# CONSIDERATION OF FUEL CONSUMPTION CAUSED BY AIRCRAFT SYSTEMS IN AIRCRAFT DESIGN

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## Abstract

*Aircraft systems contribute to the mission fuel mass of an aircraft due to their mass, power consumption and their impact on drag. The developments of new aircraft system technologies such as all-electric aircraft system architectures promise significant mission fuel mass savings due to improvements in these parameters. Besides this primary effect on fuel consumption, there exists a further potential in fuel reduction if these changes are considered in aircraft design. This paper outlines these coherences with a focus on the determination of the wing area.*


## SYMBOLS

| | |
|---|---|
| $A_1$ | coefficient for calculation of $m_{\text{wing}}$ |
| $A_f$ | aerofoil factor |
| $b$ | wingspan |
| $b_s$ | specific fuel consumption |
| $B_1$ | coefficient for calculation of $m_{\text{wing}}$ |
| $c_{\text{lam}}$ | fraction of chord of wing over which flow is laminar |
| $C_D$ | drag coefficient |
| $C_{D_0}$ | zero lift drag coefficient |
| $C_L$ | lift coefficient |
| $C_{L,\text{ max}}$ | maximum lift coefficient |
| $C_{L,\text{ opt}}$ | lift coefficient at optimum lift to drag ratio |
| $C_{L,\text{ take-off}}$ | lift coefficient at take-off |
| $D$ | drag |
| $F_{\max}$ | installed engine thrust |
| $g$ | gravitational constant |
| $h$ | altitude |
| $h_{\text{rel}}$ | relative enthaply of bleed air |
| $L$ | lift |
| $m_{\text{fuel}}$ | fuel mass |
| $m_{\text{fuel, max}}$ | maximum fuel mass |
| $m_{\text{gear}}$ | landing gear mass |
| $m_{\text{landing}}$ | maximum landing mass |
| $m_{\text{MTOW}}$ | maximum take-off mass |
| $m_{\text{systems}}$ | system mass |
| $m_{\text{wing}}$ | wing mass |
| $m_{\text{zero}}$ | zero fuel mass |
| $\dot{m}_{\text{bleed}}$ | bleed air mass flow |
| $\dot{m}_{\text{ram}}$ | ram air mass flow |
| $Ma$ | Mach number |
| $n$ | polytropic exponent |
| $N$ | maximum load factor |
| $p_{\min}$ | minimum required bleed air pressure |
| $P_{\text{el}}$ | electrical power demand |
| $R$ | mission range |
| $R_s$ | specific gas constant |
| $R_{\text{wetted}}$ | fraction of overall wetted area to reference wing area |
| $S$ | reference wing area |
| $S_{\text{approach}}$ | minimum wing area for required approach speed |
| $S_{\text{fuselage}}$ | fuselage area |
| $S_{\text{landing}}$ | minimum wing area for requested landing distance |
| $S_{\max}$ | maximum allowable wing area |
| $S_{\min}$ | minimum allowable wing area |
| $S_{\text{opt}}$ | wing area for minimum fuel consumption |
| $S_{\text{take-off}}$ | minimum wing area for required take-off distance |
| $S_{\text{tank}}$ | minimum wing area for required tank volume |
| $S_{\text{turbulence}}$ | maximum wing area for required |

| | |
|---|---|
| | sensitivity to turbulence |
| $t$ | time index |
| $T$ | ambient air temperature |
| $T_f$ | wing type factor |
| $(t/c)$ | average thickness to chord ratio of the wing |
| $(t/c)_{root}$ | thickness to chord ratio at wing root |
| $(t/c)_{tip}$ | thickness to chord ratio at wing tip |
| $v$ | true air speed |
| $v_{stall}$ | stall speed |
| $V_{tank}$ | tank volume |
| $x_{landing}$ | landing field length |
| $x_{TOFL}$ | take-off field length |
| $\rho$ | ambient air density |
| $\rho_{fuel}$ | fuel density |
| $\eta_G$ | generator efficiency |
| $\sigma$ | relative density |
| $\kappa$ | adiabatic exponent |
| $\lambda$ | wing taper ratio |
| $\varphi_{eff}$ | effective wing sweep |
| $\Lambda$ | wing aspect ratio |

## 1 Introduction

As a consequence of high fuel prices and the hard-fought aircraft market both on airframer and airliner side, aircraft manufacturers spend high efforts on the reduction of fuel consumption. Besides new technologies in structure materials, aerodynamics and propulsion, a further reduction of fuel consumption may come from new aircraft system technologies. Aircraft systems contribute to the mission fuel mass of the aircraft due to their mass, their ram air needs, and their pneumatic, hydraulic and electric power demand which is taken as bleed air or shaft power respectively from the engine. Besides this direct impact on the mission fuel mass, changes in aircraft system technologies which reduce the mission fuel mass may, under some circumstances, allow the wing area to be resized. This leads to secondary effects on the fuel consumption of the aircraft, as changing the wing area also means changes in structural mass and drag.

