MDO Assignment Group 66: Fokker 100

by

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1 Parameterization and specification of the MDO problem

1.1. Aircraft data

In order to extract the relevant data for the Fokker aircraft and to obtain an accurate description of the initial values of the variables, a planform of the aircraft (see Figure 1.1) is put into a CAD program[1] to extract the initial parameters of the aircraft (see Figure 1.2).

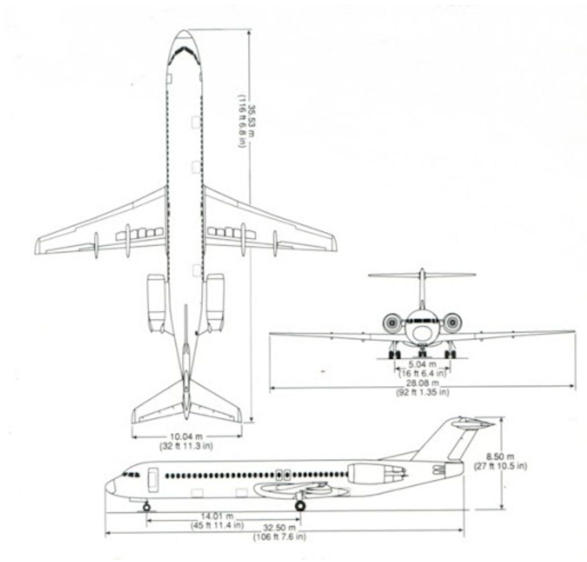

Figure 1.1: Geometrical parameters for Fokker 100 [1]

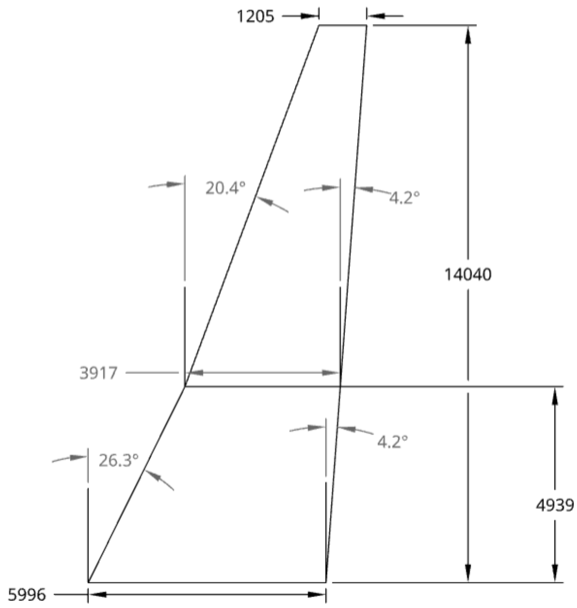

Figure 1.2: Extracted parameters of the wing in $[mm]$ from the Fokker diagrams

A nomenclature table is provided (see: Table 1.1). Each variable can be one of four types: *reference*, which is a value used in the reference aircraft but not used in this optimization; *constant*, which does not change throughout the assignment; *variable*, which can change but is not part of the design vector; and *design*, which represents variables that belong to the design vector and need to be optimized. These design vectors are explained in more detail in section 1.2. They also require bounds, which are discussed in section 1.3. All the initial values used are justified with a source.

Table 1.1: Nomenclature Table

Variable	Symbol	Type	Value	Bounds	Units
General Wing Planform					
Wing Area	S	Reference	93.50 [2]	–	m^2
Wing Semi Span	s	Reference	14.04 [2]	–	m
Mean Aerodynamic Chord	MAC	Reference	3.80 [2]	–	m
Aspect Ratio	AR	Reference	8.43 [2]	–	–
Taper Ratio	λ	Reference	0.235 [2]	–	–

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[1] `https://www.onshape.com/en/` (Accessed 28th October 2025)

