Generic stability and control for aerospace flight vehicle conceptual design

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ABSTRACT

The recent period has been filled with exceptionally interesting developments and advances, resulting in high-performance conventional and non-conventional manned and unmanned aircraft. Although those vehicles seem to comply well with specific mission performance requirements, one is still confronted with an apparent weakness to reliably stabilise and control throughout the flight envelope. Since the provision of satisfactory stability and control characteristics invariably compromises flight performance, it becomes essential to identify and integrate performance-optimal stability and control design solutions early during the flight vehicle definition phase. In particular, the conceptual design of integrated control effectors for advanced aircraft is far from being trivial. Never before have we been presented with such tremendous wealth of specialised data and information suitable for detail design of controls. In contrast, never before has it been necessary to approach any one of the primary design disciplines still as entirely ad hoc and inconsistent as in the case of designing controls during the conceptual design phase. This need initiated the development of a configuration independent (generic) stability and control methodology capable of sizing primary control effectors of fixed wing subsonic to hypersonic designs of conventional and unconventional, symmetric and asymmetric configuration layouts. This paper summarises the methodology concept and demonstrates its versatility and validity by analyzing selected stability and control characteristics of the Northrop YB-49 flying wing.

NOMENCLATURE

b	span
$\bar{c}$	mean aerodynamic chord (MAC)
C_D	drag coefficient
C_l	rolling moment coefficient
C_L	lift coefficient
C_m	pitching moment coefficient
C_n	yawing moment coefficient
C_T	thrust coefficient
C_Y	side force coefficient
g	acceleration due to gravity
h	altitude
h'_x	x-component of angular momentum vector of spinning rotor with respect to rotor mass centre
h'_y	y-component of angular momentum vector of spinning rotor with respect to rotor mass centre
h'_z	z-component of angular momentum vector of spinning rotor with respect to rotor mass centre
I_x	x-axis moment of inertia
I_y	y-axis moment of inertia
I_z	z-axis moment of inertia
I_{xy}	x-y product of inertia
I_{yz}	y-z product of inertia
I_{xz}	x-z product of inertia
K_p	roll rate gain
K_q	pitch rate gain

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K_r	yaw rate gain
K_α	angle-of-attack gain
L/D	aerodynamic efficiency
L/D_{trim}	trimmed aerodynamic efficiency
M	Mach number
m	mass
n	load factor
$\overline{q}$	dynamic pressure
S_{ref}	wing reference area
T_i	thrust per engine
V	aircraft velocity
V_{rot}	take-off rotation velocity
W	aircraft weight
x_{cg}	x-axis centre of gravity location as fraction of the mean aerodynamic chord
x_T	x-axis thrust line co-ordinate relative to the aircraft centre of gravity
y_T	y-axis thrust line co-ordinate relative to the aircraft centre of gravity
z_T	z-axis thrust line co-ordinate relative to the aircraft centre of gravity

Greek symbols

α	angle-of-attack
β	angle-of-sideslip
γ	flight path angle
δ_{DiCE}	aerodynamic directional control effector deflection angle
δ_{LaCE}	aerodynamic lateral control effector deflection angle
δ_{LoCE}	aerodynamic longitudinal control effector deflection angle
δ_{TF}	aerodynamic trim flap deflection angle
Δ	increment (perturbation) of a parameter; non-zero reference value
ζ_{sp}	short period damping ratio
$\ddot{\theta}$	instantaneous pitch acceleration during take-off rotation
τ	thrust available in percent
ϕ	bank angle
ϕ_{T_i}	vertical engine thrust line inclination angle, projection on xz-plane
ψ_{T_i}	horizontal engine thrust line inclination angle, projection on xy-plane
ω_{sp}	short period natural frequency

1.0 INTRODUCTION

This paper summarises the development objectives, system description, design capability and validation effort of a stability and control methodology. Research aim has been to transform the subject matter stability and control as today's highly advanced analysis discipline into an aircraft configuration independent (generic) design discipline, thereby providing an efficient control effector (CE) sizing capability during the early conceptual design phase not seen with current industry approaches. Overall, the true complexity of the research undertaking has been hidden in the objective to develop a generic (vehicle configuration independent) methodology concept. The requirement to size the primary control effectors of fixed wing subsonic to hypersonic designs, the inclusion of asymmetric aircraft layouts in addition to symmetric aircraft types of conventional and unconventional configuration layout, the requirement to analyse any aircraft geometry in symmetric and asymmetric flight conditions, such demands contributed significantly to the overall technical complexity and level of abstraction.

1.1 Characterisation of the conceptual design phase

The aerospace flight vehicle conceptual design phase is, in contrast to the succeeding preliminary and detail design phases, a critical step in the product development sequence due to its pre-defining function. However, this phase can be considered the least well understood part of the entire product development process due to its high level of abstraction and associated risk, multi-disciplinary design complexity, permanent shortage of design information available, and finally chronic time pressure to find solutions. Currently, important aircraft conceptual design decisions, like the sizing of primary control effectors, are still made using extremely simple analyses and heuristics. The two primary reasons for this scenario are first the absence of a modern disciplinary design tool capable of sizing control effectors performance-optimal throughout the flight envelope for classical and novel aircraft shapes (addressed here), and second the difficulty of synthesising the range of individual design disciplines in a timely manner in more than an ad-hoc fashion.

1.2 Problem statement: stability and control during the conceptual design phase

Most aircraft conceptual design environments in industry, research institutions, and academia consider stability and control a secondary rather than a primary design discipline. This fact is even more surprising when identifying its overall importance on flight safety, flight operation and certification. Clearly, there exists a wide discrepancy between the sophisticated approaches of the modern flight dynamicist during preliminary and detail design as compared to the traditional approach in use by aircraft designers during the conceptual design phase. Realising the multitude of performance-driven advancements in the field of modern aircraft design such as control configured vehicles (CCV), relaxed static stability (RSS) aircraft, morphing aircraft, the utilisation of techniques like fuel transfer and control allocation, it becomes obvious that the traditional and well established stability and control methods in use during initial design have stagnated in their evolution over virtually half of a century. As a consequence, there is a clear need to bridge the gap between the deficient stability and control design capability available during the early conceptual design phase, as compared with the sophisticated stability and control analysis proficiency available during the later preliminary to detail design phases where the overall flight vehicle arrangement is mostly frozen.

The technical advancements experienced over the last decades justify the following question: Has the traditional approach to stability and control at the conceptual design stage reached its limits of applicability? The present research investigation answers the above question in the affirmative. Clearly, only minimum research effort has been invested recently into the development of working methods for modern conceptual design applications. As a consequence, the theory and practice of stability and control at conceptual design lag behind modern performance design requirements. A remark from Root[1] in a publication from 1935 has not lost any of its sharpness today; "*...those who have studied the theory realise its inherent complication so far as any rapid application to design is concerned*." Clearly, the subject stability and control is one in which it is easy to wander off into elaborate mathematics while decoupling from its practical application.

The theory, methods, and processes currently available to design for adequate stability and control characteristics during the early conceptual design phase are not compatible with today's performance demands for advanced airframes of conventional but especially unconventional configuration layout. This leads to the quest of a renewed confluence of theory and practice in the field of flight vehicle stability and control. McRuer and Graham[2] and Obert[3] describe the distinctive historical separation between the scientists/theoreticians and tinkerers/inventors during the early eras

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TAC	Tail-Aft Configuration	symmetric
TFC	Tail-First Configuration	symmetric
TSC	Three-Surface Configuration	symmetric
FWC	Flying-Wing Configuration	symmetric
OWC	Oblique-Wing Configuration	asymmetric
OFWC	Oblique Flying-Wing Configuration	asymmetric

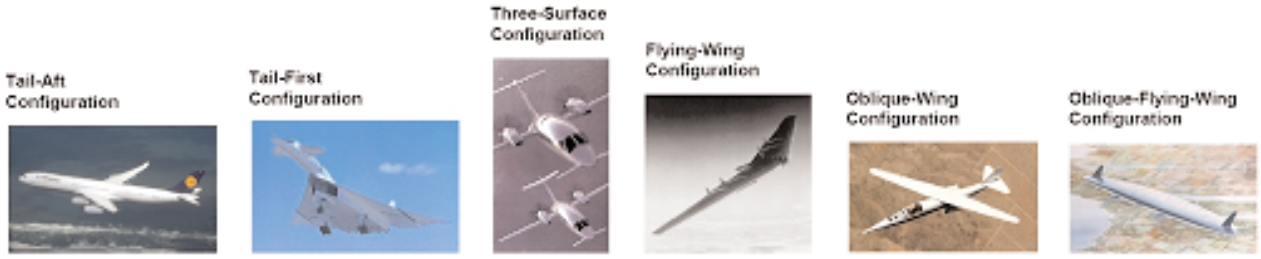


Figure 1. Range of relevant symmetric and asymmetric aircraft configurations.

of flight control development. The confluence of flight mechanics theory and practice is dated to around 1948 and was forced by the marked deficiencies in stability and control of the new jet aircraft, in particular the Northrop YB-49 control configured flying wing. McRuer and Graham remark that flight mechanics at the detail design level provides us with "*... awesome capabilities to compute, simulate, and, sometimes, to confuse. ... As a consequence, the analysts' physical means now often exceeds his mental grasp, and what he can compute may far exceed his understanding or appreciation. This can lead to an excessively empirical approach to design which is similar to the one used by the tinkerers thirty or more years ago...*"[2].

