POWERPLANT AS DESIGN VARIABLE FOR MULTI-DISCIPLINARY DESIGN AND OPTIMIZATION OF TRANSPORT AIRPLANES

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Abstract. *Successful application of multi-disciplinary design and optimization (MDO) methodology in the aeronautical industry only started in the early 2000s. This was due to the extreme complexity and the low automation on all levels of aircraft design. Thanks to the development of high-performance computer systems with lower cost and high-performance as that provided by Linux clusters, the real benefit of multi-disciplinary design and optimization frameworks could now finally be accounted to aircraft design. Nevertheless, even nowadays, usual MDO approaches still just consider the airplane under the manufacturer's point of view, minimizing production cost and maximizing performance. The airline requirements are taken into account in the MDO methodology by minimizing the direct operating costs in general form only. A different approach for aircraft optimal conceptual design considering airline needs and employing an advanced engine model is described and analyzed in this paper. In this context, a methodology for the design of an airliner better suitable for an existing small airline network was developed and two test cases were run. The MATLAB® suite was employed to code and house the present methodology. Another additional feature in the present work is the incorporation of wing and engine position in the configuration, number of engines, and tail configuration as design variables. The remained design variables are related to the wing and horizontal tail geometry. In addition, a genetic algorithm was developed, tested and validated to perform the optimization task.*

Keywords*: aircraft design, optimization, turbofan engine*

Symbols and Abbreviations

MDO	Multi-disciplinary design and optimization
DOC	Direct operating cost
IOC	Indirect operating cost
S_W	Wing area
AR_W	Wing aspect ratio
λ_W	Wing taper ratio
Λ_W	Wing sweepback angle
L_{HT}	Horizontal tail arm
λ_W	Taper ratio of the horizontal tail
V_{HT}	Volume coefficient of the horizontal tail
MTOW	Maximum take-off weight
OEW	Operating empty weight
W_{FUEL}	Weight of internal fuel
W	Weight
SFC	Thrust specific fuel consumption
GIG	Rio de Janeiro International Airport
CGH	São Paulo Central Airport (Congonhas)
SSA	International Airport of Salvador
F_D	Engine fan diameter
BPR	Engine by-pass ratio
PRC	Compression ratio provided by the compressor assembly
M	Number of Mach
MMO	Maximum Operating Mach Number
OASPL	Overall Sound Pressure Level

I. INTRODUCTION

An important factor to the calculation of balance sheet of airliners is the cost per seat mile. Deducing the cost from the revenue, the operator is then able to calculate the payback of his investment. Usually the operator's cost is split into the direct operating cost (DOC) and indirect operating cost (IOC), the latter contained items such as ticket's printing costs. The traditional airlines cost structure is under heavy fire because of increasing oil prices and intense competition from low-cost/low-fare companies, which are well represented in the way they make business by SouthWest Airlines from USA, and EasyJet, and Ryanair from Great Britain. In order to remain competitive all airlines are changing their operating procedures as well as ordering airplanes that feature the lowest operating cost possible. In this context multi-disciplinary design and optimization (MDO) is a useful tool to deliver what airlines are asking for.

The roots of MDO are found in structural optimization. This is not without good reason as in nearly all engineering systems there is a structure to which other subsystems attach, and much of the subsystem interaction involve the structure as a conduit. In the field of optimization the Nonlinear Programming (NLP) formalism was first transplanted to structural optimization practice in Schmit's seminal work on a simple three bar truss.

MDO requires a high degree of integration among the disciplines under consideration. Typical disciplines are aerodynamic, structure, loads, aeroelasticity, flight mechanics, and costs, among many others. MDO has been at the spotlight of the industry for the last 15 years but its heavy application in the aeronautical industry only started in the last five years. This can be explained by the high complexity of the aeronautical design, and the low automation at all levels of design. Since the early sixties, multi-disciplinary optimization has been a motivation of study for a great number of researches. However, only with the advent of high-speed computing in the eighties, its true benefit could be useful to the aeronautical industry. Consistent with the computer technology limitations of the early times, the first MDO efforts developed at *two levels*. At one level, structural, aerodynamics, and aircraft performance analysis codes, all processing mathematical models whose fidelity could be categorized as *medium* by the current standards, were nested in a single optimization loop operating on a very limited (a dozen or less) design variables to optimize a system level objective such as flight range. At another level, the number of disciplines was increased to a more complete set typical for a conceptual design stage at the price of lowered fidelity to gain a fairly realistic representation of the design process at that stage [Barthelemy and Haftka, 1993].

II. METHODOLOGY

II.1 Airplane

Traditionally the MDO methodology for aircraft design has focused on the fuel consumption and weight reduction. However, only few MDO frameworks incorporated concepts for aircraft design for taking into account the airline point of view and a program to model a turbofan engine adequately. The present methodology finds out the best suited airliner to an existing small airline network. The workflow of the present methodology is illustrated in **Fig. 1**. A genetic algorithm was chosen as optimizer due to its robustness and ability in dealing with global minima or maxima in poor-known search space. Airplanes are described by its chromosomes containing information such wing area, wing aspect ratio, vertical wing placement relative to fuselage and so on (please, address **Table I** and **II**).

Isikveren (2002) developed algorithms to optimize range and fuel consumption. However due to the high degree of coupling among the disciplines involved, semi-empirical formulations were employed most of time in his work. In Cavalcanti et al (2006), multi-objective optimization of wing planform was carried out by the minimization of the block time and block fuel for a given mission. In this study various method of optimization were employed and analyzed. Allan (2006) applied the same methodology as by Cavalcanti did but only for the optimization of wing airfoil geometries. In Taylor (2006), the aircraft and the network were optimized for a selected few cities with a fixed demand. The solutions demonstrated improvement in cost over the solutions obtained by traditional aircraft optimization.

The aircraft manufacturers highlight DOC as a selling point. This scenario stimulated the development of a framework able to design an aircraft that comply with the need of designing airliners with lower operational costs. In this context, the content of this paper is the description of a multi-disciplinary optimization suited to conceptual design of aircraft, which enables to lower direct operational cost for a specific network and requirement. The design methodology embraces disciplines such as weight, aerodynamic, flight mechanics, performance and costs. **Figure 1** shows the workflow for the calculation of DOC and other aircraft characteristics. DOC calculation is performed using Roskam's Methodology [Roskam, 1990]. This procedure is employed for all individuals (airplanes) generated by the genetic algorithm.