## 2 The Impact Of Aircraft Systems On Fuel Consumption

Besides the aircraft system parameters mass and drag-causing ram air for system supply, aircraft systems influence the thermodynamic engine process by taking secondary power from the engine. The term *secondary power offtake* combines shaft power offtake and bleed air offtake for the supply of aircraft systems. While shaft power is taken from one or more engine shafts and converted by generators into electrical or by pumps into hydraulical power, bleed air is consumed directly by the demanding systems. Both kinds of power offtakes result in an increased fuel consumption of the engine.

As the effect of secondary power offtake on the specific fuel consumption of the engine highly depends on the operating point of the engine (flight altitude, Mach number, engine thrust, ambient temperature) and on the engine parameters (thrust category, bypass ratio, inlet mass flow etc.), formulas for an approximation of this effect allow for very rough estimations only. The complex parametrics and continuously varying aircraft operating conditions (consideration of atmospheric parameters, engine operating conditions, mission requirements etc.) resulted in the development of the simulation tool SYSFUEL, which was first presented in [2]. Based on a simplified flight performance model, the influence of shaft power and bleed air offtake on the specific fuel consumption of the engine is calculated with the use of engine decks within each simulation step for the corresponding operating point.

Figure 1 shows a simplified flow chart of the simulation model. After defining a reference mission and the aircraft system parameters, the flight attitude is calculated for each simulation step. The flight attitude is influenced by the mass and the ram air needs of the systems, which increase the required engine thrust. Combined with atmospheric parameters and the mission point, these requirements to the engine from the flight performance are combined with requirements from the aircraft systems (shaft power offtake, bleed air offtake mass flow and stage, minimum bleed

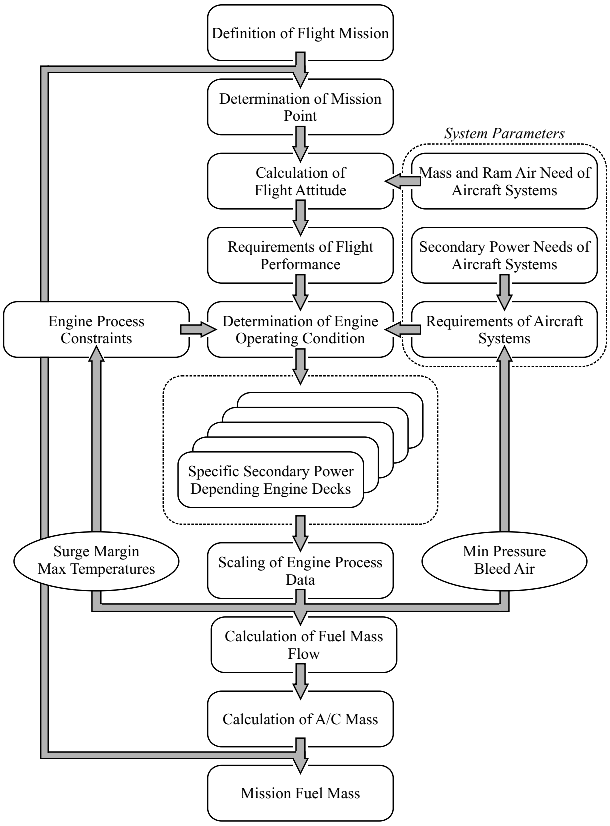


**Fig. 1** Calculation of mission fuel mass in dependency to aircraft system parameters with the simulation tool SYSFUEL.

pressure) and restraints given by the engine process itself (minimum surge margin, maximum temperatures etc.). The tool does not include an engine model, but a combination of engine decks, which each has been rendered considering separate secondary power offtakes. The output data of each engine deck is calculated separately for each operating point and represent engine data linked to a certain reference secondary power offtake. The engine data which represent the secondary power data linked to the system needs is calculated by scaling the singular data using the nodes from each deck and combining the results, as described in [2].

Besides fuel consumption values for each operating point, the engine decks also contain data related to the engine process itself, such as temperatures or surge margins. By using these parameters, it is checked within each simulation step if the engine is able to provide the required secondary power offtake to the systems without exceeding the constraints given by the engine process and the system requirements. This allows an iterative process to adapt the requirements to the engine operating point to these needs. This simulation allows for the calculation of the mission fuel mass caused by an aircraft system architecture on a defined aircraft.

Besides this primary effect, changes of the mission fuel mass due to new aircraft system technologies may by considered during aircraft design and enable adaptions of some design parameters which can result in secondary effects on the mission fuel mass. This interrelationship between the fuel consumption caused by aircraft systems and aircraft design is outlined in the next section. The shown formula and coherences must hereby not be understood as a reference to aircraft design, but they allow to highlight considerable interrelations between system caused fuel consumption and aircraft design parameters.

## 3 The Impact On Aircraft Design

The design of an aircraft is a highly iterative process. As shown in figure 2, almost all design criterias are affected directly or indirectly by the mass and the fuel consumption of the aircraft.

A helpful formula to show the effects of various design parameters in the pre-design phase is the Breguet range equation:

$$R = \frac{v}{b_s \cdot g} \cdot \frac{L}{D} \cdot \ln \frac{m_{\text{zero}} + m_{\text{fuel}}}{m_{\text{zero}}}. \tag{1}$$

Following this equation, the fuel consumption for a given mission decreases with increasing lift to drag ratio. The lift coefficient $C_{\text{L, opt}}$ at optimum lift to drag ratio can be determined graphically as shown in figure 3 for any drag polar.

