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AIRPLANE CONCEPTUAL DESIGN IMPROVEMENT THROUGH SIMPLIFIED MULTIDISCIPLINARY OPTIMIZATION USING ANALYTIC MODELS

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Abstract. *A numerical optimization procedure is proposed to obtain the best airplane configuration in the conceptual design phase of a subsonic jet transport aircraft. In order to determine the "optimal aircraft" and the corresponding "optimal trajectory" that minimizes a combination of mission parameters, some models for estimating the aircraft performance have been selected and implemented. Two degrees of freedom performance models for climb, cruise and descent flight phases are used to estimate the aircraft fuel consumption and time elapsed to perform a typical mission. Empirical models are used to evaluate the takeoff and landing field lengths which are considered as optimization constraints. To obtain an accurate prediction of the airplane drag, a comprehensive model has been selected due to its capability of handling changes in geometric and aerodynamic data. Empirical methods were applied to correct aircraft empty weight due to the changes in design parameters. The optimization criteria adopted in this work are based on linear combinations of estimated time and fuel spent to fly a typical mission. The mathematical optimization is done using a commercial code which employs a gradient method. The results presented show reasonable objective function reduction through optimization of the design and trajectory parameters.*

Keywords: *Multidisciplinary Design Optimization, Aircraft conceptual design, Aircraft performance, Aerodynamic drag.*

1. INTRODUCTION

The airplane design is a process where the skill and experience of a multidisciplinary team of engineers are required in order to obtain a solution which complies to several requirements simultaneously. It is commonly divided in three phases: conceptual, preliminary and detailed design. The conceptual design is the phase when airplane configurations which satisfy all design specifications are selected and the best among them is frozen. The choice of a best configuration suggests an opportunity of performance optimization in this phase. In preliminary design the external geometry, structures and systems are defined so that the "virtual" airplane is created. The detailed design is when the many parts of the airplane are designed in detail and the production processes are established.

One way of helping the design group to obtain the best airplane is the application of Multidisciplinary Design Optimization (MDO) as defined in many published works coordinated by the Technical Comittee for MDO in AIAA (American Institute of Aeronautics and Astronautics), and which consists in using optimization methods considering several engineering disciplines simultaneously.

Each one of the engineering disciplines may be represented by an analytic model, constituting the so-called simplified multidisciplinary optimization. This kind of analysis has become a field of research of great present interest due to the relevance and the up-to-dateness of the procedure, with many recently published works. Suzuki & Kawamura (1996), Dovi & Wrenn (1990 and 1995), Malone & Mason (1995), Pant et al. (1995), Simos & Jenkinson (1988), Wells & Shevell (1982) and Dixit & Patel (1980) are examples of works in this field.

Exploring the possibilities of simplified MDO, the objective of this work is to present a procedure to improve a conceptual design of a jet transport aircraft conceived for subsonic flight. This improvement is done through performance optimization in a typical mission profile: the objective function is a linear combination of the time spent and the fuel burnt in that mission. The optimization process search the optimum values of some geometric, aerodynamic and propulsive variables that, together with some other fixed variables, define the aircraft (Diniz,1998).

2. FORMULATION

The presented conceptual design optimization may be solved through the determination of the optimum values of some aircraft parameters, considering a mission flight trajectory with all controls fixed. In the case of a commercial transport aircraft this is a pertinent assumption because the air traffic procedures constrains the flight trajectory to a typical mission profile (Diniz,1998). However, the inclusion of some trajectory parameters in the group of variables to be optimized may imply a better optimum objective function.

2.1 Simplifying assumptions

An adequate aircraft performance estimation only requires two degrees of freedom, which are the horizontal and the vertical distances.

In the trajectory calculation considering the typical mission profile the flight controls are completely prescribed in the segments of climb and descent. During these phases $L = W \cdot \cos\gamma \approx W$, where L is the lift force, W is the aircraft weight and γ is the flight path angle. According to Boeing (1964), for typical trajectory angles of commercial transport flights, where γ is usually less than 15 degrees, the latter approximation is valid.

