Adaptable conceptual aircraft design model

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Abstract. This paper presents a new conceptual design model ACAD (Adaptable Conceptual Aircraft Design), which differs from the other models due to its considerable adaptability to the different classes of aircraft. Another significant feature is the simplicity of the process which leads to the preliminary design outputs and also allowing a substantial autonomy in design choices. The model performs the aircraft design in terms of total weight, weight of aircraft subsystems, airplane and engine performances, and basic aircraft configuration layout. Optimization processes were implemented to calculate the wing aspect ratio and to perform the design requirements fulfillment. In order to evaluate the model outcomes, different test cases are presented: a STOL ultralight airplane, a new commuter with open-rotor engines and a last generation fighter.

Keywords: conceptual design; weight estimation; aircraft performances; design optimization; design requirements

1. Introduction

Many Aircraft Conceptual Design models have been developed during the past decades. Roskam (2003), Raymer (2012) and Jenkinson et al. (1999) are some of the most highly valued models. The research group to which the author belongs has been interested in this subject for many years. Its results (Chiesa and Maggiore 1995, Chiesa et al. 2000, Antona et al. 2009, Chiesa et al. 2012) represent the background of the ACAD model. These models are widely used and provide reliable and consistent results. The ACAD Model, which is discussed here, does not aim to overcome these models in terms of quantity and accuracy of results, but seeks to add flexibility and simplicity of use. The number of data input can be considered smaller than those of the other conceptual design tools and, withstand standard airplanes knowledge, the data required are not difficult to define. Moreover, the model is implemented in Microsoft® Excel work sheets (without visual basic macro) in order to simplify its use and keep the users constantly aware about the design process and avoid “black box” automatic procedures. The other main feature, the model flexibility, consists in providing the user with the capability to design a wide variety of aircraft categories (a good example is Azamatov et al. 2010). This is not a common feature, in fact several tools have been developed to design one or a small number of specific class of airplane, e.g. Jenkinson et al. (1999) model focuses on civil jet aircraft. The model process, formulas and input data are briefly described through the conceptual design of three different test cases here:

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Two-seat Short Takeoff and Landing (STOL) Ultralight Aircraft. The main requirement for this test case is the maximum take-off weight ($W_{MTO}$) which must not be greater than 450 Kg due to airworthiness regulations. For this aircraft category, the weight limit differs for each country, due to the lack of common regulations. Consequently the strictest one is considered in this study. Moreover, given the STOL requirement, the ultralight can operate from very short length airfields (about 100 meters).

Fast and Green Regional Aircraft. This aircraft has to carry 90 passengers for short-haul flights. The fuel consumption has to be similar to regional turboprop, but the cruise speed ($V_{CR}$) must be greater (at least 800km/h). Considering these requirements this new regional aircraft is equipped with Prop-Fan engines (McDonald et al. 2008, Lee et al. 2009).

Fifth Generation Fighter. This is a valuable test case, as it includes all model features. The payload weight ($W_{PL}$) and combat radius requirements, which are respectively 1200 kg of weapons and 1200 km, classify that airplane as a light fighter. It should be a new segment of this class of fighter since the others, which are already flying, such as Lockheed Martin F-22, F-35 and Sukhoi PAK FA, have greater weapon weight and radius which means a greater $W_{MTO}$ (respectively of 38, 32 and 37 tons). Combat turn and super cruise requirements are also considered in the conceptual design process.

These test cases were selected in order to emphasize the ACAD model flexibility. As it is possible to understand from their descriptions, they notably differ in terms of $W_{MTO}$, $V_{CR}$, mission profile, configuration, wing loading, class of propulsion system and basically all the remaining airplane characteristics.

The present work starts with the description of the test case design requirements and the analysis of the model design process. Then it continues with the examination of the aspect ratio optimization subroutine and the matching chart subroutine necessary to verify the design requirements fulfillment. Finally, the test case results are analyzed.

### 2. Main design process

The test cases main requirements are listed in Table 1. In addition to the payload and crew weight requirements, which were already defined in the tests case description, the cruise range

<table>
<thead>
<tr>
<th></th>
<th>STOL Ultralight</th>
<th>Fast Regional</th>
<th>Fifth Gen Fighter</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload and crew</td>
<td>195 (2 persons + baggage)</td>
<td>9570 (90 pax + crew + baggage)</td>
<td>1300 (weapons + pilot)</td>
</tr>
<tr>
<td>weight [kg]</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>D$_{TO}$ [m]</td>
<td>100</td>
<td>1350</td>
<td>550</td>
</tr>
<tr>
<td>D$_{LND}$ [m]</td>
<td>75</td>
<td>1000</td>
<td>550</td>
</tr>
<tr>
<td>V$_{CR}$ [km/h]</td>
<td>110</td>
<td>800</td>
<td>900</td>
</tr>
<tr>
<td>R$_{CR}$ [km]</td>
<td>450</td>
<td>1200</td>
<td>2400</td>
</tr>
<tr>
<td>Supersonic Cruise</td>
<td>-</td>
<td>-</td>
<td>1.4 M @ 6000 m</td>
</tr>
<tr>
<td>parameters</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Combat Turn</td>
<td>-</td>
<td>-</td>
<td>1.4 M @ 1000 m @ 5 g</td>
</tr>
<tr>
<td>parameters</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
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\( (R_{CR}) \), the takeoff distance \((D_{TO})\) and landing distance \((D_{LND})\) are now assumed.

The passenger and crew weight for civil aircraft test cases is defined, starting from (Roskam 2003), considering a weight of 93 kg and a baggage of about 9 kg per passenger (or per crew member). With regard to the military test case, 100 kg of pilot weight and 1200 kg for weapons are taken in to account. The additional requirements for the Fifth Gen. Fighter test case, which are supersonic cruise and combat turn (combat turn definition is the same utilized in Saha et al. 2008), are quantified according to typical military aircraft performance. As regard the combat turn parameters, it is worth noting that they are able to fully define the sustained turn maneuver. The angle of bank can be obtained from the load factor during turn \((n_{turn})\) and the turn radius from bank angle and turn speed \((V_{turn})\).

The flow chart in Fig. 1 summarizes the whole design process. The design requirements, which have been previously defined, have to be integrated with attempted design parameters in order to determine all the needed variables. These guess data are the \(W_{MTO}\) the parasite drag coefficient \((C_{d0})\) and the wing aspect ratio \((AR)\). With basic knowledge of aircraft design, it is not difficult to assume guess values for these data. In any case, the model is able to converge to the proper value through iterative loops even if the guess data are far from rational ones.