At cruise condition, the lift equals approximately the weight of the aircraft:

$$L \approx (m_{\text{zero}} + m_{\text{fuel}}(t)) \cdot g = \frac{1}{2} \cdot \rho \cdot S \cdot C_L \cdot v^2. \tag{2}$$

As can be seen in equation (2), the air density has to decrease continuously during flight for maintaining an optimum lift coefficient during cruise, as the aircraft total mass decreases due to fuel

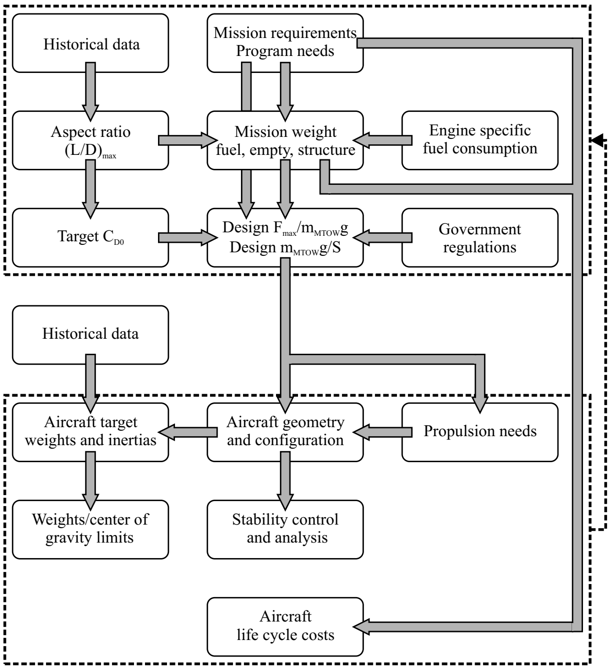


**Fig. 2** Design process of a subsonic transport category aircraft [1].

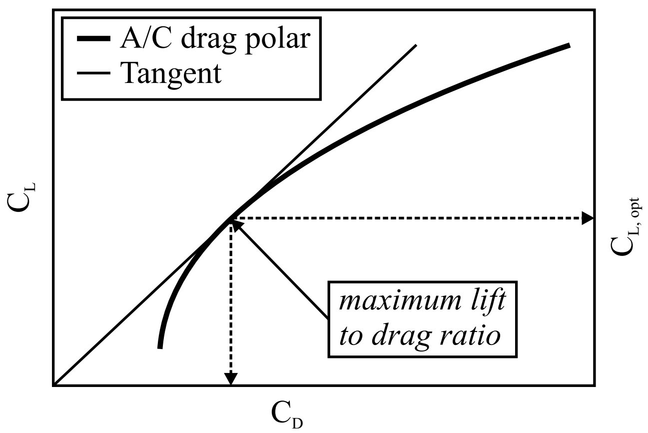


**Fig. 3** Graphical determining of $C_{L,\,opt}$ from a given drag polar.

consumption. This leads to a cruise profile which is often called *continuous climb*. In this case, the flight velocity if flying at constant Mach number will change as well, as speed is linked to the Mach number and temperature (which depends on flight altitude):

$$v = Ma \cdot \sqrt{\kappa \cdot R_S \cdot T}. \tag{3}$$

Atmospheric parameters in dependency to altitude can be calculated using the International Standard Atmosphere (ISA). Equations for temperature and density are

$$T = \begin{cases} T_N + \frac{dT}{dh} \cdot h & , h < 11000\text{m} \\ T_N & , h \geq 11000\text{m} \end{cases} \tag{4}$$

and

$$\rho = \begin{cases} \rho_N \cdot \left(1 + \frac{dT}{dh} \cdot \frac{h}{T_N}\right)^{\frac{1}{n-1}} & , h < 11000\text{m} \\ \rho_N \cdot \exp\left((h - h_N) \cdot \frac{-g}{R_S \cdot T_N}\right) & , h \geq 11000\text{m}. \end{cases} \tag{5}$$

Parameters with index N are constant up to and above 11000m and are given in the ISA. Up to an altitude of 11000m, the required flight altitude for flying at optimum lift to drag ratio can be calculated using equations (2) to (5):

$$\rho_N \cdot \left(1 + \frac{dT}{dh} \cdot \frac{h(t)}{T_N}\right)^{\frac{1}{n-1}} \cdot \left(T_N + \frac{dT}{dh} \cdot h(t)\right) - \frac{2 \cdot (m_{zero} + m_{fuel}(t)) \cdot g}{S \cdot C_{L,\,opt} \cdot Ma^2 \cdot \kappa \cdot R_S} = 0 \tag{6}$$

This equation can be solved numerically for the optimum altitude $h$. Above 11000m, equations (2) to (5) become

$$h(t) = h_N - \frac{R_S \cdot T_N}{g} \cdot \ln\left(\frac{2 \cdot (m_{zero} + m_{fuel}(t)) \cdot g}{T_N \cdot S \cdot C_{L,\,opt} \cdot Ma^2 \cdot \kappa \cdot R_S \cdot \rho_N}\right). \tag{7}$$

As shown in equations (1) to (7), the required flight altitude is also a function of the wing area. Thus, changes of the wing area during the design phase will result in a different altitude level during the continuous climb profile. Assuming no changes of the aircraft mass and drag due to changes of the wing area, the optimum flight altitude level during cruise will increase with increasing wing area. However, changes in flight altitude are limited by the service ceiling, which depends on many factors like engine technology or aerodynamics.