1.3 Methodology development motivation

What is the motivation and technical rationale to develop a generic stability and control methodology? Cook accentuates the integration of stability and control into conceptual design by calling for the renewed confluence of theory and practice in his 1999 paper titled The New Age of Flight Control[4]. Cook's outlook into the future has been motivation and justification for the present research investigation. '*So what of the future? It is no longer appropriate to regard the flight control system as a 'bolt-on' extra whose role is to rectify the legacy of stability and control deficiencies left by the airframe designer, and designers who continue to subscribe to this view simply have their 'heads in the sand'; and we all know about the flying qualities of the ostrich! The recognition by the Wright brothers of the critical importance of stability and control still applies today. The provision of good stability and control properties in an aeroplane must be at the top of the designer's agenda, for without that commitment the best laid plans will surely fail.*'

Advanced flight vehicle development and research programs like the B787 and A350, Blended-Wing-Body (BWB) flying wing aircraft, Supersonic Commercial Transport (SCT), Supersonic Business Jet (SSBJ), Single-Stage To Orbit (SSTO) vehicles, Two-Stages To Orbit (TSTO) vehicles and others, may offer promising performance potential but are particular demanding to assure certifiable but at the same time performance-optimal stability and control characteristics throughout the flight envelope. In order to enable an efficient comparison of the inherent design risks associated with conventional versus novel flight vehicles, and being able to unlock hidden design potential as early as possible in the design process, one has to strive for performance-optimal stability and control solutions (e.g. minimum trim drag, smallest control effectors, light-weight actuation system) resulting in true control-configured vehicles. The availability of a configuration-independent (generic) stability and control method for the early conceptual design phase is a prerequisite to enable consistent comparisons of wider ranges of flight vehicle configurations and their stability and control strategies on an impartial and rational basis. This finally helps rationalising the optimistic or skeptical 'opinions' in which advanced flight vehicle ideas are viewed today.

1.4 Development objectives

In an effort to address the problems stated above, a joint research undertaking has been initiated by several aircraft manufacturers in 1995 to develop a generic stability and control methodology. The principal investigator of this research project and first author of this paper was tasked to develop a methodology capable of consistently sizing control effectors of fixed-wing flight vehicles of conventional but in particular non-conventional (novel) configuration shapes during the conceptual design phase[5,6]. This paper presents an overview of the fully integrated and functional methodology.

The following illustrates the problem of performing meaningful stability and control work during the initial design phase. The primary control effectors (CE) of any fixed wing flight vehicle are not dimensioned for the relatively easy to quantify cruise design point requiring only minimum control power for trim, but are instead sized for the control power demanding 'grey-areas' of the flight envelope where non-linear aerodynamics, aerodynamic and inertia coupling, asymmetric flight conditions, and other complications do prevail. As a consequence, with the availability of the classical thus restricted CE sizing toolbox, the design team is usually challenged to generate or assume not readily available design data during the early conceptual design phase. Torenbeek[7] comments on the overall tail sizing problem: "*During the preliminary design stage, the tail surfaces may present one of the most difficult problems in the dimensioning of the main parts of the aircraft, and this, in turn, may lead to many iterations.*" Overall, it has been a clear development objective to arrive at a flight vehicle configuration-independent (generic) conceptual design methodology applicable to the range of flight vehicles shown in Fig. 1 with the consequent intent "*...things should be as simple as possible, but no simpler...*".

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2.0 METHODOLOGY BACKGROUND

2.1 Stability and control tasks

The primary control effectors (CEs), as installed in any flight vehicle type, have to secure the aircraft's intended flight path by providing adequate stability, control, and trim around each axis. Those tasks can only be realised when the complex balance between control power and the inherent airframe stabilities (static, dynamic, manoeuvre) are secured during the conceptual design phase. Over the decades of flight vehicle development, the inherently stable airframe of classical configuration arrangement has been transformed into an inherently indifferent or even unstable design of advanced configuration layout, incorporating powered controls, a flight control system, artificial feel, fly-by-wire signaling, and control allocation, just to mention some of the key technologies involved. Although not always recognised as such, stability and control is the single most critical requirement for flight safety and certification.

2.2 Stability and control techniques reviewed

Stability and control assessment in aircraft design can range from statistical volume coefficients and 1-DOF models (typical for the conceptual design phase) to full 6-DOF flight simulation based on the complete aerodynamic and inertia representation of the aircraft with integrated flight control law determination (typical for the detail design phase). However, the statistical volume coefficient approach restricts the designer to interpolate between earlier designs, thereby limiting design diversity. Reduced order models usually have not been developed for novel flight vehicles. In contrast, higher order flight simulation approaches during the preliminary to detail design phase require input data not available during the early conceptual design phase. Thus, neither of these extreme settings is acceptable for a consistent analysis and comparison of advanced conventional and unconventional aircraft during the conceptual design phase.

In the context of the present research investigation, the spectrum of stability and control tools, procedures, and best practices relevant for the conceptual design phase has been surveyed in detail and will not be repeated here[5,8]. From this investigation, one observation is the fact, that conceptual design oriented stability and control methods have stagnated in capability and development effort although the flight vehicle continues to evolve.

There is a clear trend to size the primary control effectors by using statistical data, resulting in the determination of volume coefficients for the longitudinal and directional CEs as a first step. If practiced at all in follow-on steps, static stability, dynamic stability, and selected quasi-static design-critical flight conditions crosscheck the initial tail volume assumptions using a non-consistent library of reduced order models. Most importantly, those 'classical' CE sizing procedures have evolved for the conventional tail-aft configuration (TAC) only, therefore are based on well-known design principles. The resulting CE hardware design proposals are intended to be conservative with the aim to cover all design-critical operational eventualities. This subsequently leads to over-engineered individual CEs, resulting in an overall performance-compromised aircraft. Unfortunately, it regularly happens that CEs are ill-dimensioned (e.g. too small), having overlooked a particular design-critical flight condition pertinent to this specific aircraft type. This may result in design-parameter coupling effects not captured by the available statistical database, adversely altering the stability and control characteristics while being invisible to the designer. A multitude of historical aircraft projects document the late discovery of deficient stability and control characteristics, thereby translating into non-reparable operational limitations. When reflecting such end-of-the-life-cycle scenario back to the responsible pre-defining conceptual design phase, it is interesting to note that the 'classical' thus accepted CE sizing procedures are mostly applicable to TACs only. In addition, the commonly accepted approaches are usually limited to statically stable designs only, thus fails to provide reliable design solutions for indifferent or unstable designs around any axis. As a consequence, the statistics and reduced order based approaches rarely offer any reliable design guidance when diverting from the classical TAC towards unconventional aircraft arrangements.

2.3 Methodology development specification

At the launch of the present research undertaking in the mid 1990s, the procedures proposed and implemented by Lee *et al*[9] and Kay *et al*[10] represented the 'state of the art' in addressing stability and control during the conceptual design phase[5,8]. Both implementations already couple vortex lattice aerodynamic prediction with reduced-order stability and control analysis to test a vehicle's control power and stability over a range of flight conditions. Although both approaches already implement aerodynamic prediction tools of generic nature, the stability and control analysis is still highly non-generic, thereby limited to symmetric TAC and symmetric flight conditions due to the decoupling of the longitudinal from the lateral-directional motions. This poses a particular dilemma since CEs are naturally sized at the extremities of the flight envelope. Consequently, the commonly adopted symmetry requirement in stability and control analysis prevents consistent dimensioning of the primary CEs of symmetric conventional and unconventional aircraft configurations in asymmetric flight conditions. Overall, the approaches by Lee *et al* and Kay *et al* have been the starting point for the development of the generic stability and control methodology.