As the flow chart describes (see Fig. 1), the fuel weight \((W_{F})\) is the first parameter which is calculated. The model determines \(W_{F}\) as composed of three terms. The first term is the fuel necessary to perform the taxi, takeoff, climb, descent and landing phases \((W_{F\_OTHER})\). It is calculated by Eq. (1) which uses statistical data (Roskam 2003) that are different for each aircraft category and are parameterized by \(k_{F}\) coefficient described in Table 2. These data take into account the different mission profiles of each category and the different engine specific fuel consumption \((SFC)\).

\[
W_{F\_OTHER} = W_{MTO} \cdot k_{F} \tag{1}
\]

The second term (the most substantial) is the fuel necessary to perform the cruise mission phase \((W_{F\_CR})\). The estimation is carried out through Eq. (2) where thrust \((T)\) or power \((P)\) required and the cruise time are multiplied by the engine SFC. The \(T\) or \(P\) required is defined through the lift to drag ratio calculated at cruise condition \((E_{CR})\). The aircraft weight is defined as the \(W_{MTO}\) minus the half \(W_{F\_CR}\) and a fraction of \(W_{F\_OTHER}\) which considers only the fuel burned during taxi, takeoff and climb.

\[
W_{F\_CR} = \left( W_{MTO} - \frac{W_{DPL}}{2} - \frac{W_{F\_CR}}{2} \right) \cdot \frac{R_{CR}}{E_{CR} \cdot V_{CR}} \cdot SFC
\]

\[
\Rightarrow W_{F\_CR} = \frac{R_{CR} \cdot SFC}{E_{CR} \cdot V_{CR} \cdot \left( W_{MTO} - \frac{W_{DPL}}{2} \right)} \tag{2}
\]

For military aircraft, i.e. fighters, bombers and in some cases cargos (e.g. airdrop mission) that have to carry a dropping payload, the half dropping payload weight \((W_{DPL})\) is also subtracted to \(W_{MTO}\). In this way, the model takes into account that the second half of the cruise phase is performed without the dropping payload. Therefore, it is worth noting that all parameters, which are used to carry out the \(W_{F\_CR}\) estimation, are assumed considering the airplane in the middle of its cruise phase. This is a model simplification which is useful in order to reduce the number of input and consequently the workload of design process. The input data are \(R_{CR}\) and \(V_{CR}\), which can
Fig. 1 ACAD model process flow chart
Adaptable conceptual aircraft design model

Table 2 Fuel fraction for taxi, takeoff, climb, descent and landing

<table>
<thead>
<tr>
<th>Aircraft category</th>
<th>Fuel weight for taxi, takeoff, climb, descent and landing [% of $W_{MTO}$]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Homebuilt</td>
<td>2.1</td>
</tr>
<tr>
<td>General aviation</td>
<td>3.6</td>
</tr>
<tr>
<td>Regional turboprop</td>
<td>5.4</td>
</tr>
<tr>
<td>Transport jet</td>
<td>6.1</td>
</tr>
<tr>
<td>Fighter</td>
<td>14</td>
</tr>
</tbody>
</table>

be considered as design requirements, and the engine SFC, which can be estimated using statistical data provided in the model.

The last term is used only for military aircraft which perform a combat action during their mission profile. The model uses the Eq. (3) which estimates the fuel necessary for combat action ($W_{F,\text{COMBAT}}$) as the product of engine SFC at maximum thrust ($SFC_{\text{COMBAT}}$), the thrust required and the time duration of the combat action ($t_{\text{COMBAT}}$).

$$W_{F,\text{COMBAT}} = SFC_{\text{COMBAT}} \cdot \left( \frac{T}{W} \right)_{\text{COMBAT}} \cdot t_{\text{COMBAT}} \cdot \left( W_{MTO} - \frac{W_{F,\text{CR}}}{2} - \frac{4}{5} W_{F,\text{OTHER}} \right) \tag{3}$$

These parameters may be different than those related to the other mission phases. The thrust to weight ratio, from which the thrust is obtained, has been considered a specific design requirement and $SFC_{\text{COMBAT}}$ has to be specified since it can be very different from the cruise SFC, for example, when the designed aircraft turns on the engine afterburner.

Finally, the total fuel weight is estimated using the Eq. (4)

$$W_F = W_{F,\text{CR}} + W_{F,\text{COMBAT}} + W_{F,\text{OTHER}} \tag{4}$$

It is worth noting that the ACAD model is able to design aircraft with different kind of propulsion, in particular, aircraft powered by jet or propeller engine. Therefore the model automatically expresses the Eq. (2) in terms of thrust, for jet aircraft, or in terms of power when a propeller engine is selected. To carry out the conversion, the model utilizes the Eq. (5) to transform the power SFC ($SFC_p$) expressed as $[\text{lb} \cdot \text{h} / \text{hp} \cdot \text{h}]$ to thrust SFC ($SFC_T$) expressed in $[\text{lb} / \text{h}]$ also taking into account the propeller efficiency ($\eta_P$)

$$SFC_T = \frac{SFC_p \cdot V_{CR}}{550 \cdot \eta_P} \tag{5}$$

According to the model flow chart (see Fig. 1), after fuel weight estimation, it is now possible to calculate the aircraft wing surface ($S_w$) and the $Cd_\beta$. With the aim of creating a simple design process, the model uses statistical equations by Roskam (2003). The necessary input, which is functional to estimate the $Cd_\beta$, is the following: aircraft class, $W_{MTO}$ and $S_w$. The aircraft class is known by the designer as $W_{MTO}$ was previously attempted. The $S_w$, on the contrary, has to be calculated. This is performed considering the airworthiness regulation (FAR 23 / 25), the lift equation and, once more, the statistical equations by Roskam (2003) which correlates, through Eq.
(6) and (7), the $D_{LND}$ with the aircraft approach speed ($V_{App}$) using stall the speed at landing configuration ($V_{S,LND}$).

$$D_{LND} = 0.5136 \cdot V_{S,LND}^2$$  \hspace{1cm} (6)

$$V_{App} = 1.3 \cdot V_{S,LND}$$  \hspace{1cm} (7)

The Eq. (8) defines the landing weight ($W_{LND}$) as $W_{MTO}$ to which the half $W_F$ and $W_{DPL}$ (for a military aircraft) is subtracted. Obtaining the $S_W$ from Eq. (9), after having assumed a value of maximum lift coefficient at landing configuration ($C_{l_{MAX,LND}}$) and following Roskam’s methodology (Roskam 2003), the statistical figure of $C_{d0}$ can be now estimated. The term $\rho_{LND}$ represents the air density at airport altitude. The model performs the first iterative loop comparing the $C_{d0}$ calculated value with the guess value. Until the value becomes stable, the model recalculate the $W_F$, $S_W$ and $C_{d0}$ estimation blocks.

$$W_{LND} = W_{MTO} - \frac{W_F}{2} - W_{DPL}$$  \hspace{1cm} (8)

$$\frac{W_{LND}}{S_W} = 0.5 \cdot \rho_{LND} \cdot V_{APP}^2 \cdot C_{l_{MAX,LND}} \Rightarrow S_W$$  \hspace{1cm} (9)

The $AR$ is a main design parameter since its importance in influencing, primarily, the $V_{CR}$ at best range flight condition, $W_F$ and the wing weight ($W_w$). Therefore, the $AR$ definition is not a simple process and a choice led by statistical analysis would over reduce the model flexibility. For this reason, the $AR$ calculation is particularly accurate in ACAD model and it is performed by specific model subroutine which is described in a specific chapter of this work. The model carries out the second iterative cycle as long as the $AR$ is stabilized.