Figure 4 demonstrates these coherences by considering two alternative system architecture configurations A and B, with architecture B having a lower fuel consumption and a smaller wing area. After starting the continuous climb in the cruise phase, the aircraft with the configuration A reaches the maximum ceiling at $t_1$ and continues cruise from this point at constant altitude. According to equation (2) and because of further fuel consumption, this results in a lift coefficient, and thus lift to drag ratio, beyond the optimum value. The lower fuel consumption of configuration B allows for a smaller wing area, why this architecture reaches the maximum ceiling later at $t_2 > t_1$, and the aircraft is able to maintain the continuous climb profile and thus optimum lift to drag ratio longer than architecture A.

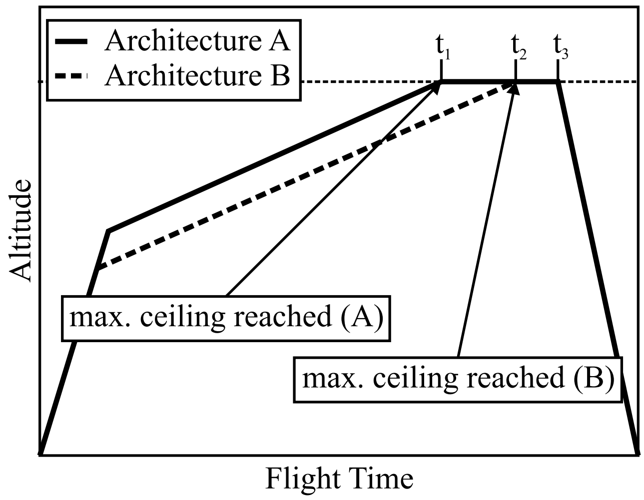


**Fig. 4** Simplified lapse of flight altitude during reference mission.

It is obvious that a change of the wing area to maintain an optimum lift to drag ratio may be preferable to a change of the altitude level (not to be confounded with the change of altitude during the continuous climb, which is performed in both cases) for some conditions as shown in figure 4. However, as a variation of the wing area will also affect the total mass and drag of the aircraft, the determination of the optimum wing area results in a complex optimization task.

Besides the effect on the optimum wing area, changes of the mission fuel mass (and, thus, changes of the total aircraft mass) will also impact other systems such as the landing gear. The landing gear mass can be considered as a fraction of the maximum take-off mass of the aircraft, what results again in the need for an iterative calculation process. According to [3], this fraction can be estimated as

$$\frac{m_{\text{gear}}}{m_{\text{MTOW}}} = 0.04 \tag{8}$$

for a transport aircraft with two main landing gear units. Although other aircraft systems or structures may be affected as well by changes of the wing area or the maximum take-off mass, this paper is limited to considerations concerning the wing and the landing gear.

## 3.1 Wing area variation constraints

Although the optimum lift to drag ratio is an important factor for the determination of the wing area, several constraints have to be considered. Depending on the mission requirements, dimensioning factors for the wing area may be –among others– the fuel tank volume, the approach speed, the maximum take-off or landing distance, sensitivity to turbulence or installation space for equipment like high-lift systems or spoiler actuation. Another criterion for the wing area is the use of the same wing for different aircraft of one aircraft family as for the A330-200/300 and A340-200/300. As each criterion has to be fulfilled, it depends on the requirements to the aircraft which is the sizing criterion of the wing area:

$$\begin{aligned} S_{\text{min}} &= \max\{S_{\text{tank}}, S_{\text{approach}}, S_{\text{take-off}}, S_{\text{landing}}, \ldots\} \\ S_{\text{max}} &= \min\{S_{\text{turbulence}}, \ldots\} \end{aligned} \tag{9}$$

### 3.1.1 *Minimum approach speed*

As the maximum lift coefficient is limited by the high-lift system (flaps and slats), equation (2) defines a minimum wing area at a given speed. For the approach this equation can be written, considering an minimum approach speed and maximum lift coefficient, as

$$S_{\text{approach}} \geq \frac{2 \cdot m_{\text{landing}} \cdot g}{\rho \cdot C_{\text{L, max}} \cdot (1.3 \cdot v_{\text{stall}})^2}, \tag{10}$$

where the factor 1.3 between approach speed and stall speed reflects a safety margin.

