Advances in Aircraft and Spacecraft Science, *Vol. 1, No. 1 (2014) 043-067*
DOI: http://dx.doi.org/10.12989/aas.2014.1.1.043

Adaptable conceptual aircraft design model

Marco Fioriti*

Department Mechanical and Aerospace Engineering, Politecnico di Torino, Turin, Italy

(Received February 22, 2013, Revised August 31, 2013, Accepted September 01, 2013)

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:

*Corresponding author, Researcher, Ph.D., E-mail: marco.fioriti@polito.it

Copyright © 2014 Techno-Press, Ltd.
http://www.techno-press.org/?journal=aas&subpage=7 ISSN: 2287-528X (Print), 2287-5271 (Online)

- 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 1 Test cases main design requirements

	STOL Ultralight	Fast Regional	Fifth Gen Fighter
Payload and crew weight [kg]	195 (2 persons + baggage)	9570 (90 pax + crew + baggage)	1300 (weapons + pilot)
D_{TO} [m]	100	1350	550
D_{LND} [m]	75	1000	550
V_{CR} [km/h]	110	800	900
R_{CR} [km]	450	1200	2400
Supersonic Cruise parameters	-	-	1.4 M @ 6000 m
Combat Turn parameters	-	-	1.4 M @ 1000 m @ 5 g

(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 (Cd_0) 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.

$$\begin{aligned} W_{F\_CR} &= \left(W_{MTO} - \frac{W_{DPL}}{2} - \frac{W_{F\_CR}}{2}\right) \cdot \frac{R_{CR} \cdot SFC}{E_{CR} \cdot V_{CR}} \\ \Rightarrow W_{F\_CR} &= \frac{\frac{R_{CR} \cdot SFC}{E_{CR} \cdot V_{CR}}}{1 + \frac{1}{2} \cdot \frac{R_{CR} \cdot SFC}{E_{CR} \cdot V_{CR}}} \cdot \left(W_{MTO} - \frac{W_{DPL}}{2}\right) \end{aligned} \tag{2}$$

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

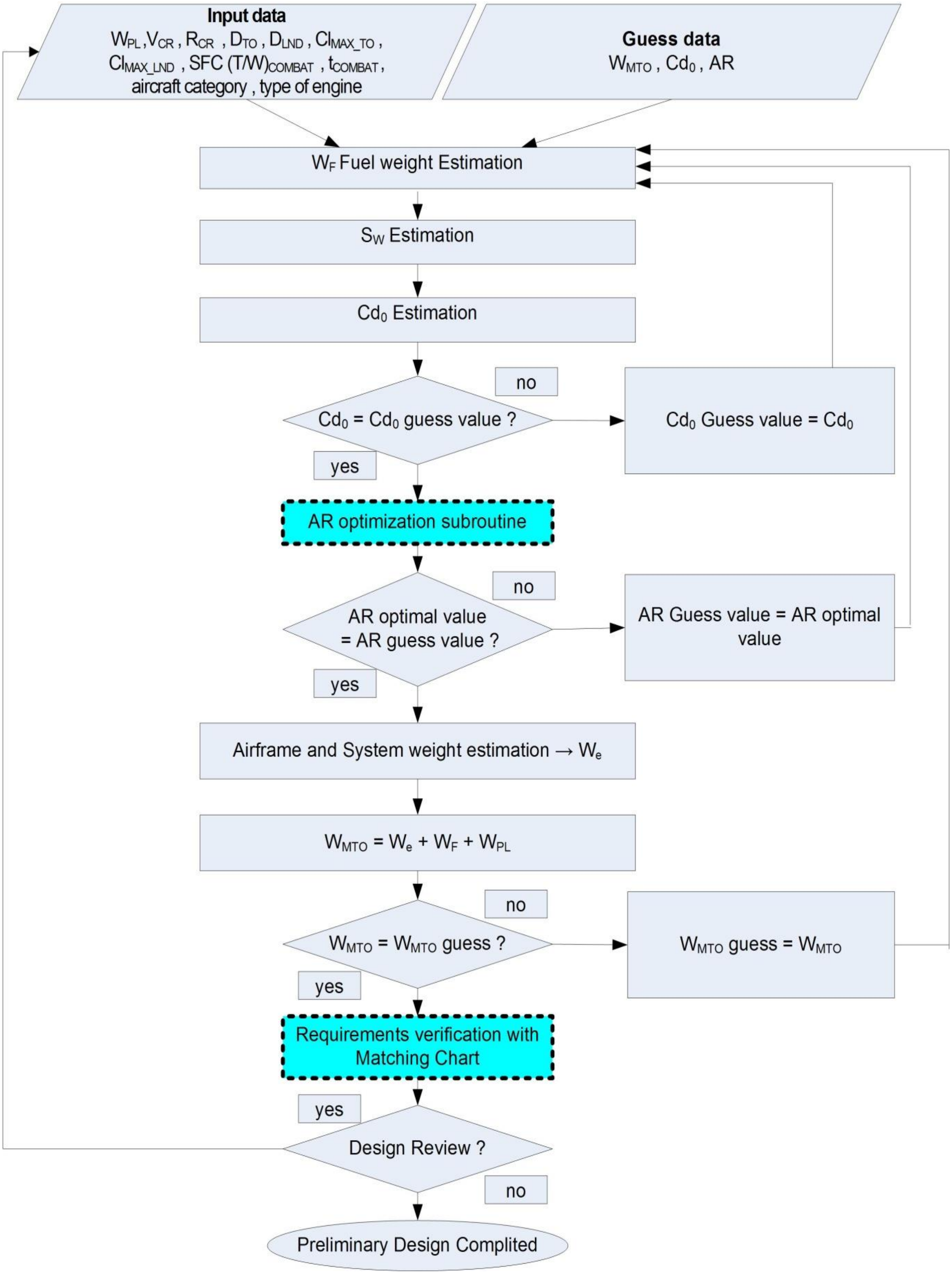


Fig. 1 ACAD model process flow chart

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

k_F parameter	
Aircraft category	Fuel weight for taxi, takeoff, climb, descent and landing [% of W_{MTO}]
Homebuilt	2.1
General aviation	3.6
Regional turboprop	5.4
Transport jet	6.1
Fighter	14