Having researched and understood the CE sizing dilemma[5], it has been specified that the methodology be capable of trimming, stabilising, and controlling the flight vehicle in six-degrees-of-freedom. The control allocation problem has to be addressed. Airframes of stable, indifferent, or unstable layout have to be accessible by emulating a flight control system (FCS) capable of restoring airframe stiffness and airframe damping. The aerodynamic control effectors (CEs) have to be defined by specifying the volume coefficient (geometry), stability derivatives (aerodynamics), and the deflection angles (operation) required to comply with pre-defined design and certification requirements. Overall, the methodology is required to balance control power with inherent airframe stability levels in a performance-optimal way, spanning from the 'conventional' symmetric tail-aft configuration (TAC) to the 'unconventional' asymmetric oblique flying-wing configuration (OFWC) of subsonic to hypersonic designs.

In general, the method has to be functional in two different calculation modes, if required. Although its main capability and strength is accessible when integrated into a multidisciplinary design environment, it must be possible to execute it in the stand-alone mode which is the principal mode for validation and calibration of the calculation routines.

3.0 SYSTEM DESCRIPTION

3.1 Methodology

Accepting the methodology development specification as a hard guideline and target at the same time, a research strategy has been devised and systematically executed with the aim to assemble industry-rated state-of-the-art understanding of the interdependent subject matters related to stability and control: a) geometry and mass properties, b) aerodynamics, and c) flight test expertise and associated sub-topics[5,8,11,12,13,14]. The outcome has been a unique conceptual design methodology coupling non-linear generic aerodynamic prediction with a generic stability and control algorithm for the rapid assessment of:

1. **trimmed aerodynamics** in all axes for symmetric and asymmetric flight conditions and flight vehicles, thereby enabling vehicle sizing and performance assessments;
2. **control power requirements** governed by key design-constraining flight conditions (DCFCs) identified throughout the

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mission profile, enabling performance-optimal sizing of the primary control effectors (CEc) consisting of the (a) longitudinal control effector(s) (LoCE), (b) directional control effector(s) (DiCE) and (c) lateral control effector(s) (LaCE);

3. **static and dynamic stability** assessment in all axes to enable true control-configured vehicle (CCV) design by assessing the aircraft's open and closed loop flying qualities, flight control system (FCS) emulation for the estimation of first-order feedback gains and associated control power requirements for stability augmentation (SAS).

The enabling methodology consists of six logic modules:

(1) input,

(2) aerodynamic prediction,

(3) steady state control analysis,

(4) trimmed aerodynamics,

(5) static, dynamic, and manoeuvre stability, and

(6) output. Each of these modules has been developed to keep the tool as generic as possible to ensure applicability to a wide variety of aircraft configurations.

The top level organisation scheme is presented with Fig. 2. The sequence of input-analysis-output is systematically executed for each individual design test case defined for a particular design constraining flight condition (DCFC) (e.g., steady straight line flight), formulated for a particular mission segment (e.g., climb segment defined with altitude, speed, etc.). Any number of design test cases can be defined for each individual DCFC by specifying permutations of aircraft configuration setting (CS) (e.g., flap setting, trim setting) and failure conditions (FC) (e.g., loss of left aileron, loss of one hydraulic system) at which the aircraft's limits and requirements are evaluated. The vehicle's primary control effectors are then sized iteratively to meet the requirements specified. The six input-analysis-output components are discussed below.

3.2 Methodology initiation

The initiation of the methodology consist of three parts:

(1) aircraft data and evaluation matrix defining for each mission segment the group of design-constraining flight conditions (DCFC) consisting of a set of individual test cases,

(2) control allocation scheduling, and

(3) aerodynamic input, see Fig. 3.

(1) *Aircraft Data and DCFC*: This information defines the aircraft model and sweep of design test cases the aircraft will be tested for.

The aircraft definition:

- **geometry** (b_{ref}, c_{ref}, S_{ref}, etc.);
- **propulsion** (T_i, ψT_i, φT_i, x_T, y_T, z_T);
- **weight and balance** (W, $x_{c.g.}$, I_x, I_y, I_z, I_{xz}, I_{xy}, I_{yz}).

The DCFC definition:

- mission segment flight condition variables (h, M, etc.);
- DCFCs tested (steady-state straight-line flight, steady-state turning flight, steady-state pull-up push-over flight, steady-state rolling performance, quasi steady-state take-off rotation) consist each of a set of design test cases that can be examined sequentially;
- vehicle design and certification requirements and constraints ($\omega_{s.p.}$, $\zeta_{s.p.}$, $\delta_{LoCE\ max}$, $\delta_{LaCE\ max}$, $\delta_{DiCE\ max}$, etc.).

(2) *Control Allocation*: Some aircraft configurations utilise redundant CEs to improve a particular performance objective[5]. For example, the three-surface configuration (TSC) Piaggio P180 Avanti utilises both a canard and tailplane to trim to a range of angles-of-attack and centre-of-gravity locations at constant velocity and altitude at which a specific performance objective can be obtained by varying the longitudinal load distribution between the canard, wing, and horizontal tail accordingly, see Fig. 4. In contrast, traditional tail-aft aircraft (TAC) show less opera-

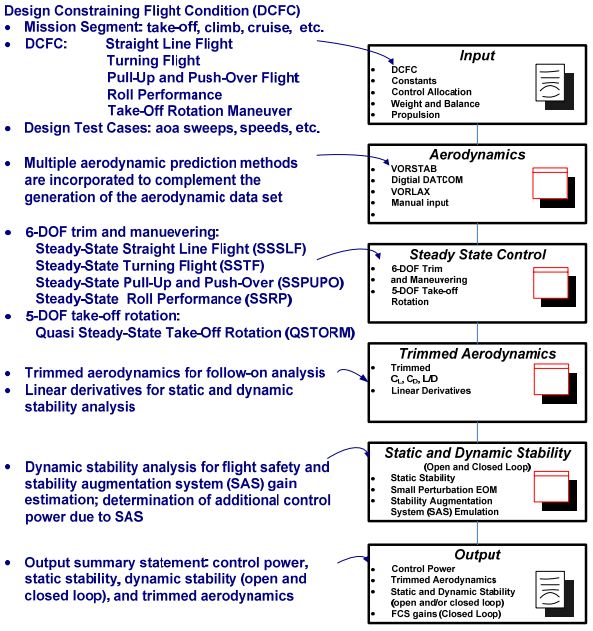


Figure 2. Overall methodology.

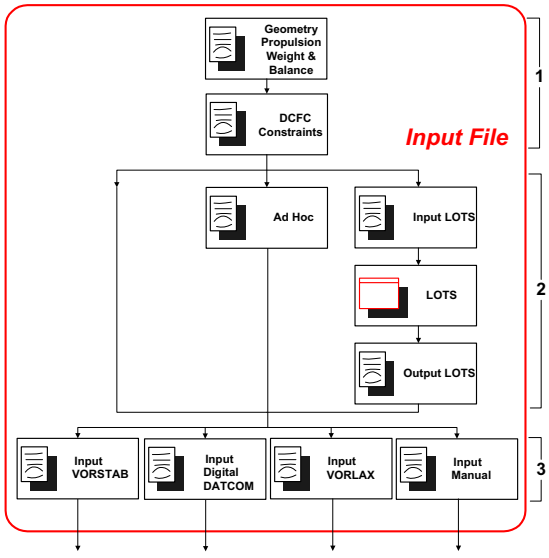


Figure 3. Methodology initiation.

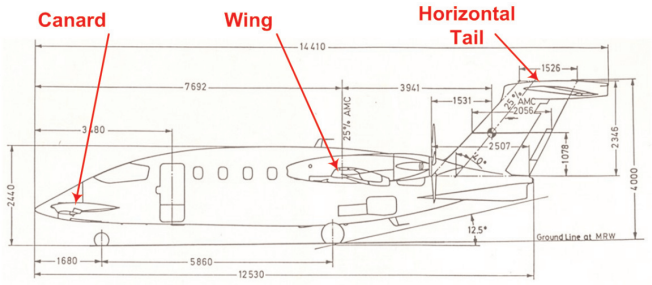


Figure 4. Three-surface configuration (TSC), Piaggio P180 Avanti.

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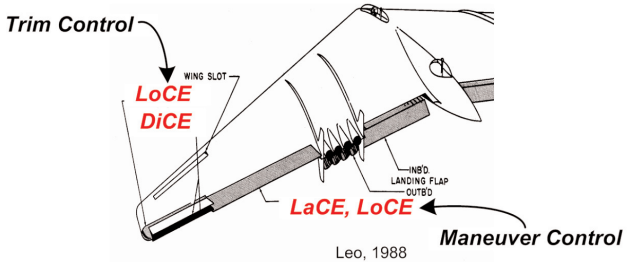


Figure 5. Longitudinal control allocation scheme implemented with the Northrop YB-49.