2.2 Simplified formulation: simultaneous optimization

The incorporation of the simplifying assumptions in the optimization problem presented in the Introduction leads to the formulation of a simultaneous optimization which may be expressed as a parametric optimization problem, where the unknown vector is $X = \{ \Omega^T \; h_d \; \rho_d \; h_a \; \rho_a \}$ and the vector of design parameters is $\Omega = \{ S \; A \; \lambda \; t/c \; \Lambda_{1/4} \; F_T \}$. The elements of these vector are $\boldsymbol{S}$ as the wing area; $\boldsymbol{A}$ as the wing aspect ratio; λ as the wing taper ratio; $^t/_c$ as the wing thickness to chord ratio (at M.A.C.=mean aerodynamic chord); $\Lambda^1/_4$ as the sweepback angle at the wing quarter-chord line; $\boldsymbol{F_T}$ as the thrust factor ($T_{available} = F_T$ x T_{engine}); $\boldsymbol{h_d}$ as the flight altitude of the cruise to destination; $\boldsymbol{\rho_d}$ as the maximum specific range reduction at the cruise to destination; $\boldsymbol{h_a}$ as the flight altitude of the cruise to alternate; $\boldsymbol{\rho_a}$ as the maximum specific range reduction at the cruise to alternate.

Given this vector it is possible to integrate explicitly the equations of motion using the methods shown in Section 3 .

After that the objective function, which is a linear combination of the block time (*BT*) and the block fuel (*BF*), may be evaluated :

$$IP(\Omega, h_d , \rho_d, h_a, \rho_a) = \alpha \times BF(\Omega, h_d ,\rho_d, h_a, \rho_a) + \beta \times BT(\Omega, h_d , \rho_d, h_a, \rho_a) \qquad (1)$$

where α and β are scalar weight coefficients which correspond to the respective specific costs. The block fuel is the weight of fuel used in the flight from the origin to the destination airports and the block time is the elapsed time in this same flight .

The minimization of the performance index (*IP*) is subject to some inequality constraints:
a) optimization variables limits, $X_{INFi} \leq X_i \leq X_{SUPi}$, $i = 1, N_{NVO}$, where N_{NVO} is the number of the optimization variables; X_{INFi} is the lower limit of the variable X_i and X_{SUPi} is the upper limit of the variable X_i ;
b) takeoff constraint, $T.O.F.L.(X) \leq RUNWAY$, where *T.O.F.L* .is the takeoff field length and *RUNWAY* is the available runway length;
c) landing constraint, $L.F.L. (X) \leq RUNWAY$, where *L.F.L.* is the landing field length.

The integration of the equations of motion is done by the Euler method using the derivatives obtained from the simplified models for performance estimation. The flight segments are arranged in the mission typical profile resulting in the calculation of the flight trajectory .

Note that this is a solution of a simultaneous optimization of aircraft design and flight trajectory parameters .

Because a linear combination of two objective functions of different kind relates them to a single scalar objective function which is a cost measure, it is not necessary to use the methods of a multi-objective optimization, where the objective function is a vector (Dovi & Wrenn, 1990 and 1995; Malone & Mason, 1995).

The formulation of a simultaneous optimization can be further simplified reducing the unknown vector to the aircraft design parameters, $X = \Omega^T$. The objective function becomes $IP(\Omega) = \alpha \times BF(\Omega) + \beta \times BT(\Omega)$ and the inequality constraints are expressed by: a) $\Omega_{INFi} \leq \Omega_i \leq \Omega_{SUPi}$, $i = 1, N_{PARAM}$; b) $T.O.F.L.(\Omega) \leq RUNWAY$ and c) $L.F.L. (\Omega) \leq RUNWAY$.

3. APPLICATIONS AND RELATED METHODOLOGY

3.1 Study Configuration

The basic aircraft, from which are generated all the initial designs for the optimization calculations, is a subsonic jet transport aircraft, low wing, with two high bypass ratio turbofans installed in the rear fuselage. Each one of the engines is rated at 3380 kgf at sea level and ISA. It should carry 50 passengers to at least 1500 km.