As shown in Fig. 1, the model performs a simple airframe and systems weight estimation. The model uses a set of weight estimation relationships (WERs). This set of WERs were developed by the research group in which the author works, that has always been interested in weight estimation topic since its extremely relevance in the conceptual design subject (Chiesa 1977, Chiesa et al. 2000, Chiesa and Viola 2007). The WERs are listed below (Eq. from Eq. (10) to Eq. (14))

$$W_W = \frac{W_W}{S_W} \cdot S_W \cdot K_{comp} \cdot K_{WP} \cdot K_{delta}$$  \hspace{1cm} (10)

$$W_{tail} = K_{tail} \cdot W_W$$  \hspace{1cm} (11)

$$W_{fus} = \frac{L_{fus}}{1000} \cdot n_{max} \cdot W_{MTO} \cdot K_{MTO} \cdot K_{comp} \cdot K_{delta}$$  \hspace{1cm} (12)

$$W_{inst \eng} = W_{eng} \cdot K_{eng} \cdot n_e$$  \hspace{1cm} (13)

$$W_{sys} = \sum_i K_{sys,i} \cdot W_{MTO}$$  \hspace{1cm} (14)
The wing weight is estimated with Eq. (10) which considers parameters such as wing weight to wing area ratio, wing area, AR, tapered ratio ($\lambda$), maximum load factor ($n_{max}$) and the airframe material. The model employs three different wing weight to wing area ratios, which are suitable for each of the following aircraft classes: military, civil transport and light airplane. In Figs. 2 and Fig. 3 the different ratios are shown.

Eq. (11) performs the tail weight ($W_{\text{tail}}$) estimation using a statistical ratio between $W_{\text{tail}}$ and $W_W$. This ratio can vary from 0.1 to 0.3 or more and represents, in that order, the ratio for a simple tail and the one for a complex multi fin tail. This ratio also depends on tail to wing surface value which varies in relation to different tail-to-airplane c.g. (center of gravity) distance. The other coefficients, $K_{\text{comp}}$, $K_{\text{WP}}$, $K_{\text{delta}}$, are used to modify the equation to take in to account, respectively, the effect, in terms of wing weight, of composite material, AR and delta wing configuration. The
Table 3 WERs coefficients

<table>
<thead>
<tr>
<th>Aircraft element</th>
<th>Coefficient</th>
<th>Coefficient range</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>$K_{\text{comp}}$</td>
<td>$1 \div 0.65$</td>
<td>% of composite material</td>
</tr>
<tr>
<td>Wing</td>
<td>$K_{\text{wp}}$</td>
<td>Calculated by Eq. (15)</td>
<td>Wing plane shape (AR and $\lambda$)</td>
</tr>
<tr>
<td>Wing</td>
<td>$K_{\text{delta}}$</td>
<td>$1 \div 0.9$</td>
<td>Delta wing configuration</td>
</tr>
<tr>
<td>Tail</td>
<td>$K_{\text{tail}}$</td>
<td>$0.1 \div 0.3$</td>
<td>Tail complexity</td>
</tr>
<tr>
<td>Fuselage</td>
<td>$K_{\text{comp}}$</td>
<td>$1 \div 0.65$</td>
<td>% of composite material</td>
</tr>
<tr>
<td>Fuselage</td>
<td>$K_{\text{delta}}$</td>
<td>$1 \div 0.9$</td>
<td>Delta wing configuration</td>
</tr>
<tr>
<td>Engine</td>
<td>$K_{\text{eng}}$</td>
<td>$1.1 \div 1.25$</td>
<td>Aircraft configuration (from fighter to civil transport)</td>
</tr>
</tbody>
</table>

Table 4 WERs coefficients for aircraft system

<table>
<thead>
<tr>
<th>Aircraft System</th>
<th>Coefficient range</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>Landing gear system</td>
<td>$0.022 \div 0.045$</td>
<td>Complexity (fixed or retractable)</td>
</tr>
<tr>
<td>Flight control system</td>
<td>$0.015 \div 0.04$</td>
<td>Complexity (flight control)</td>
</tr>
<tr>
<td>Hydraulic system</td>
<td>$0.005 \div 0.03$</td>
<td>Complexity (flight control and landing gear)</td>
</tr>
<tr>
<td>Electric system</td>
<td>$0.020 \div 0.04$</td>
<td>Complexity (flight control and avionics)</td>
</tr>
<tr>
<td>Fuel system</td>
<td>$0.015 \div 0.02$</td>
<td>n. engines and presence of after burner</td>
</tr>
<tr>
<td>Air conditioning system</td>
<td>$0.005 \div 0.07$</td>
<td>n. passengers</td>
</tr>
<tr>
<td>Avionic system</td>
<td>$0.030 \div 0.06$</td>
<td>Aircraft role (transport, trainer, fighter, multirole)</td>
</tr>
<tr>
<td>Engine system</td>
<td>$0.005 \div 0.015$</td>
<td>n. engines</td>
</tr>
<tr>
<td>Furnishing</td>
<td>$0.005 \div 0.04$</td>
<td>Passenger comfort and flight duration</td>
</tr>
</tbody>
</table>

$K_{WP}$ coefficient can be evaluated through Eq. (15).

$$K_{WP} = (0.04 \cdot AR + 0.6) \cdot [1 - 0.4 \cdot (0.5 - \lambda)]$$ (15)

The fuselage weight ($W_{\text{fus}}$) is estimated by Eq. (12) and, as for Eq. (10), it incorporates some coefficients which are able to modify the results considering fuselage material, dimensions (fuselage length $l_{\text{fus}}$), configuration and $n_{\text{max}}$. The engine weight ($W_{\text{eng}}$) (see Eq. (13)), as well as the number of engines ($n_e$), are input and through statistical graphs the model suggests turbofan, turbojet, turboprop and piston engine weight using its thrust or power as selection parameter so as to estimate the installed engine weight ($W_{\text{inst,eng}}$). Eq. (14) is used to calculate the airplane sub systems weight ($W_{\text{sys}}$). For each subsystem, the model provides a range of system weight to $W_{\text{MTO}}$ ratio ($K_{\text{sys}}$). These ratios vary with the subsystem complexity.

In Tables 3 and 4, respectively, the airframe and systems coefficients are described.

Totaling Eqs. (10), (11), (12), (13) and (14), the aircraft empty weight ($W_e$) can be now estimated. Therefore, summing up $W_f$ which was calculated before and $W_{PL}$ that was defined as design requirement, the model achieves the airplane $W_{\text{MTO}}$. As result of first iterative loop, the calculated $W_{\text{MTO}}$ probably differs from the value attempted at the start of the design process. However, as shown in Fig. 1, replacing the attempted value with the calculated one, in some loops the two values converge.