MTOW and OEW calculations are performed iteratively according to Roskam's Class II methodology (1999).

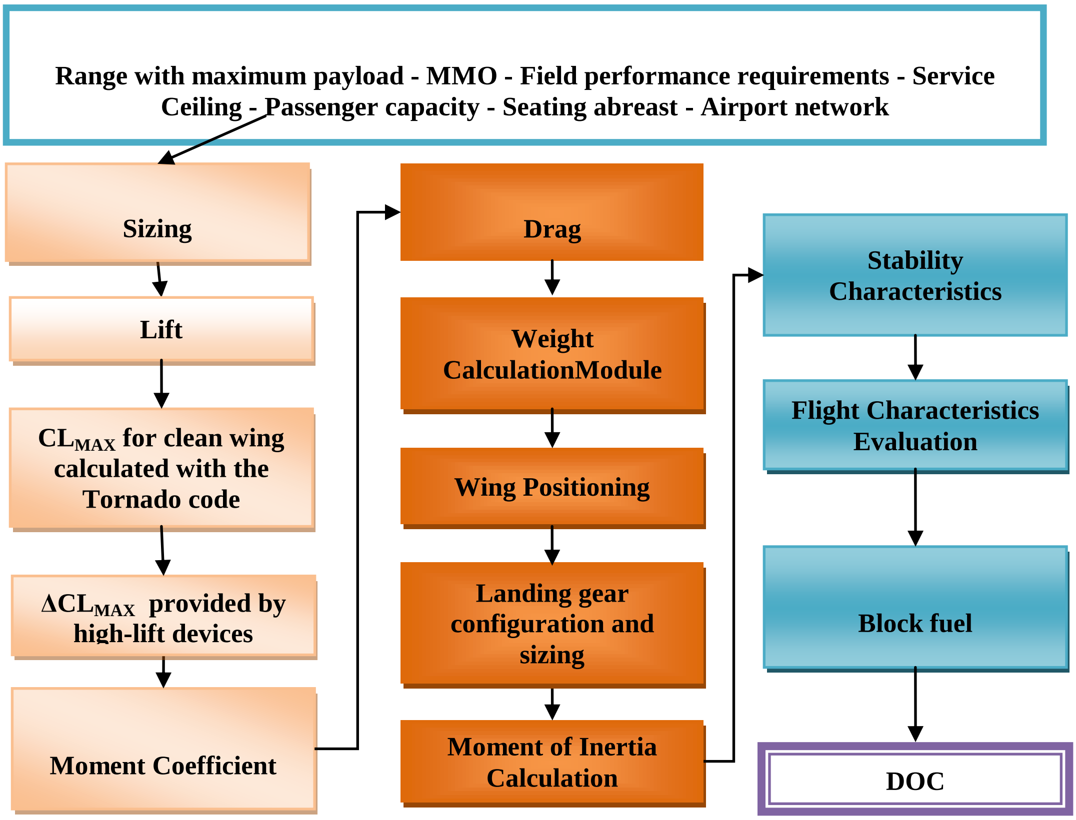

Figure 1 – Workflow for DOC calculation of given airplane configuration.

An in-house genetic algorithm code was developed and employed in the optimization framework. Genetic algorithms are a heuristic optimization approach, which is based on the theory of natural selection developed by Charles Darwin. Genetic algorithms are now of widespread use in optimization problems [Benson, 1995], [Cavalcanti et al, 2006], [van Keule and Haftka, 1993]. The genetic characteristics of an individual are stored in a chromosome. In this work, the related chromosome registers configuration variables such as wing aspect ratio, wing taper ratio, and wing area. The genetic algorithm that was employed in the present simulations is able to handle both discrete and continuous variables.

Table I - Part of chromosome that is related to aircraft geometry

S_W	AR_W	λ_W	Λ_W	l_{HT}	λ_{HT}	V_{HT}	F_D	PRC	BPR

Table II - Part of chromosome that is linked to aircraft topology.

Engine location	Number of Engines	seating Abreast	Wing Vertical Position	Tail Configuration	Number of Corridors

The initial population is randomically created from a data bank containing characteristics of existing aircraft. The best individuals in a given generation are selected according to the degree of compliance to the objective function and constraints. In the present work the objective function was set to be the direct operating cost. Aircraft that possess lower DOC are selected to compose a new population. In order to keep the diversity in the population after the reproduction some individuals suffer mutation. This process is repeated, i.e. some iterations called generations are run, until some monitored parameters fulfill some prescribed criteria. The design variables and their related lower and upper boundaries are listed in Tables **III** and **IV**, the latter containing that variables related to the airplane topology.

Table III – Design variables.

Symbol	Variable	Lower Boundary	Upper boundary
S_W	Wing planform area	50	250
AR_W	Wing aspect ratio	6	11
λ_W	Wing taper ratio	0.25	0.5
Λ_W	Wing quarter-chord sweep	20°	35°
λ_{HT}	Horizontal tail taper ratio	0.3	0.6
l_{HT}	Tail arm	-	-
V_{HT}	Volume coefficient of the horizontal tail	0.85	-
-	Seating Abreast	4	6
-	Number of engines	2	4
F_D	Fan diameter (m)	1	2
CPR	Pressure ratio provided by the compressor assembly	12	28
BPR	Engine by-pass ratio	5	7

Table IV – Discrete design variables.

Variable	Assignment
Engine location	1 - Two underwing engines 2 - Two engines integrated into the fuselage 3 - Three engines located at rear fuselage 4 - Four underwing engines
Vertical wing position	1 - Low 2 - High
Tail configuration	1 - Conventional 2 - T-Tail
Number of corridors	1 to 2

The present methodology contemplates the integration of some disciplines for modeling the airliner: interior layout; cross section calculation; weight estimation; aerodynamic; performance; stability and control; and operating cost.

For the DOC calculation it is necessary to define the network that will be analyzed. For the present work, a small network of a Brazilian's airliner operator and it is showed in the **Table V**. The direct operational cost (DOC) is calculated following the methodology presented on Roskam [Roskam, 1990].

Table V – Airline network.