Variable	Symbol	Type	Value	Bounds	Units
1/4 Chord Sweep	$\Lambda_{c/4}$	Reference	17.45 [2]	–	deg
Airfoil	–	Reference	WSTN1700 [3]	–	–
Characteristic Shape Coefficients	N_1, N_2	Constant	0.5, 1 [4]	–	–
Upper Bernstein Coefficients	A_{u_i}	Design	–	–	–
Lower Bernstein Coefficients	A_{l_i}	Design	–	–	–
Position of Leading Edge Spar	$\%_{LE}$	Design	17.5 [5]	15–20	–
Position of Trailing Edge Spar	$\%_{TE}$	Design	57.5 [5]	55–60	–
Inboard Wing Planform					
Inboard Span	s_{in}	Constant	4.939 [1]	–	m
Tank Span	s_{tank}	Variable	11.934 [1]	–	m
Inboard Leading Edge Sweep	$\Lambda_{LE_{in}}$	Design	26.3 [1]	0.01–37.5	deg
Inboard Trailing Edge Sweep	$\Lambda_{TE_{in}}$	Constant	4.2 [1]	–	deg
Inboard Root Chord	$c_{r_{in}}$	Design	5.996 [1]	5.11–6.10	m
Inboard Tip Chord	$c_{t_{in}}$	Variable	3.917 [1]	–	m
Inboard Dihedral	Γ_{in}	Constant	2.78 [1]	–	deg
Wing Incidence Angle	i_w	Constant	0 [3]	–	deg
Tank Offset	d_{tank}	Variable	–	–	m
Outboard Wing Planform					
Outboard Span	s_{out}	Design	9.101 [1]	7.48–9.88	m
Outboard Leading Edge Sweep	$\Lambda_{LE_{out}}$	Variable	20.4 [1]	–	deg
Outboard Trailing Edge Sweep	$\Lambda_{TE_{out}}$	Constant	4.2 [1]	–	deg
Outboard Tip Chord	$c_{t_{out}}$	Variable	1.205 [1]	–	m
Outboard Dihedral	Γ_{out}	Constant	2.78 [1]	–	deg
Tip Twist Angle	ϵ_t	Constant	2 [3]	–	deg
Operational Conditions					
Range	R	Variable	–	–	m
Reference Range	R_{ref}	Variable	2450196 [6]	–	m
Cruise Speed	V_{cr}	Variable	233.56 [6]	–	m/s
Cruise Mach Number	M_{cr}	Design	0.7877 [6]	0.709–0.866	–
Altitude	h_{cr}	Design	10668 [6]	9619–11735	m
Maximum Take Off Weight	$W_{TO_{max}}$	Variable	45810 [7]	–	kg
Wing Weight	W_W	Variable	–	–	kg
Wing Less Weight	W_{A-W}	Constant	–	–	kg
Fuel Weight	W_f	Variable	10731 [7]	–	kg
Design Weight	W_{des}	Variable	140087	–	kg
Reference Wing Loading	W/S_{ref}	Constant	–	–	kg
Design Lift/Weight	L_{des}, W_{des}	Variable	40087 [7]	–	kg
Material Properties					
Aluminum Elasticity Modulus	E_{al}	Constant	70×10^3 [5]	–	N/mm^2
Aluminum Tension Yield Stress	$\sigma_{y_{tens}}$	Constant	295 [5]	–	N/mm^2
Aluminum Compression Yield Stress	$\sigma_{y_{comp}}$	Constant	295 [5]	–	N/mm^2
Aluminum Density	ρ_{al}	Constant	2800 [5]	–	kg/m^3
Other Variables					
Propulsive Efficiency Factor	η	Variable	–	–	–
Specific Fuel Consumption	C_T	Constant	1.8639×10^{-4} [5]	–	N/(N·s)
Tank Volume Factor	f_{tank}	Constant	0.93 [5]	–	–

