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Handling Qualities Optimization in Aircraft Conceptual Design

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By including handling qualities considerations early in the design process it is possible to design statically unstable aircraft with a stability augmentation system which have reduced fuel burn. The present paper describes a general regional pole placement method based on a modified Routh-Hurwitz criterion, applicable to both augmented and unaugmented aircraft designs. The method efficiently computes aerodynamic stability and control derivatives and evaluates their impact on the handling qualities. Furthermore, it ensures controllability in all flight regimes and optimizes longitudinal handling qualities by sizing the horizontal tailplane, positioning the wing, and synthesizing a suitable stability augmentation system using modern control design techniques. The methodology has been applied to a single-aisle, midrange aircraft. The results showed that the methodology is indeed capable of achieving the desired handling qualities level for both augmented and unaugmented configurations. Furthermore, when simultaneously designing a stability augmentation system, it was shown that it is possible reduce the horizontal tail size of an augmented aircraft by 22% for the same control anticipation parameter while increasing the short-period damping ratio by 38% with respect to the unaugmented case.

Nomenclature

Latin Symbols

A	State matrix [~]
AR	Aspect ratio [~]
b	Span [m]
B	input matrix [~]
C	output matrix [~]
$\bar{c}$	MAC [m]
c	Chord [m] or constraint [~]
C_D	Drag coefficient [~]
C_f	Skin friction coeff. [~]
C_L	Lift coefficient [~]
C_m	Pitching moment coefficient [~]
D	Feedthrough matrix [~]
D_c	Non-dimensional derivative [~]
J	Objective function [~]
k_α	AoA feedback gain [rad/rad]
k_q	Pitch rate feedback gain [rad/rad/s]
K_Y^2	Non-dimensional inertia [~]
l	arm length [m]
L	Thickness ratio factor [~]
q	dynamic pressure [Pa]
q^*	State weight factor [~]
Q	State weight matrix [~]

R	Control weight matrix [~]
R_{ls}	Lifting surface correction factor [~]
R_{array}	Modified Routh array [~]
S	Surface area [m ²]
t	Thickness [m]
T_{θ_2}	Incidence lag constant [~]
$\mathbf{u}$	input vector [~]
V	speed [m/s]
W	Weight [kg]
x	Design var. [~] or long. position [m]
$\mathbf{y}$	Output vector [~]

Greek Symbols

α	Angle of attack [rad]
δ_e	Elevator deflection [rad]
ϵ	Downwash angle [rad]
ζ	Damping ratio [~]
λ	Taper ratio [~]
Λ	Sweep angle [deg]
ρ^*	Control weight factor [~]
ω_n	Natural frequency [rad/s]

Acronyms

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CAP	Control anticipation parameter [$\sim$]	cg	Center of gravity
MAC	Mean aerodynamic chord	h	horizontal tailplane
LE	Leading edge	i	induced
NP	Neutral point	ph	phugoid
TO	Take-off	pos	position
		sp	short period
		w	wing
		wet	wetted
subscripts			
0	zero-lift or initial		
ac	aerodynamic center		

I. Introduction

The term *handling qualities* has been used throughout the years to indicate different aspects of the flight characteristics of an aircraft. Cooper and Harper pioneered the improvement of the idea of a unified framework for the evaluation of handling qualities based on pilot rating scales, defining them as “those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot is able to perform the tasks required in support of an aircraft role”.¹ With the advent of Automatic Flight Control Systems it became clear that it would have been possible to redesign an aircraft exploiting this new technology, achieving smaller stabilizing surfaces and therefore improvements in terms of parasitic drag, weight and fuel consumption. Additionally, referring to handling qualities, Soban states that “Although a relatively small fraction of life cycle costs are spent during the preliminary design phase, mistakes and misjudgments during this phase prove costly, and sometimes financially disastrous, to fix at later dates.”² Hence, the objective of this research is the creation of a general tool for the optimization of aircraft handling qualities in the conceptual design phase. This is highly relevant because it allows for the investigation of safer and more fuel-efficient configurations earlier in the design process, as well as lessens the chances of costly redesigns at later stages. The presented tool handles both unaugmented designs (bare airframe), as well as augmented design, in which a suitable stability augmentation system is included. To this end, a common stability paradigm, based on a modified Routh-Hurwitz criterion, has been implemented as a set of nonlinear constraints on the design space. In essence, the criterion has been transformed from a test to a design procedure, taking the form of a general, polynomial-based regional pole placement method. The methodology applies to conventional aircraft configurations, specifically by sizing the horizontal tailplane and positioning the wing. The tool is focused on optimization of the short period motion, while the objective functions to be minimized are tailplane induced drag, zero-lift drag, and weight.

The proposed tool is fitted as a module within the Initiator, a conceptual and preliminary design tool developed at TU Delft. The initial philosophy that forms the basis of this tool is found in the *Design and Engineering Engine* (DEE),³ which was later on followed by the development of the *Initiator*.⁴ Starting from user-specified top-level requirements and a chosen configuration, the tool can generate a first estimation of the aircraft geometry and conduct an evaluation of the associated performance parameters. A schematic flowchart representing the process diagram of the Initiator is illustrated in Figure C. The new module is positioned at the end of the Class 2 convergence loop, since it requires inputs from various modules located within the Class 1 convergence loop, while the final design of the tail and the position of the wing, which is the module’s output, are required in the mission analysis module as well as the Class 2.5 convergence loop. Since the tool is focused on the short period motion, specific metrics for the evaluation of the aircraft performance in this respect have been selected. Firstly, one of the most widely used criterion for evaluating longitudinal handling qualities: the Control Anticipation Parameter,⁵ or CAP. The assessment is done by introducing a parameter which is compared against the damping ratio ζ_{sp} of the aircraft, and is expressed as follows:

$$CAP = \frac{\omega_{n_{sp}}^2}{n_\alpha} \simeq \frac{\omega_{n_{sp}}^2}{C_{L_{trim}}/\alpha} \quad (1)$$

In words, the CAP can be interpreted as “the ratio of an aircraft’s initial pitching acceleration to its change in steady state normal acceleration”.⁶ The core of this approach resides in the term n_α , which is constant with airspeed and therefore does not depend on the conditions of the test being carried out. The magnitude of the CAP determines the abruptness and intensity of pitching acceleration of the aircraft following an elevator input, with respect to the trimmed state normal acceleration. It can also be condensed in the

Figure 1. Initiator convergence N2 diagram

statement *does the nose follow the stick?*.