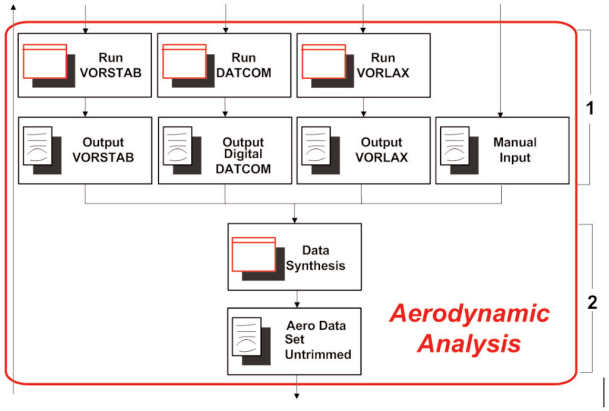


Figure 6. Aerodynamic methodology.

Table 1
Aerodynamic data organisation

Static Derivatives	Dynamic Derivatives	LoCE Increments	LaCE Increments	DiCE Increments
$C_L = f(\alpha,\beta)$	$C_{L_\alpha} = f(\alpha,\beta)$	$\Delta C_L = f(\alpha,\delta)$	$\Delta C_L = f(\alpha,\delta)$	$\Delta C_L = f(\alpha,\delta)$
$C_D = f(\alpha,\beta)$	$C_{m_\alpha} = f(\alpha,\beta)$	$\Delta C_D = f(\alpha,\delta)$	$\Delta C_D = f(\alpha,\delta)$	$\Delta C_D = f(\alpha,\delta)$
$C_m = f(\alpha,\beta)$	$C_{L_q} = f(\alpha,\beta)$	$\Delta C_m = f(\alpha,\delta)$	$\Delta C_m = f(\alpha,\delta)$	$\Delta C_m = f(\alpha,\delta)$
$C_{Y_\beta} = f(\alpha,\beta)$	$C_{m_q} = f(\alpha,\beta)$	$\Delta C_Y = f(\alpha,\delta)$	$\Delta C_Y = f(\alpha,\delta)$	$\Delta C_Y = f(\alpha,\delta)$
$C_{l_\beta} = f(\alpha,\beta)$	$C_{Y_p} = f(\alpha,\beta)$	$\Delta C_l = f(\alpha,\delta)$	$\Delta C_l = f(\alpha,\delta)$	$\Delta C_l = f(\alpha,\delta)$
$C_{n_\beta} = f(\alpha,\beta)$	$C_{l_p} = f(\alpha,\beta)$	$\Delta C_n = f(\alpha,\delta)$	$\Delta C_n = f(\alpha,\delta)$	$\Delta C_n = f(\alpha,\delta)$
	$C_{n_p} = f(\alpha,\beta)$			
	$C_{n_r} = f(\alpha,\beta)$			
	$C_{l_r} = f(\alpha,\beta)$			
	$C_{n_r} = f(\alpha,\beta)$			

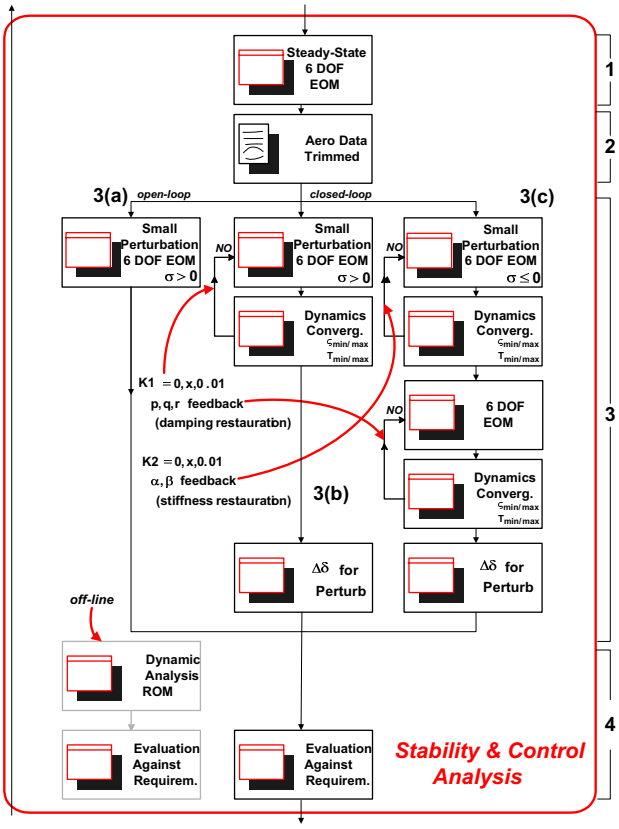


Figure 7. Stability and control methodology.

Table 2
Steady-state equation of motion assumptions

	Assumptions
1	The Earth s treated as flat and stationary in inertia space.
2	Equations are valid for any orthogonal axis system fixed at the c.g. of the aircraft.
3	Aircraft is a rigid body having any number of rigid spinning rotors.
4	Spinning rotors have a constant angular velocity relative to the body axis.
5	Wind velocity is zero.
6	Steady state flight, no acceleration.
7	No plane of symmetry, i.e. full moment of inertia matrix. I_{xy} and I_{zy} are included.
8	Aerodynamic cross coupling included.

tional flexibility due to the availability of only a single auxiliary CE. Consequently, the minimum induced trim drag design point is obtained at only one specific angle-of-attack and centre-of-gravity location at constant velocity and altitude. Thus, the three-surface configuration (TSC) presents an over-determined system (due to the second auxiliary horizontal control effector) for longitudinal stability, control, and trim, thereby giving the designer the freedom to schedule the over determined LoCE's to either minimise trim drag, minimise control surface deflections, etc. The logic chosen to schedule the longitudinal control effectors is referred to as control allocation.

The methodology addresses the control allocation issue by utilising two methods. First, the designer can manually schedule the longitudinal, directional and lateral CEs based on experience. In the methodology this is referred to as the '*ad hoc*' method[5]. Second, the Linear Optimum Trim Solution (LOTS) methodology[15] solves for a minimum longitudinal trim drag solution, applicable to three-surface configurations (TSC) and two-surface configurations (TAC, TFC) with thrust vectoring. The implementation adopted in the present context uses a Lagrange formulation to obtain a longitudinal load distribution that minimises a quadratic trim drag model[10].

Referring to the Northrop YB-49 flying wing case study, longitudinal control allocation is required due to the availability of two independent longitudinal CEs located at the trailing edge of the wing, being the elevon and trim flap on the outer wing panel, see Fig. 5. The YB-49 uses the trim flap to trim the vehicle longitudinally for straight-line flight, whereby the elevons are used for manoeuvreing control (pitch and roll). This manual allocation of the longitudinal CEs represents the 'ad-hoc' approach in scheduling the CEs.

(3) *Aerodynamics*: The designers can opt for four selected methods providing aerodynamic input: (1) VORSTAB[16], a non-linear vortex lattice method; (2) Digital DATCOM[17], a digital semi-empirical handbook method; (3) VORLAX[18], a linear vortex lattice method; (4) manual data input (aerodynamic look-up tables); (5) any combination of the above. The input files for VORSTAB, Digital DATCOM and VORLAX are assembled based upon the geometry of the aircraft. These tools are utilised to develop the initially un-trimmed aerodynamic data set.

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3.3 Aerodynamic analysis

The conceptual design phase places a premium on low-fidelity aerodynamic simulation tools that can be rapidly applied with the limited geometric data inevitably available. Clearly, non-linear potential flow methods are considered most appropriate for the estimation of aerodynamic coefficients and derivatives. As detailed in Refs 5 and 19, VORSTAB has been chosen as the primary aerodynamic estimation tool for the methodology, justified by its unique ability to handle a wide variety of flight vehicle configurations coupled to its superior non-linear aerodynamic modeling capability. This rare ability is critical for the sizing task of aerodynamic control effectors (CEs) which are typically dimensioned to meet stability and control requirements in the 'non-linear corners' of the flight envelope (e.g., final approach at low dynamic pressure and high angles of attack (aoa)).

However, the non-linear VLM VORSTAB challenges with certain limitations as seen with every aerodynamic estimation code. The primary drawback is its inability to model unsteady aerodynamics, a limitation inherent to any vortex lattice method (VLM). Thus, critical derivatives for dynamic analysis, such as $C_{m_{\dot{q}}}$ and $C_{m_{\dot{\alpha}}}$, are not computed. These derivatives have a significant effect on the longitudinal damping of certain aircraft configurations such as the tail-aft configuration (TAC), thus must be estimated before dynamic stability can be addressed. In addition, VORSTAB is restricted in modeling asymmetric flight vehicle configurations consisting of combinations of vertical surfaces and a fuselage with an asymmetric wing layout, although the oblique flying wing (OFWC) can be assessed. To overcome some of these inherent deficiencies, VORSTAB is complemented with Digital DATCOM and VORLAX. Digital DATCOM is utilised to generate the unsteady aerodynamic derivatives and serves to model other control effecter types, whereby VORLAX contributes to model asymmetric vehicle configurations such as the oblique wing configuration (OWC). For information related to individual modeling capabilities of VORSTAB, Digital DATCOM and VORLAX, see Refs 5 and 20. In addition to this complimentary set of aerodynamic estimation tools, the methodology allows for manual input of aerodynamic data in a generic data file, enabling users to utilise numerical aerodynamic information from miscellaneous sources. The organisation of the aerodynamic data generation is detailed in Fig. 6.