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Variable	Symbol	Type	Value	Bounds	Units
Fuel Density	ρ_f	Constant	0.81715×10^3 [5]	–	kg/m^3
Lift Coefficient	C_L	Variable	–	–	–
Drag Coefficient	C_D	Variable	–	–	–
Load Factor	n_{max}	Constant	2.5 [5]	–	–
Rib Pitch	θ_{rib}	Constant	0.5 [5]	–	m

Calculations must be performed for the initial values of these variables, including the calculations for the design lift, weight, and critical Mach number:

$$L_{des}\left(\bar{x}\right) = W_{des}\left(\bar{x}\right) = \sqrt{W_{TO_{max}}\left(\bar{x}\right) \cdot \left[W_{TO_{max}}\left(\bar{x}\right) - W_{fuel}\left(\bar{x}\right)\right]} \tag{1.1}$$

$$W_{des}\left(\bar{x}\right) = \sqrt{45810 \cdot [45810 - 10731]} = \sqrt{1606968990} \approx 40087.02 \text{ kg} \tag{1.2}$$

From the cruise altitude of 10668 m and using the ISA assumption, the cruise temperature T_{cr} is found to be 218.808 K and the density ρ_{cr} is found to be 0.37957 kg/m^3. Thus, the cruise Mach number becomes:

$$M_{cr} = \frac{V_{cr}}{a_{cr}} = \frac{V_{cr}}{\sqrt{\gamma R T_{cr}}} = 0.7877019$$

1.2. Design Vector

To define the full vector, three design vectors are first defined, which are then compiled into a full design vector, seen below:

$$\mathbf{x} = \left[M_{cr}, h_{cr}, \Lambda_{LE_{in}}, c_{r_{in}}, s_{out}, \%_{LE}, \%_{TE}, A_{u_1}, A_{u_2}, A_{u_3}, A_{u_4}, A_{u_5}, A_{l_1}, A_{l_2}, A_{l_3}, A_{l_4}, A_{l_5},\right]$$

1.2.1. Performance Design Vector

The first vector that needs to be defined is the performance design vector, which includes the cruise Mach number M_{cr} and the cruise altitude h_{cr}:

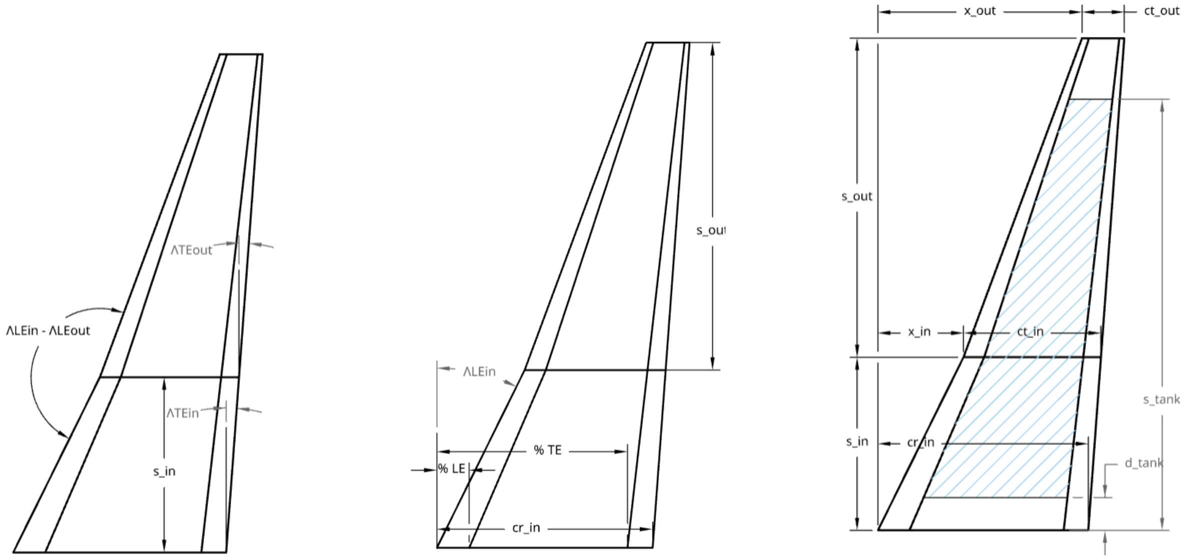
$$\mathbf{x_{per}} = \left[M_{cr}, h_{cr}\right]$$