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_COMBAT}) as the product of engine SFC at maximum thrust (SFC_{COMBAT}), the thrust required and the time duration of the combat action (t_{COMBAT}).

$$W_{F\_COMBAT} = SFC_{COMBAT} \cdot \left(\frac{T}{W}\right)_{COMBAT} \cdot t_{COMBAT} \cdot \left(W_{MTO} - \frac{W_{F\_CR}}{2} - \frac{4}{5} W_{F\_OTHER}\right) \tag{3}$$

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_{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)

$$W_F = W_{F\_CR} + W_{F\_COMBAT} + W_{F\_OTHER} \tag{4}$$

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 $\left[\frac{lb}{hp \cdot h}\right]$ to thrust SFC (SFC_T) expressed in $\left[\frac{lb}{lb \cdot h}\right]$ also taking into account the propeller efficiency (η_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_0. 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_0, 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}).

$$D_{LND} = 0.5136 \cdot V_{S\_LND}^2 \tag{6}$$

$$V_{App} = 1.3 \cdot V_{S\_LND} \tag{7}$$

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 ($Cl_{MAX\ LND}$) and following Roskam's methodology (Roskam 2003), the statistical figure of Cd_0 can be now estimated. The term ρ_{LND} represents the air density at airport altitude. The model performs the first iterative loop comparing the Cd_0 calculated value with the guess value. Until the value becomes stable, the model recalculates the W_F, S_W and Cd_0 estimation blocks.

$$W_{LND} = W_{MTO} - \frac{W_F}{2} - W_{DPL} \tag{8}$$

$$\frac{W_{LND}}{S_W} = 0.5 \cdot \rho_{LND} \cdot V_{APP}^2 \cdot Cl_{MAX\_LND} \implies S_W \tag{9}$$

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))

$$W_W = \frac{W_W}{S_W} \cdot S_W \cdot K_{comp} \cdot K_{WP} \cdot K_{delta} \tag{10}$$

$$W_{tail} = K_{tail} \cdot W_W \tag{11}$$

$$W_{fus} = \frac{L_{fus}}{1000} \cdot n_{max} \cdot W_{MTO} \cdot K_{MTO} \cdot K_{comp} \cdot K_{delta} \tag{12}$$

$$W_{inst\ eng} = W_{eng} \cdot K_{eng} \cdot n_e \tag{13}$$

$$W_{sys} = \sum_i K_{sys\_i} \cdot W_{MTO} \tag{14}$$

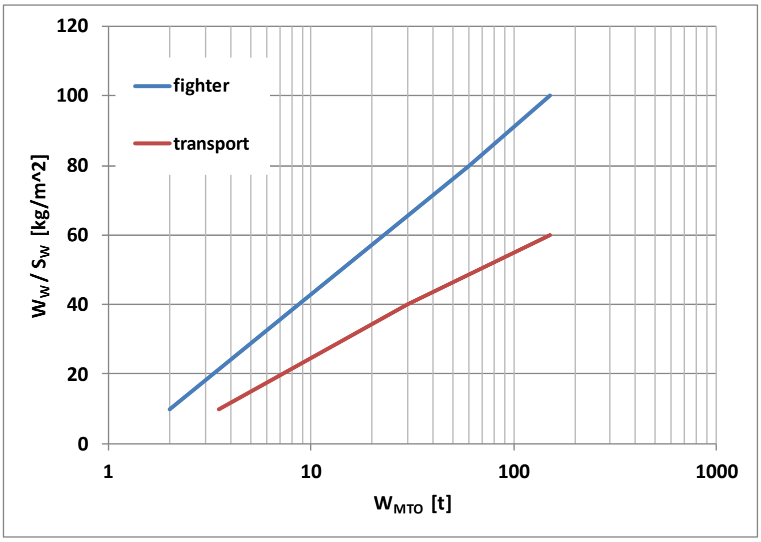


Fig. 2 W_W to S_W ratio (Military and civil transport aircraft)

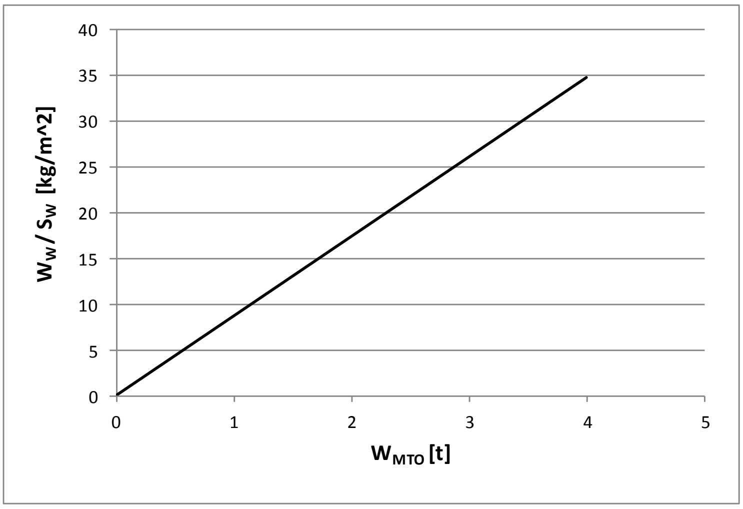


Fig. 3 W_W to S_W ratio (ultralight and general aviation aircraft)

The wing weight is estimated with Eq. (10) which considers parameters such as wing weight to wing area ratio, wing area, AR, taped ratio (λ), 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. the different ratios are shown.