DEPARTURE	ARRIVAL	Stage Length [nm]	Runway length [m]
Rio de Janeiro (GIG)	CGH	207	3180 (GIG)
Congonhas (CGH)	GIG	207	1435 (CGH)
GIG	Salvador (SSA)	653	1520 (SSA)

II.2 Turbofan Engine Modeling

An engine calculation module was developed in the present work in MATLAB® language. The theory and methodology of this module were based on that made available by on NASA's Website [Benson, 1995]. This module outputs the fuel consumption and thrust provided that the altitude, Mach number, ISA condition, and throttle are properly inputted. The classical one-dimensional thermodynamics is the basis of the calculation procedure. Two steps were built in the procedure. The first is for design and the second is for analysis. In the design mode all engine characteristics are raised at a given design point. In the analysis step it is then possible to calculate the engine thrust and fuel flow rate from the geometric characteristics obtained in the first step. **Figure 3** illustrated the different components of a typical turbofan. The numbering contained in that figure will be employed by the equations that will follow in this section.

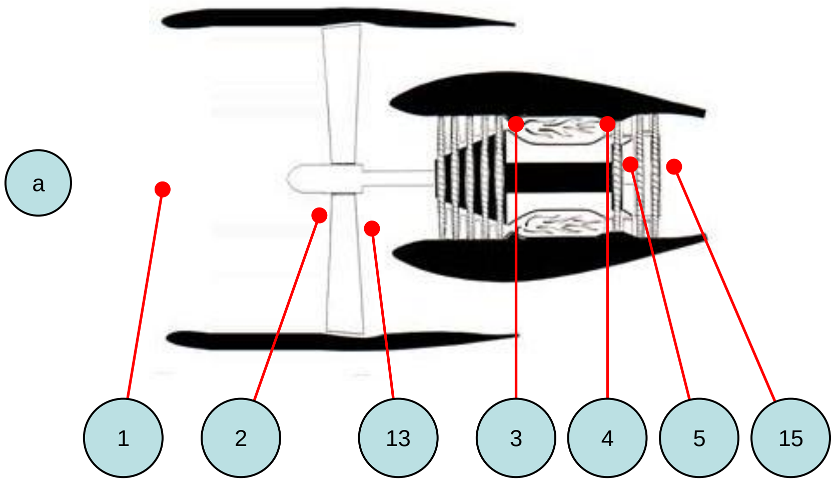

Fig. 2 – Engine parts and reference points for the calculation procedure.

In the design step, the following condition is adopted:

Mach number = 0; altitude = 0 m; and throttle setting = 100 %.

The analysis process starts with the total pressure and total temperature calculation in the engine inlet.

$$\frac{T_{01}}{T_a} = 1 + \frac{(\gamma - 1)}{2} \cdot M^2 \quad (9) \qquad\qquad \frac{P_{01}}{P_a} = \left(\frac{T_{01}}{T_a}\right)^{\frac{\gamma}{\gamma-1}} \quad (1)$$

The condition at the engine inlet is described by P_{01} and T_{01}. P_a and T_a are obtained by a atmosphere calculation routine. The next thing to do is the evaluation of the airflow path in the inlet. There the total temperature remains unchanged and the total pressure is obtained by using a recovery pressure coefficient. Thus

$$T_{02} = T_{01} \text{ and } P_{02} = \alpha \cdot P_{01} \quad (2)$$

After the airflow passed through the engine fan the following relation can be written

$$T_{013} - T_{02} = \frac{T_{02}}{\eta_{13}} \cdot \left[\left(\frac{P_{013}}{P_{02}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \quad (3) \qquad\qquad \frac{P_{013}}{P_{02}} = \text{Prfan} \quad (4)$$

Prfan is the pressure ratio provided by the fan and η_{13} is the fan efficiency. In the present work *Prfan* and η_{13} were kept constant assuming values of 2 and 0.92, respectively. Similarly for the airflow through the compressor, one can write

$$T_{03} - T_{02} = \frac{T_{02}}{\eta_3} \cdot \left[\left(\frac{P_{03}}{P_{02}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \quad (5) \qquad\qquad \frac{P_{03}}{P_{02}} = prc \quad (6)$$

Prc is the compression ratio due to the compressor work. This is a design variable in the MDO process of the present work. The combustion chamber exhaust temperature is obtained by a direct relationship between the throttle setting (ψ) and the maximum admissible temperature at this stage, which was assumed to be 1388 K. Therefore, the following equations can be derived

$$T_{04} = \psi \cdot 1388 \quad (7) \qquad \frac{P_{04}}{P_{03}} = prcomb \quad (8)$$

Indeed, the temperature at turbine inlet shall be considered as varying with altitude for a more realistic modeling of the engine. *Prcomb* is the total pressure drop in the combustion chamber and was assumed to be 0.99.

In the present work the engine configuration is composed of two turbines. Two coaxial shafts link the turbines to their respective compressors.

$$\Delta h_{lower\_turb} = -\Delta h_{fan} = -Cp \cdot (T_{013} - T_{02}) \quad (9)$$

$$\Delta h_{hight\_turb} = -\Delta h_{compressor} = -Cp \cdot (T_{03} - T_{013}) \quad (10)$$

By the enthalpy variation caused by the turbines the temperature variation can then be calculated

$$T_{05} = T_{04} + \frac{\Delta h_{lower\_turb}}{Cp} \quad (11) \qquad T_{015} = T_{05} + \frac{\Delta h_{hight\_turb}}{Cp} \quad (12)$$

From the calculated temperatures the pressure ratio for the turbines can be obtained

$$\frac{P_{05}}{P_{04}} = \left[1 + \frac{1}{\eta_{turb}} \cdot \left(\frac{T_{05}}{T_{04}} - 1\right)\right]^{\frac{\gamma}{\gamma-1}} \quad (13) \qquad \frac{P_{015}}{P_{05}} = \left[1 + \frac{1}{\eta_{turb}} \cdot \left(\frac{T_{015}}{T_{05}} - 1\right)\right]^{\frac{\gamma}{\gamma-1}} \quad (14)$$

Thus, the thermodynamic properties along the engine could be evaluated for the design point. They will be useful for the next step.
The area can of the engine core be calculated by using the fan area and the by-pass ratio. Next is the calculation of the S_{8_RATIO} , the ratio between the turbine area to the core one.

$$S_{8\_RATIO} = \min(0.75 \cdot \frac{\sqrt{ETR}}{EPR};1.0) \quad (15)$$

ETR and EPR are the engine temperature and pressure ratio, respectively. The area of the engine nozzle is obtained as follow

$$S_8 = S_{8\_RATIO} \cdot S_{NUCLEO} \quad (16)$$

The area of the low- and high-pressure turbines can be then calculated

$$S_{4\_HIGHT} = S_8 \cdot \frac{(P_{05}/P_{04}) \cdot (P_{015}/P_{05})}{\sqrt{(T_{05}/T_{04}) \cdot (T_{015}/T_{05})}} \quad (17)$$

$$S_{4\_LOW} = S_8 \cdot \frac{P_{015}/P_{05}}{\sqrt{T_{015}/T_{05}}} \quad (18)$$