(1) *Aerodynamic Analysis*: The three codes are operated in different ways and thus require their own unique integration schemes, see Ref. 46.

(2) *Data Synthesis*: Once the appropriate aerodynamic data has been generated, the data contained in the separate output files for the individual methods are combined into a consistent untrimmed aerodynamic input file (lookup tables) for follow-on analysis. At a set Mach number, coefficients and derivatives are tabulated as a function of angle-of-attack, side-slip angle and control surface deflection angles, see Table 1. This organisation includes all the typical aerodynamic effects as well as the aerodynamic cross coupling effects that become significant for asymmetric vehicles but in particular asymmetric flight conditions of any aircraft configuration.

3.4 Stability and control analysis

At the heart of the methodology is the generic stability and control analysis module. This module is based upon a generic mathematical framework consisting of steady state and dynamic analysis[5,6,21]. The stability and control analysis methodology is shown in Fig. 7 and contains four basic steps:

(1) solve trim equations of motion to determine control power available,

(2) produce trimmed aerodynamics,

(3) solve the small perturbation equations of motion, and

(4) compare results to requirements.

(1) *Steady State 6-DOF Trim Equations of Motion*:
Assessing a vehicle's control power requires examining the critical flight conditions and manoeuvres that impose the greatest demand on the control effectors; these have been termed design-constraining flight conditions (DCFC)[5,6]. In general, any aircraft experiences design-critical flight conditions in the form of five basic manoeuvres throughout its mission profile:

- **Straight-Line Flight**: Sustained flight at a constant heading and glide path angle (cruise, climb, descent, one-engine inoperative, crosswind landing, sideslip conditions, etc.).
- **Pull-Up/Push-Over Manoeuvre**: longitudinal motion pitching motion in the nose-up or nose-down direction (speed recovery, load factor manoeuvreing capability, tailplane mis-trim, etc.).
- **Roll Performance**: rolling motion about the body or stability axis primarily induced by the LaCE (time to roll, roll acceleration, adverse yaw coupling, etc.).
- **Turning Flight**: inducing and maintaining a turning motion characterised by a change of heading. (horisontal flight turn coordination, climbing and descending flight turn coordination, sideslip conditions, etc.).
- **Take-Off Rotation**: a pitching motion initiated by the LoCE during the take-off run to lift the nose wheel off the ground (pitch rate and acceleration, failure conditions, etc.).

The methodology models these DCFC manoeuvres by solving the coupled (both aerodynamic and inertial) steady-state 6-DOF equations of motion. These equations are based upon a set of custom derived *General Euler Equations of Motion with Spinner Rotors*[5,6,20,21,22,23] with the assumptions listed in Table 2. Equations (1) to (6) show the set of equations written in stability axis.

$$\left(-C_{T_D}\tau + C_D(\alpha,\beta) + C_{D_q}(\alpha,\beta)\frac{q\bar{c}}{2V} + \Delta C_{D_{LoCE}}(\alpha,\delta_{LoCE}) + \Delta C_{D_{LaCE}}(\alpha,\delta_{LaCE}) + \Delta C_{D_{DiCE}}(\alpha,\delta_{DiCE})\right)\bar{q}S_{ref} - \\ - mg\sin\gamma - mrV\sin\beta = 0 \quad \ldots(1)$$

$$\left(C_{T_Y}\tau + C_Y(\alpha,\beta) + C_{Y_r}(\alpha,\beta)\frac{rb}{2V} + C_{Y_P}(\alpha,\beta)\frac{pb}{2V} + \Delta C_{Y_{LoCE}}(\alpha,\delta_{LoCE}) + \Delta C_{Y_{LaCE}}(\alpha,\delta_{LaCE}) + \Delta C_{Y_{DiCE}}(\alpha,\delta_{DiCE})\right)\bar{q}S_{ref} - \\ - mg\cos\gamma\sin\phi - mrV = 0 \quad \ldots(2)$$

$$-\left(-C_{T_L}\tau + C_L(\alpha,\beta) + C_{L_q}(\alpha,\beta)\frac{q\bar{c}}{2V} + \Delta C_{L_{LoCE}}(\alpha,\delta_{LoCE}) + \Delta C_{L_{LaCE}}(\alpha,\delta_{LaCE}) + \Delta C_{L_{DiCE}}(\alpha,\delta_{DiCE})\right)\bar{q}S_{ref} \\ - mg\cos\gamma\cos\phi + mqV - mpV\sin\beta = 0 \quad \ldots(3)$$

$$\left(C_{T_l}\tau + C_l(\alpha,\beta) + C_{l_r}(\alpha,\beta)\frac{rb}{2V} + C_{l_p}(\alpha,\beta)\frac{pb}{2V} + \Delta C_{l_{LoCE}}(\alpha,\delta_{LoCE}) + \Delta C_{l_{LaCE}}(\alpha,\delta_{LaCE}) + \Delta C_{l_{DiCE}}(\alpha,\delta_{DiCE})\right)\bar{q}S_{ref}b + \\ + I_{yz}(q^2 - r^2) + I_{zx}pq - I_{xy}rp + (I_y - I_z)qr - qh'_z + rh'_y = 0 \quad \ldots(4)$$

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$$\left(C_{T_m}\tau + C_m(\alpha,\beta) + C_{m_q}(\alpha,\beta)\frac{q\bar{c}}{2V} + \Delta C_{m_{LoCE}}(\alpha,\delta_{LoCE}) + \Delta C_{m_{LaCE}}(\alpha,\delta_{LaCE}) + \Delta C_{m_{DiCE}}(\alpha,\delta_{DiCE})\right)\bar{q}S_{ref}\bar{c} + \quad \ldots (5)$$
$$+ I_{zx}(r^2 - p^2) + I_{xy}qr - I_{yz}pq + (I_z - I_x)rp - rh'_x + ph'_z = 0$$

$$\left(C_{T_n}\tau + C_n(\alpha,\beta) + C_{n_r}(\alpha,\beta)\frac{rb}{2V} + C_{n_p}(\alpha,\beta)\frac{pb}{2V} + \Delta C_{n_{LoCE}}(\alpha,\delta_{LoCE}) + \Delta C_{n_{LaCE}}(\alpha,\delta_{LaCE}) + \Delta C_{n_{DiCE}}(\alpha,\delta_{DiCE})\right)\bar{q}S_{ref}b + \quad \ldots (6)$$
$$+ I_{zx}(p^2 - q^2) + I_{yz}rp - I_{zx}qr + (I_x - I_y)pq - ph'_y + qh'_x = 0$$

These equations are distinctive in that they account for

(1) aerodynamic cross-coupling,

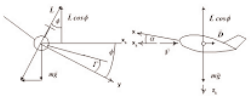
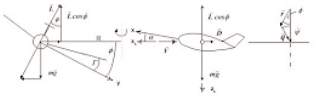
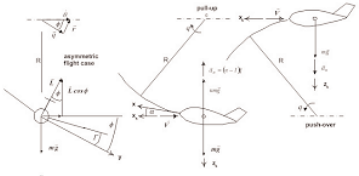
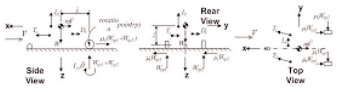
(2) inertial cross-coupling, and

(3) spinning rotor cross-coupling.

These effects are typically neglected during the conceptual design phase due to the small effect on symmetric vehicles like the TAC at symmetric flight conditions. However, those effects are significant when analyzing symmetric configurations in asymmetric DCFC (e.g., maximum side-slip, one-engine inoperative) and asymmetric configurations such as the oblique flying wing (OFW). Thus, the set of equations can be considered generic trim equations of motion (EOM) because they can be applied to both symmetric and asymmetric vehicles and symmetric and asymmetric flight conditions, allowing the designer to explore configurations at flight conditions not mastered by traditional stability and control tools during the conceptual design phase.