### 3.1.2 Maximum take-off and landing distance

The maximum take-off distance is limited by the available maximum take-off field length, which is defined in the FAR 25 regulations as the combination of the lift-off distance and the distance needed to reach 35ft altitude. According to [1], a possible approximation to estimate the required wing area is

$$S_{\text{take-off}} \geq 37.5 \cdot \frac{g^2 \cdot m_{\text{MTOW}}^2}{x_{\text{TOFL}} \cdot \sigma \cdot C_{\text{L, take-off}} \cdot F_{\max}}. \tag{11}$$

Assuming a defined installed engine thrust independent from aircraft system changes, the required wing area is linked directly to system mass and mission fuel mass changes.

An approximation for the wing area required for a given landing distance is given in the same source by

$$S_{\text{landing}} \geq 0.3559 \cdot \frac{g \cdot m_{\text{landing}}}{x_{\text{landing}} \cdot \rho \cdot C_{\text{L, max}}}. \tag{12}$$

The authorities provide several constraints for these requirements, such as consideration of engine failure or system failures during take-off and landing, which are taken into consideration in the presented formula.

### 3.1.3 Sensitivity to turbulence in cruise

The minimization of discomfort to both crew and passengers caused by turbulences may be another criterion for wing sizing. The sensitivity of the aircraft to turbulence is linked with the wing loading. A possible approach for a wing area which is satisfactory in this respect is given in [3] as

$$S_{\text{turb}} \leq \frac{m_{\text{MTOW}} \cdot g \cdot \left(0.32 + \frac{0.16 \cdot \Lambda}{\cos\varphi_{\text{eff}}}\right)}{2.7 \cdot v_{\text{dive}} \cdot \Lambda} \cdot \left(1 - \left(Ma \cdot \cos\varphi_{\text{eff}}\right)^2\right)^{0.5}. \tag{13}$$

Unlike the other requirements, this criterion provides a maximum value for the wing area, which must not be exceeded during the variation of the wing area.

### 3.1.4 Tank volume

Figure 5 shows a typical payload-range-diagram for an airliner. Mission A defines the design point for maximum payload at a certain mission range. The shown missions A and B are typical requirements to aircraft design. The required fuel mass grows with increasing range up to mission A, where the aircraft take-off mass reaches its limit. Between missions A and B, the increasing fuel mass has to be balanced with decreased payload due to the maximum allowed take-off mass of the aircraft. With the maximum fuel capacity due to the tank volume reached at point B, a further range increase can only be achieved by reducing payload further. If resizing the wing area it has to be ensured that the available tank volume does not reduce the aircraft range at given payload.

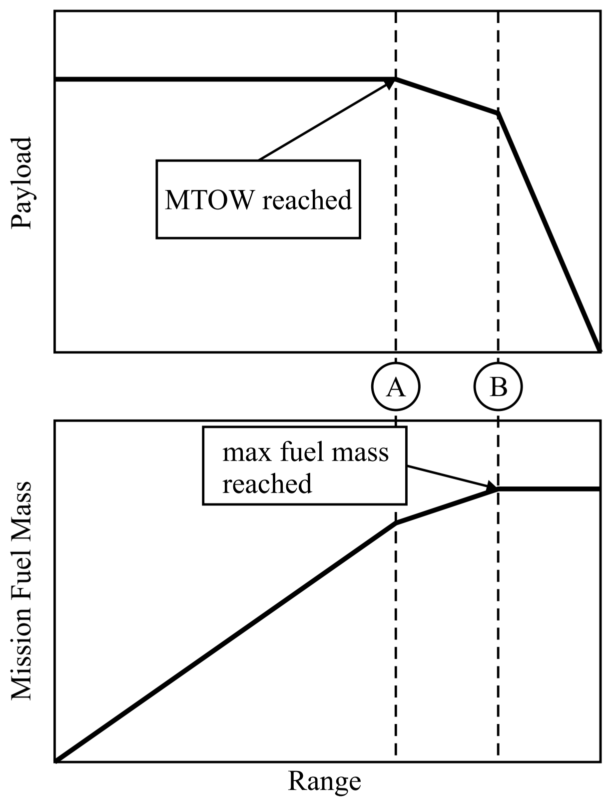


**Fig. 5** Payload-range-diagram for a typical airliner.

According to [4], a first approximation of the tank volume for a straight tapered, swept wing design can be obtained by using

$$V_{\text{tank}} = 0.54 \cdot \frac{S^2}{b} \cdot (t/c)_{\text{root}} \cdot \frac{1+\lambda \cdot \sqrt{\tau} + \lambda^2 \cdot \tau}{(1+\lambda)^2} \tag{14}$$

with

$$\tau = \frac{(t/c)_{\text{tip}}}{(t/c)_{\text{root}}} \tag{15}$$

and

$$V_{\text{tank}} = \frac{m_{\text{fuel, max}}}{\rho_{\text{fuel}}}. \tag{16}$$

The maximum mission fuel mass $m_{\text{fuel, max}}$ must also include all reserves, e.g. for a missed approach, flight to an alternate airport or holding. This leads to a minimum wing area of

$$S_{\text{tank}} \geq \sqrt{\frac{m_{\text{fuel, max}} \cdot b \cdot (1+\lambda)^2}{0.54 \cdot \rho_{\text{fuel}} \cdot (t/c)_{\text{root}} \cdot \left(1+\lambda \cdot \sqrt{\tau} + \lambda^2 \cdot \tau\right)}}. \tag{17}$$

## 3.2 Effects of wing area variations on mass and drag

The previous section describes both the constraints and the motivation for adapting the wing area if the mission fuel mass of the aircraft changes. As the wing area is linked to several geometric parameters, it has to be defined which parameters are kept constant while scaling the wing area. In this paper, the aspect ratio of the wing is assumed to be kept constant:

$$\Lambda = \frac{b^2}{S} = \text{const.} \tag{18}$$

This leads to the assumption that a change of the wing area is considerd as a change of the wing span width $b$.