Eq. (11) performs the tail weight (W_{tail}) estimation using a statistical ratio between W_{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_{comp}, K_{WP}, K_{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

Aircraft element	Coefficient	Coefficient range	Rationale
Wing	K_{comp}	1 ÷ 0.65	% of composite material
Wing	K_{wp}	Calculated by Eq.(15)	Wing plane shape (AR and λ)
Wing	K_{delta}	1 ÷ 0.9	Delta wing configuration
Tail	K_{tail}	0.1 ÷ 0.3	Tail complexity
Fuselage	K_{comp}	1 ÷ 0.65	% of composite material
Fuselage	K_{delta}	1 ÷ 0.9	Delta wing configuration
Engine	K_{eng}	1.1 ÷ 1.25	Aircraft configuration (from fighter to civil transport)

Table 4 WERs coefficients for aircraft system

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

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

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

The fuselage weight (W_{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_{fus}), configuration and n_{max}. The engine weight (W_{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_{inst_eng}). Eq. (14) is used to calculate the airplane sub systems weight (W_{sys}). For each subsystem, the model provides a range of system weight to W_{MTO} ratio (K_{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_{MTO}. As result of first iterative loop, the calculated W_{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 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 ρ_{CR} is the air density at cruise altitude) the value of Cl_{CR} (Cl at cruise condition) is calculated. Through Cl_{CR}, the AR guess value, Cd_0, Oswald's coefficient (*e*) and Eq. (17) the model calculates a new figure of Cd_{CR} (drag coefficient at cruise condition).

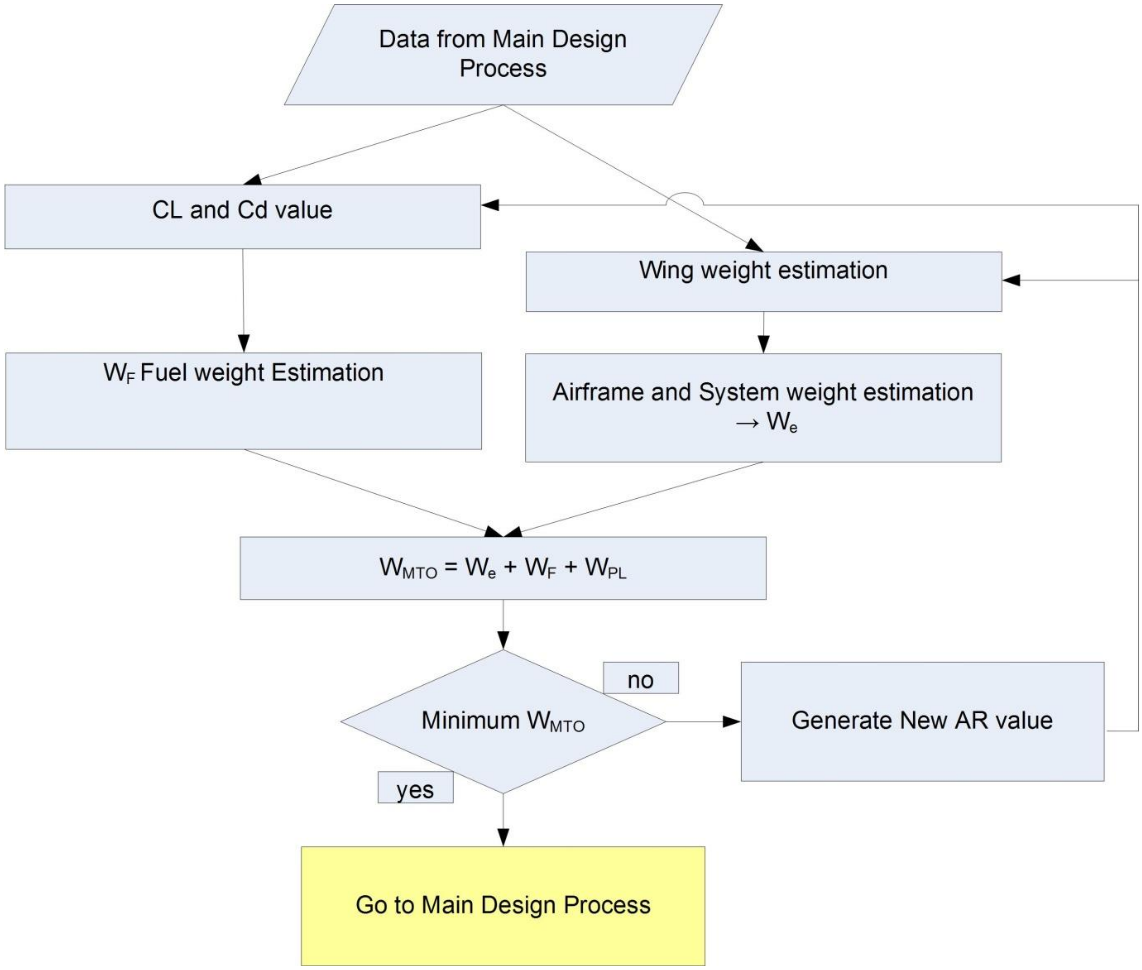


Fig. 4 Aspect ratio optimization subroutine – process flow chart

$$Cl_{CR} = \frac{2 \cdot \left(W_{MTO} - \frac{W_{DPL}}{2} - \frac{W_{F\_CR}}{2}\right)}{\rho_{CR} \cdot S_W \cdot V_{CR}^2} \tag{16}$$

$$Cd_{CR} = Cd_0 + \frac{Cl_{CR}^2}{\pi \cdot AR \cdot e} \tag{17}$$

The new value of Cd_{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 Cd_{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