The design problem which is treated in this work is referred as "reverse design problem" (Dixit & Patel, 1980). In this kind of problem, the airplane design is conducted around a known engine.

The thrust and the fuel flow data are tabled as function of the engine operation mode, of the altitude and of the flight speed for a given deviation from the standard atmosphere. All data is taken from the engine manufacturer computer deck (Rolls-Royce,1998).

The fixed data which define the geometry and the aerodynamic configuration of the study configuration are shown in Diniz (1998). These data together with the optimized parameters allow the calculation of the airplane drag polar according to the method referenced in this section.

3.2 Analytic Models

The objective function of the optimization procedure is a performance figure of the aircraft. Therefore it is necessary the selection of methods which allow the performance estimation as function of the design parameters and which are fast enough to permit many optimization iterations.

Performance calculations require aircraft drag data, usually calculated from drag polars. The modification of some design parameter lead to a new drag polar which must detect the resultant aircraft drag change.

The aircraft performance is also affected by the aircraft weight. In general, the changes in the design parameters lead to changes in the aircraft empty weight.

All models used in the objective function calculation have known fidelity, which means they are all validated.

Drag Model. The aircraft performance estimation requires the availability of the drag polars to be used in each flight phase. Then the drag polars estimation is an essencial part of the conceptual design of an aircraft. The chosen method obtains the subsonic and incompressible drag polar. The compressible drag correction, taken as function of the Mach number and the lift coefficient, is then added in each point of the trajectory .

The subsonic drag polar calculation method must be able to translate in terms of aerodynamic drag the changes in the design parameters. However, most of the available methods for drag polar preliminar estimation are based on the empirical / statistical evaluation of the drag polars of existing aircraft. The chosen methodology was developed by Torenbeek (1982) and calculates thirty one drag components divided in four groups : induced drag, profile drag, interference corrections and the drag due to protuberances and surface imperfections.

Compressibility corrections. The "drag creep", when $M < M_{DIV}$, is given by the equation:

$$\Delta C_D = \frac{0.002}{\left\{1+\eta\frac{M_{DIV}-M}{\Delta M}\right\}} \quad . \qquad (2)$$

The "drag rise" , when $M > M_{DIV}$, is given by :

$$\Delta C_D = 0.002\left\{1+\frac{M-M_{DIV}}{\Delta M}\right\}^{\eta} \quad , \qquad (3)$$

where M_{DIV} is the drag divergence Mach number; η =0.2 and ΔM=0.06 . These values of η and ΔM are default values used by the program. However, they may be changed if better information is available.

Validation of the drag prediction method. The drag prediction method was implemented by Torenbeek et al. (1983), where the validation results are discussed involving many airplanes of many classes, for wich geometric and aerodynamic data were available .

All calculation routines are based on classical methods and experimental data. Realistic results were found for subcritical polars (generally within 10 to 15 counts; 1 count = 0.0001), but the precision is poorer for high subsonic speeds, especially at high angles of attack .

The program presented by Torenbeek et al. (1983) was used to obtain the theoretical drag polars for many airplanes from EMBRAER as well as for many airplanes for which experimental drag polars were available on that reference. The geometric and aerodynamic data for these airplanes were obtained from three-view drawings and from several editions of the "Jane's All the World Aircraft". The results of this validation are shown in Embraer (1992), which shows that for jet-transport aircraft the percentual error in the total drag evaluated at a typical C_L is not greater than 6%.

Comparison of the compressibility correction results with flight data shows that calculated M_{DIV} is smaller than flight test values (Torenbeek et al., 1983).

Equivalent wing. The "equivalent" straight-tapered wing planform is a suitable representation of the true wing planform because it has the same overall characteristics of wing-body combinations with cranks or notches. Using this concept makes possible to employ the considerable mass of generalised data already published for straight-tapered wings in terms of their planform geometry .