A second metric for short period handling qualities evaluation stems from military specifications. Therein, aircraft are divided according to their characteristics and the required values for dynamic stability parameters are established with respect to different flight phases. Specifically, aircraft are divided in four classes, while three distinct flight phases are considered.⁷ Concerning the short period motion, an example of the different requirements for the various levels of handling qualities, for class III aircraft, are shown in Table 1.

Table 1. Short period damping ratio requirements - MIL-F-8785C⁷

Flight Phase	Level 1		Level 2		Level 3	
	Min	Max	Min	Max	Min	Max
A	0.35	1.30	0.25	2.00	0.1	—
B	0.30	2.00	0.20	2.00	0.1	—
C	0.50	1.30	0.35	2.00	0.25	—

The goal in this case is to achieve values of short period damping ratio such that the aircraft falls into the level 1 boundary. Hence, by combining CAP and ζ_{sp} it is possible to optimize a configuration as to achieve satisfactory longitudinal handling qualities. Although there are many more different methods of evaluating such flight characteristics, the present research is concerned solely with the aforementioned two.

II. Modified Routh-Hurwitz Criterion

The classical Routh-Hurwitz criterion allows for the determination of the stability of a system by analyzing the sign of the coefficients of its characteristic polynomial.⁸ This method can be specialized in order to identify new conditions on the sign of the coefficients so as to locate a specific region of the complex plane, which correspond to the desired modal response characteristics. Two conditions may be identified in the imaginary plane:

- The real part of the eigenvalues λ of the system is less than a prescribed value $x_1 = -c$.
- The eigenvalues are inside a cone of amplitude $2\alpha^*$, such that the damping ratio ζ is within the values $(\cos \alpha^*, 1)$.

In other words, the method defines a precise area of the plane, as shown in Figure 2, in which the roots of the characteristic polynomial should be located to achieve the desired modal response characteristics. Hence, the methodology identifies the necessary and sufficient conditions for an aircraft to possess the desired handling qualities in terms of modal response.

Figure 2. Modified Routh-Hurwitz criterion - allowable pole locations

Graphically, α^* indicates the angle between the horizontal axis and each of the lines denoted with blue in Figure 2. The region in is then bounded within a cone of amplitude $2\alpha^*$ and a vertical line at $x_1 = -c$. To have a better understading of the modified Routh-Hurwitz criterion let us consider a characteristic polynomial in a general form:

$$p(z) = p_0 z^n + p_1 z^{n-1} + \dots + p_n \quad (2)$$

To satisfy the first condition, let us apply the classical Routh-Hurwitz criterion to the polynomial obtained by translating Equation 2, such that $z = z_1 + x_1$:

$$p(z) = p_0(z_1 + x_1)^n + p_1(z_1 + x_1)^{n-1} + \dots + p_n \quad (3)$$

A Routh array similar to the one that can be found in literature¹⁰ is then created, based on the new polynomial. If $p(z)$ satisfies the Routh conditions on the signs of the coefficients, then all the eigenvalues λ_i will be to the left of a straight line $x_1 = -c$ in the complex plane. The second condition, which is that the solution of Equation 2 are within a cone of amplitude $2\alpha^*$, can be satisfied by referring to the following modified Routh array:

$$R_{\text{array,modified}} = \begin{cases} a_0 & a_1 & a_2 & \dots \\ b_0 & b_1 & b_2 & \dots \\ c_0 & c_1 & c_2 & \dots \\ \vdots & \vdots & \vdots & \ddots \end{cases} \quad (4)$$

With:

$$c_0 = \frac{a_0 b_1 - b_0 a_1}{b_0}, \quad c_1 = \frac{a_0 b_2 - b_0 a_2}{b_0}, \quad \dots \quad (5)$$

For the sake of brevity the full mathematical derivation of this array is not presented. For the interested reader, the complete description can be found in the work of Teofilatto.¹⁰ In contrast with the usual Routh array, the first two rows are computed for $i = 1, \dots, n$ as:

$$a_i = (-1)^i p_i \cos(n - i)\alpha^* \quad (6)$$

$$b_i = (-1)^{i+1} p_i \sin(n - i)\alpha^* \quad (7)$$

The second condition is satisfied by the usual test, or that $p_i > 0$ and, taking the first column of the modified array (Eq. 4), we have all entries with equal sign. If these two conditions are met, then the system has roots in a specific region in the complex plane, which can be chosen a priori as a function of the characteristics of the system which are of interest. Specifically this is done by setting:

$$\zeta_{sp} \geq \cos \alpha^* \quad (8)$$

$$\omega_{n_{sp}} \geq -\frac{x_1}{\cos \alpha^*} \quad (9)$$

Hence, it is possible to investigate whether a given aircraft configuration has roots in the region of the complex plane identified by the conditions on the signs of the characteristic polynomial (Eq. 2). It must be noted that this methodology, just as the original Routh-Hurwitz criterion, still has the characteristics of a test. Both can be used to assess the stability of a system, to a different extent respectively, and indeed the modified Routh-Hurwitz criterion has been applied as to give preliminary information on suitable design changes needed to fulfill a certain desired level of handling qualities.⁹

In general, however, the scope of this method is still rather limited. Nevertheless, there is the potential to enhance this method, by translating its test nature into a design procedure which could be included in an optimization framework. This can be done, qualitatively, through the following procedure:

1. Choose the desired ζ_{sp} and $\omega_{n_{sp}}$
2. Obtain the required α^* and x_1
3. Derive the modified Routh array (Eq. 4)
4. Implement the conditions on the signs of the polynomial coefficients as nonlinear inequality constraints on the design space

Hence, by casting the results of the modified Routh-Hurwitz criterion as constraints on the design space, it is possible to investigate only the configurations which satisfy the desired level of handling qualities, broadly intended as optimal combination of ζ_{sp} and CAP. The practical implementation of the aforementioned procedure is detailed in Section 4. The inclusion in an optimization framework allows for an increase in the design variables which can be considered, the only limitation being the eventual increment in the dimensionality of the problem. It must also be noted that there is no practical limit to the order of the polynomials that can be analyzed. The conditions on the coefficients of the characteristic polynomial are simple algebraic relations, which are readily solved. Hence this method is highly efficient in terms of computational time, as well as expandable to include higher order dynamics.