After the engine is calculated its thrust and fuel flow can be obtained for different conditions. For this, the points 1 and 2 are recalculated according to the new conditions. The turbine inlet temperature is calculated using the same procedure as described in step one. It is assumed that the turbine Mach number is one. Thus, it can be derived

$$\frac{S_8}{S_{4\_LOW}} - \sqrt{\frac{T_{05}}{T_{04}}} \cdot \left[1 + \frac{1}{\eta_{turb}}\left(\frac{T_{05}}{T_{04}} - 1\right)\right]^{-\frac{\gamma}{\gamma-1}} = 0 \quad (19)$$

$$\frac{S_{4\_LOW}}{S_{4\_HIGHT}} - \sqrt{\frac{T_{015}}{T_{05}}} \cdot \left[1 + \frac{1}{\eta_{turb}}\left(\frac{T_{015}}{T_{05}} - 1\right)\right]^{-\frac{\gamma}{\gamma-1}} = 0 \quad (20)$$

The temperature ratios are calculated by solving **Equations 19** and **20**. Thus, by using the temperature T_{04}, which was previously calculated from the throttle settings, T_{015} and T_{05} can be obtained. Pressure ratios are calculated from these temperature ratios and also from the already known the efficiency factor (**Eqs. 21** and **22**).

$$\frac{P_{05}}{P_{04}} = \left[1 + \frac{1}{\eta_{turb}} \cdot \left(\frac{T_{05}}{T_{04}} - 1\right)\right]^{\frac{\gamma}{\gamma-1}} \tag{21}$$

$$\frac{P_{015}}{P_{05}} = \left[1 + \frac{1}{\eta_{turb}} \cdot \left(\frac{T_{015}}{T_{05}} - 1\right)\right]^{\frac{\gamma}{\gamma-1}} \tag{22}$$

By using the temperature variation at the low- and high pressure turbines the enthalpy variation can be calculated. If the turbine enthalpy variation is known the enthalpy at the compressor and fan assemblies can be also obtained.

$$Cp \cdot (T_{015} - T_{05}) = \Delta h_{lower\_turb} = -\Delta h_{fan} = -Cp \cdot (T_{013} - T_{02}) \tag{23}$$

$$Cp \cdot (T_{05} - T_{04}) = \Delta h_{hight\_turb} = -\Delta h_{compressor} = -Cp \cdot (T_{03} - T_{013}) \tag{24}$$

The fan inlet temperature, T_{02}, is already known and it is used in the calculation of T_{013} and T_{03}.
Again the pressure ratios can be obtained in the same way that was done for the turbine. Provide all pressure ratios are at this time known, one can calculate the pressures along the engine.

The engine thrust can be calculated by using the mass flow rate and the nozzle area. Thus

$$\dot{m} = S_8 \cdot P_{08} \cdot \sqrt{\frac{\gamma}{T_{08} \cdot R}} \cdot \left(1 + \frac{1}{2}(\gamma - 1)\right)^{-\frac{\gamma+1}{2(\gamma-1)}} \tag{25}$$

The exhaust hot gas speed can be calculated by

$$U_8 = \left[\frac{2R}{\gamma - 1} \cdot \gamma \cdot T_{08} \cdot \left(1 - \frac{P_1}{P_{08}}^{\frac{\gamma-1}{\gamma}}\right)\right]^{1/2} \tag{26}$$

The speed of the cold air coming from the engine fan is calculated by

$$U_{13} = \left[\frac{2R}{\gamma - 1} \cdot \gamma \cdot T_{013} \cdot \left(1 - \frac{P_1}{P_{013}}^{\frac{\gamma-1}{\gamma}}\right)\right]^{1/2} \tag{27}$$

The specific thrust is obtained using the following relation

$$\frac{T}{\dot{m}} = (U_8 - U_0) + bpr \cdot (U_{13} - U_0) + \frac{(P_\alpha - P_1) \cdot S_8}{\dot{m}} + \frac{(P_\beta - P_1) \cdot bpr \cdot S_{núcleo}}{\dot{m}} \tag{28}$$

P_α and P_β are

$$P_\alpha = \begin{cases} P_1 \leftrightarrow (P_{08}/P_1) \leq 1.893 \\ 0.52828 \cdot P_{08} \leftrightarrow (P_{08}/P_1) > 1.893 \end{cases}$$

$$P_\beta = \begin{cases} P_1 \leftrightarrow (P_{013}/P_1) \leq 1.893 \\ 0.52828 \cdot P_{013} \leftrightarrow (P_{013}/P_1) > 1.893 \end{cases}$$

The mass ratio between the fuel mass pumped into the combustion chamber and the airflow is given by

$$f = \frac{T_{04} - T_{03}}{\frac{\eta_{camara} \cdot PC}{Cp} - T_{04}} \tag{29}$$

The fuel flow and the thrust are calculated by using **Eqs. (27)** and **(28)**

$$FF = \dot{m} \cdot f \tag{30}$$

$$T = \dot{m} \cdot \left(\frac{T}{\dot{m}} \right) \tag{31}$$

III RESULTS

III.1 Turbofan Engine Model

A set of graphs were generated to properly evaluate if the methodology for modeling a turbofan engine was adequately implemented. In **Figure 3** it can be easily observed that the thrust increases considerably when the by-pass ratio is increased.