The concept of the equivalent wing planform, detailed in Engineering Sciences Data (1976), was first developed by ESDU to estimate the location of the aerodynamic centre for a wing-body combination. Since that time it has been employed as a standard procedure in numerous applications within ESDU and has proved very satisfactory.

Performance Models. In order to obtain an adequate performance estimation to be used in an optimization procedure, it is necessary to use reduced order models. These models allow fast performance estimates without compromising precision. To obtain an adequate performance prediction it is not necessary a complete dynamic model of the airplane, with six degrees of freedom. The procedure proposed in this work uses reduced order models, with two translational degrees of freedom, the horizontal and vertical distances. Similar models are shown in Boeing (1964) .

The climb model calculates the rate of climb as function of thrust, drag, speed and weight. The thrust is set to maximum climb rate and is function of speed, altitude and temperature. The speed is prescribed as function of altitude and temperature, usually as the maximum rate of climb speed.

The descent model is analogous to the climb model. Thrust is set to the flight idle rate and the speed is the maximum operational speed, V_{MO}. However if a constant rate of descent is specified instead of the thrust mode, the required thrust to allow a permanent descent at V_{MO} and with the specified rate of descent is calculated. If it is smaller than the flight idle thrust, then thrust is set to the flight idle value and the rate of descent is calculated.

The cruise models refer to permanent level flight, where lift equals weight and thrust equals drag. The maximum cruise mode is obtained with the speed where the maximum cruise thrust equals drag. The economical cruise is obtained through the percentual reduction of the maximum specific range in order to achieve a greater speed. The maximum endurance speed is obtained as the speed which implies the minimum fuel consumption for a given flight condition.

Trajectory Calculation. The flight trajectory is composed according to the typical mission profile. First, a climb is performed to achieve the cruise altitude after which is calculated a cruise trajectory at the specified mode assuming a range equal to the distance to be flown less the climb distance. A descent is performed to the final altitude resulting in a total flown distance. Then, a iteration is done re-estimating the cruise distance to be flown until the sum of the cruise, climb and descent distances are equal to the airports distance within a given tolerance .

Validation of the Performance Estimation Method. The performance models were implemented as computer programs in order to be used in the optimization procedure. The output of these programs were compared to the EMB-145 Performance Manual, Embraer (1997). It was used the proper engine data and the flight-test drag polars incorporate the compressibility effect. The results showed satisfactory agreement as shown in Diniz (1998).

Flight Regulations. The airplane operation in a country is subject to regulations issued by the aeronautical authorithy of that country. In Brazil, there are the R.B.H.A. ("Regulamentos Brasileiros de Homologação Aeronáutica"); in the USA, there are the F.A.R. (Federal Aviation Regulations); in most of Europe, the J.A.R. (Joint Aviation Regulations) are adopted; and so on.

The certification of the aircraft design requires compliance to the takeoff and landing regulations as those in FAR-part25 (Federal Aviation Regulations, 1982), which are taken into account in the optimization procedure proposed in this work .

The route flight is also regulated, being required the presentation of a flight plan before takeoff. The FAR-part 121 (Federal Aviation Regulations, 1986) require the consideration of a mission profile, with an alternate airport, holding time and additional fuel reserves taken as function of the range to be flown.

Weight Correction Methods. The weight corrections are evaluated through empirical methods which relate a given component weight to the aircraft takeoff weight, to some wing design parameters and to the engine thrust.

Two methods were selected: Roskam (1985b) and Beltramo et al. (1977). There were identificated thirteen aircraft weight components affected by the design variables in the first method. In the second method there were identificated eight weight components to be modified. The original weights of these components were determined and the corrections due to the new values of the design variables are then evaluated and added to these original values.

Both methods were calibrated to give no correction at the reference configuration, taken to be the following values of the design variables: wing area = 51.18 m^2 , aspect ratio = 7.8 , taper ratio = 0.25 , thickness over chord ratio = 0.12 , sweep-back angle at quarter chord line = 22.73 deg , cruise thrust factor = 1.0.