To solve the generic trim EOMs for each of the five basic manoeuvres described previously, the following steps are usually taken:

Table 3
Steady-state manoeuvre (DCFC) overview[5]

Description	Diagram
Steady-State Straight-Line Flight	
Steady-State Turning Flight	
Steady-State Pull-Up/Push-Over Flight	
Steady-State Roll Performance	
Quasi-Steady State Take-Off Rotation	

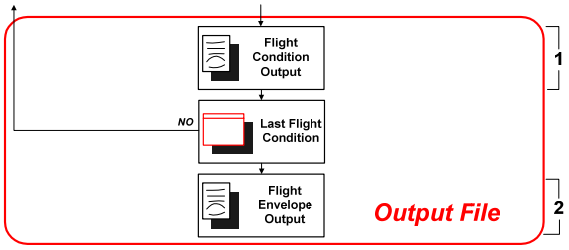


Figure 8. Deliverables definition.

1. The steady-state pitch (q), roll (p) and yaw (r) rates are typically specified for each manoeuvre. These are defined from the kinematics equations. For example, during steady-state straight line flight $q = p = r = 0$.
2. The appropriate attitude (α, β, ϕ, *etc.*) and control surface deflections (δ_{LoCE}, δ_{LaCE}, δ_{DiCE}) required are solved iteratively for the prescribed angular velocities.

The five basic steady-state manoeuvres (DCFCs) are graphically depicted in Table 3. For each DCFC, multiple permutations (design test cases) can be defined including sweeps of sideslip angle, speed, altitude, failure conditions, configuration setting, etc., to reliably size the primary control effectors. For more information on design-relevant DCFC, see Refs 5 and 20.

(2) *Trimmed Aerodynamics*: The methodology first solves for the steady state trim condition, second produces trimmed aerodynamics, and third addresses the dynamic behavior of the vehicle. Trimmed aerodynamics is essentially required for correct control power assessment, dynamic stability analysis around a trim point, accurate performance calculations leading to realistic comparison of different flight vehicle configurations (e.g., TAC *vs* FWC) based on trimmed *L/D*. In particular, the methodology generates this data by

1. solving for the 6-DOF steady-state straight line flight angle-of-attack, side-slip angle, and control surface deflections required (control power assessment);
2. interpolating the trimmed aerodynamic data from the untrimmed aerodynamic data set at the conditions defined in the 6-DOF steady-state straight line flight calculations;

Specification	
Span	172 ft
Wing Area	4,000 ft²
Airfoil Section	NACA 64A019 (root) NACA 64A018 (tip)
Max TO Weight	196,000 lbs
Max speed	430 kts
Range	3,000 to 4000 nm
Propulsion	8 Allison J-35-A-15 Turbojets
Thrust	4,000 lbs each (SL)

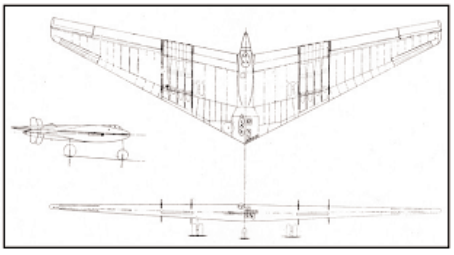


Figure 9. Three-view of the Northrop YB-49.

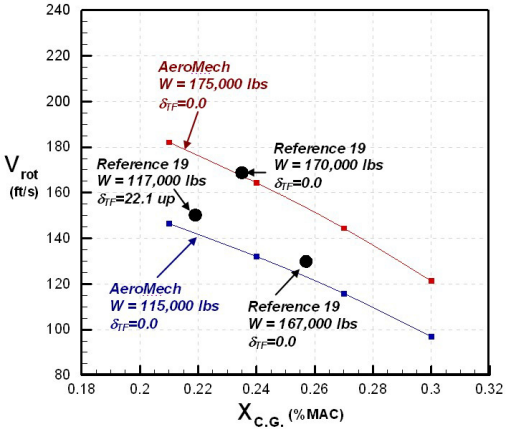


Figure 10. Effects of relaxed longitudinal static stability on the YB-49 take-off rotation speed.

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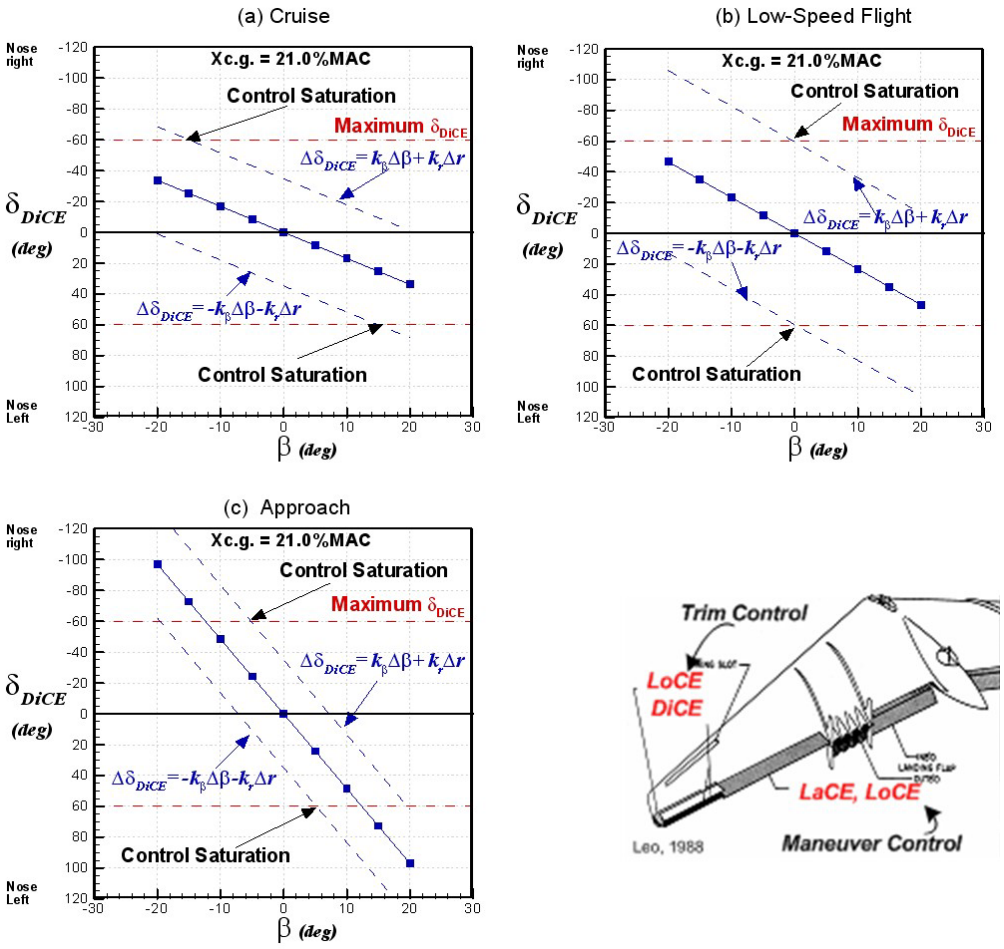


Figure 11. YB-49 DiCE (drag rudder) trim deflection required during cruise, low-speed flight, and approach.

3. calculating the linear derivatives about the trim point using a centre difference method.

(3) *Dynamic Stability and Control Analysis*: With the trim point and linear trimmed aerodynamics around this point defined, the designer predicts the vehicle dynamic behavior using the small perturbation equations of motion. It is important to quantify a vehicle's dynamic stability characteristics early during the conceptual design phase, thereby providing timely design feedback to enforce compliance with flight safety and performance requirements. In the case of aircraft with relaxed static stability in any axis, a stability augmentation system (SAS) is emulated for the axis concerned, thereby estimating the additional control power required due to the SAS function.

References 5, 20 and 21 discuss the derivation and application of the coupled small perturbation equations of motion for conceptual design applications which retain the aerodynamic, inertial, and control cross coupling terms, all being of significance for asymmetric flight conditions. For validation purposes, the present context only discusses implementation results generated with the decoupled small perturbation equations of motion. Considered to be an interim development step, the decoupled formulation currently provides a higher level of physical visibility and experience database for aircraft in symmetric and asymmetric flight conditions, thus serves as a datum to determine the magnitude of the cross-coupling effects found when compared with effects estimated using the fully coupled formulation. Reference 46 discusses available implementations of the classic decoupled small-perturbation EOM, resulting in the selection of the formulation provided by Abzug[24]. This code combines the classic decoupled (longitudinal and lateral/directional) small perturbation equations of motion with an analogue flight controller model to compute the Eigenvalues of the total system. The analog controller model allows the conceptual designer to estimate additional control power requirements due to the need for a stability augmentation system (SAS). Although the detailed choice of the controller architecture depends on the particular aircraft type under consideration, the purpose of the tool in the present context is not to design an advanced flight control system for the aircraft, but to emulate only a controller for the estimation of control power requirements. For additional information see Refs 8, 20, and 24.