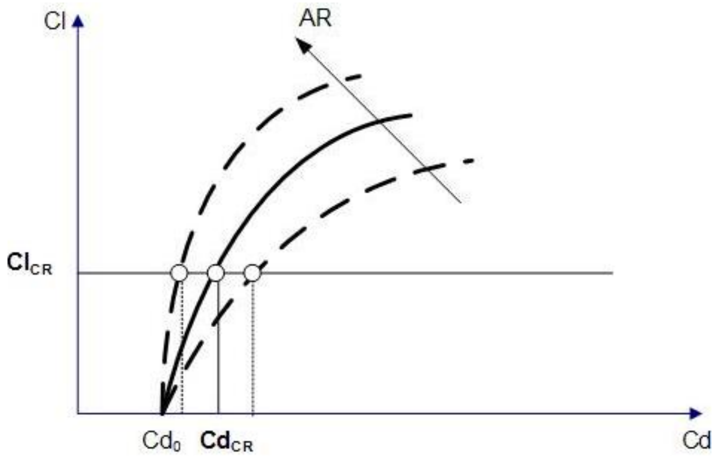


Fig. 5 Effect of the AR on Cd_{CR}

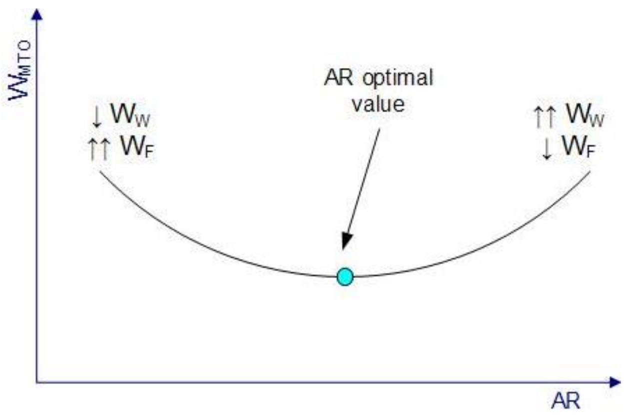


Fig. 6 AR effects on W_{MTO}

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_W 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).

