuming the neutral axis parallel to the airfoil chord should be re-derived. Also, the proposed method makes use of an empirical equation to account for the aeroelastics effect on the structure sizing. This relation would need to be modified to account for the adverse aeroelastics effect acting on forward swept wings. Otherwise, the proposed sizing method should be expanded to account for the aerolastic effects in a more physics based manner. Similarly, the relations proposed here to estimate secondary weights and non-optimum weights would need to be reconsidered in case of non-conventional configurations. Either a physics based sizing approach should be developed also for the non-primary weight contributions, or new sets of empirical equations should be derived.

The superior accuracy of the class II & 1/2 proposed in this paper is claimed to depend on two main contributions, namely the derivation of a more accurate and design sensitive definition of the effective distance parameter $\eta_t$, and the use of more realistic aerodynamic load estimations (via the quasi-3D analysis tool). Both of them, actually, are enabled by the geometry generator embedded in EMWET. It can be argued that a class II & 1/2 weight estimation method would not need to include dedicated geometry and load generators, as far as it is able to handle accurate (geometry and load) data provided externally. Hence, the method should be sensitive to the accuracy of geometric and load data, but not responsible for their generation. However, at the relatively early phase of the design process where class II & 1/2 methods are generally used, these data are often not readily available to the designer. Thereby, the need to generate them autonomously, as demonstrated by the various methods discussed at the beginning of the paper. However, when oversimplified approaches are used to generate these data, the quality of the weight prediction can be quite negatively affected. The positive effect of using the embedded geometry generator (to support the deployment of the advanced analytical sizing method proposed in this paper) and load estimation tool are highlighted in the following two subsections.

### 6.1. Effect of $\eta_t$ on the accuracy of wing weight estimation

As discussed in Section 4.2, a new method to calculate the airfoil effective distance parameter $\eta_t$ has been developed. This new method was compared with Shanley's and Torenbeek's method for the wing weight estimation of the Fokker 50 and Boeing 737. The results are summarized in Table 5.

### 6.2. Effect of lift distribution on the accuracy of wing weight estimation

Another study is done to investigate the effect of the lift distribution on the accuracy of the weight estimation. As mentioned in Section 2, several of the existing methods use empirical equations to estimate the lift distribution on the wing. Classic examples are those of lift distribution proportional to chord, and elliptical lift distribution. To compare the results obtained using these two assumed lift distributions with those achieved using a more realistic aerodynamic analysis, EMWET was run three times. In the first two cases, the VLM solver was switched off and the lift distribution imposed analytically. The results are shown in Table 6.

**Table 5**
Wing weight estimation using various methods to calculate $\eta_t$.

| Aircraft | Error of wing weight estimation (%) | | |
|---|---|---|---|
| | Torenbeek method | Shanley method | Elham method |
| Fokker 50 | 4.37 | 2.39 | −0.72 |
| Boeing 737-200 | 5.24 | 3.47 | 0.15 |

**Table 6**
Wing weight estimation using various methods to calculate lift distribution.

| Aircraft | Error of wing weight estimation (%) | | |
|---|---|---|---|
| | Elliptical | Proportional to chord | VLM |
| Fokker 50 | 1.54 | 3.11 | −0.72 |
| Boeing 737-200 | 3.58 | −0.72 | 0.15 |

From Table 6 one can observe that for the Fokker 50 aircraft the result of wing weight estimation using elliptical lift distribution is more close to reality, whereas, for the Boeing 737, the lift distribution proportional to chord is a better choice. Indeed, the Fokker 50 has a straight tapered wing that has a lift distribution very close to the optimum elliptical lift distribution, while the Boeing 737 has a swept wing, for which a lift distribution proportional to local chord seems to be more realistic. However, when EMWET makes use of the VLM tool to compute the lift distribution, the quality of the weight estimation is always better.

## 7. Conclusion

The various classes of weight estimation methods used in the aircraft design process have been discussed. In particular, a number of class II & 1/2 wing weight estimation methods from literature have been reviewed to assess their accuracy and identify the main causes of their limitations. Based on the outcome of this investigation, a new advanced quasi-analytical wing weight estimation method has been developed, which is able to account the effect of many detailed design choices (such as airfoil shape, spars position, type of stringers, etc.) on the wing weight. The proposed method was demonstrated to have an outstanding level of accuracy, with an average error in the order of 2%. This is much higher than all the other considered quasi-analytical weight estimation methods, whose average error is often in excess of 25%. Even more, the accuracy of the proposed method was proven to be independent of aircraft size, typology and manufacturer.

Similarly to other class II & 1/2 wing weight estimation methods, the one proposed in this paper is based on the elementary sizing of the primary wing box structure to calculate the amount of material required to resist the applied loads. The secondary weights (weight of flaps, slats, etc.) and the non-optimum weights (weight penalties due to joints, attachments, cutouts, etc.) are estimated using various empirical methods.

The novelty elements of the proposed method consist in a new analytical method to size the wing box structure and an advanced modeling and analysis system to obtain accurate loads estimations. In particular, an improved analytical formulation of the so-called wing panel effective distanced has been derived, which is an essential element to make the proposed weight estimation method sensitive to the aerodynamic shape of the wing. Besides, the overall wing box sizing methods has been extended to composite structures. A quasi-3D aerodynamic analysis tool has been developed to provide the wing box analytical sizing approach with a realistic load distribution, able to account also for the presence of winglets and tailplanes. Furthermore, the quasi-3D tool is able to compute the aerodynamic pitching moment generated by the various wing sections, such that the non-negligible torsional contribution of supercritical airfoils, for example, can be accounted as well. A dedicated geometry generator represents the last key element in the implementation of the proposed method. This allows the user to model any type of wing geometry and then takes care of feeding the various analysis and sizing modules with detailed and consistent geometry datasets. This combination of accurate geometry models, improved loads estimation and advanced analytical structural sizing techniques represents not only the novelty of the proposed method, with respect to other class II & 1/2 methods, but the very reason to its accuracy.

Apart from its outstanding accuracy, typical of higher class weight estimation methods, and its ability to assess the effect of detailed design decisions (from planform and airfoils shape to structure layout and material selection), the proposed tool maintains the high computational efficiency of lower class methods. Although some class II methods exist with acceptable level of accuracy, and even lower computational cost than any class II & 1/2 method, these empirical methods are only suitable when very little high level design parameters (such as MTOW, wing area and span) are available and cannot help designers assessing the effect of other design decisions such as airfoil shape, internal structure layout, etc. As such, the new developed class II & 1/2 method also qualifies as suitable tool to support multidisciplinary design optimization studies.

Beside that, since some thickness distribution of skin panels and spars web are obtained during the sizing process, the EMWET can be used not only as a weight estimator but also as a simple sizing tool, whose output could be used to start a more sophisticated FEM-based sizing process.

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