Changing the wing area generally leads to changes of the wing mass and the zero lift drag. Mass changes due to variations of the wing area can be estimated according to [3] using

$$\begin{aligned} m_{\text{wing}} = \quad & \left(A_1 - B_1 \cdot m_{\text{MTOW}} \cdot 10^{-3}\right) \\ & \cdot \left[\Lambda^{0.5} \cdot S^{1.5} \cdot \sec\varphi_{\text{eff}} \cdot \left(\frac{1+2\cdot\lambda}{3+3\cdot\lambda}\right)\right. \\ & \cdot \frac{m_{\text{MTOW}}}{S} \\ & \left. \cdot (1.65 \cdot N)^{0.3} \cdot \left(\frac{v_{\text{dive}}}{(t/c)_{\text{root}}}\right)^{0.5}\right]^{0.9} \end{aligned} \tag{19}$$

For this paper, the wing mass results of this equation have been multiplied with an adaption factor gathered from comparison with a built reference aircraft.

An approximation for the estimation of the zero lift drag coefficient is also given in [3] by

$$\begin{aligned} C_{D_0} = 0.005 \cdot & \left(1 - \frac{2 \cdot c_{\text{lam}}}{R_{\text{wetted}}}\right) \cdot \tau \cdot R_{\text{wetted}} \cdot T_{\text{f}} \cdot S^{-0.1} \\ \cdot & \left[1 - 0.2 \cdot Ma + 0.12 \cdot \left(\frac{Ma \cdot (\cos\varphi_{\text{eff}})^{0.5}}{A_{\text{f}} - (t/c)}\right)^{20}\right] \end{aligned} \tag{20}$$

with

$$\tau = \frac{R_{\text{wetted}} - 2}{R_{\text{wetted}}} + \frac{1.9}{R_{\text{wetted}}} \cdot \left(1 + 0.526 \cdot \left(\frac{(t/c)}{0.25}\right)^3\right) \tag{21}$$

and

$$R_{\text{wetted}} = \frac{S_{\text{wetted}}}{S} = \frac{S_{\text{fuselage}} + 2 \cdot S}{S} = \frac{S_{\text{fuselage}}}{S} + 2. \tag{22}$$

As the fuselage is assumed to remain unchanged, an increased wing area leads to a reduced factor $R_{\text{wetted}}$ and, thus, to a reduced zero lift drag coefficient. However the zero drag increases in that case, as the product $S \cdot C_{D_0}$ increases. The changes of the zero lift drag coefficient also influence the optimum lift to drag ratio.

Figure 6a shows the impact of changes of the wing area on wing mass and the zero lift drag coefficient. In the special case that the wing area is sized due to the required tank volume given in equation (17), and thus maximum mission fuel mass, it is possible to express changes of these parameters in dependency to the mission fuel changes by combining equations (14) to (22). This interrelationship is shown in figure 6b.

As a consequence of this, reductions of the mission fuel mass, primarily caused by system effects, offer the potential for a further reduction of the required mission fuel by adapting the wing area to a decreased required tank volume if this has been the sizing parameter. This secondary effect is shown in the following section.

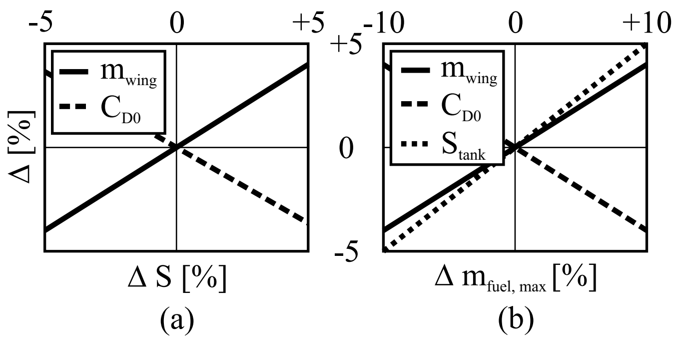


**Fig. 6** Effects of wing area and maximum mission fuel mass changes on mass and zero lift drag coefficient of the wing.

### 3.3 The secondary effect of wing area variations on fuel consumption

To calculate the effect of wing area adaption due to system parameter changes on fuel consumption, the simulation tool SYSFUEL shown in figure 1 has been extended by implementing a wing area adaption function, which is shown in figure 7.