At this point of the design process, the designed airplane is consistent since whole input and
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calculated data are coherent. However, it is not yet possible to determine if the aircraft is capable to meet all design requirements. In more details, considering the requirements listed on Table 1, the aircraft fulfill the cruise requirement \( V_{CR}, R_{CR} \) and altitude), the landing requirement \( D_{LND} \) for selected airport altitude) and it is able to carry the defined payload. The takeoff requirement \( D_{TO} \) for selected airport altitude) and, only for military aircraft, the combat turn and the supersonic cruise requirements are not yet verified. As shown in Fig. 1, the ACAD model uses a specific subroutine, which is deeply described later, to check the requirements fulfillment. This subroutine graphically displays the design requirements and how the aircraft meets them. If the aircraft does not satisfy or over satisfy the requirements, a design review is needed and it is performed by varying appropriately the aircraft parameters and repeating the design process. Otherwise the design can be considered complete, coherent with the requirements and optimized.

3. Aspect ratio optimization subroutine

In the ACAD model, wing AR is considered as one of the major design parameters. AR influences greatly the airplane performances, \( W_{W}, W_{F} \) and, as consequence, airplane \( W_{MTO} \). The AR optimization subroutine is implemented in the model using the SOLVER add-in of Microsoft® Excel. The nonlinear solving algorithm used in the SOLVER add-in is the Generalized Reduced Gradient method (GRG2). This function permits to define a target parameter, which is used to evaluate the optimization process, and a set of other variables which have to be optimized to modify the target parameter which is directly or indirectly influenced by them. Others parameters can be introduced as constraints. Consequently, the SOLVER subroutine performs the optimization process even keeping in mind the constraint compliance.

Focus on AR optimization subroutine and considering its flow chart presented in Fig.4, the target, the variable to be optimized and the constraints are defined as follows:

- The variable that has to be optimized is the wing AR. This wing parameter cannot be easily calculated because is related in different manner to \( V_{CR}, W_{F}, W_{W} \) and consequently to aircraft \( W_{MTO} \). Therefore, the AR can be determined only after the optimization process.
- \( V_{CR} \) is a design requirement and is defined as constraint and the SOLVER subroutine identifies only solutions which achieve this speed.
- \( W_{MTO} \) is chosen as target parameter (Lombardi et al. 2006). In this way, using this all-comprehensive design parameter, it is possible to evaluate the whole effect of the different AR figures.

As described in Fig. 4, the optimization process uses an iterative calculation where different AR figures are assumed. After this process, the AR value which minimizes \( W_{MTO} \) is chosen. At the process beginning, an AR value is attempted. Substituting the AR value into Eq. (10), a new wing weight figure is estimated. Considering invariable the other parameter used in Eq. (10), it is worth noting that when the AR value increases \( W_{W} \) also increases. It is known that a wing with a greater wing span and the same surface has a strengthened structure to withstand the additional flectional momentum. A strengthened structure means a greater wing weight.

At the same time, using Eq. (16) (where \( \rho_{CR} \) is the air density at cruise altitude) the value of \( C_{l_{CR}} \) (Cl at cruise condition) is calculated. Through \( C_{l_{CR}} \), the AR guess value, \( C_{d_{0}}, Oswald's \) coefficient \( (e) \) and Eq. (17) the model calculates a new figure of \( C_{d_{CR}} \) (drag coefficient at cruise condition).
\[
C_l_{CR} = \frac{2 \cdot \left( W_{MTO} - \frac{W_{DPL}}{2} - \frac{W_{F, CR}}{2} \right)}{\rho_{CR} \cdot S_w \cdot V_{CR}^2}
\]

(16)

\[
C_d_{CR} = C_d_0 + \frac{C_l_{CR}^2}{\pi \cdot AR \cdot e}
\]

(17)

The new value of \( C_d_{CR} \) leads to a different lift to drag ratio \((L/D)\) which is used to calculate \( W_{F, CR} \) through Eq. (2). Taking into account Eqs. (16) and (17), it is worth noting that \( L/D \) is influenced by \( AR \) and \( L/D \) increases with the wing \( AR \). This produces a \( C_d_{CR} \) reduction and hence a lower weight of fuel burned (see Fig. 5).

As seen in the main loop (see Fig. 1), the model calculates a new value of aircraft \( W_{MTO} \) and, until the lower value of it is found, the model attempts the above calculation with other values of
AR. In Fig. 6 the effect of the wing AR on $W_{MTO}$ is shown. An increase in AR produces an increment of $W_{MTO}$ due to higher wing weight and, at the same time, a decrease of $W_{MTO}$ due to the reduction of $W_F$. Moreover, $W_{MTO}$ reduction due to fuel saving is lower than its increment caused by $W_F$ rise. The opposite effects are produced by lower AR values.

It is worth to note that an increment of fuel or wing weight generates an increment of the weight of other aircraft components. For example, if the wing were heavier, probably, the fuselage should be strengthened to withstand the additional load. The same could be said for landing gear and other airframe parts and aircraft subsystems. For this reason the $W_{MTO}$ increment is greater than just the increment of fuel or wing weight. Therefore, the difference in terms of $W_{MTO}$, caused by AR variation, is not negligible and, also for that reason, the $W_{MTO}$ can be considered an excellent target parameter for the AR optimization process.

For supersonic aircraft, another requirement has to be taken into account. AR variation, with constant $S_w$, determines a variation of wing span ($b$). During supersonic flight, the wing has to be inside the aircraft Mach cone, therefore, not all values of $b$, hence the AR, are allowed. To facilitate the requirement achievement, the ACAD model draws a simplify aircraft plan view (see Fig. 7).
Fig. 7 Mach cone requirement

Fig. 8 Design requirements fulfillment
4. Matching chart subroutine

The second half of the main design process involves the design requirements fulfillment as depicted in Fig. 8.

In this part of the design process, $D_{TO}$, $D_{LND}$, $V_{CR}$, climb speed and the other requirements related to military aircraft, such as super-cruise and combat turn capabilities, are expressed in terms of thrust (or power, for propeller driven aircraft) to weight ratio and wing loading. In order to simplify the evaluation of requirements fulfillment a graphical approach is implemented. Therefore, the design requirements are depicted on the same diagram which is identified, in the ACAD model, as matching chart. As shown in Fig. 8, the axes of the matching chart are: thrust (or power) to weight ratio and the wing loading calculated in takeoff condition. This means that the $D_{LND}$ requirement, the $V_{CR}$ requirement etc. are recalculated considering the $W_{MTO}$ and the thrust (or power) which the engine should generate at sea level altitude. In this manner, it is possible to compare the design requirements, which have to be satisfied in the different flight phases, in the same graph. If one or more requirements were not satisfied, it would be necessary to carry out a design review. It consists in modifying the aircraft parameters which influence the requirements. This issue is discussed later in this chapter.

Considering the matching chart, it is worth noting that a value of thrust (or power) to weight ratio can be identified. Knowing the aircraft weight, the engine maximum thrust (or power) can be calculated and, by this, it is possible to review the engine weight estimation which was attempted in the first step of the design process.