1.2.2. Wing Planform Vector

In Figure 1.3a, the parameters of the wing that are assumed to remain constant are shown. One major assumption is that the sweep of the trailing edge of the outboard wing is kept fixed. The selected design variables are presented in Figure 1.3b, which include: the sweep of the inboard leading edge, the positions of the leading and trailing edge spars relative to the chord, the inboard chord length, and the outboard spar position. Finally, Figure 1.3c shows the geometry generated according to the requirements of the EMWET and Q3D software. In this figure, the position of the fuel tank is defined by s_{tank}, which corresponds to 85$\%$ of the total wing span. The variable d_{tank} represents the displacement of the tank, which is adjusted to accommodate the maximum fuel weight defined by the fuel-emission constraints (see section 1.4). It is positioned away from the centerline of the aircraft to reduce the bending moments of the wing.

Thus, the resulting design vector for the wing planform is the following:

$$\mathbf{x_{wp}} = \left[\Lambda_{LE_{in}}, c_{r_{in}}, s_{out}, \%_{LE}, \%_{TE},\right]$$



(a) Design Constants **(b)** Design Vector **(c)** Wing Geometry

Figure 1.3: Wing planform

1.2.3. Airfoil Vector

The Fokker F100 uses a combination of the WSTN4700 airfoil at the wing kink and the WSTN1700 at the wing root [3]. Since these specific airfoils cannot be found online, the RAE 2822 is used as a reasonable approximation[2]. This airfoil is then converted into CST curves. Following the MDO lectures [4], the values of N_1 and N_2 are chosen to be 1 and 0.5, respectively. To obtain the airfoils Bernstein coefficients, a MATLAB script is used to fit the CST representation to the RAE 2822 airfoil, as shown in Figure 1.4.

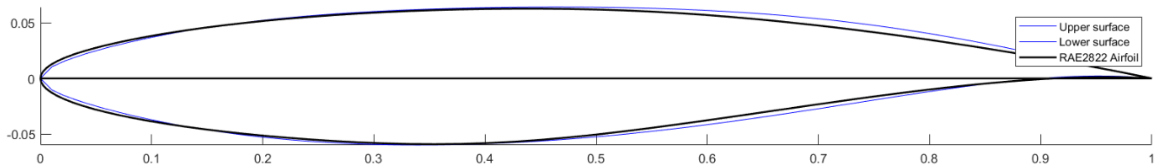


Figure 1.4: RAE 2822 Airfoil Fit

Thus, for this airfoil the design vector is the following:

$$\mathbf{x_{wp}} = \left[A_{u_1}, A_{u_2}, A_{u_3}, A_{u_4}, A_{u_5}, A_{l_1}, A_{l_2}, A_{l_3}, A_{l_4}, A_{l_5},\right]$$

1.3. Design Vector Bounds

The design variable bounds for the performance design vector are specified in the assignment [5]. For the wing planform, the inboard leading-edge sweep angle $\Lambda_{LE_{in}}$ is bounded between the minimum value required by Q3D and EMWET ($0.01°$[5]) and the maximum quarter-chord sweep angle typically reported for commercial airliners ($37.5°$ [8]), which is approximated to be the same as the leading edge.