III. Methodology

A. Optimization Framework

The appropriate set-up of the optimization stems from two main requirements which have to be met, namely computational efficiency and robustness. The need for the former is dictated by the requirements imposed by the Initiator: the module itself should run in about ten seconds and while a certain margin is allowed, nevertheless implementing a fast optimization is of capital importance. The second consideration entails that the module has to cope with a variety of designs and configurations without hindering the design procedure. This is especially true in this case, since casting the handling qualities requirements as constraints on the design space has its drawbacks. Certain configurations produced by the Initiator will not initially satisfy these constraints, which means that they do not possess the desired handling qualities. Hence, the occurrence of unfeasible initial design vectors might hinder the optimization procedure,¹¹ which is to be avoided.

1. Optimization Structure

In Figure 3 the high-level structure of the optimization is presented. As can be seen, the optimization is divided in two steps denoted by *preliminary* and *main*. This approach is used to ensure the robustness of the framework. The preliminary optimization, which is run using the gradient-based optimization algorithm `fmincon`, has the purpose of finding an initial design vector $x_{0_{init}}$ which is feasible. Thus only the nonlinear

constraints have to be satisfied, while the objective function is set to zero. The preliminary optimization problem can be therefore posed in the following way:

$$\begin{aligned} \min \quad & J_i = 0 & i = 1, 2, 3 \\ \text{s.t} \quad & c_{\text{ineq}_j}(x_k) \leq 0 & j = 1, \dots, m \\ & lb_k \leq x_k \leq ub_k & k = 1, \dots, n \end{aligned}$$

Figure 3. High level optimization architecture.

The sole purpose of this optimization procedure is to act as a filter: if no feasible design vector is found then a logical statement evaluates the exit flag and the optimization procedure is interrupted, feeding through the baseline configurations to the subsequent modules. This approach was devised to make sure that the handling qualities module would not hinder the convergence of the Initiator, as well as making the module itself more robust and adaptable to different initial designs.

The main optimization takes as input the design vector $x_{0_{\text{init}}}$ provided by the preliminary optimization and generates as output the final design vector x_{opt} . Furthermore, in contrast with the previous procedure, it aims at minimizing an objective function comprised of three components, which will be detailed in Section 2. It must be noted that the solver is also different: for the preliminary optimization `fmincon` was used, while in the main optimization the solver is `fgoalattain`. This choice of the latter was dictated by the fact that it allows for the definition of three separate performance indexes, instead of compounding all the components into one with the aid of weights. Instead, the weight can be defined independently for every component, indicating the relative strength of each objective function. The main optimization problem can therefore be defined in the following way:

$$\min \gamma = \begin{cases} J_i(x_k) - \text{weight} \cdot \gamma \leq \text{goal} & i = 1, 2, 3 \\ c_{\text{ineq}_j}(x_k) \leq 0 & j = 1, \dots, m \\ lb_k \leq x_k \leq ub_k & k = 1, \dots, n \end{cases}$$

2. Objective Functions

As stated in Section I, the novel approach of this research consists of implementing the handling qualities requirements as constraints on the design space, rather than embedding them in an objective function. This allows for the definition of a performance-based objective function, with the aim of achieving the required flight-dynamic characteristics while concurrently seeking performance benefits.

Three distinct performance indexes are used in the main optimization, i.e. horizontal tailplane weight W_h , horizontal tailplane induced drag $C_{D_i}|_h$ and horizontal tailplane zero-lift drag $C_{D_0}|_h$. In order to keep the computational time at a minimum, the three performance indexes are calculated using handbook methods (empirical formulas). As will be explained in Section 3, this approach, albeit computationally inexpensive compared to more refined methods, presents some drawbacks and will directly impact the choice of the design vector as well as the results.

Horizontal Tailplane Weight The tailplane weight is computed using Torenbeek's method,¹² which estimates the tailplane weight as function of area S_h , aircraft dive speed V_D and half-chord sweep angle $\Lambda_{0.5h}$:

$$W_h = K_h S_h \left[3.81 \cdot \frac{S_h^2 V_D}{1000 \sqrt{\cos(\Lambda_{0.5h})}} - 0.287 \right] \quad (10)$$

where K_h is a correction factor for composite structures. Although it was found to produce quite accurate results,¹³ this formulation is insensitive to several design parameters, such as the taper ratio, which would otherwise have a considerable impact on the weight of the tailplane.

Horizontal tailplane induced drag The second objective functions is the tailplane induced drag, which is computed using the well-known definition:

$$C_{D_i}|_h = K C_{l_h}^2 \quad (11)$$

where the factor K is defined as:

$$K = \frac{1}{\pi e A R_h} \quad (12)$$

In Equation 12 the term e is the Oswald factor, which in the module is determined through an empirical correlation prior to the optimization procedure. The lift coefficient of the horizontal tailplane in cruise condition is obtained through the condition of longitudinal equilibrium and equality of weight and lift.¹² The equation, in non-dimensional form, is defined as follows:

$$C_{l_h} = \frac{S}{S_h} \frac{q}{q_h} \frac{\bar{c}}{l_h} \left(C_{m_{ac, wb}} + C_{L_{wb}} \cdot \frac{x_{cg} - x_{ac, wb}}{\bar{c}} \right) \quad (13)$$

Where $\frac{x_{cg} - x_{ac, wb}}{\bar{c}}$ is the static margin of wing plus fuselage. The location of the aerodynamic center is computed using a two-step method: first the aerodynamic center of the wing is found through Torenbeek's carpet plots,¹² as a function of taper, sweep, and aspect ratio. Secondly, the presence of the fuselage is taken into account through a correction factor. Furthermore, the pitching moment of the wing is computed using the DATCOM method,¹⁴ and again a correction factor is used to determine the contribution of the fuselage.¹²

Horizontal tailplane zero-lift drag Finally, the zero-lift drag coefficient of the horizontal tailplane has been estimated using the following empirical relation:¹⁶

$$C_{D_0}|_h = R_{wf} R_{ls} C_{D_f} \left(1 + L \cdot \left(\frac{t}{c} \right) + 100 \cdot \left(\frac{t}{c} \right)^4 \right) \frac{S_{h_{wet}}}{S_h} \quad (14)$$

The various terms appearing in Equation 14 are detailed in the list of symbols. However, it must be noted that zero-lift drag is dependent on the Reynolds and Mach number through the tailplane turbulent flat plate coefficient C_{D_f} . This term is computed within the parasitic drag estimation module in the Initiator. The remaining terms are mostly dependent on the geometry of the surface, i.e. the ratio between wetted area and reference area, sweep, and thickness distribution. Furthermore, the presence of the fuselage, with the related interference effects, is also taken into account through the correction factor R_{wf} .