Starting with the wing area of the reference aircraft, a first calculation is done for the mission fuel mass changes considering the parameters of the new system architecture. As these parameters lead to changes of the take-off mass of the aircraft for the design mission, changes of the wing mass and landing gear mass have to be considered as well, as these are linked directly to the maximum take-off mass, see equations (8) and (19). These interrelationships result in an iterative process, as the mass changes involve a new mission fuel mass again. The iteration is repeated until the changes of the effect of maximum take-off mass changes on the wing and landing gear mass is negligible. This process, which leads to the determination of the mission fuel mass for a defined wing area, is repeated for each variation step of the wing area, until a variation of the wing area does not further reduce the mission fuel mass or violates any constraints. Figure 8 shows the evolution of the optimum wing area and the according mission fuel mass for the case of a wing area adaption to a decreased mission fuel mass.

The potential mission fuel mass reduction due to an adaptation of the wing area is shown

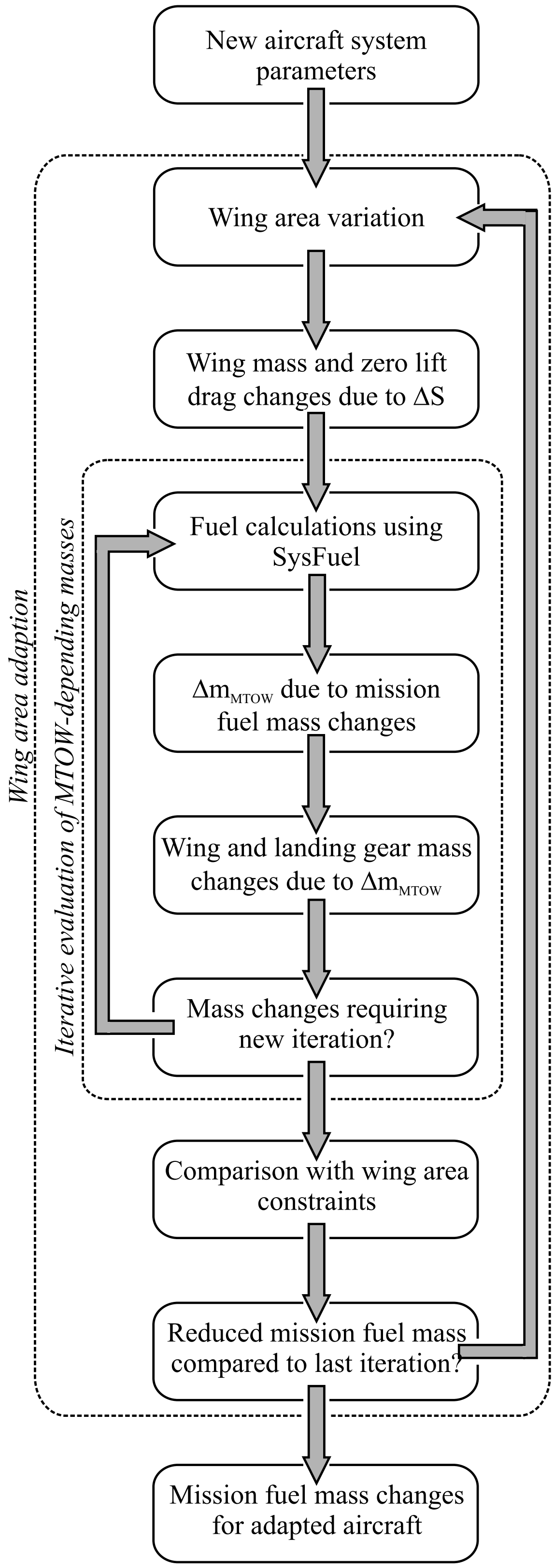


**Fig. 7** Implementation of the wing area adaption process in SYSFUEL.

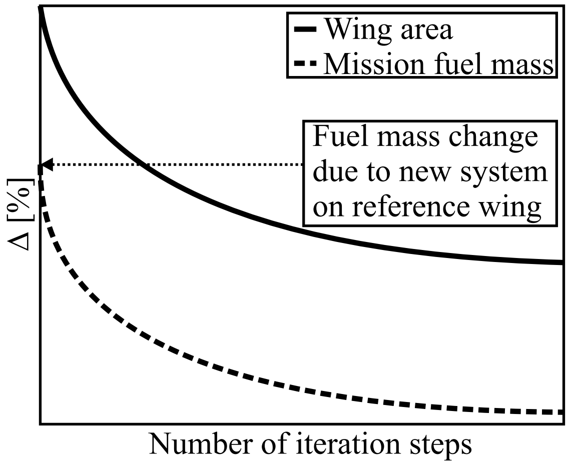


**Fig. 8** Evolution of optimum wing area and according mission fuel mass.

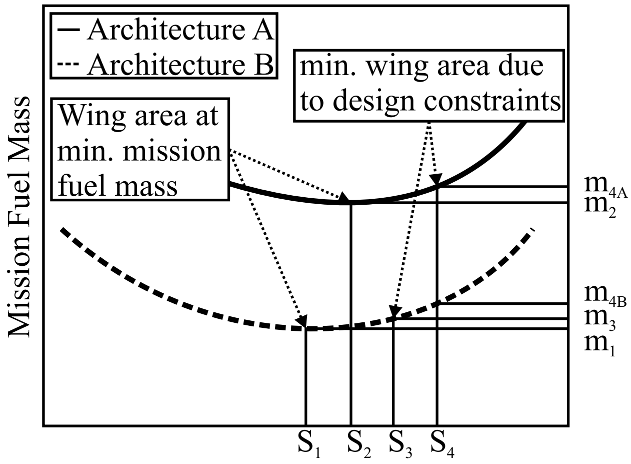


**Fig. 9** Fuel reduction due to new aircraft system parameters for unadapted and resized wing area.

in figure 9 for the system configurations A (reference) and B again. The wing area $S_2$, which would theoretically lead to the lowest mission fuel mass for the aircraft with architecture A, can not be realised due to the constraints given in section 3.1. This leads to the reference wing area $S_4$. An implementation of the new aircraft system technology B on the aircraft with an unchanged wing area would lead to a reduction of the mission fuel mass from $m_{4A}$ to $m_{4B}$. An adaption of the wing area as described in this section reduces –considering all secondary effects and constraints– the wing area to $S_3$ and the according mission fuel mass to $m_3$. Likewise the reference architecture, the wing area which would result in a minimum mission fuel mass can not be realised due to the constraints.

It should also be mentioned that the optimum wing area may also increase for alternate system architectures, depending on the particular parameters.

**Table 1** Nomenclature of mission fuel mass parameters in figure 9.

| | |
|---|---|
| $m_1$ | mission fuel mass of architecture B at minimum fuel wing area $S_1$ |
| $m_2$ | mission fuel mass of architecture A at minimum fuel wing area $S_2$ |
| $m_3$ | mission fuel mass of architecture B at wing area $S_3$ due to wing area constraints |
| $m_{4A}$ | mission fuel mass of architecture A at wing area $S_4$ due to wing area constraints |
| $m_{4B}$ | mission fuel mass of architecture B at wing area $S_4$ (unchanged reference wing) due to wing area constraints |