The $D_{LND}$ requirement is displayed on the matching chart as a vertical line which identifies a specific value of wing loading (see line n.1 on Fig. 9 (a) and Eq. (18)).

\[
\frac{W}{S_w} \leq \frac{W_{MTO}}{S_w}
\]  

(18)

As seen in section 2, this requirement imposes, by Eqs. (6) and (7), a maximum $V_{App}$ which defines, through Eq. (9), a minimum value of $S_w$. Combining this value of $S_w$ with aircraft $W_{MTO}$, the wing loading obtained represents the landing requirement evaluated in takeoff condition. It embodies the maximum wing loading, at takeoff weight, allowed to satisfy the $D_{LND}$ requirement. Therefore, lower values, e.g. lighter aircraft or greater $S_w$, would be permitted. Moreover, no thrust

---

Fig. 9 Matching chart. Influence of design parameters on requirements
(or power) to weight value is needed for this requirement. Another important issue is the identification of the design parameters which are able to influence the landing requirement. By Eqs. (6), (7), (8) and (9) the only design parameters which modified the landing requirement are the maximum lift coefficient in landing configuration \( \left( \text{Cl}_{\text{MAX,LND}} \right) \) by varying the type of high lift device, and evidently the \( D_{\text{LND}} \). The effects of the increment of these parameters are included in Fig. 9 (b).

Eqs. (19) and (20) are used to determine the \( D_{\text{TO}} \) requirement for, respectively, jet and propeller driven aircraft. Eq. (19) is a statistical equation from Roskam (2003) and Eq. (20) is an adaptation for propeller driven aircraft where \( T_{\text{SL}} \), \( P_{\text{SL}} \) and \( \sigma \) are, respectively, the engine thrust at sea level, the engine power at sea level and the ratio between the air density at airport altitude and the air density at sea level \( (\rho_{\text{SL}}) \). Taking in mind the airworthiness regulations, takeoff speed \( (V_{\text{TO}}) \) is obtained by increasing the stall speed in takeoff configuration \( (V_{\text{S,TO}}) \) of 20\% (see Eq. (21)). \( V_{\text{S,TO}} \) is calculated from lift equation incorporated in Eq. (21) where the \( \rho_{\text{TO}} \) is the air density at airport altitude. Considering the matching chart, the takeoff requirement is represented by a line from axis origin (see line n.2 on Fig. 9(a)) and embodies the minimum thrust to weight ratio, for a giving wing loading, necessary to perform the takeoff within the distance required. Likewise the landing case, the design parameters which are able to modify the T/W requirement are the maximum lift coefficient in takeoff configuration \( \left( \text{Cl}_{\text{MAX,TO}} \right) \) and the \( D_{\text{TO}} \) as can be seen in Fig. 9(b).

\[
\frac{T_{\text{SL}}}{W_{\text{MTO}}} \geq 2.33 \cdot \frac{\frac{W_{\text{MTO}}}{s_w}}{\sigma \cdot \text{Cl}_{\text{MAX,TO}} \cdot D_{\text{TO}}} \tag{19}
\]

\[
\frac{P_{\text{SL}}}{W_{\text{MTO}}} \geq \left[ 2.33 \cdot \frac{\frac{W_{\text{MTO}}}{s_w}}{\sigma \cdot \text{Cl}_{\text{MAX,TO}} \cdot D_{\text{TO}}} \right] \cdot \frac{V_{\text{TO}}}{\eta_P} \tag{20}
\]

\[
V_{\text{TO}} = 1.2 \cdot V_{\text{S,TO}} = 1.2 \cdot \sqrt{\frac{2 \cdot W_{\text{MTO}}}{\rho_{\text{TO}} \cdot S \cdot \text{Cl}_{\text{MAX,TO}}}} \tag{21}
\]

The cruise requirement is displayed as a hyperbola (see curve n.3 of Fig.9 (a)) which is defined by Eqs. (22) and (23) where the term \( \zeta \) is the engine de-rate which take into account that the engine cannot be utilized at maximum thrust or power during cruise (e.g. a value of 0.8, that represents the 80\% of the maximum thrust or power, is commonly accepted). All points above this curve are allowable in accordance to the requirement. These equations resulting by drag formula calculated at cruise condition.

\[
\frac{T_{\text{SL}}}{W_{\text{MTO}}} \geq \frac{1}{2} \cdot \rho_{\text{CR}} \cdot \frac{V_{\text{CR}}^2}{\zeta \cdot F_{Zj}} \cdot \frac{4}{3} \cdot \frac{C_d_0}{S_w} \cdot \frac{W_{\text{MTO}}}{S_w} \tag{22}
\]

\[
\frac{P_{\text{SL}}}{W_{\text{MTO}}} \geq \frac{1}{2} \cdot \rho_{\text{CR}} \cdot \frac{V_{\text{CR}}^3}{\zeta \cdot F_{Z} \cdot \eta_P} \cdot \frac{2}{3} \cdot \frac{C_d_0}{S_w} \cdot \frac{W_{\text{MTO}}}{S_w} \tag{23}
\]
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The thrust or power is converted (Roskam 2003 and Raymer 2012) to sea level engine performance using the following coefficients (Eq. (24), (25) and (26)), respectively, for jet, piston and turboprop engine.

\[ T_{CR} = T_{SL} \cdot \frac{\rho_{CR}}{\rho} \Rightarrow F_{Z,J} = \frac{\rho_{CR}}{\rho} \] (24)

\[ P_{CR} = P_{SL} \cdot \left( \frac{\rho_{CR}}{\rho} - \frac{1}{7.55} \frac{\rho_{CR}}{\rho} \right) \Rightarrow F_{Z,PP} = \left( \frac{\rho_{CR}}{\rho} - \frac{1}{7.55} \frac{\rho_{CR}}{\rho} \right) \] (25)

\[ P_{CR} = P_{SL} \cdot \frac{\rho_{CR}}{\rho} \Rightarrow F_{Z,TP} = \frac{\rho_{CR}}{\rho} \] (26)

\( V_{CR}, \rho_{CR} \) and \( C_{d0} \), when modified, determine an effect on the cruise requirement. As seen on Fig. 9(b), the \( T/W \) increases with the augmentation of these parameters.

The forth requirement, which is applied only in case of multiengine aircraft, concerns the climb gradient (\( \gamma \)) in case of OEI (one engine inoperative) condition. In particular, this model considers the climb gradient required by FAR 23 or FAR 25 during takeoff segment. Examining Eqs. (27) and (28), it is easy to note that they define only a \( T/W \) requirement, consequently, the climb gradient corresponds to a horizontal line on the matching chart (see line n.4 on Fig. 9(a)). The climb requirement is influenced by lift to drag ratio in takeoff configuration (\( E_{TO} \)) and for this reason the design parameters identified are \( C_{l_{MAX,TO}} \) and \( C_{d0} \) as Fig. 9(b) shows.

\[ \frac{T_{SL}}{W_{MTO}} \geq \frac{n_{e}}{n_{e} - 1} \cdot \left( \frac{1}{E_{TO}} + \sin \gamma \right) \] (27)

\[ \frac{P_{SL}}{W_{MTO}} \geq \frac{n_{e}}{n_{e} - 1} \cdot \left( \frac{1}{E_{TO}} + \sin \gamma \right) \cdot \frac{V_{TO}}{\eta_{p}} \] (28)

The ACAD model is able to design also military aircraft which, beside other characteristics, can flight at supersonic speed and can be equipped with engine with afterburner. Therefore, being this aircraft more complex than the civil ones, the ACAD model has to consider new requirements. These new design constraints are: supersonic cruise, combat turn and combat thrust to weight ratio.