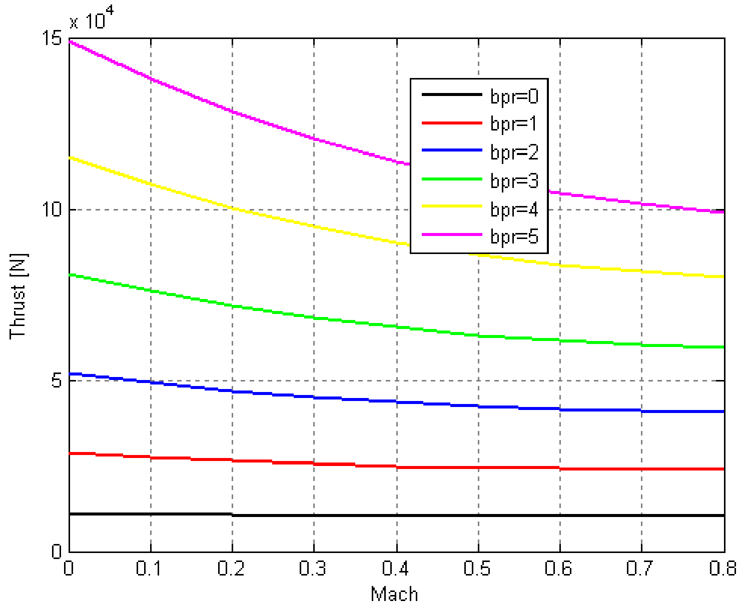

Figure 3 – Thrust dependence on Mach number when varying the by-pass ratio.

However, by increasing the Mach number the following pattern can be observed: the higher the by-pass ratio, the higher the loss in thrust. This can more easily seen in **Fig. 4**, where the thrust ratio number is plotted against Mach number**.** The graph at left in **Figure 4** shows typical trends of thrust vs. speed for turbojets and turbofans with varying bypass ratio at sea level and all curves in both graphs are in very good agreement.

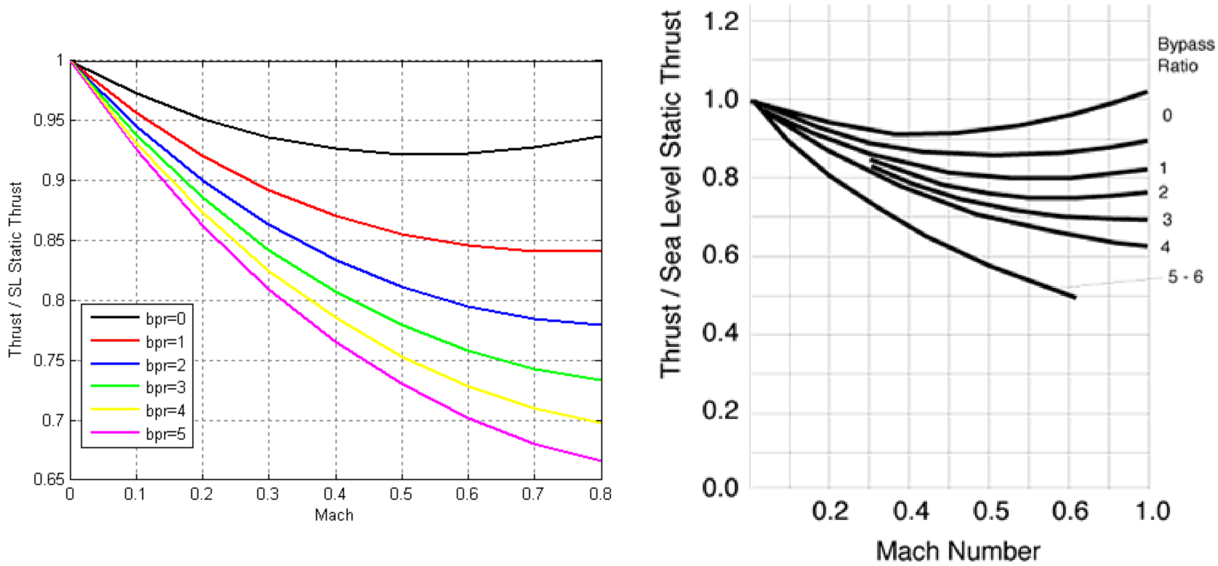

Figure 4 - Thrust ratio vs. Mach number at sea level for several by-pass ratios.

The engine module results were compared to the ones from the NASA EngineSim routine. **Table VI** summarizes the comparison between the codes. **Table VII** shows a direct comparison of the prediction capability of the engine module by confronting its results with actual engine data. The engine weight was calculated a methodology proposed by Raymer (1989). In addition, actual engine performance differs from the basic engine data in a number of other ways. The air bled from the compressor for air conditioning, the power extracted for hydraulic pumps and alternators, and inlet and exhaust duct losses reduce engine thrust. The exact amount depends, of course, on the requirements of the accessories, the engine size, and the inlet and duct design, but reasonable estimates for conventional inlets are

1) Thrust is reduced by 3.5% below engine specification levels 2) Specific fuel consumption is increased by 2%

During the take-off the air conditioning bleed is often shut-off automatically to avoid the thrust loss. The remaining thrust loss is about 1%. If a long or curved (S-bend) inlet is involved as in center engine installations, an additional thrust loss of 3% and a specific fuel consumption increase of 1-1/2% may be assumed. This additional loss applies only to the affected engine

Table VI – NASA's EngimeSim vs. the engine module of the present work.

PRC	Diameter	By-pass Ratio	EngineSim 1.7a / Present Calculation / Error			
10	0,5	1	21871	1160	7807	539
			21953	1163	7864	543
			0.37%	0.26%	0.72%	0.74%
10	0,5	4	14448	464	4313	215
			14504	465	4342	217
			0.39%	0.22%	0.67%	0.92%
10	1	1	87484	4642	31231	2157
			87811	4653	31455	2171
			0.37%	0.24%	0.71%	0.64%
10	1	4	57793	1856	17254	862
			58016	1861	17368	868
			0.38%	0.27%	0.66%	0.69%
10	1,5	1	196840	10445	70271	4854
			197576	10469	70773	4884
			0.37%	0.23%	0.71%	0.61%
10	1,5	4	130034	4178	38822	1941
			130537	4188	39079	1953
			0.39%	0.24%	0.66%	0.61%

Table VII – Comparison with actual engines.