A remark must be made about the applicability of Equation 14: it is generally valid for hydraulically smooth surfaces. Any roughness and imperfections introduce additional drag, which however is not taken into account. It must, therefore, be borne in mind that the results are likely to be underestimated. Moreover, the wetted area of the tailplane $S_{h_{wet}}$ is defined as:

$$S_{h_{wet}} = 2 \left[1 + 0.5 \cdot \left(\frac{t}{c} \right)_{\max} \right] b \bar{c} \quad (15)$$

Finally, the three components are normalized using their respective values for the initial design (subscript 0). Thus the three components of the objective function are defined according to the following definitions:

$$J_1 = \frac{W_h}{W_{h_0}}, \quad J_2 = \frac{C_{D_0}}{C_{D_{0,0}}}, \quad J_3 = \frac{C_{D_i}}{C_{D_{i,0}}} \quad (16)$$

3. Sensitivity Study

In order to set-up the optimization procedure in a correct way, a sensitivity analysis is performed. The method used involves assessing the logarithmic sensitivity¹⁵ of several aircraft geometrical parameters on the objective functions, in order to determine their relative strengths and selecting a suitable set of design variables. Through the sensitivity study it was possible to infer that some parameters, which play a crucial role in the design of a horizontal tailplane, do not impact the objective function in a coherent way, and in some cases, have almost no effect. This is mainly due to the empirical nature of the objective functions, which therefore do not account for every variable. To clarify, in Table 2, the value of the objective functions' gradient with respect to five different tailplane design parameters is shown. If the gradient is close to 1, then the variables has a detectable impact on the objective function, while a value close to zero indicates almost no effect.

Table 2. Gradient of objective functions for various design variables

$x \downarrow \frac{dF}{dx} \rightarrow$	$\frac{dW_h}{dx}$	$\frac{dC_{D_0}}{dx}$	$\frac{dC_{D_i}}{dx}$
S_h	1.19	1.01	-1.92
AR_h	0.018	$-9.8 \cdot 10^{-3}$	-1.25
λ_h	0.01	$-7.7 \cdot 10^{-3}$	-0.12
Λ_{LE_h}	0.07	-0.08	-0.22
x_{pos_w}	0	0	30

It is immediate to see how the impact of several variables is unevenly distributed. To ensure that the optimization problem is well-posed and there are no design variables which might take on unrealistic values, the set has been reduced to just S_h and x_{pos_w} . Together, these two parameters are used to determine the horizontal tail arm, and are therefore deemed crucial. The remainder of the geometrical parameters would require more refined analysis methods for the objective functions in order to be included in the design vector which, at this stage, are yet to be implemented.

B. Module implementation

The bulk of the procedure is implemented in the file which contains the nonlinear constraints (`NonLinConstraints.m`). An overview of the calculation procedures is presented in Figure 4, followed by a presentation of the methodology for each sub-module.

Figure 4. Module architecture - NonLinConstraints.m

1. Stability and Control Derivatives Estimation

The stability and control derivatives required for the implementation of the Equations of Motion are computed using a simplified version¹⁹ of the DATCOM method.¹⁴ Only a limited set of derivatives are calculated, specifically the ones required to define the short-period reduced-order model, which will be introduced in Section 2. Care is taken to compute these derivatives at the correct flight condition, depending on which flight phase the optimization is run for. It must be noted that a conversion between the stability axis and the body axis is required before the derivatives can be applied to the system of Equations of Motions shown in Section 2. It must be noted that this methodology is computationally very efficient and in general provides acceptable results in the subsonic flight domain.²⁰ In some cases, however, it might also prove less accurate compared to other methods,²¹ such as Vortex Lattice or panel methods.

2. Flight dynamics model

As introduced in Section I, the scope of this work is to develop a tool capable of optimizing the handling qualities of an aircraft, specifically regarding the short period oscillation. In order to obtain a compact module, the linearized Equations of Motion have been reduced using the short-period approximation. The hypotheses for this simplification are the following:

- $V_0 = \text{constant}$
- $\gamma_0 = 0$

The first assumption entails that $\hat{u} = 0$ and therefore the forces in the X -direction remain in equilibrium during the entire motion. The second assumption states that the initial flight condition is considered to be level, which also implies that the term C_{X_0} is equal to 0, since there is no weight component in the X -direction. As a consequence, the pitch-angle (θ) contribution is neglected, which, in turn, allows for the omittance of the kinematic relations.²² Finally, the Equations of Motion approximated for the short period

motion can be expressed in the following form:²²

$$\begin{bmatrix} C_{Z_\alpha} + (C_{Z_\dot{\alpha}} - 2\mu_c)D_c & C_{Z_q} + 2\mu_c \\ C_{m_\alpha} + C_{m_{\dot{\alpha}}}D_c & C_{m_q} - 2\mu_c K_y^2 D_c \end{bmatrix} \begin{bmatrix} \alpha \\ \frac{q\bar{c}}{V} \end{bmatrix} = \begin{bmatrix} -C_{Z_{\delta_e}} \\ -C_{m_{\delta_e}} \end{bmatrix} \delta_e \quad (17)$$

By rearranging Equations 17²² it is possible to obtain a state-space form of this reduced-order model:

$$\dot{\mathbf{x}} = A \mathbf{x} + B \mathbf{u} \quad (18)$$

$$\mathbf{y} = C \mathbf{x} + D \mathbf{u} \quad (19)$$

Where:

$$A = \begin{bmatrix} z_\alpha & z_q \\ m_\alpha & m_q \end{bmatrix}, \quad B = \begin{bmatrix} z_{\delta_e} \\ m_{\delta_e} \end{bmatrix}, \quad C = \begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix}, \quad D = \begin{bmatrix} 0 \\ 0 \end{bmatrix} \quad (20)$$

The state-space system is then directly fed to the pole-placement module in case the design is unaugmented (i.e. no controller is present). Otherwise it is the input to the stability-augmentation-system design module, outlined in the following section.