The requirement number 5 is the \( T/W \) required during combat phase, which is defined as a horizontal line on the matching chart (see line n.5 on Fig. 10(a)) and which is defined by Eq. (29). This is a value of \( T/W \) directly defined by the designer. For example, in case of air-superiority aircraft design, it would be suitable to consider a \( T/W \) greater than one to obtain adequate maneuverability during combat.

\[ \frac{T}{W} \geq \left( \frac{T}{W} \right)_{COMBAT} \] (29)

The combat turn is a typical requirement for military aircraft and it is defined by \( V_{\text{turn}} \), \( n_{\text{turn}} \) and turn altitude (i.e. air density at turn altitude \( \rho_{\text{turn}} \)). In particular, the combat turn performed at
supersonic speed with afterburner turned on is considered. This requirement is defined by Eq. (30) (Saha et al. 2008) and is depicted by curve number 6 on Fig. 10(a).

$$\frac{T_{SL}}{W_{MTO}} \geq \frac{\beta}{F_{ZM, JW}} \left[ K \cdot n_{\text{turn}}^{\frac{1}{2}} \cdot \beta \cdot \frac{\rho_{\text{turn}} \cdot V_{\text{turn}}^2}{S_W} + \frac{(C_{d_0})_{M>1}}{\frac{\beta}{\rho_{\text{turn}} \cdot V_{\text{turn}}^2}} \cdot \frac{W_{MTO}}{S_W} \right]$$

(30)

In order to use Eq. (30), it is necessary to calculate: the \((C_{d_0})_{M>0}\), the drag due to lift factor \((K)\) during supersonic flight and \(\beta\) (fuel fraction) which can be estimated from Table 2. Moreover, it is also necessary to calculate the thrust correction factor, which takes into account altitude and Mach number \((M)\) with afterburner turned on. To simplify the design process, the \(C_{d_0}\) at supersonic speed is calculated as the double of the subsonic one (see Eq. (31)). This rough approximation finds support by statistical data presented on Roskam (2003). The \(K\) and the thrust correction factors is calculated by Eq. (32) (Raymer 2012) and (33) (Saha et al. 2008). The combat turn requirement is influenced by design parameters such as \(C_{d_0}\), wing sweep angle \((\Lambda_{LE})\), turn altitude, speed and load factor (see Fig. 10(b)).

\[(C_{d_0})_{M>1} \cong 2 \cdot (C_{d_0})_{M<1}\]

(31)

\[K = \frac{AR(M^2 - 1)}{4AR\sqrt{M^2 - 1} - 2} \cos \Lambda_{LE}\]

(32)

\[F_{ZM, JW} = [0.952 + 0.3(|M - 0.4|)^2] \left(\frac{\rho_{\text{turn}}}{\rho_{SL}}\right)^{0.7}\]

(33)

The last requirement taken into account in the ACAD design model is the supersonic cruise (or supersonic) performed without afterburner. It is defined by Eq. (34) and regarding the \(C_{d_0}\), \(K\) and

\[(C_{d_0})_{M>0} \cong 2 \cdot (C_{d_0})_{M<1}\]

(34)
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$\beta$, it is still applicable what was defined for combat turn requirement. The $\rho_{\text{SCR}}$ and $V_{\text{SCR}}$ are the air density at supercruise altitude and the aircraft speed.

$$\frac{T_{\text{SL}}}{W_{\text{MTO}}} \geq \frac{\beta}{F_{ZM, JD}} \left[ K \frac{\beta}{2} \rho_{\text{SCR}} V_{\text{SCR}}^2 \frac{W_{\text{MTO}}}{S_W} + \frac{(C_{d_0})_{M>1}}{2 \rho_{\text{SCR}} V_{\text{SCR}}^2} \frac{W_{\text{MTO}}}{S_W} \right]$$ (34)

$$F_{ZM, JD} = 0.76 \cdot [0.907 + 0.262(|M - 0.5|)^{1.5}] \left(\frac{\rho_{\text{SCR}}}{\rho_{\text{SL}}}\right)^{0.7}$$ (35)

Whereas, the thrust correction factor is now calculated through Eq. (35) (Saha et al. 2008) which considers the afterburner turned off. As can be seen in Fig. 10(b), the parameters that influence the supercruise requirement are the same considered for combat turn requirement.

5. Limits of the ACAD model flexibility

Through the ACAD model, it is possible to obtain typical conceptual design outputs, such as $W_{\text{MTO}}$, $W_e$, engine power or thrust, aircraft speeds, wing parameters and other data, for different aircraft classes. The model, as described in the previous chapters, performs its calculation through different equations which sometimes use a statistical database. In particular, this occurs when $C_{d_0}$, $S_W$, airframe and systems weight are calculated. The database used is quite comprehensive and includes the following aircraft classes:

- Ultralight and general aviation
- Regional aircraft (propeller and jet engine driven)
- Business jet
- Civil transport jet ($W_{\text{MTO}} < 150,000 \text{ kg}$)
- Patrol aircraft or UAV
- Military and civil cargo (propeller and jet engine driven) ($W_{\text{MTO}} < 150,000 \text{ kg}$)
- Military trainer
- Fighter or UCAV
- Bomber ($W_{\text{MTO}} < 150,000 \text{ kg}$)

Accordingly with the list, ACAD model is not able to design, with adequate reliability, aircraft with a $W_{\text{MTO}}$ greater than 150,000 kg or particular aircraft classes, such as space tourism, hypersonic, special mission (e.g. AWACS, ELINT, SIGINT). The reasons for model inaccuracy or inadequacy are various and are here explained. The wing WER is function of the $W_W$ / $S_W$ parameter, which is calculated using the statistical diagram shown in Fig. 2. The database from which the diagram is built includes aircraft with a $W_{\text{MTO}}$ up to 150,000 kg. Therefore the use of the model for aircraft with a $W_{\text{MTO}}$ greater than 150,000 kg could lead to inaccurate results. Special mission aircraft, such as AWACS, use avionic sensors, antennas, and other equipments which have a great influence on the design of the aircraft. The model database not include this aircraft class, therefore, the $C_{d_0}$ calculation is not able to take into account the drag increment due to antennas and sensors and the range of the weight coefficient of the avionic system WER (see Table 4) is not wide enough to consider the significant weight of special equipments. Hypersonic and space tourism aircraft are other classes which are not included in the model database. The $C_{d_0}$ is
estimated in a reliable way only up to Mach 2.5 and the model is not developed to design aircraft which consist of several stages as for space tourism aircraft. On the other hand, most of these limits can be overtake simply extending the model database. The model process is compatible with the database extension and this will be carried out in the future in order to improve the model flexibility. In contrast, the changes that would be required to allow the model to design space tourism aircraft are quite challenging because the model process is not designed for this purpose. Therefore, in order to design this aircraft class, a new and specific design model would be necessary.