		Thrust @ TO (lbf)	SFC TO	airflow TO (lb/s)	Thrust @ Cruise (lbf)	SFC @ Cruise	Weight (lbf)
Engine Manufacturer	PRC Bypass Diameter (m)	Actual Data Engine Module Error (%)					
CF6-45A	21	46500	-	1393	11250	0,63	8768
GE	4,64	46221	-	1371	11223	0,60	9412
	2,2	-0.6%	-	-1.6%	-0.2%	-5.1%	7.3%
PW2237	17	36600	-	1210	6500	0,58	7185
Pratt Whitney	5,8	35180	-	1145	7400	0,59	6488
	2,01	-3.9%	-	-5.4%	13.8%	0.8%	-9.7%
CFM56-5A1	17	25000	0,33	852	5000	0,60	4995
GE	6	25535	0,33	856	5351	0,60	4605
	1,73	2.1%	1.1%	0.5%	7.0%	0,3%	-7.8%
Tay 620	8	13850	-	410	-	0.69	3135
Rolls Royce	3,04	14035	-	355	4575	0.65	2673
	1,118	1.3%	-	-13.4%	-	-5.7%	-14.7%
BR710A1-10	15	14750	0,39	435	-	-	3520
BMW / RR	4,2	14792	0,40	428	-	-	2688
	1,23	0.3%	3.2%	-1.6%	-	-	-23.6%

III.2 Airliner model

In order to evaluate the accuracy of the present methodology for estimating airliner weight a MATLAB® code called *Aviao Aeronautico* (AA) was developed. *Aviao Aeronautico* features no routine to perform any optimization. AA utilizes improved Class II weight estimation of structural parts and aircraft systems. It is just an airplane configuration calculator. AA features the engine model described before as a built in module. When the overall characteristics of the CRJ-200LR regional jet [Jane's] are inputted into the code, a MTOW of 24,011 kg is calculated for this plane. Indeed, a very small deviation, 0.6 %, when compared to the actual MTOW of 24,154 kg (**Table VIII**). For the Fokker 100 airliner fitted with Rolls&Royce Tay 620 engines, we obtain with *Aviao Aeronautico* a MTOW of 43,097 kg, an insignificant difference when compared to the actual figure of 43,090 kg of the standard Fokker 100 [Mattos]. The calculated empty weight is 24,957 kg, again very close to the actual value of 24,593 kg. **Table VIII** resumes the validation effort undertook for some airliners of different categories.

Table VIII – Weight figures that were estimated by *Avião Aeronáutico* (AA) for some airliners still operating with airlines.

Airplane	MTOW (kg)	OEW (kg)	MTOW (AA) (kg)	OEW (AA) (kg)
Fokker 100 Standard with R&R Tay 620 engines	43,090	24,593	43,097	24,957
Bombardier CRJ-200LR	24,154	13,835	24,011	14,238
Boeing 737-300 with CFM56-3B1 engines	56,473	31,480	55,491	31,121
Boeing 757-200	115,650	62,100	115,501	59,544

III. 3 Airplane Optimization

For the optimization task were selected two 108-seater twinjet airliners equipped with engines belonging to different generations and with distinct typical-passenger maximum range. The first airliner is fitted with Rolls-Royce Tay-620 engines and has a maximum range of 1400 nm; the second one is fitted with CF34-10 turbofans and its 108-passenger range is 1800 nm. For each airplane two optimization tasks were carried out. In the first task the engine remains unchanged; in remained optimization task the engine was allowed to be designed concurrently with the airframe.

III.3.1 Test Case I

The first airliner that was optimized is closely related to the Fokker 100 with Rolls-Royce Tay 620 low-bypass turbofans. For comparison purposes the Fokker 100 with Tay 620 engines presents a MTOW of 43,090 kg. The Fokker 100 airliner has a maximum range with 107 passengers of 1290 nm, slighter lower than the requirement posted for Test Case I. In the first task the optimized airplane after 200 generations presents two underwing engines and a conventional tail configuration (**Fig. 5**). If the engine and airframe are simultaneously designed to better suit the objective of lowest DOC for the flight legs under consideration, a twinjet underwing configuration with conventional tail surfaces is also born. However, a higher by-pass engine when compared to the previous optimization task was found to be the best one (**Fig. 6**). It is normal since the Fokker 100 airliner is a relative old design and available engines presented lower by-pass ratios at that time when compared with modern ones. Variable values for the optimal airplane are listed in **Table IX**. Weight break down for the optimized airliners of Test Case I can be seen in **Table X**. **Figure 7** presents the MTOW history in the optimization process with the genetic algorithm that was employed in present simulations.

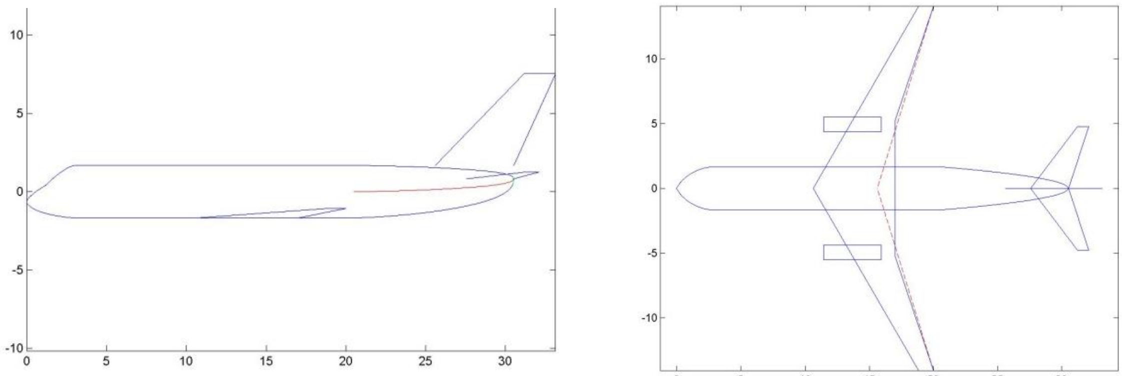

Figure 5 – Optimized airplane of Test Case I with R&R Tay 620 engines.

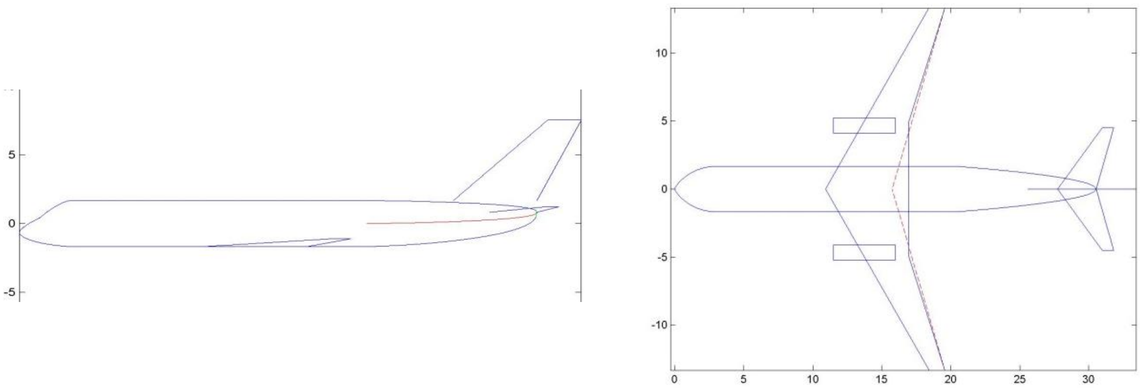

Figure 6 – Optimized airplane of Test Case I with optimal engine.