The bounds for the inboard root chord $c_{r_{in}}$ and the outboard span s_{out} are derived from a specified aspect ratio range AR for standard airliners (between 6.6 and 9.4 [9]). Using the definition of aspect ratio:

$$AR = \frac{b^2}{S}, \tag{1.3}$$

[2] `http://airfoiltools.com/airfoil/details?airfoil=rae2822-il`

and the known wing area $S = 93.50$ m^2 and inboard span $s_{in} = 4.939$ m for the reference aircraft, the outboard semi-span is:

$$s_{out} = \frac{\sqrt{S \cdot AR}}{2} - s_{in}. \tag{1.4}$$

Assuming the wing taper ratio of the reference aircraft $\lambda = 0.235$, the root chord is related to the wing area and span by Equation 1.5.

$$c_r = \frac{2S}{b(1+\lambda)}, \tag{1.5}$$

The average value for $\%_{LE}$ and $\%_{TE}$ is based on the values given by the assignment [5], and the value is taken to be 17.5 for the leading edge bounded between 15 and 20, and 57.5 for the trailing edge bounded between 55 and 60.

For the airfoil design vector, the upper surface of a supercritical airfoil normally exhibits a convex camber line, as reported in [10]. For this reason, the corresponding upper-surface CST coefficients are positive, bounded between 50 $\%$ and 150 $\%$ of their value. For the lower surface the Bernstein coefficients are bounded between 50 $\%$ and 150 $\%$, with the exception that all the lower values are negative for the upper bound, to encompass all possible values for the airfoil design. Figure 1.5 shows a representation of the lower and upper bounds of the CST airfoil.

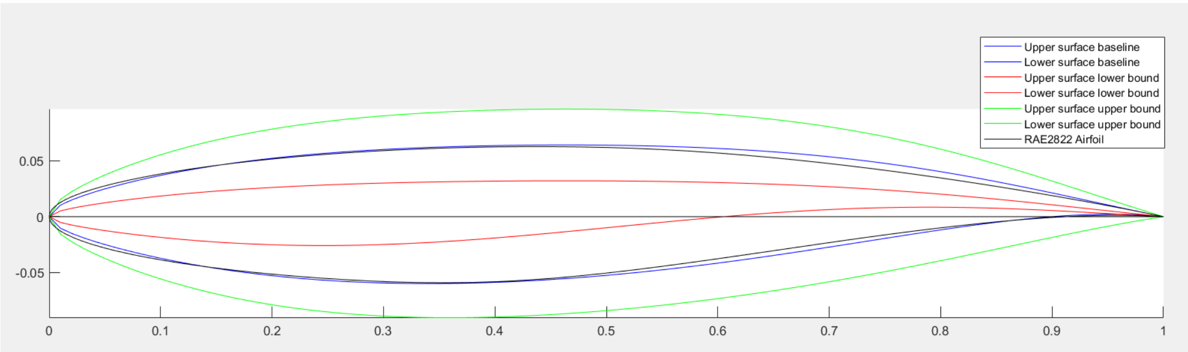


Figure 1.5: Airfoil bounds

An overview of the design initial values and bounds is provided in Table 1.2.

Table 1.2: Design Variables, Initial Values, and Bounds

Symbol	Initial Value	Lower	Upper
	Performance Variables		
M_{cr}	0.7877	0.709	0.866
h_{cr}	10668 m	9619 m	11735 m
	Wing Planform Variables		
$\Lambda_{LE_{in}}$	26.3°	0.01°	37.5°
$c_{r_{in}}$	5.996 m	5.11 m	6.10 m
s_{out}	9.101 m	7.48 m	9.88 m
$\%_{LE}$	17.5%	15%	20%
$\%_{TE}$	57.5%	55%	60%

Symbol	Initial Value	Lower	Upper
	Airfoil CST Variables		
A_{u_1}	0.1	0.05	0.15
A_{u_2}	0.2	0.10	0.30
A_{u_3}	0.1	0.05	0.15
A_{u_4}	0.3	0.15	0.45
A_{u_5}	0.2	0.10	0.30
A_{l_1}	-0.1	-0.05	-0.15
A_{l_2}	-0.2	-0.10	-0.30
A_{l_3}	-0.13	-0.075	-0.185
A_{l_4}	-0.2	-0.10	-0.30
A_{l_5}	0.1	0.05	-0.15