3. Stability Augmentation System Design

Once the state-space system of the plant has been obtained, it is possible to use state feedback to synthesize a stability augmentation system, or SAS. Specifically, the inner SAS loop is made up by a pitch damper and an angle-of-attack feedback, as shown in Figure 5. For this purpose, two different approaches are considered: the first one entails including the feedback gains in the design vector. The second approach makes use of the Linear Quadratic Regulator (LQR) for their computation. For both the unaugmented and the augmented case, the ideal actuator assumption is used, which at the conceptual level is deemed acceptable.^{36, 37}

Method 1 For this approach, the control law is defined in the following form:

$$\mathbf{u} = -k\mathbf{x} + \delta_e = - \begin{bmatrix} k_\alpha & k_q \end{bmatrix} \begin{bmatrix} \alpha \\ \frac{q\bar{c}}{V} \end{bmatrix} + \delta_e \quad (21)$$

where k_α and k_q are the gains for the angle of attack and pitch rate respectively. It must be noted that these gains are implemented as design variables. The state space system then becomes:

$$\dot{\mathbf{x}} = A\mathbf{x} + B\mathbf{u} = A\mathbf{x} + B(-k\mathbf{x} + \delta_e) = (A - Bk)\mathbf{x} + B\delta_e \quad (22)$$

The matrix $A_c = (A - Bk)$ is the plant matrix describing the closed-loop system, which is subsequently fed to the pole-placement submodule.

Method 2 The control law implemented for the linear quadratic regulator is defined as:

$$\mathbf{u} = -k\mathbf{x} \quad (23)$$

The gains k are determined such that the control law minimizes the following quadratic performance index:

$$J(\mathbf{u}) = \int_0^\infty (\mathbf{x}^T Q \mathbf{x} + \mathbf{u}^T R \mathbf{u}) dt \quad (24)$$

As opposed to the state feedback design, in which the gains are part of the design vector, the design variables for the LQR are the weight factors of the matrices Q and R , i.e. q_1^* , q_2^* and ρ^* . This approach allows for the automation of the design procedure, which usually requires manual tuning of these weight factors in order to achieve the desired response.²³ Specifically, the matrices are defined as:

$$Q = \begin{bmatrix} q_1^* & 0 \\ 0 & q_2^* \end{bmatrix} \quad R = \rho^*[1] \quad (25)$$

The design of the controller is carried out using the `lqr` function in Matlab, by first providing the matrices and the state-space system of the model. The results is a 1×2 vector containing the required gains:

$$k = [k_\alpha \quad k_q] \quad (26)$$

The augmented matrix is then formed using the same approach as illustrated in Equation 22 and passed on as input to the pole-placement submodule.

4. Regional Pole Placement

This submodule accepts as input the state matrix, for both unaugmented and augmented designs. This matrix can be used to compute the determinant of the short-period reduced-order model, and hence obtain the characteristic polynomial of the system. For the two different options, augmented and unaugmented, the determinant can be computed in the following way:

$$\text{Unaugmented configuration} \rightarrow \det(A - \lambda \mathbf{I}) = p_0 \lambda^2 + p_1 \lambda + p_2 = 0 \quad (27)$$

$$\text{Augmented configuration} \rightarrow \det(A_c - \lambda \mathbf{I}) = p_0 \lambda^2 + p_1 \lambda + p_2 = 0 \quad (28)$$

Note that the coefficients p_0 , p_1 , and p_2 are a function of the aircraft stability and control derivatives, as well as mass, inertia characteristics and, eventually, feedback gains. As was shown in Section II, the coefficients must satisfy a specific set of conditions to ensure that the design complies with the required handling qualities. First, the amplitude of the semi-cone which determines the boundary of the allowable pole region in the complex plane is set, such that:

$$\alpha^* = \arccos(\zeta_{sp}) \quad (29)$$

This directly determines the minimum required short period damping ratio. Secondly, the lower limit for the real part of the eigenvalues is set, as function of the required control anticipation parameter (CAP). To this end, the short period natural frequency corresponding to the required minimum CAP^{25,26} is obtained, such that:

$$\omega_{n_{sp}} = \sqrt{\frac{\text{CAP}_{\min} \cdot V_0}{T_{\theta_2} \cdot g}} \quad (30)$$

It must be noted that this definition of the CAP²⁶ differs slightly from the basic form shown in Section I. Nevertheless, it describes the same metric. It is therefore possible to compute the required real part of the eigenvalues as follows:

$$x_1 = -\omega_{n_{sp}} \zeta_{sp} = -\omega_{n_{sp}} \cos \alpha^* \quad (31)$$

A remark must be made about the incidence lag constant T_{θ_2} : it is obtained by implementing the full-order longitudinal equations of motion in the flight dynamics submodule and extracting the pitch transfer function in the familiar zero-pole-gain form:²⁵

$$\frac{\theta(s)}{\delta_e(s)} = \frac{k_\theta [s + (1/T_{\theta_1})] [s + (1/T_{\theta_2})]}{(s^2 + 2\zeta_{ph}\omega_{n_{ph}}s + \omega_{n_{ph}}^2) (s^2 + 2\zeta_{sp}\omega_{n_{sp}}s + \omega_{n_{sp}}^2)} \quad (32)$$

It is then possible to compute T_{θ_2} in order to obtain the required $\omega_{n_{sp}}$ in Equation 30.

The next step entails the creation of the Routh matrices through the procedure detailed in Section II, for both conditions ($Re(\lambda) \leq -x_1$ and $\zeta_{sp} \geq \cos \alpha^*$). The final matrices have the following form:

$$R_{array,x_1} = \begin{bmatrix} p_0 & a_2 & a_3 \\ b_1 & b_2 & b_3 \\ c_1 & c_2 & c_3 \end{bmatrix}, \quad R_{array,\alpha} = \begin{bmatrix} a_0 & a_1 & a_1 \\ b_0 & b_1 & b_2 \\ -c_0 & c_1 & c_2 \\ -d_0 & d_1 & d_2 \\ e_0 & e_1 & e_2 \\ f_0 & f_1 & f_2 \end{bmatrix} \quad (33)$$

Having defined all the required matrices, it is now possible to express the conditions on the polynomial coefficients in the form of inequality constraints. The first condition that must be met entails that the coefficients must all have equal sign, or, in other terms:

$$c_{ineq}(s) = \text{sign}(p_0) = \text{sign}(p_s) \quad \text{for } s = 1, \dots, n \quad (34)$$

This yields a number of constraints equal to the order of the polynomial considered. Secondly, the sign conditions on the coefficients of the matrices can be synthesized in the following way: in order for the poles of the system to be located inside the cone of amplitude $2\alpha^*$, the coefficients in the first column of the matrix $R_{array,\alpha}$ need to have the same sign as a_0 . In order for the poles to be located to the left of a straight vertical line at $x_1 = -c$, the coefficients in the first column of the matrix R_{array,x_1} need to have the same sign as p_0 . The same applies to the first two rows. In terms of inequality constraints, for the condition $Re(\lambda) \leq -x_1$, considering $p_0 > 0$ we have:

$$c_{ineq}(i) = -R_{array,x_1}(i+1, 1) \leq 0 \quad \text{for } i = 1, 2 \quad (35)$$

$$c_{ineq}(j) = -R_{array,x_1}(j, 2) \leq 0 \quad \text{for } j = 1, 2 \quad (36)$$

For the condition $\zeta_{sp} \geq \cos \alpha$ a similar procedure applies:

$$c_{ineq}(k) = -R_{array,\alpha}(k+2, 1) \leq 0 \quad \text{for } k = 1, \dots, 4 \quad (37)$$

Finally the complete set of inequality constraints for the pole placement method is obtained: it consists of eight elements plus the two conditions on the coefficients shown in Equation 34, or $c_{ineq} = c_{ineq}(s) + c_{ineq}(i) + c_{ineq}(j) + c_{ineq}(k)$. If the solution is feasible, it automatically has the desired dynamic characteristics, or, in some cases, especially for augmented designs, the design could also exceed the minimum requirements.