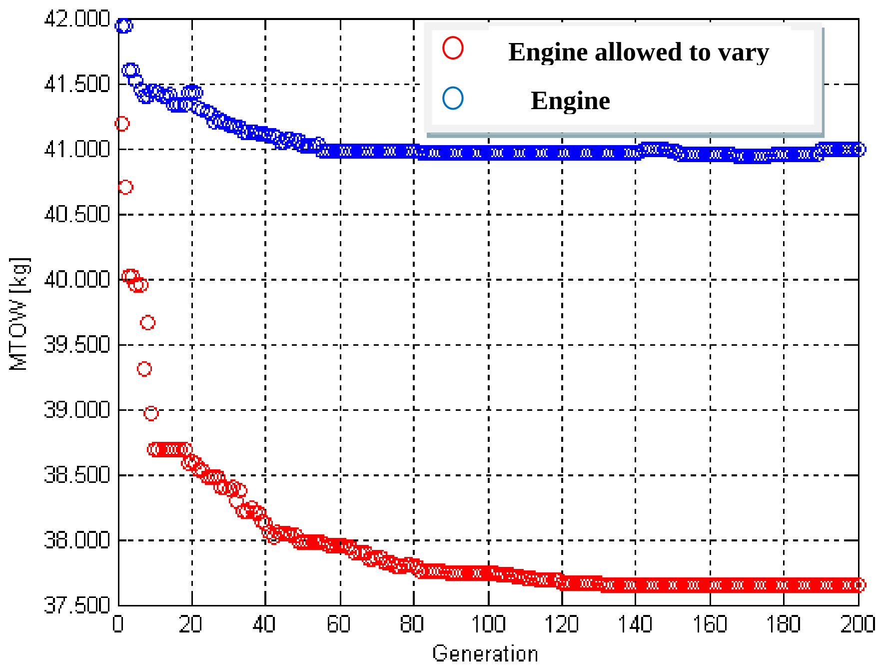

Figure 7 – MTOW history for the Test Case I optimization tasks. Convergence for the fixed engine test case are displayed in blue.

Table IX – Optimal design variables for the Test Case I airliner.

		Tay 620	Optimal Engine
Wing	Area	81.00 m²	75.30 m²
	Aspect ratio	9.79	9.41
	Taper ratio	0.25	0.25
	1/4 chord sweepback angle	27.58 º	26.59 º
Horizontal Tail	Taper ratio	0.30	0.30
	Arm	14.83 m	14.83 m
	Volume coefficient	0.90	0.90
Powerplant	Fan diameter	1.138 m	1.14 m
	Compressor pressure ratio	8	14.14
	Bypass ratio	3.04	5.23
Toplogy	Number of engines	2	2
	Engine location	Asa	Asa
	Wing vertical position	low	low
	Seating abreast	5	5
	Tail configuration	Conventional	Conventional
Direct Operating Cost	DOC	6.86 US$/nm	6.41 US$/nm

Table X – Weight figures for optimized airliners of Test Case I (all figures in kg).

	Tay 620 Engines	Optimal Engine
MTOW	40,995	37,646
MZF	33,512	31,908
OEW	21,070	19,497
Fuel	7,556	5,796
Wing	3,459	3,153
Horizontal Tail	464	425
Vertical Tail	468	468
Fuselage	4,199	4,108
Nacelle	1004	1033
Engine	1,247	948
Landing Gear	1,364	1,254

III.3.2 Test Case II

The second airliner that was optimized is closely related to the EMBRAER 190STD, which is fitted with General Electric CF34-10E turbofans. For comparison purposes the EMBRAER 190STD presents a MTOW of 47,790 kg and a 108-passenger maximum range of 1800 nm. In the first task the optimized airplane after 200 generations presents two underwing engines and a conventional tail configuration (**Fig. 8**). If the engine and airframe are simultaneously designed for better suit the objective of lowest DOC for the three five flight legs under consideration a twinjet underwing configuration with conventional tail surfaces emerges (**Fig. 9**). However, a lower by-pass engine when compared to the previous optimization task was found to be the best one (**Table XI**). Weight break down for the optimized airliners of Test Case I can be seen in **Table XII**. **Figure 10** presents the MTOW history in the optimization process with the genetic algorithm employed in the simulations. Both airliners in

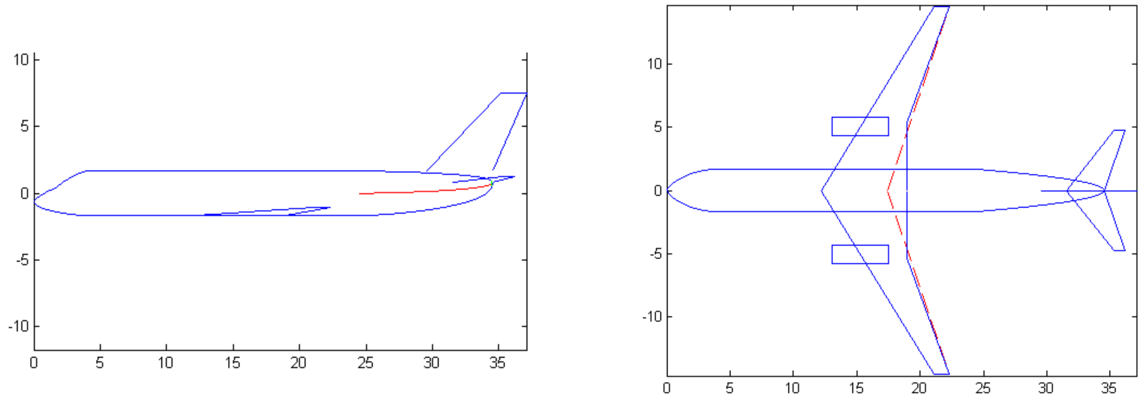

Figure 8 – Optimal 108-seater airliner with CF34-10 engines.