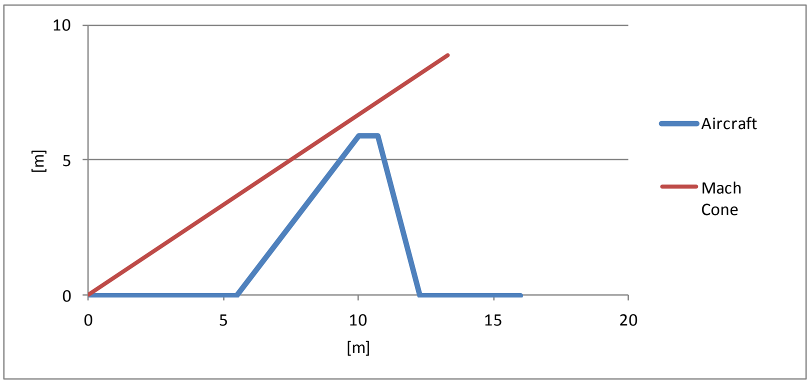


Fig. 7 Mach cone requirement

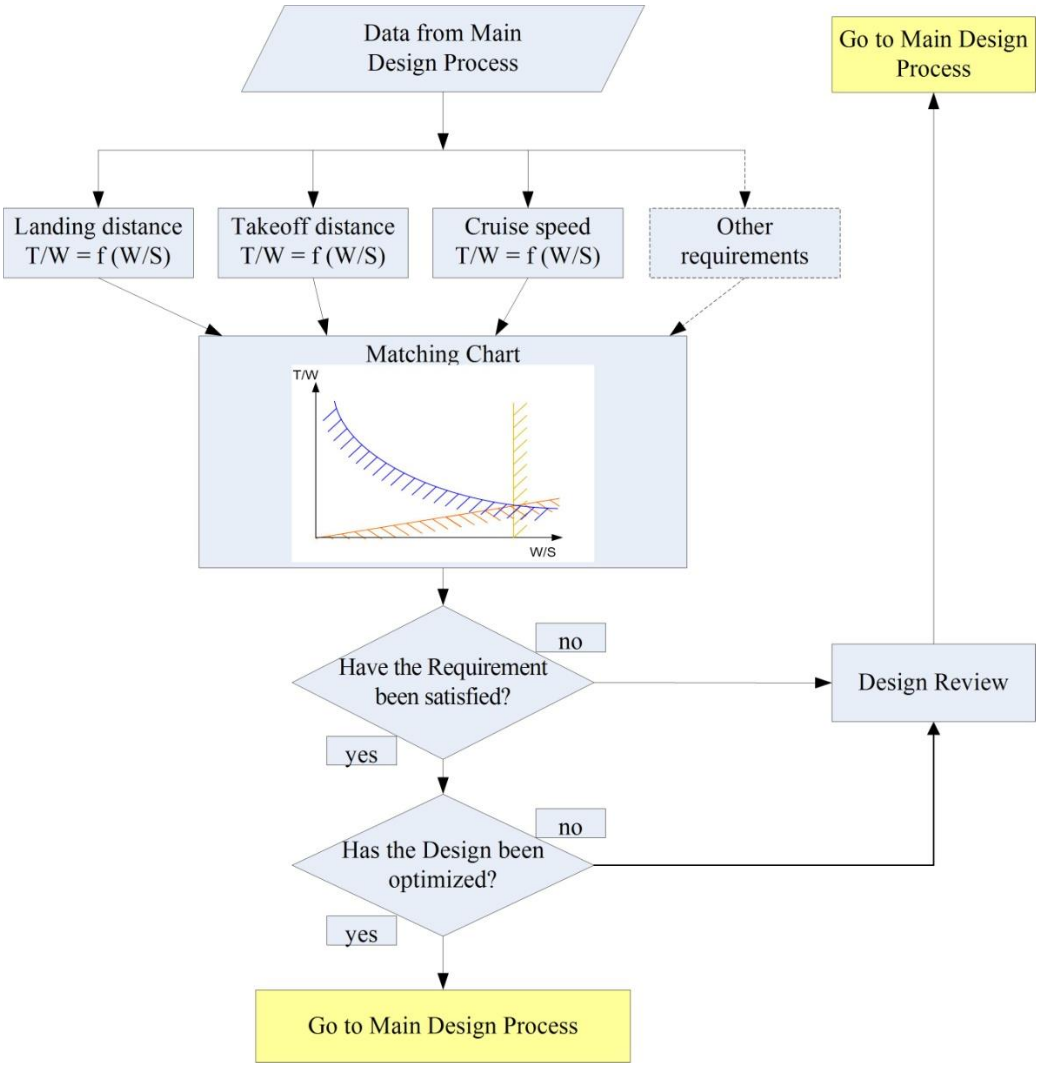


Fig. 8 Design requirements fulfillment

4. Matching chart subroutine

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

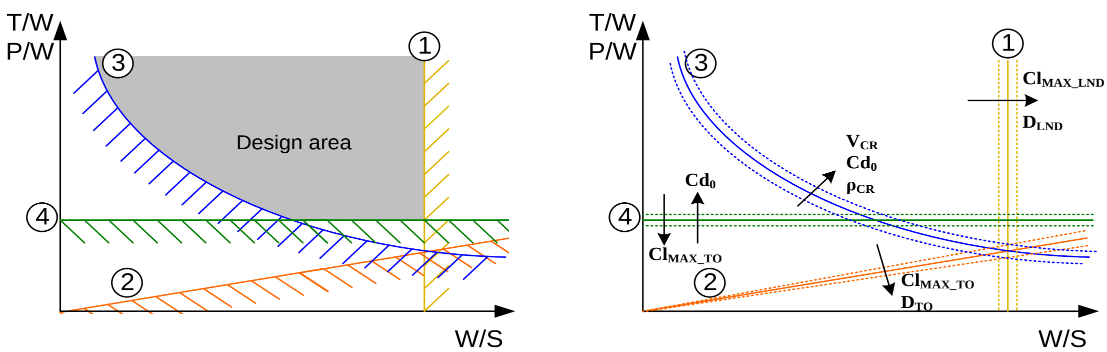
In this part of the design process, D_{TO}, D_{LND}, V_{CR}, climb speed and the other requirements related to military aircraft, such as super-cruise and combat turn capabilities, are expressed in terms of thrust (or power, for propeller driven aircraft) to weight ratio and wing loading. In order to 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)).

$$\frac{W}{S_W} \leq \frac{W_{MTO}}{S_W} \tag{18}$$

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



(a) (b)

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 (Cl_{MAX_LND}) by varying the type of high lift device, and evidently the D_{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_{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_{SL}, P_{SL}, and σ 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 (ρ_{SL}). Taking in mind the airworthiness regulations, takeoff speed (V_{TO}) is obtained by increasing the stall speed in takeoff configuration (V_{S_TO}) of 20% (see Eq. (21)). V_{S_TO} is calculated from lift equation incorporated in Eq. (21) where the ρ_{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 (Cl_{MAX_TO}) and the D_{TO} as can be seen in Fig. 9(b).

$$\frac{T_{SL}}{W_{MTO}} \geq 2.33 \cdot \frac{\frac{W_{MTO}}{S_W}}{\sigma \cdot Cl_{MAX\_TO} \cdot D_{TO}} \tag{19}$$

$$\frac{P_{SL}}{W_{MTO}} \geq \left[2.33 \cdot \frac{\frac{W_{MTO}}{S_W}}{\sigma \cdot Cl_{MAX\_TO} \cdot D_{TO}}\right] \cdot \frac{V_{TO}}{\eta_P} \tag{20}$$

$$V_{TO} = 1.2 \cdot V_{S\_TO} = 1.2 \cdot \sqrt{\frac{2 \cdot W_{MTO}}{\rho_{TO} \cdot S_W \cdot Cl_{MAX\_TO}}} \tag{21}$$

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 ζ 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.

$$\frac{T_{SL}}{W_{MTO}} \geq \frac{\frac{1}{2} \cdot \rho_{CR} \cdot {V_{CR}}^2 \cdot \frac{4}{3} \cdot Cd_0}{\zeta \cdot F_{Z_J}} \cdot \frac{S_W}{W_{MTO}} \tag{22}$$

$$\frac{P_{SL}}{W_{MTO}} \geq \frac{\frac{1}{2} \cdot \rho_{CR} \cdot {V_{CR}}^3 \cdot 2 \cdot Cd_0}{\zeta \cdot F_Z \cdot \eta_P} \cdot \frac{S_W}{W_{MTO}} \tag{23}$$

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

$$T_{CR} = T_{SL} \cdot \frac{\rho_{CR}}{\rho} \implies F_{Z\_J} = \frac{\rho_{CR}}{\rho} \tag{24}$$

$$P_{CR} = P_{SL} \cdot \left(\frac{\rho_{CR}}{\rho} - \frac{1 - \frac{\rho_{CR}}{\rho}}{7.55} \right) \implies F_{Z\_PP} = \left(\frac{\rho_{CR}}{\rho} - \frac{1 - \frac{\rho_{CR}}{\rho}}{7.55} \right) \tag{25}$$

$$P_{CR} = P_{SL} \cdot \frac{\rho_{CR}}{\rho} \implies F_{Z\_TP} = \frac{\rho_{CR}}{\rho} \tag{26}$$

V_{CR}, ρ_{CR} and Cd_0, 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 (γ) 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 Cl_{MAX_TO} and Cd_0 as Fig. 9(b) shows.

$$\frac{T_{SL}}{W_{MTO}} \geq \frac{n_e}{n_e - 1} \cdot \left(\frac{1}{E_{TO}} + \sin\gamma \right) \tag{27}$$

$$\frac{P_{SL}}{W_{MTO}} \geq \frac{n_e}{n_e - 1} \cdot \left(\frac{1}{E_{TO}} + \sin\gamma \right) \cdot \frac{V_{TO}}{\eta_P} \tag{28}$$

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.

$$\frac{T}{W} \geq \left(\frac{T}{W} \right)_{COMBAT} \tag{29}$$

The combat turn is a typical requirement for military aircraft and it is defined by V_{turn}, n_{turn} and turn altitude (i.e. air density at turn altitude ρ_{turn}). In particular, the combat turn performed at