5. Stability and Controllability

The approach chosen to ensuring static stability and controllability is based on Torenbeek's X-plots.¹² While the required dynamic flight characteristics are guaranteed through the regional pole placement method discussed in the previous section, it must be ensured that the aircraft remains controllable and statically stable in all the relevant flight conditions. Specifically, the X-plot methodology is used to compute the minimum tail area which satisfies the required center-of-gravity range, as computed in the class-2 weight estimation module in the Initiator. The most aft position dictates the stability limit, or when the c.g. location overlaps the aircraft aerodynamic center location minus a static margin, which is usually between 5% and 10% of the MAC.^{12,17,18} The most forward c.g. position is critical to determine whether enough control power is available to reach the airplane $C_{L_{max}}$ with flaps down in landing configuration. Another condition, which usually is less limiting, is the availability of enough control power in order to rotate the aircraft at takeoff. The maximum forward and aft positions of the c.g are implemented as non-linear constraints in the optimization problem, adding to the previously derived eleven constraints pertaining to the pole placement method:

$$c_{ineq}(11) = \bar{x}_{cg_{aft,max}} - \bar{x}_{cg_{NP}} \leq 0 \quad (38)$$

$$c_{ineq}(12) = \max(\bar{x}_{cg_{stall}}, \bar{x}_{cg_{TO}}) - \bar{x}_{cg_{fwd,max}} \leq 0 \quad (39)$$

The implementation of these conditions as constraints on the design space implies that, if the solution is feasible, controllability and static stability are simultaneously ensured.

IV. Verification and validation

Verification of the module is of vital importance as it ensures the credibility of the results. Unfortunately useful validation data regarding handling qualities optimization was not found in open literature, and therefore the model in its entirety cannot be validated. Furthermore validation within the Initiator is very difficult, due to uncertainties related to the methods that are used. Nevertheless the separate parts of the module were checked extensively by hand calculations, to ensure the correct working. Regarding validation, the approach was twofold: firstly the aerodynamic derivatives calculation has been verified against data found in open literature. Secondly the flight dynamics model was validated using Phalanx,²⁷ an in-house flight mechanics tool developed at TU Delft.

The aerodynamic stability and control derivatives have been compared against results found in open literature¹⁹ for two different aircraft, namely Boeing B747-100 and Learjet 24. For the B747-100, in cruise conditions and for the Learjet 24, in cruise conditions and maximum takeoff weight the results are shown in Figure 6 and 7, respectively.

Figure 6. Aerodynamic derivatives data - Boeing B747-100 (cruise - MTOM)¹⁹

Figure 7. Aerodynamic derivatives data - Learjet 24 (cruise - MTOM)¹⁹

The results are all within acceptable range and, most notably, for both aircraft the error pattern is the same, considering that all the derivatives are overestimated. No definitive explanation for this occurrence has been found yet, but it would not seem imputable to a systematic error in the inputs for the validation procedure. Using the derivatives estimated by the handling qualities optimization module, the Class-2 weight estimation results and the inertia estimation results coming from the Initiator, three input files for Phalanx were created in order to validate the flight dynamics module described in Section 2. The three aircraft considered are a regional airliner, a mid-range single-aisle airliner and a long-range twin-aisle airliner, namely the Fokker 100, the Airbus A320-200 and the Boeing B747-100, respectively. This choice was dictated by the need to investigate the impact of the aircraft size and configuration on the accuracy of the flight dynamics submodule. The results are summarized in Table 3.

Table 3. Validation results

	A320-200			B747-100			F-100		
Param.	Initiator	Phalanx	Err.	Initiator	Phalanx	Err.	Initiator	Phalanx	Err.
ζ_{sp}	0.41	0.4	1.7%	0.49	0.485	1%	0.66	0.64	3%
$\omega_{n_{sp}}$	1.23	1.22	0.8%	1.37	1.34	2.2%	3.87	3.75	3%
CAP	0.15	0.16	4.3%	0.1	0.09	3%	0.163	0.158	3%

It can be seen the damping ratio is overpredicted in all three case, while the natural frequency and CAP are consistently underpredicted. The values are all within 4% of relative error, which is considered acceptable considering that such discrepancy might be attributed to the use of the short-period reduced-order model

(Equation 17), which naturally introduces a small source of inaccuracy.^{22, 24}

V. Demonstration

A. Optimization Setup

For the test case, a baseline configuration is chosen: a single-aisle midrange aircraft designed within the Initiator, based on an Airbus A320-200. The optimization procedure is run one time for each of the user options which have been specified in Section III (*Unaugmented*, *Augmented method 1*, *Augmented method 2*). The only flight condition analyzed is cruise, with maximum takeoff weight configuration.

The first option (*Unaugmented*) of the module available to the user allows for handling qualities optimization purely through resizing the horizontal tailplane and repositioning the wing. The geometrical design variables are the tail area S_h and the wing longitudinal position x_{pos_w} , as discussed in Section 3. Other tail planform parameters such as the leading edge sweep, taper ratio, and aspect ratio are kept constant and equal to the baseline value computed in previous Initiator modules.

The second option (*Augmented method 1*) entails the concurrent design of a pitch damper and α -feedback to augment the stability of the baseline aircraft. This approach includes the feedback gains in the design vector, as described in Section 3. The geometrical parameters that define the horizontal tailplane remain unvaried with respect to the previous case.