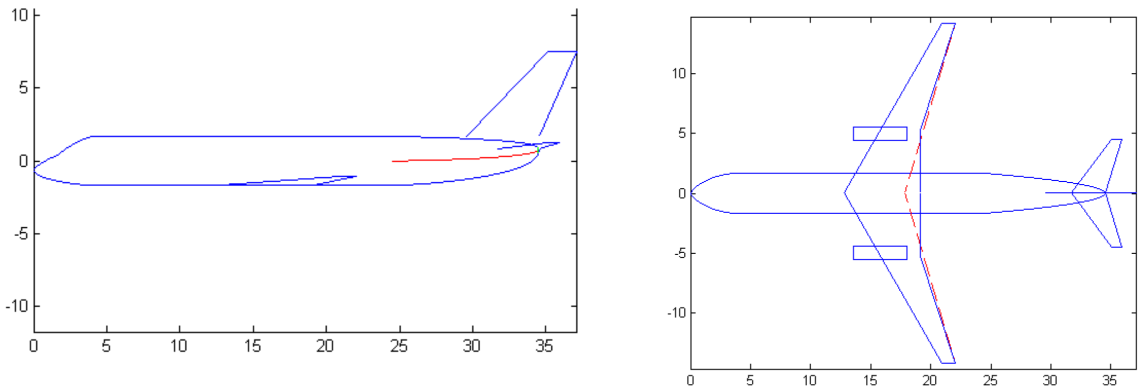

Figure 9 – Optimal airliner for the Test Case II.

Table XI – Design variables for Test Case II.

		CF34-10	Optimal Engine
Wing	Area	88.1 m²	81.9 m²
	Aspect Ratio	9.5	9.8
	Taper Ratio	0.3	0.3
	1/4 chord Sweepback Angle	28.9 º	27.0 º
Horizontal tail	Taper Ratio	0.3	0.3
	Arm	17.0 m	16.7 m
	Volume Coefficient	0.9	0.9
Powerplant	Fan Diameter	1.5 m	1.2 m
	Compressor Pressure Ratio	14.5	18.4
	By-pass Ratio	5.4	4.9
Topology	Number of Engines	2	2
	Engine Locatiion	Wing	Wing
	Wing Vertical Location	Low	Low
	Seating Abreast	5	5
	Tail Configuration	Conventional	Conventional
Cost	DOC	7.25 US$/nm	6.80 US$/nm

Table XII – Weights for Test Case II (All figures in kg).

	Motor Fixo	Optimal Engine
MTOW	44,261	41,167
MZF	36,484	34,201
OEW	24,013	21,757
Fuel	7,864	7,042
Wing	3,807	3,535
Horizontal Tail	473	406
Vertical Tail	468	468
Fuselage	5,124	5,073
Nacelle	1,228	1,065
Engine	1,534	1,069
Landing Gear	1,471	1,370

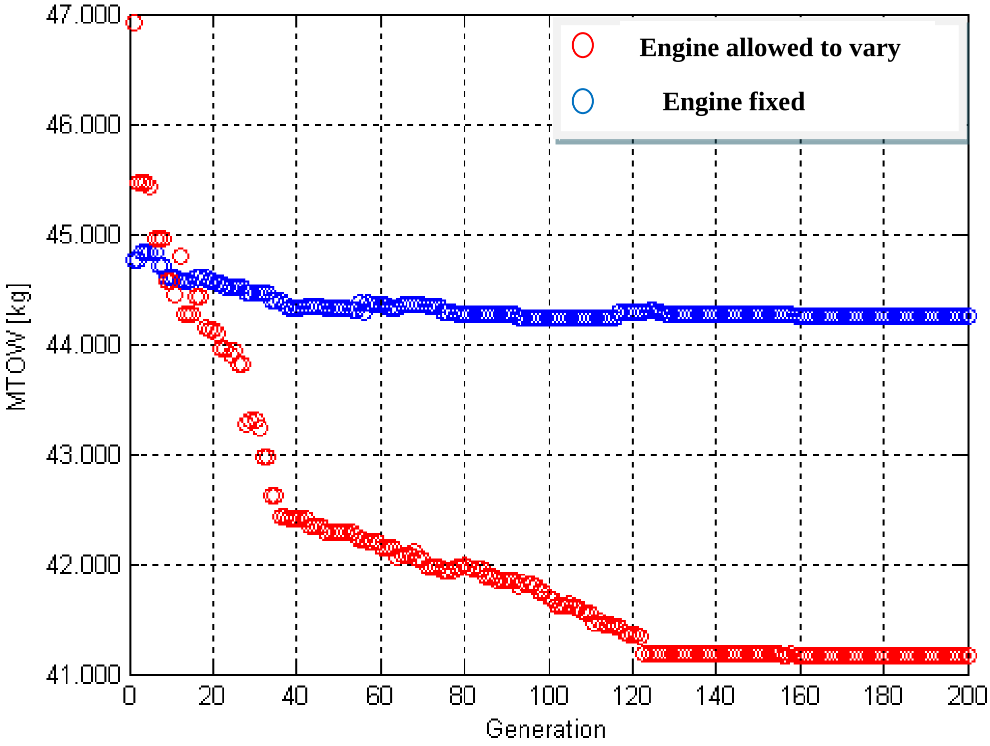

Figure 10 – MTOW history for the Test Case II optimization tasks.

III. CONCLUDING REMARKS

Due to increased competition in the aerospace sector, it is of the utmost importance to design new airliners with lower operational costs and more comfort than those from an earlier generation. The present multidisciplinary optimization framework is a powerful tool to assist the design task fulfilling such requirements.

In both test cases run in the present work, the resulting optimized configurations were fitted with engines presenting higher compression ratios than the reference ones. Concerning the powerplant installation, this outcome indicates that existing engines could incorporate additional compressor stages in order to achieve the required compression ratios. Thus, the optimization task provides a solid basis for the airframe manufacturer to deal with the engine supplier. For example, the airframe manufacturer could ask the supplier for modification of an existing engine rather than developing a new one. This would represent significantly lower development risks because such engine will probably be available to the airframe manufacturer on time.

Some improvements to the present methodology are currently underway. The incorporation of airframe and engine noise routines will enable the design of airplanes under specified noise constraints. A better interior layout as well as wings composed of any other airfoil instead of the NACA ones are also being implemented.

Further work will be focused on the integrated design of the aircraft and the network. In other words, the aircraft fleet and the network will be designed simultaneously [Taylor, 2006]. This will provide the aircraft manufacturers with a powerful tool for understanding the real customer needs and incorporate them into the design.

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