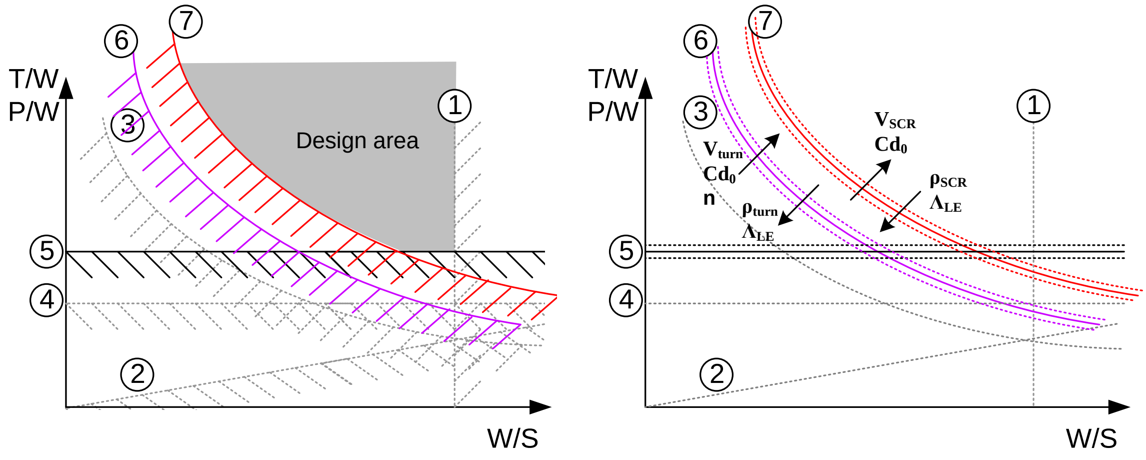


Fig. 10 Matching chart for military aircraft. Design parameters vs. requirements

supersonic speed with afterburner turned on is considered. This requirement is defines by Eq. (30) (Saha *et al*. 2008) and is depicted by curve number 6 on Fig. 10(a).

$$\frac{T_{SL}}{W_{MTO}} \geq \frac{\beta}{F_{ZM\_JW}}\left[K \cdot n_{turn}^2 \frac{\beta}{\frac{1}{2}\cdot \rho_{turn} \cdot V_{turn}^2}\frac{W_{MTO}}{S_W} + \frac{(Cd_0)_{M>1}}{\frac{\beta}{\frac{1}{2}\cdot\rho_{turn}\cdot V_{turn}^2}\cdot\frac{W_{MTO}}{S_W}}\right] \tag{30}$$

In order to use Eq. (30), it is necessary to calculate: the $(Cd_0)_{M>0}$, the drag due to lift factor (K) during supersonic flight and β (fuel fraction) which can be estimated from Table 2. Moreover, it is also necessary to calculate the thrust correction factor, which takes in to account altitude and Mach number (M) with afterburner turned on. To simplify the design process, the Cd_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 Cd_0, wing sweep angle (Λ_{LE}), turn altitude, speed and load factor (see Fig. 10(b)).

$$(Cd_0)_{M>1} \cong 2\cdot(Cd_0)_{M<1} \tag{31}$$

$$K = \frac{AR(M^2-1)}{4AR\sqrt{M^2-1}-2}\cos\Lambda_{LE} \tag{32}$$

$$F_{ZM\_JW} = [0.952 + 0.3(|M-0.4|)^2]\left(\frac{\rho_{turn}}{\rho_{SL}}\right)^{0.7} \tag{33}$$

The last requirement taken in to account in the ACAD design model is the supersonic cruise (or supercruise) performed without afterburner. It is defined by Eq. (34) and regarding the Cd_0, K and

β, it is still applicable what was defined for combat turn requirement. The ρ_{SCR} and V_{SCR} are the air density at supercruise altitude and the aircraft speed.

$$\frac{T_{SL}}{W_{MTO}} \geq \frac{\beta}{F_{ZM\_JD}}\left[K\frac{\beta}{\frac{1}{2}\cdot\rho_{SCR}\cdot V_{SCR}^2}\cdot\frac{W_{MTO}}{S_W}+\frac{(Cd_0)_{M>1}}{\frac{\beta}{\frac{1}{2}\cdot\rho_{SCR}\cdot V_{SCR}^2}\cdot\frac{W_{MTO}}{S_W}}\right] \tag{34}$$

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

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_{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 Cd_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_{MTO} < 150,000 kg)
- Patrol aircraft or *UAV*
- Military and civil cargo (propeller and jet engine driven) (W_{MTO} < 150,000 kg)
- Military trainer
- Fighter or *UCAV*
- Bomber (W_{MTO} < 150,000 kg)

Accordingly with the list, ACAD model is not able to design, with adequate reliability, aircraft with a W_{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_{MTO} up to 150,000 kg. Therefore the use of the model for aircraft with a W_{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 Cd_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 Cd_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

	STOL Ultralight	Fast Regional	Fifth Generation Fighter
Performance			
L/D	13.8	16.2	11.3
Cd_0	0.0225	0.023	0.0174
W/S max [kg/m^2]	35	425	340
P/W max [kW/kg]	0.07	0.23	-
T/W max [kg/kg]	-	-	1.1
V_{App} [m/s]	14.7	54	40
V_{TO} [m/s]	17	63	61
P_{SL} [kW]	45	2940 x 2	-
T_{SL} [kg]	-	-	8350 x 2

Table 6 Test cases results. Weight and geometric characteristics

	STOL Ultralight	Fast Regional	Fifth Generation Fighter
Weight estimation			
Airframe weight [kg]	141	3290	3121
W_{sys} [kg]	52	5589	2895
W_{inst_eng} [kg]	33	1980	2176
W_F [kg]	36	2685	5613
W_e [kg]	226	10859	8191
W_{MTO} [kg]	448	23000	15102
Geometric characteristics			
S_W [m^2]	12.8	54	44
AR	6.4	9	3.2
λ	1	0.35	0.1
b [m]	9	22.1	11.8
Λ_{LE} [°]	0	0	27
l_{fus}	6	30	16

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*, λ, 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.