The last user-defined option (*Augmented method 2*) involves the design of the same stability augmentation system as in the previous case. However, it makes use of the Linear Quadratic Regulator to compute the feedback gains. The design vector is then augmented with the weight factors required for the matrices Q and R , as defined in Section 3. Clearly, with respect to the previous case in which the gains were part of the design vector, there is less control over the final solution. The Linear Quadratic Regulator aims at minimizing a time-domain performance index. Hence the gains will not be selected only with the purpose of satisfying the required short-period damping ratio and CAP, but also to achieve a satisfactory time response of the augmented system.

In Table 4 the general data regarding the optimization setup is presented. The objective functions are equal for every case and normalized with respect to the initial values, computed before the start of the optimization for the baseline aircraft. The initial design variables with the respective bounds are presented in Table 5.

Table 4. Optimization setup data

Case	Unaugmented	Augmented (method 1)	Augmented (method 2)
Objectives	$\min \left(\frac{W_h}{(W_{h0})}, \frac{C_{D0}}{C_{D0,0}}, \frac{C_{Di}}{C_{Di,0}} \right)$	$\min \left(\frac{W_h}{(W_{h0})}, \frac{C_{D0}}{C_{D0,0}}, \frac{C_{Di}}{C_{Di,0}} \right)$	$\min \left(\frac{W_h}{(W_{h0})}, \frac{C_{D0}}{C_{D0,0}}, \frac{C_{Di}}{C_{Di,0}} \right)$
HQ Constraints	$\zeta_{\text{sp}} \geq 0.55, \text{CAP} \geq 0.3$	$\zeta_{\text{sp}} \geq 0.76, \text{CAP} \geq 0.3$	$\zeta_{\text{sp}} \geq 0.76, \text{CAP} \geq 0.3$
SAS design	none	Design gains	LQR
Design vector	S_h, x_{pos_w}	$S_h, x_{\text{pos}_w}, k_\alpha, k_q$	$S_h, x_{\text{pos}_w}, \rho^*, q_1^*, q_2^*$

Table 5. Optimization setup data

Variable	$\frac{S_h}{S_{h0}}$	$\frac{x_{\text{pos}_w}}{x_{\text{pos}_{w0}}}$	k_α [rad/rad]	k_q [rad/rad/s]	ρ^*	q_1^*	q_2^*
Initial	1	1	1	1	1	1	1
Lower bound	0.1	0.95	-1	-50	0.001	0.001	0.001
Upper bound	$+\infty$	1.05	1	1	$+\infty$	$+\infty$	$+\infty$

B. Results of Uncycled Design Study

In this section, the horizontal tailplane is designed without the blue feedback loop shown in Figure C. This means that the change in weight and drag of the horizontal tail plane does not influence the maximum take-off mass, engine thrust, or wing size. This so-called *uncycled* design demonstrates the effect of the proposed

methods on the tail size, tail drag, tail weight, and wing location. The same initial aircraft configuration has been run three times, one for each user-defined option. The results are subsequently compared to the baseline design in order to assess the effectiveness of the handling qualities optimization tool. The baseline design of the tailplane has been carried out by means of a tail volume method.

Unaugmented design optimization As shown in Table 6 and Figure 8, the optimized tail area is 43 % larger than the baseline, while the wing has been shifted forward by 5 % with respect to the baseline position, which puts this design variable on the lower bound. This directly translates into an increase of 43 % in tailplane weight, 50 % increase in zero-lift drag and a reduction of 70 % in induced drag. These results are easily analyzed, since they are a direct consequence of the need for a larger horizontal tail arm and tail area to improve the relevant aerodynamic derivatives, specifically C_{m_q} and C_{l_α} , which are required to achieve the prescribed ζ_{sp} and CAP. Indeed the optimization has been successful in identifying a design which complies with the specified handling qualities requirements. This is clearly shown in Figure 9: the pitch response following an initial disturbance in angle of attack shows reduced oscillations with a faster decay, while the effect of the increased CAP is evident from the abruptness of the response in the first few seconds after the disturbance.

Table 6. Unaugmented optimization results

Param.	$\frac{S_h}{S_{h0}}$	$\frac{x_{pos_w}}{x_{pos_w,0}}$	$\frac{W_h}{W_{h0}}$	$\frac{C_{D0}}{C_{D0,0}}$	$\frac{C_{Di}}{C_{Di,0}}$	ζ_{sp}	CAP
Initial	1	1	1	1	1	0.41	0.15
Final	1.43	0.95	1.43	1.5	0.3	0.55	0.3
Diff.	+43%	-5%	+43%	+43%	-70%	+34%	+100%

Figure 8. Configuration comparison - Unaugmented

Figure 9. Pitch rate response - Unaugmented

Augmented design optimization - method 1 By designing a suitable stability augmentation system using *method 1* the tail area has been reduced by 22 %. Moreover, the wing has been shifted forward by 3 % with respect to the baseline, which moves the solution away from the lower bound. An explanation for this choice of the optimizer is found by considering that there is a hard limit on the most aft center of gravity location attainable, specifically selected to ensure at least neutral stability for safety reasons. A reduction in tail area shifts the aircraft aerodynamic center close the aft c.g position, and is limited by the inequality constraints imposed to enforce the neutral stability margin. In contrast with the previous case, all the geometrical modifications contribute to a net reduction in tailplane weight, zero-lift drag and induced drag by 22 %, 26 % and 55 % respectively.

Table 7. Augmented (method 1) optimization results

Param.	$\frac{S_h}{S_{h0}}$	$\frac{x_{posw}}{x_{posw,0}}$	k_α	k_q	$\frac{W_h}{W_{h0}}$	$\frac{C_{D0}}{C_{D0,0}}$	$\frac{C_{Di}}{C_{Di,0}}$	ζ_{sp}	CAP
Initial	1	1	1	1	1	1	1	0.41	0.15
Final	0.78	0.97	-0.3	-31	0.78	0.74	0.45	0.76	0.3
Diff.	-22%	-3%	-	-	-22%	-26%	-55%	+85%	+100%

Remarkable improvements are achieved also in terms of time response following an initial disturbance, as shown in Figure 11: ζ_{sp} has been increased to 0.76, while the CAP is set to 0.3. The oscillations are almost completely damped out within 4 seconds from the disturbance, while the peak in pitch rate is also considerably reduced.

Figure 10. Configuration comparison - Augmented (method 1)

Figure 11. Pitch rate response - Augmented (method 1)

The benefit of the stability augmentation system presence is therefore quite apparent: the aircraft presents a remarkably improved response in spite of the reduction in tail area, while achieving a noticeable reduction in tailplane weight, skin friction drag and induced drag.