Takeoff Field Length Estimation. In order to calculate the takeoff field length to be used as a constraint in the optimization procedure, it was built a correlation: *TOFL = 1.39 x*

$TOP25$, where $TOFL$ is the takeoff field length and $TOP25$ = (W/S) / (T*$C_{L,LO}$ *(T_0/W)). The data used in this correlation were obtained from flight test.

The lift coefficient at lift-off, $C_{L,LO}$ is equal to: $C_{L,LO} = C_{Lmax)cruise} + \Delta C_{Lmax)takeoff\ flaps}$, where $C_{L,LO}$ is affected by the design variables through the $C_{Lmax)cruise}$. Loftin (1980) and Roskam (1985a) give examples of relations using this parameter.

The consideration of the takeoff constraint is important in the cases where the optimum wing area is smaller than the one required to allow the takeoff in the available runway.

Landing Field Length Estimation. Based on theoretical formulations of the calculation of the landing ground run, it is expected a relation between the square of the stall speed and the landing ground run. The approach speed is defined as: $V_A = 1.3 \times V_{SL}$, where V_{SL} is the stall speed at landing configuration. Then, $S_{FL} = k \, x \, V_A{}^2$.

Considering ,

$$V_{SL} = \sqrt{\frac{2W}{\rho\, S\, C_{L\,max)landing}}} \tag{4}$$

and $C_{Lmax)landing} = C_{Lmax)cruise} + \Delta C_{Lmax)\ landing\ flaps}$. The design variables affect $C_{lmax)landing}$ through the $C_{Lmax)cruise}$.

It was obtained a relation of the landing field length as function of the square of the approach speed from fligth test data.

The consideration of the landing constraint guarantee that the optimum wing area allow landing on the available runway.

3.3 Optimization

Optimization method. The non-linear parameter optimization problem which is consequence of the presented formulation is solved by the subroutine CONMIN (Vanderplaats, 1973). In order to determine the parameters that imply an optimal non-linear performance index, it is used a method that searches the smaller gradient of each variable with respect to the performance index, also taking into account some non-linear constraints which are functions of these variables. The basic algorithm is the Method of Feasible Directions, which can be found in Himmelblau (1972). The termination criterion is the satisfaction of a given tolerance in repeated iteractions, that is, the optimum point is found when the objective function has values within a given tolerance in some repeated iterations.

Objective functions. An optimization analysis involve a function whose minimum or maximum value is searched. There are three objective functions (also called performance indexes) used in the procedure:

1. minimization of the block fuel which is the fuel spent in the flight from the origin to the destination airports, without considering the reserves;

2. minimization of the block time which is the time spent in the flight from the origin to the destination airports;

3. minimization of a linear combination of the block fuel (BF) and the block time (BT), IP = α.BF + β.BT, where α is the price of the fuel kilogram, taken to be α = (US$ 0.336)/ kg and β is the cost of the flight minute, taken to be β = (US$18.48)/min. In fact, β is the D.O.C. for a given route less the fuel consumption cost. It is assumed the hypothesis that the aircraft price do not change as the performance is optimized, and then the β value is not also changed.

The consideration of the time cost may allow the choice for a faster flight even at the expense of a greater fuel consumption.

4. ANALYSIS OF THE RESULTS

All decision variables are subject to lower and upper limits, as shown in Table 1.

Table 1: Side constraints of the decision variables

VARIABLE	LOWER BOUND	UPPER BOUND
Area (m^2)	40	60
Aspect Ratio	7	10
Taper ratio	0.2	0.4
Thickness over chord ratio	0.110	0.160
Sweepback angle (deg)	18.	25.
Cruise thrust factor	0.8	1.2

The evidence that the optimization procedure finds the global optima is shown in Figure 1. It can be seen that different initial designs lead to approximately equal values of the objective function. All three types of objective function were analysed in Diniz (1998), but only the general case is shown here.