1.4. Objective Function

Equation 1.6 is the objective function that is optimized.

$$R = \frac{V}{C_T} \cdot \frac{L}{D} \cdot \ln\left(\frac{W_{\text{start-cr}}}{W_{\text{end-cr}}}\right) \tag{1.6}$$

The velocity of the aircraft V is given by the Mach number and the flight altitude, as indicated in Equation 1.7. M_{cr} is included in the design vector and T_{cr} is determined by the standard atmosphere model [3], seen in Equation 1.8:

$$V_{cr} = M_{cr}\sqrt{\gamma R T_{cr}} \tag{1.7}$$

$$T = T_0 - L \cdot h \tag{1.8}$$

The specific fuel consumption for the current design point C_T is given by the baseline fuel consumption $\boldsymbol{C_T}$ and the propulsive efficiency factor η:

$$C_T = \frac{\boldsymbol{C_T}}{\eta} \tag{1.9}$$

$$\eta = 1 \cdot \exp\left(-\frac{(V - V_{\text{cr,ref}})^2}{2 \cdot 70^2} - \frac{(h - h_{\text{cr,ref}})^2}{2 \cdot 2500^2}\right) \tag{1.10}$$

The lift-over-drag ratio is defined via Q3D calculations of $C_{L,wing}$ and $C_{D,wing}$, and the additional drag term $C_{D,A-W}$ is given as a constant:

$$\frac{L}{D} = \frac{C_L}{C_D} = \frac{C_{L,wing}}{C_{D,wing} + C_{D,A-W}} \tag{1.11}$$

The fraction of the start and end cruise weight is given by Equation 1.12. The maximum take-off weight is defined by Equation 1.13.

$$\frac{W_{\text{end-cr}}}{W_{\text{start-cr}}} = \frac{1 - \frac{W_{\text{fuel}}}{W_{\text{TO,max}}}}{0.938} \tag{1.12}$$

$$W_{\text{TO\_max}}(\bar{x}) = W_{A-W} + W_{\text{fuel}}(\bar{x}) + W_{\text{str,wing}}(\bar{x}) \tag{1.13}$$

The aircraft fuel weight must not exceed the value of the reference aircraft,

$$W_f \leq W_{f_{\text{ref}}}. \tag{1.14}$$

Because the available fuel volume depends on the wing geometry, the maximum fuel that can be stored, $W_{f_{\text{storage}}}$, may be larger or smaller than the reference requirement. If the wing can store more than the reference fuel weight, the fuel tank is reduced in volume so that the storable fuel matches $W_{f_{\text{ref}}}$. Let d_{needed} denote the displacement required to reduce $W_{f_{\text{storage}}}$ down to $W_{f_{\text{ref}}}$. Then the tank displacement variable is defined as:

$$d_{\text{tank}} = \begin{cases} d_{\text{needed}}, & \text{if } W_{f_{\text{storage}}} > W_{f_{\text{ref}}}, \\ 0, & \text{if } W_{f_{\text{storage}}} \leq W_{f_{\text{ref}}}. \end{cases} \tag{1.15}$$

[3] https://www.aviationhunt.com/standard-atmosphere-calculator/

1.5. Inequality constraints

Constraints on the fuel weight are given by both the maximum fuel weight, which is the fuel weight for the reference aircraft and the maximum tank capacity, thus:

$$W_f \leq W_{f_{ref}} \tag{1.16}$$

$$\frac{W_f(x)}{\rho_f} \leq V_{tank}(x) \cdot f_{tank} \tag{1.17}$$

Thus, to maximize range while meeting this constraints, the fuel weight will be:

$$W_f = \min\left(W_{f_{ref}}, V_{tank} \cdot f_{tank} \cdot \rho_f\right) \tag{1.18}$$