Augmented design optimization - method 2 Lastly, the third user-specified option entails the computation of the feedback gains using a Linear Quadratic Regulation (*method 2*). Analyzing the results, the main feature that can be recognized is that both the optimized tail area and wing position values are the same as in the previous case (i.e. *method 1*). The same holds for the final objective functions, which show the same decrease in tailplane weight, zero-lift drag and induced drag.

Table 8. Augmented (method 2) optimization results

Param.	$\frac{S_h}{S_{h0}}$	$\frac{x_{posw}}{x_{posw,0}}$	ρ^*	q_1^*	q_2^*	$\frac{W_h}{W_{h0}}$	$\frac{C_{D0}}{C_{D0,0}}$	$\frac{C_{Di}}{C_{Di,0}}$	ζ_{sp}	CAP
Initial	1	1	1	1	1	1	1	1	0.41	0.15
Final	0.78	0.97	0.05	80	0.1	0.78	0.74	0.45	0.76	0.38
Diff.	-22%	-3%	-	-	-	-22%	-26%	-55%	+85%	+150%

The feedback gains differ considerably from the ones computed with the previous method. In fact, the final result yields $k_\alpha = -0.5$ [rad/rad] and $k_q = -40.6$ [rad/rad/s]. The gains are increased in magnitude due to the Linear Quadratic Regulator, which aims at minimizing the time-domain performance index in Equation 24. This has a profound effect on the modal response characteristics and CAP of the configuration. It can be noticed how ζ_{sp} has been achieved precisely, while the CAP has been overachieved by 50%. This result reflects on the pitch rate response of the aircraft following an initial disturbance in angle of attack.

Figure 12. Configuration comparison - Augmented (method 2)

Figure 13. Pitch rate response - Augmented (method 2)

C. Results of Cycled Design Study

In this section, the impact of the newly developed handling qualities optimization module is assessed with respect to the overall performance of a converged aircraft design. The design is *cycled* according to the green feedback loop shown in until the maximum take-off mass converges within 1%. The objective is to get the big picture in terms of the complete design procedure: in order to do that, the Initiator has been run to convergence on the aforementioned single-aisle midrange aircraft for varying short period damping ratios and Control Anticipation Parameters. Since the final purpose of the module is to be included at the conceptual design stage, it is meaningful to assess the impact that the different optimization methods have on the final configurations, and to extrapolate general trends which can be used to characterize the overall design methodology. Specifically, the convergence study has been performed from $\zeta_{sp_{min}}$ ranging from 0.41, which is the baseline value, to 0.7. Furthermore, two different CAP levels have been investigated: 0.15, i.e the baseline value, and 0.3. For each design point, the L/D_{max} and MTOM of the converged aircraft are obtained and evaluated with respect to the baseline values.

Figure 14. CAP = 0.13 - Unaugmented

Figure 15. CAP = 0.3 - Augmented

A term which appears in Figure 14 needs to be introduced: the *breakeven point*. In this context, this denomination was defined to indicate the short-period damping-ratio values for which, at a given CAP, the L/D_{max} and MTOM of the optimized configuration are equal to the respective baseline values. In other words, it marks the maximum handling qualities level, intended broadly as a combination of ζ_{sp} and CAP, which can be achieved without incurring a degradation of the performance of the aircraft in terms of lift-over-drag and maximum takeoff mass.

In Figure 14 it is possible to see that requiring a CAP of 0.3 has a substantial effect. Specifically, for $L/D_{\max}$ the breakeven point occurs at $\zeta_{\text{sp}} = 0.52$, while for MTOM the value is now $\zeta_{\text{sp}} = 0.51$. Furthermore, for ζ_{sp} equal to the initial value, a doubling of the CAP does not entail a degradation in performance. On the contrary, the optimized configuration shows a higher aerodynamic efficiency and lower MTOM. It must be noted that for CAP = 0.15 the trend, which is not shown here, is remarkably similar, with the breakeven points shifted to higher ζ_{sp} values. Having discussed the unaugmented optimization method, it is now relevant to assess what happens if the configuration is optimized by taking into account the presence of a stability augmentation system.

In Figure 15 the outcome of the analysis in terms of $L/D_{\max}$ and MTOM for the optimized augmented configuration is presented. One feature stands out: the performance of the aircraft is independent of the desired level of damping ratio for CAP = 0.3. This is a remarkable result, which gives profound insights on the effect of including stability augmentation systems in the conceptual design process. Clearly, the optimizer minimizes the tail area and positions the wing such as to obtain the highest improvements in terms of performance while the desired level of handling qualities is achieved just by adjusting the feedback gains accordingly. Hence, a general rule can be inferred by the evaluation of the Initiator convergence: if an aircraft is to be optimized for handling qualities by simultaneously designing a stability augmentation system, the final configuration will achieve the maximum benefits, in terms of performance, independently of the required short period damping ratio and CAP. In other words, the optimizer will reduce the tail area and reposition the wing as to minimize weight and drag to the full extent right from the start, while the compliance with the handling qualities requirements is delegated to the stability augmentation system. The limiting factor that determines the maximum possible achievement in terms of performance benefit was found to be the static stability margin requirements with aft c.g. The tail area cannot be reduced further than the value for which there is neutral stability. By relaxing this requirement, it would be possible to obtain even higher savings in terms of tailplane mass and drag.

VI. Conclusions

A method has been conceived based on Routh-Hurwitz criteria to size the horizontal tailplane and position the wing in longitudinal direction such as to comply with handling quality requirements in addition to stability and control requirements. The method has been extended to include the automated design of two types of control laws in order to demonstrate the effect of a control augmentation system on the aforementioned geometric design variables. It was shown that imposing requirements on short-period damping ratio and control anticipation parameter had a significant effect on the tail size and wing location for the unaugmented design. In the presented study, a 43% tail size increase was shown with respect to a baseline design based on volume coefficients. Furthermore, it was shown that with the addition of the augmentation systems, the tail size could be reduced to a size 22% below the baseline value, while the short-period damping ratio increased with 85% with respect to the baseline value. The static stability margin was shown to actively constrain the minimum tail size. Both $L/D_{\max}$ and MTOM were found to be insensitive to changes in control-anticipation parameter and short-period damping ratio. The compliance with the handling qualities requirements was achieved by suitable changes in feedback gains, while the aircraft geometry was modified to achieve minimum drag and weight. It was found that the factor which limits the achievable performance benefits is the stability requirement, embodied by the static margin at aft center-of-gravity location. Hence, by relaxing this requirement, higher performance benefits could be achieved.

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