Table 5 Test cases results. Performance

<table>
<thead>
<tr>
<th>Performance</th>
<th>STOL Ultralight</th>
<th>Fast Regional</th>
<th>Fifth Generation Fighter</th>
</tr>
</thead>
<tbody>
<tr>
<td>L/D</td>
<td>13.8</td>
<td>16.2</td>
<td>11.3</td>
</tr>
<tr>
<td>C_d</td>
<td>0.0225</td>
<td>0.023</td>
<td>0.0174</td>
</tr>
<tr>
<td>W/S max [kg/m²]</td>
<td>35</td>
<td>425</td>
<td>340</td>
</tr>
<tr>
<td>P/W max [kW/kg]</td>
<td>0.07</td>
<td>0.23</td>
<td>-</td>
</tr>
<tr>
<td>T/W max [kg/kg]</td>
<td>-</td>
<td>-</td>
<td>1.1</td>
</tr>
<tr>
<td>V_{App} [m/s]</td>
<td>14.7</td>
<td>54</td>
<td>40</td>
</tr>
<tr>
<td>V_{TO} [m/s]</td>
<td>17</td>
<td>63</td>
<td>61</td>
</tr>
<tr>
<td>P_{SL} [kW]</td>
<td>45</td>
<td>2940 x 2</td>
<td>-</td>
</tr>
<tr>
<td>T_{SL} [kg]</td>
<td>-</td>
<td>-</td>
<td>8350 x 2</td>
</tr>
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</table>

Table 6 Test cases results. Weight and geometric characteristics

<table>
<thead>
<tr>
<th>Weight estimation</th>
<th>STOL Ultralight</th>
<th>Fast Regional</th>
<th>Fifth Generation Fighter</th>
</tr>
</thead>
<tbody>
<tr>
<td>Airframe weight [kg]</td>
<td>141</td>
<td>3290</td>
<td>3121</td>
</tr>
<tr>
<td>W_{sys} [kg]</td>
<td>52</td>
<td>5589</td>
<td>2895</td>
</tr>
<tr>
<td>W_{inst_eng} [kg]</td>
<td>33</td>
<td>1980</td>
<td>2176</td>
</tr>
<tr>
<td>W_F [kg]</td>
<td>36</td>
<td>2685</td>
<td>5613</td>
</tr>
<tr>
<td>W_e [kg]</td>
<td>226</td>
<td>10859</td>
<td>8191</td>
</tr>
<tr>
<td>W_{MTO} [kg]</td>
<td>448</td>
<td>23000</td>
<td>15102</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Geometric characteristics</th>
<th>STOL Ultralight</th>
<th>Fast Regional</th>
<th>Fifth Generation Fighter</th>
</tr>
</thead>
<tbody>
<tr>
<td>S_W [m²]</td>
<td>12.8</td>
<td>54</td>
<td>44</td>
</tr>
<tr>
<td>AR</td>
<td>6.4</td>
<td>9</td>
<td>3.2</td>
</tr>
<tr>
<td>λ</td>
<td>1</td>
<td>0.35</td>
<td>0.1</td>
</tr>
<tr>
<td>b [m]</td>
<td>9</td>
<td>22.1</td>
<td>11.8</td>
</tr>
<tr>
<td>A_LE [°]</td>
<td>0</td>
<td>0</td>
<td>27</td>
</tr>
<tr>
<td>I_{tas}</td>
<td>6</td>
<td>30</td>
<td>16</td>
</tr>
</tbody>
</table>
6. Analysis of the test cases results

In Tables 5 and 6, some of test cases results are listed. In details, the weight of the major aircraft components, aircraft $W_e$ and $W_{MTO}$ and other performance and geometric parameters such as lift to drag ratio, speed, $AR$, $\lambda$, maximum $T/W$ (or $P/W$) and $W/S$, are selected as validation parameters. The results of each test case are here analyzed.

6.1 Two-seat STOL ultralight (LSA category)

The STOL Ultralight meets the regulation requirements, having a $W_{MTO}$ of 448 kg and a stall speed less than 18 m/s. The system weight is 52 kg and should not be considered excessive since the presence of a simple, but complete VFR avionic system, gear brakes, flight control cables, engine starter, battery and two seats as furnishing. The engine weight is relatively low because a light two-cylinder two-stroke engine was chosen. Many ultralight aircraft use this kind of engine since its high power to weight ratio and low cost, despite it has generally a high fuel consumption and short service life. The matching chart displayed on Fig. 11 shows that the power required to takeoff and to perform the cruise phase is less than 45 kW. In fact, to achieve STOL operation, the aircraft is equipped with flaps along the whole wing aperture and fixed slats. These high lift devices reduce the power required for takeoff and $S_w$ necessary for landing. It is worth noting that in matching chart, for reference purpose only, the wing loading at landing is displayed as vertical and dashed line.

6.2 Fast and green regional aircraft

This regional aircraft test case can be considered a new aircraft segment for some aspects.
Indeed, in order to satisfy the $V_{CR}$ requirement which is higher than the regional turboprop aircraft, it would be necessary to employ a turbofan engine. On the other hand, an aircraft driven by turbofan engine would not meet the low pollution requirement. For this reason a Propfan (or Open-rotor) engine was chosen. At the present, there are only prototypes of this class of engine, but their advantages would be significant. They should maintain elevated propulsive efficiency at faster speed than turboprop engine with the same $SFC$. Focusing on Table 6, it is worth noting that
the $W_{MTO}$ figure is comparable with other regional aircraft with lower payload. This is due to the use of composite material for the airframe instead of aluminum. In comparison to the regional turboprop, another difference is the value of the wing AR. This is lower than the typical value since the greater $V_{CR}$ of the designed aircraft. In Fig. 12 the matching chart is displayed. It is possible to note that the cruise and takeoff requirements are closer and they are satisfied with acceptable value of $P/W$ ratio. The climb requirement in $OEI$ condition is met with lower $P/W$.

6.3 Fifth generation fighter

The weights, performances and the other characteristics of the fifth generation fighter, listed in Tables 5 and 6, can be considered in line with comparable aircraft. The matching chart (see Fig.13) includes all design requirements. The takeoff, climb (in $OEI$ condition) and subsonic cruise requirements are negligible if compared with the other military specific requirements. It is important to note that the supercruise requirement, which has to be performed without afterburner, identifies the maximum engine dry thrust. The combat $T/W$ and combat turn requirements, on the contrary, identify the maximum engine wet thrust, i.e. the engine thrust with afterburner turned on. Another remarkable issue which can be observed in the matching chart is the huge difference between the wing loading during takeoff and the one calculated during landing. This is due to the greater fuel consumed respect the other test cases

6. Conclusions

The aim of this work is to describe a new aircraft conceptual design (ACAD) model in terms of process, equations and results. Many accurate and well-established design models already exist; nevertheless the ACAD model has the ambition to provide reliable output even using a simple process and to maintain an adequate flexibility to design different aircraft categories. In order to verify these model capabilities, the study was carried out through the design of three different test cases.