The simultaneous optimization leads to smaller value of the *IP* (peformance index or objective function) than the design parameters optimization, that is, the consideration of some trajectory parameters in the optimization process leads to better optima.

Inspecting the effects of the takeoff field-length constraint, it is observed that the decrease of the available runway make the wing area to increase (Figure 2).

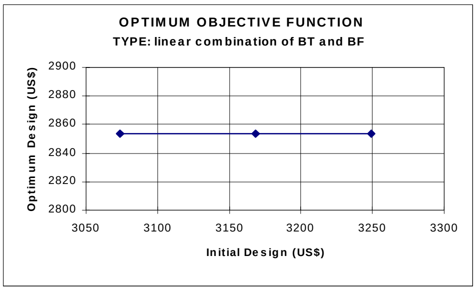


Figure 1 - Global Optima

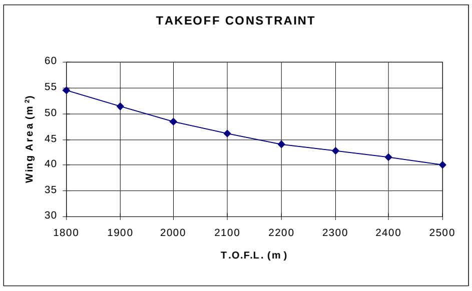
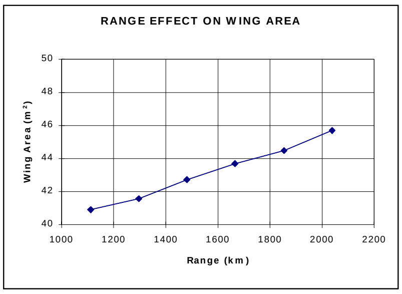


Figure 2 - Takeoff constraint

When considering different missions with increasing range, Figure 3 shows that the wing area also increases. The increase in wing area leads to smaller C_L and the resultant induced drag reduction exceeds the increase in the profile drag. The sweep angle decrease as the mission range is increased, which may be explained by the fact that the maximum cruise speed decrease as the aircraft weight increases and then for greater ranges the average speed should be smaller. This allows a smaller optimum sweep angle, as shown in Figure 4.



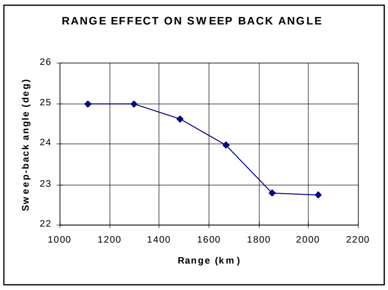


Figure 3 - Range x S **Figure 4 - Range x $\Lambda_{1/4}$**

5. CONCLUSIONS

The objetive of this work was to apply and test a procedure which helps to improve the design of a jet transport aircraft conceived for subsonic flight. This improvement was done through an optimization procedure which determines a set of optimum design parameters that corresponds to a minimum value of an objective function that is the linear combination of the block fuel and block time .

The results presented in section 4 showed sensible reduction of the optimum objective functions with respect to those of the initial designs. Moreover, these results show that the equivalent wing and the engine size optimization allow this sensible improvement .

The analytic models fidelity showed to be adequate for the accomplishment of the proposed objective. The performance models validation is shown in Diniz (1998). The validation of the subsonic drag polar prediction method is presented by the authors in Torenbeek et al. (1983). The compressibility effect model is theoretical but conservative. The weight correction methods require no validation due to their statistical nature .

The types of the objective functions used in the procedure imply the determination of strictly technical optimum configurations. In this work, the design and manufacturing costs are not taken into account, and they would allow a more rigorous determination of the optimum designs, which, in the present solution, are better technical choices than the initial designs but not necessarily the best economical ones .

The gradient optimization method showed satisfactory results even considering its well known characteristic of sttoping the search in a point of local optimum. The optimization domain is well behaved with respect to the decision variables. However, there is no way of guaranteeing the global optima determination. Anyway, even not beeing global optima, the sub-optimum points presented considerable objective function reduction.

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