Finally, a constraint on the maximum wing loading is enforced, requiring it to be lesser or equal than the wing loading of the reference aircraft:

$$W/S_{max} \leq W/S_{ref} \tag{1.19}$$

1.6. Extended Design Structure Matrix

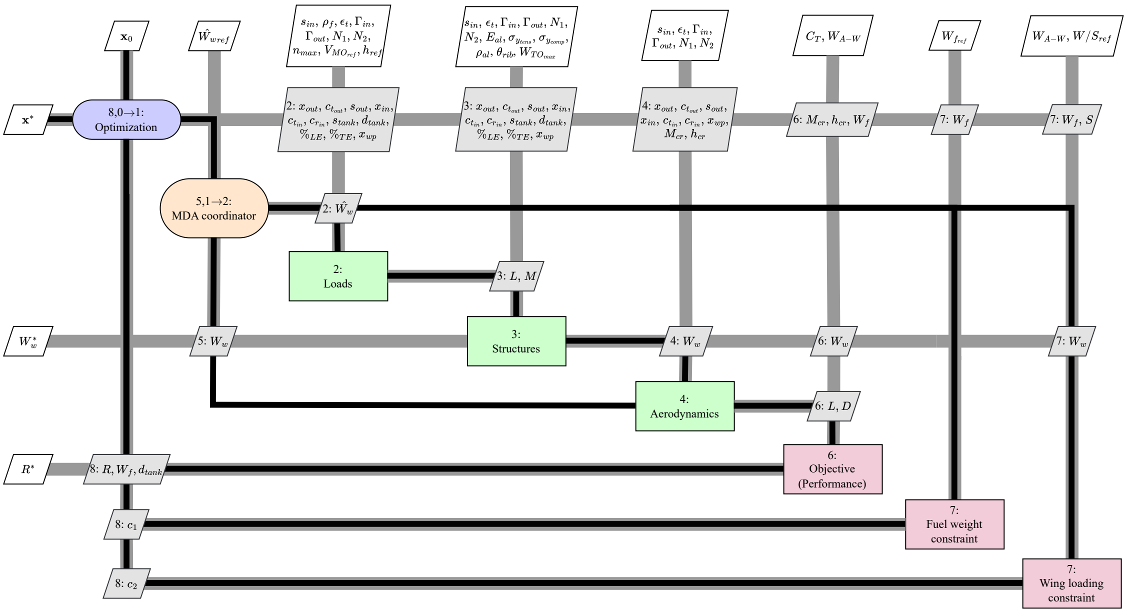


Figure 1.6: Extended Design Structure Matrix

For this Extended Design Structure Matrix (XDSM) four disciplines were utilized. The last one being Performance which is also the objective function and was explained in section 1.4. The other three are: loads, structures and aerodynamics.

1.6.1. Loads

The loads discipline is in charge of estimating the loads that the wing will endure during operation. To do so, it receives the wing's geometry, a reference flight condition at the maximum load factor, and the wing's weight distribution. With these parameters, the loads discipline estimates the different load distributions due to the fuel tanks, inviscid aerodynamics (calculated with Q3D) and the wing itself. Finally, it outputs the lift and moment distribution which the structure will need to withstand.

1.6.2. Structures

The structures discipline is in charge of determining the wing's weight. This discipline receives the same inputs from the optimizer as Loads. Additionally, it requires the outputs from Loads as inputs - the lift and moment distributions. With the use of EMWET, this discipline's only output is the wing's weight.

1.6.3. Aerodynamics

Finally, the Aerodynamics discipline uses as inputs the wing's geometry and the optimizer's cruise condition. Running Q3D with viscid conditions, gives us at last the lift and drag which is used by the objective function.

References

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