## 4 Example Calculation

The potential wing area adaption benefit is shown in this section by evaluating the fuel consumption of two alternative system architecture technologies A and B for a long-range aircraft. Table 2 gives an overview on relevant aircraft data, while table 3 lists relevant system parameters. The example calculation compares a conventional environmental control system architecture to an electrified bleedless one, with the system parameters taken from [2] with the additional assumption of no mass increase.

The design mission in this example requires a wing area determined by the required mission fuel mass. The new system architecture B results in a mission fuel mass reduction of 1.4% and, thus, allows for a smaller layout design of the wing area. The iterative wing area adaption process described in the previous section indicates a further reduction of the wing area by 0.9%, what results in a 2.3% reduced mission fuel mass and in a reduction of the maximum take-off mass of 1.6% considering all secondary effects.

**Table 2** Reference mission, aircraft and engine data.

| | |
|---|---|
| Max. take-off mass [t] | 275 |
| Reference wing area [$m^2$] | 462.5 |
| Cruise Mach number [-] | 0.86 |
| Max. range [nm] | 8000 |
| Service ceiling [ft] | 39000 |
| Installed engine thrust [kN] | 2·300 |
| Bypass ratio (take-off) [-] | 5.1 |

**Table 3** Mission fuel mass relevant parameters of system architectures A and B.

| | Architecture A | Architecture B |
|---|---|---|
| $m_{\text{systems}}$ [kg] | incl. | incl. |
| $\dot{m}_{\text{ram}}$ [kg/s] | 1.8 | 3.8 |
| $\dot{m}_{\text{bleed}}$ [kg/s] | 2.2 | 0.0 |
| $h_{\text{rel}}$ [-] | 0.3 | - |
| $p_{\text{min}}$ [bar] | 2.6 | - |
| $P_{\text{el}}$ [kW] | - | 300 |
| $\eta_G$ [-] | - | 0.9 |

**Table 4** Wing area, take-off and mission fuel mass changes due to system technology change without ($B_1$) and with wing area adaption ($B_2$).

| | $B_1$ | $B_2$ |
|---|---|---|
| $\Delta m_{\text{MTOW}}$ [%] | -0.7 | -1.6 |
| $\Delta S$ [%] | n/a | -0.9 |
| $\Delta m_{\text{fuel}}$ [%] | -1.4 | -2.3 |

## 5 Conclusion

The reduction of mission fuel mass due to new aircraft system technologies can be amplified by consideration of its coherences in aircraft design.

The quantitative impact has shown to be highly sensitive to design parameters of the aircraft, such as design mass, maximum speed or service ceiling. For specific aircraft design parameters, resizing the wing area with respect to a new mission fuel mass showed a considerable effect in an example calculation. It is evident that the potential benefit of wing area changes is highly linked to the mass and drag sensitivities to the wing area. Due to the simple applied mass and drag approximation models in this paper, the results can diverge significantly for other cases. It seems to be essential to replace these general approximations with mass and drag models linked to the particular reference aircraft in each case. A general quantitative statement on the impact of fuel consumption caused by aircraft systems on aircraft design can not be given, as this process may be dominated by other considerations in each specific case. Nevertheless, especially for long-range missions, the presented coherences can become considerable under certain conditions, and the extension of the simulation model has shown a huge potential for further studies.

## LITERATURE


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