When designing a high-performance aircraft, the designer has to determine if a stability augmentation system (SAS) is a requirement, and if so, what level of complexity is required with implications on control power, system complexity, weight and cost issues, and others. Since the degree of longitudinal and lateral/directional static stability is a primary design variable, the methodology provides for each axis separately three analysis options: 3(a) open loop, 3(b) closed loop damping restoration, and 3(c) closed loop stiffness and damping restoration, see Fig. 7.

3(a) *Open Loop*: If no stability augmentation is desired, then only the open loop behavior of the vehicle is determined. In this case, the vehicle dynamic behavior can only be altered through adjusting the airframe geometry and mass properties.

3(b) *Closed-Loop Damping Restoration* (SAS): This branch represents the stability augmentation system (SAS) function for an

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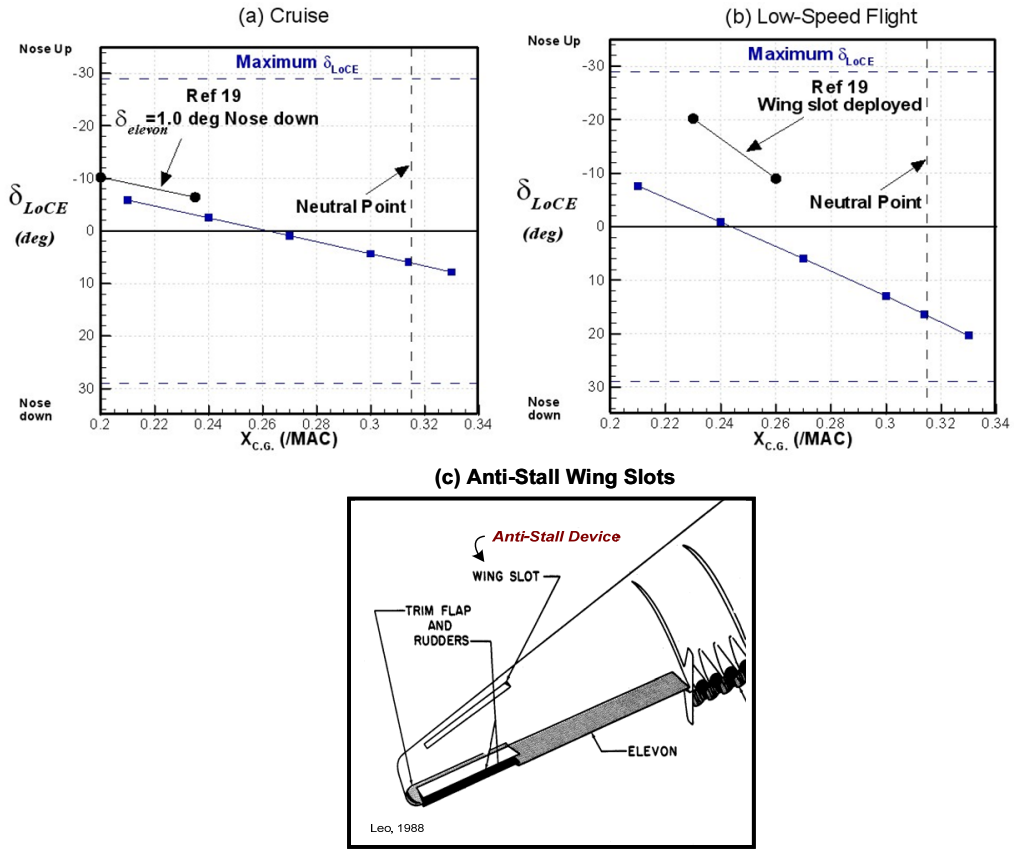


Figure 12. Comparison of generated steady-state straight line flight 1-g trim LoCE deflection results with YB-49 flight test data for the (a) cruise and (b) low-speed mission segments; (c) YB-49 wing slot anti-stall device deployed with landing gear.

aircraft with relaxed static stability. In this case the vehicle is still statically stable but does not meet certain dynamic stability requirements. Thus, pitch, roll and yaw feedback can be used to artificially increase dynamic stability until the requirements are met. These feedback gains require additional control power as seen with Equations (7) through (9). From a predefined perturbation from trim, additional restoring control surface deflection is estimated with;

$$\Delta\delta_{LoCE} = \Delta q \cdot K_q \quad \ldots (7)$$

$$\Delta\delta_{LaCE} = \Delta p \cdot K_p \quad \ldots (8)$$

$$\Delta\delta_{DiCE} = \Delta r \cdot K_r \quad \ldots (9)$$

3(c) *Closed-Loop Stiffness and Damping Restoration* (SAS): This branch represents the SAS requirements for an inherently unstable aircraft in any axes if required. Such a vehicle requires the SAS to first artificially restore stiffness and then damping. This is to be accomplished by (a) angle-of-attack and side-slip feedback to restore stiffness, then (b) pitch, roll, and yaw feedback to restore damping accordingly. Additional control power is required for this type of stability augmentation as shown with Equations 10 through 12. In practice, angle-of-attack and side-slip feedback should be used sparingly since angle-of-attack and side-slip angle sensors are prone to noise.

$$\Delta\delta_{LoCE} = \Delta q \cdot K_q + \Delta\alpha \cdot K_\alpha \quad \ldots (10)$$

$$\Delta\delta_{LaCE} = \Delta p \cdot K_p \quad \ldots (11)$$

$$\Delta\delta_{DiCE} = \Delta r \cdot K_r + \Delta\beta \cdot K_\beta \quad \ldots (12)$$

3.5 Methodology deliverables definition

When combining the modules estimating steady state control power, trimmed aerodynamics, and static and dynamic stability and associated control power requirements, a large and diverse data set can be produced quickly. Proper organisation and visualisation of this data is required to provide meaningful information to the conceptual designer. To accomplish this, the methodology deliverables definition is organised into two principal parts: a) individual DCFC output and b) summary output over the entire flight envelope, see Fig. 8.

Each DCFC analysed has an individual detailed output file, providing information related to control power and stability characteristics for each individual design test case. Having executed all of the design-constraining flight conditions (DCFC) throughout the mission segments, a summary flight envelope output file is created,

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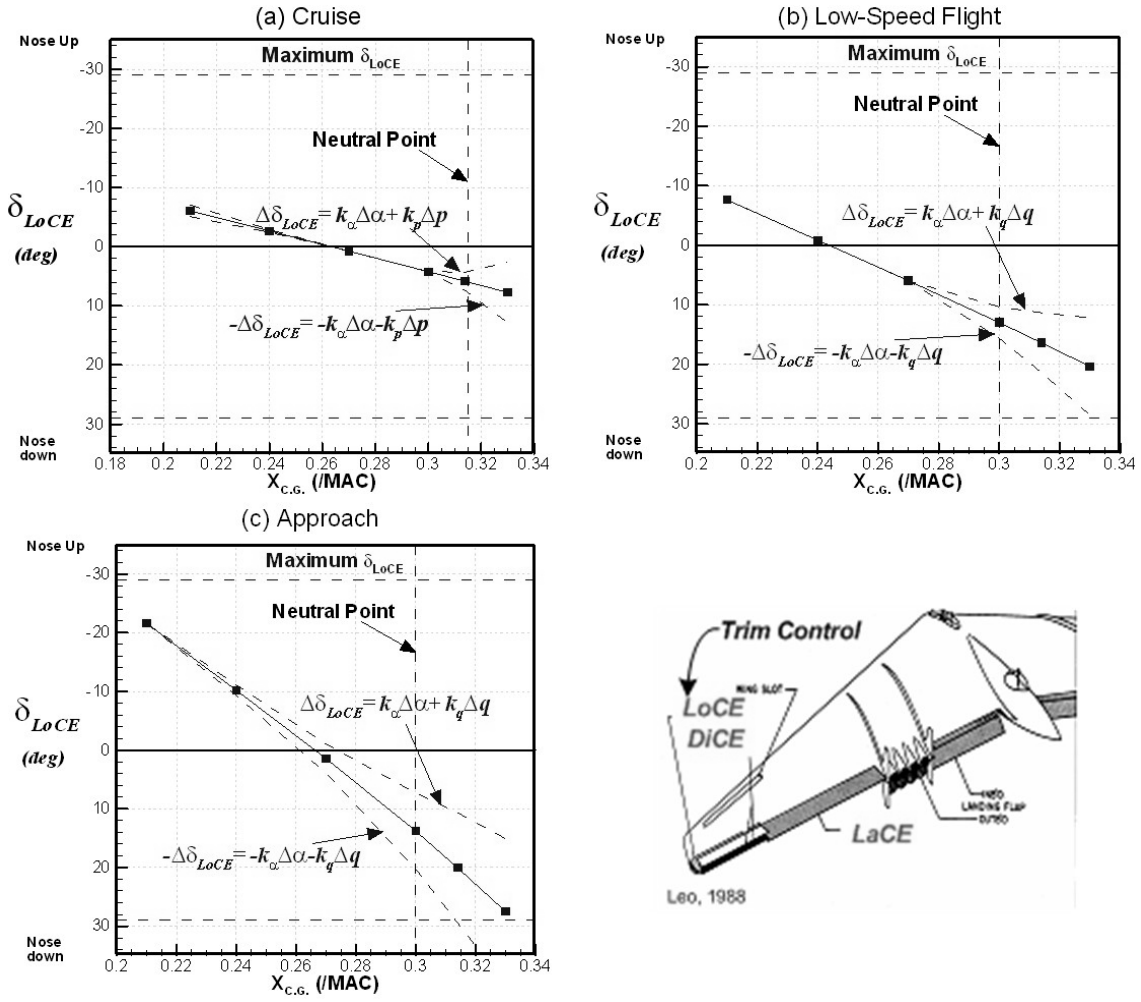