The ACAD model findings can be summarize as follow:

- The test cases results are consistent and in line with the similar existing aircraft. The model output data cover different aircraft characteristics including weights, performances, geometric and layout features.
- The model has demonstrated its flexibility and adaptability to design different aircraft categories (i.e. ultralight, civil transport and military fighter aircraft).
- The model is easy to use since the relatively low number of required data which are basically the aircraft design requirements that are not difficult to hypothesize. Moreover, the model is implemented in Microsoft® Excel® spreadsheet without visual basic macro keeping the design process always visible to the user.
- New WERs of aircraft systems and main airframe components are introduced. The wing WER was accurately designed adapting it to different aircraft categories and making it susceptible to AR and $\lambda$ variation.
- A new wing AR optimization subroutine is described. In this way, different to other design models, the wing AR is not statistically evaluated but it is calculated optimizing the aircraft $W_{MTO}$.

Finally, the ACAD model uses, through the matching chart, a graphical and simple method to
verify the design requirements achievement.

Acknowledgements

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References


EC
### Nomenclature

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>AR</td>
<td>Wing aspect ratio</td>
</tr>
<tr>
<td>AWACS</td>
<td>Airborne warning and control system</td>
</tr>
<tr>
<td>b</td>
<td>Wingspan</td>
</tr>
<tr>
<td>Cd</td>
<td>Drag coefficient</td>
</tr>
<tr>
<td>$Cd_{CR}$</td>
<td>Drag coefficient at cruise condition</td>
</tr>
<tr>
<td>$Cd_{0}$</td>
<td>Parasite drag coefficient</td>
</tr>
<tr>
<td>Cl</td>
<td>Lift coefficient</td>
</tr>
<tr>
<td>$Cl_{CR}$</td>
<td>Lift coefficient at cruise condition</td>
</tr>
<tr>
<td>$Cl_{MAX,LND}$</td>
<td>Maximum lift coefficient, landing configuration</td>
</tr>
<tr>
<td>$Cl_{MAX,TO}$</td>
<td>Maximum lift coefficient, takeoff configuration</td>
</tr>
<tr>
<td>$D_{LND}$</td>
<td>Landing distance</td>
</tr>
<tr>
<td>$D_{TO}$</td>
<td>Takeoff distance</td>
</tr>
<tr>
<td>e</td>
<td>Oswald’s coefficient</td>
</tr>
<tr>
<td>$E_{CR}$</td>
<td>Lift to drag ratio during cruise</td>
</tr>
<tr>
<td>ELINT</td>
<td>Electronic signals intelligence</td>
</tr>
<tr>
<td>$E_{TO}$</td>
<td>Aerodynamic efficiency, take off configuration</td>
</tr>
<tr>
<td>$F_{Z,J}$</td>
<td>Altitude correction factor for jet engine</td>
</tr>
<tr>
<td>$F_{Z,PP}$</td>
<td>Altitude correction factor for piston propeller engine</td>
</tr>
<tr>
<td>$F_{Z,TP}$</td>
<td>Altitude correction factor for turboprop engine</td>
</tr>
<tr>
<td>$F_{ZM,JD}$</td>
<td>Altitude and Mach number correction factor for jet engine without afterburner</td>
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<tr>
<td>$F_{ZM,JW}$</td>
<td>Altitude and Mach number correction factor for jet engine with afterburner</td>
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<tr>
<td>K</td>
<td>Drag due to lift factor</td>
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<tr>
<td>$K_{comp}$</td>
<td>Composite material coefficient</td>
</tr>
<tr>
<td>$K_{delta}$</td>
<td>Delta wing configuration coefficient</td>
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<td>$K_{eng}$</td>
<td>Installation coefficient (installed engine WER)</td>
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<td>$k_f$</td>
<td>Fuel fraction parameter</td>
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<td>$K_{sys}$</td>
<td>Systems weight coefficient (systems WER)</td>
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<td>$K_{tail}$</td>
<td>Tail weight coefficient (tail WER)</td>
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<td>$K_{TOW}$</td>
<td>Maximum takeoff weight coefficient (fuselage WER)</td>
</tr>
<tr>
<td>$K_{WP}$</td>
<td>Wing planform coefficient (wing WER)</td>
</tr>
<tr>
<td>$l_{ fus}$</td>
<td>Fuselage length</td>
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<tr>
<td>LSA</td>
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<td>M</td>
<td>Mach number</td>
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</tr>
<tr>
<td>$n_{turn}$</td>
<td>Load factor during turn maneuver</td>
</tr>
</tbody>
</table>
OEI One engine inoperative
P$_{SL}$ Sea level engine power
R$_{CR}$ Cruise range
SIGINT Signals intelligence
Sin$_{y}$ Climb gradient
SFC Engine specific fuel consumption
SFC$_{COMBAT}$ Engine specific fuel consumption during combat operation
SFC$_{P}$ Engine specific fuel consumption (power)
SFC$_{T}$ Engine specific fuel consumption (thrust)
STOL Short takeoff and landing
S$_{w}$ Wing surface
t$_{COMBAT}$ Combat time
T$_{SL}$ Sea level engine thrust
T/W Thrust to weight ratio
(T/W)$_{COMBAT}$ T/W during combat operation
UAV Unmanned Air Vehicle
UCAV Unmanned Combat Air Vehicle
V$_{App}$ Approach speed
V$_{CR}$ Cruise speed
V$_{S,LND}$ Stall speed, landing configuration
V$_{S,TO}$ Stall speed, takeoff configuration
V$_{SCR}$ Supersonic cruise speed
V$_{TO}$ Takeoff speed
V$_{turn}$ Combat turn speed
W$_{DPL}$ Dropped payload weight
W$_{e}$ Aircraft empty weight
W$_{eng}$ Engine weight
WER Weight estimating relationship
W$_{F}$ Total fuel weight
W$_{F,COMBAT}$ Fuel weight required during combat operation
W$_{F,CR}$ Fuel weight required during cruise
W$_{F,OTHER}$ Fuel weight required during the takeoff, climb, descent and landing
W$_{fus}$ Fuselage weight
W$_{inst\,eng}$ Installed engine weight
W$_{LND}$ Landing weight
W$_{MTO}$ Maximum takeoff weight
W$_{PL}$ Payload weight
Adaptable conceptual aircraft design model

\begin{itemize}
  \item $W_{sys}$: System weight
  \item $W_{tail}$: Tail weight
  \item $W_{W}$: Wing weight
  \item $\Lambda_{LE}$: Wing sweep angle (leading edge sweep angle)
  \item $\beta$: Fuel fraction
  \item $\zeta$: Throttle position (percentage)
  \item $\eta_p$: Propeller propulsion efficiency
  \item $\lambda$: Wing taped ratio
  \item $\rho$: Air density at sea level
  \item $\rho_{CR}$: Air density at cruise altitude
  \item $\rho_{LND, TO}$: Air density at airport altitude
  \item $\rho_{SCR}$: Air density at super cruise altitude
  \item $\rho_{SL}$: Air density at sea level
  \item $\rho_{turn}$: Air density at combat turn altitude
  \item $\sigma$: Air density ratio
\end{itemize}