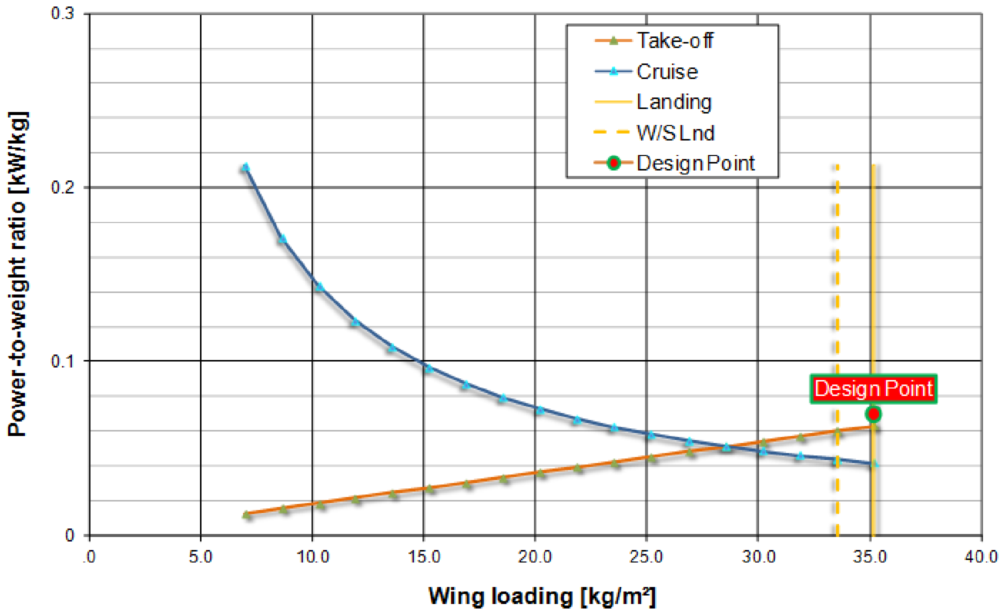


Fig. 11 STOL Ultralight - Matching Chart

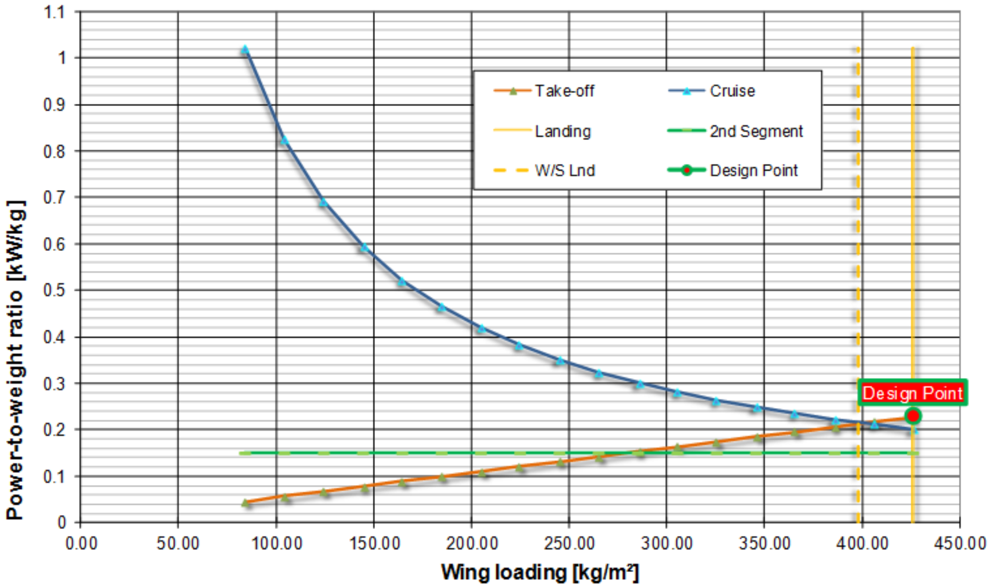


Fig. 12 Fast and green regional transport – Matching Chart

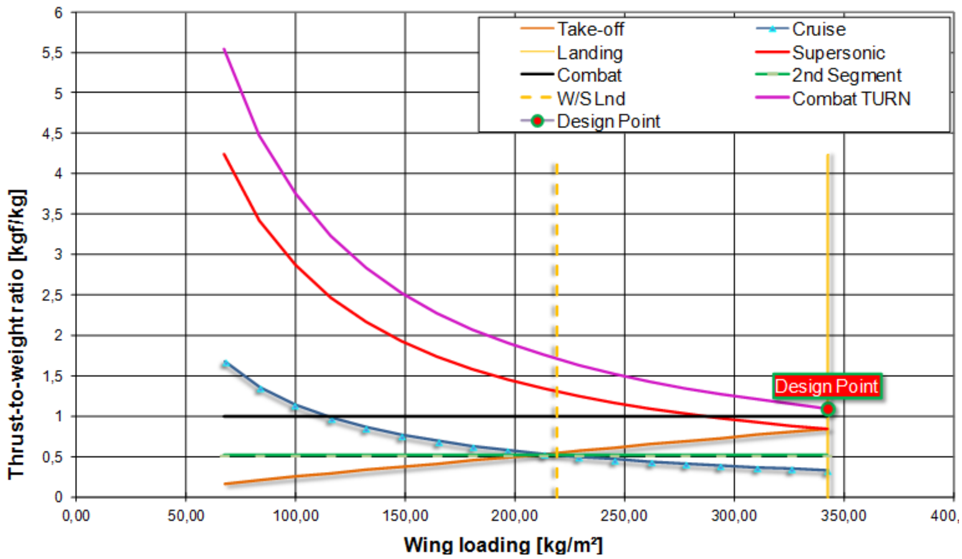


Fig. 13 Fifth generation fighter – Matching Chart

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 *WER*s 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 λ 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

The author has pleasure in acknowledging invaluable support and technical comments of Sergio Chiesa Full Professor of Aircraft System Design of Polytechnic of Turin. Moreover, the author acknowledges the whole research team, which is led by Prof. S. Chiesa, for all the previous works that have been the background of ACAD model.

References

Antona, E. Chiesa, S. Corpino, S. and Viola, N. (in Italian) (2009), "L'avamprogetto dei velivoli", *Atti dell'Accademia delle Scienze di Torino*

Azamatov, A. Woo-Lee, J. and Byun, Y.H. (2012),"Comprehensive aircraft configuration design tool for integrated product and process development", *Elsevier Adv. in Engi. Software*, **42**,35-49

Chiesa, S. (in Italian) (1977), "Sulla previsione del peso degli impianti di bordo nella fase preliminare del progetto dei velivoli", *Ingegneria, n°1-2*.

Chiesa, S. Borello, L. and Maggiore P. (2000), "An academic experience about aircraft design: affordable advanced jet trainer", *Proceeding of 2000 22nd ICAS Congress,* Harrogate, UK, August-September.

Chiesa, S. Corpino, S. Maggiore, P. and Pasquino, M. (2000), "The weight estimation in aerospace engineering at the Polytechnic of Turin", *Proceeding of 2000 SAWE Europe Conference*, Bremen, November.

Chiesa, S. and Maggiore, P. (1995), "Conceptual design in aeronautics: an academic viewpoint", *Proceeding of '95 ICED 10th*, Praga, August.

Chiesa, S. Fioriti, M. and Viola, N. (2012), *Methodology for an Integrated Definition of a System and Its Subsystems: The Case-Study of an Airplane and Its Subsystems* Chapter of the book: *Sys. Engi. - Practice and Theory*, ISBN: 978-953-51-0322-6, INTECH

Chiesa, S. and Viola, N. (2007), "Sub-systems equipments weight and volume first estimation; a tool for aircraft conceptual design", *Int' J. Mech. Control*, **8**(1),813-820

Jenkinson, L.R. Simpkin, P. and Rhodes, D. (1999), *Civil Jet Aircraft Design (2nd Edition),* American Institute of Aeronautics and Astronautics, Reston, VA.

Lee, D.S. Fahey, D.W. Forster, P.M. Newton, P.J. Wit, R.C.N. Lim, L.L. Owen, B. and Sausen, R. (2009), "Aviation and global climate change in the 21st century", *Elsevier Atmosferic Envi.*, **43**(22-23),3520-3537

Lombardi, G. Mengali, G. and Beux F. (2006), "A hybrid genetic based optimization procedure for aircraft conceptual analysis", J. Optim. Engi., **7**(2), 151-171

McDonald, C.F. Massardo, A.F. Rodgers, C. and Stone, A. (2008), "Recuperated gas turbine aeroengines, part II: engine design studies following early development testing", *Aircraft Engi. Aerospace Tech.*, **80** (3), 280- 294

Raymer, D.P. (2012), *Aircraft Design: A Conceptual Approach (5thEdition),* American Institute of Aeronautics and Astronautics, Washington, DC.

Roskam, J. (2003), *Airplane Design Part I: Preliminary Sizing of Airplanes (2ndEdition),* DARcorporation, Lawrence, KS.

Saha, U. K. Mitra, M. Menon, S. J. John, N. T. Gajapathi, S. S. and Behera P. (2008), "Preliminary design analysis of a lightweight combat aircraft", *Proce. the Insti. Mech. Engi., Part G: J. Aerospace Engi.*, **222,** 507-513

Nomenclature

AR	Wing aspect ratio
AWACS	Airborne warning and control system
b	Wingspan
Cd	Drag coefficient
Cd_{CR}	Drag coefficient at cruise condition
Cd_0	Parasite drag coefficient
Cl	Lift coefficient
Cl_{CR}	Lift coefficient at cruise condition
Cl_{MAX_LND}	Maximum lift coefficient, landing configuration
Cl_{MAX_TO}	Maximum lift coefficient, takeoff configuration
D_{LND}	Landing distance
D_{TO}	Takeoff distance
e	Oswald's coefficient
E_{CR}	Lift to drag ratio during cruise
ELINT	Electronic signals intelligence
E_{TO}	Aerodynamic efficiency, take off configuration
F_{Z_J}	Altitude correction factor for jet engine
F_{Z_PP}	Altitude correction factor for piston propeller engine
F_{Z_TP}	Altitude correction factor for turboprop engine
F_{ZM_JD}	Altitude and Mach number correction factor for jet engine without afterburner
F_{ZM_JW}	Altitude and Mach number correction factor for jet engine with afterburner
K	Drag due to lift factor
K_{comp}	Composite material coefficient
K_{delta}	Delta wing configuration coefficient
K_{eng}	Installation coefficient (installed engine WER)
k_F	Fuel fraction parameter
K_{sys}	Systems weight coefficient (systems WER)
K_{tail}	Tail weight coefficient (tail WER)
K_{TOW}	Maximum takeoff weight coefficient (fuselage WER)
K_{WP}	Wing planform coefficient (wing WER)
l_{fus}	Fuselage length
LSA	Light-sport aircraft
M	Mach number
n_{max}	Maximum load factor
n_e	Engine number
n_{turn}	Load factor during turn maneuver

OEI	One engine inoperative
P_{SL}	Sea level engine power
R_{CR}	Cruise range
SIGINT	Signals intelligence
Sinγ	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

W_{sys}	System weight
W_{tail}	Tail weight
W_W	Wing weight
Λ_{LE}	Wing sweep angle (leading edge sweep angle)
β	Fuel fraction
ζ	Throttle position (percentage)
η_P	Propeller propulsion efficiency
λ	Wing taped ratio
ρ	Air density at sea level
ρ_{CR}	Air density at cruise altitude
ρ_{LND}, ρ_{TO}	Air density at airport altitude
ρ_{SCR}	Air density at super cruise altitude
ρ_{SL}	Air density at sea level
ρ_{turn}	Air density at combat turn altitude
σ	Air density ratio