Figure 13. Effects of relaxed longitudinal static stability on the required trim flap deflection angle.

detailing how control power and stability characteristics change over the flight envelope. This enables the designer to finally identify the design critical corners of the flight envelope, where either control power is lacking or marginal, where poor flying and handling qualities can be expected, where it is recommended to review initial design decisions. Having generated such information enables the designer to incorporate informed design modifications which have been explored via trade and optimisation studies.

4.0 VALIDATION EXAMPLE: NORTHROP YB-49 FLYING WING

4.1 Design case study selected

The Northrop YB-49 flying wing design has been selected as a design case study to demonstrate the application, overall correctness, and accuracy of the data generated. The flying wing configuration (FWC) is of particular interest:

1. because this highly integrated configuration arrangement is, from a design perspective, more demanding compared to the disintegrated TAC, thereby enables to demonstrate the methodology's advanced and generic (configuration-independent) analysis capability;
2. because of its prospective performance potential as a next-generation efficient civil and/or military future transportation system beyond the potential seen with the highly evolved TAC;
3. because of the availability of certain geometry, performance, and stability and control flight test data[25,26,27,28] for validation purposes.

The Northrop YB-49 specification is provided with Fig. 9. This study concisely analyses three design-constraining flight conditions (DCFC) discussing the original YB-49 airframe as a point design. In addition, the design parameter solution space is quantified and visualised beyond the original YB-49 point design via design trade studies. However, this only abbreviated and therefore incomplete discussion of the YB-49 design does not necessarily propose final design recommendations related to neither the vintage YB-49 nor a future blended-wing body FWC, although certain design conclusions can be easily drawn. A dedicated Northrop YB-49 design assessment and resulting design guidelines for future FWCs are reserved for a follow-on study.

The quantification of the following three YB-49 specific DCFCs is organised such to provide useful feedback to conceptual designers for the three main purposes of stability and control in conceptual design: (1) control effector sizing, (2) estimation of trimmed aerodynamics, and (3) characterisation of static and dynamic stability and resulting control effector design implications:

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1. **DCFC:** ***take-off rotation manoeuvre*** – demonstration of the quasi steady state take-off rotation module and the effect of relaxed longitudinal static stability on YB-49 take-off performance.
2. **DCFC:** ***steady-state straight line flight*** – demonstration of the trim, static stability, dynamic stability and SAS modules on 1-g trim during various mission segments with non-zero sideslip angle while maintaining heading, on YB-49 directional control power (DiCE) available versus required.
3. **DCFC:** ***steady-state straight line flight*** – demonstration of the trim, static stability, and dynamic stability modules and the effects of relaxed longitudinal static stability on YB-49 longitudinal control power (LoCE) available versus required.

4.2 DCFC: Take-off rotation manoeuvre – longitudinal control power estimation (Northrop YB-49)

The take-off rotation DCFC has been analysed to calculate the aircraft take-off rotation speed for elevons deflected upward at 20° (maximum deflection) to command an instantaneous pitch acceleration of 6°deg/s^2. This manoeuvre has been analysed for two gross take-off weights, 115,000lb and 175,000lb, with the c.g. longitudinal location ranging from 21% to 30% MAC. The neutral point is located at 30% MAC during this manoeuvre. The YB-49 take-off rotation manoeuvre results are presented in Fig. 10. This manoeuvre has been analysed for the maximum and minimum take-off gross weights tested in Ref. 28, assuming zero trim flap deflection (δ_{TF} = 0·0deg) to show the full range of longitudinal control effector (LoCE) YB-49 take-off rotation capability. The original flight test reference data available shows scatter because of the variation of trim flap settings selected during flight testing for different gross weights and c.g. locations.

These inconsistencies aside, the results agree well with flight test data. A pronounced reduction of the take-off rotation velocity when moving the centre of gravity aft. Moving the c.g. longitudinally initiates two primary effects, where the first effect is dominating. First, the aft c.g. shift reduces the load on the nose gear (this reduces the opposing moment produced by the main landing gear), but secondly shortens at the same time the moment arm between the aerodynamic centre (*ac*) of the LoCE to the c.g. The overall effect is that an aft c.g. shift reduces the rotation velocity required. Summarising, relaxed longitudinal static stability reduces the rotation velocity while decreasing the dynamic stability once airborne. However, the YB-49 demonstrates adequate control power to artificially stabilise the aircraft once airborne as shown with the 1-g trim DCFC during the initial climb mission segment, see Ref. 14. Concluding, this DCFC does not require relaxing static longitudinal stability since the YB-49 does not suffer from excessive rotation speeds as quantified by flight test data[25,26,27,28] and numerically confirmed with the methodology presented.

4.3 DCFC: Steady-state straight line flight – directional control power estimation (Northrop YB-49)

In general, the steady state 1-g trim sideslip flight condition requires the DiCE to actively oppose and balance the restoring yawing moment produced by the inherent airframe (directional static stability) to maintain a given heading. The results for the YB-49 cruise, low-speed flight and approach mission segments are shown in Fig. 11. Note that these results only visualise one single three-dimensional trim point defined by a unique setting of not only the DiCE, but also the LoCE and LaCE, and other flight variables like thrust setting and speed, parameters not shown for simplicity in Fig. 11. For a detailed discussion of aircraft sideslip piloting techniques, see Refs 5 and 22.

The additional SAS control power demands for satisfying MIL-Spec[29] Level 1 dynamic stability requirements are shown for typical maximum disturbance values of $\Delta\beta = \pm 5°$ and $\Delta r = \pm 10°$deg/s, see Roskam[30]. The effects of relaxed longitudinal static stability are small for the required DiCE deflections for trim, thus only the forward c.g. of 21% MAC is shown.

The drag rudders show sufficient control power to tolerate sideslip divergences during the cruise mission segment, but are limited during the low-speed flight and approach mission segments. Figure 11(b) indicates that the SAS does not satisfy Level 1 Dutch roll requirements during low-speed flight for the zero side-slip condition due to inadequate directional control power physically available. Thus the DiCE saturates for any side-slip angle and cannot provide adequate artificial dynamic stability at this flight condition. During approach, see Fig. 11(c), the Level 1 requirement can be satisfied by the SAS for up to 5° sideslip angle. Beyond 5° sideslip angle the DiCEs saturate. Therefore, the YB-49 DiCEs (drag rudders) can only deliver adequate trim and stability augmentation during the approach phase in moderate cross-winds. Note that the DiCEs appear to have more control power available during the approach mission segment compared to the low-speed mission segment, since the MIL-Spec[29] requirements are slightly lower for Dutch roll during the approach phase, thereby requiring less stability augmentation.

Concluding, the DiCEs haves adequate control power to trim and stabilise the YB-49 for 1-g trimmed flight during the cruising mission segment, but cannot provide adequate stability augmentation for both, the low-speed flight and approach mission segments.

4.4 DCFC: Steady-state straight line flight – longitudinal control power estimation (Northrop YB-49)

The YB-49 utilises the outer wing panel trim flaps (combined with drag rudder) to maintain longitudinal trimmed flight. Therefore, the trim flaps are defined to be the primary LoCE for this DCFC during the cruise, low-speed flight, and approach mission segments (the elevons serve, in contrast, to manoeuvre the aircraft). For validation purposes, we are first comparing simulated results with available YB-49 flight test data[27,28] for the cruise and low-speed flight mission segments, see Fig. 12. For both mission segments, when shifting the c.g. aft, the methodology correctly estimates the required LoCE deflection to trim as confirmed with YB-49 flight test data. The discrepancy observed in Fig. 12(a) can be attributed to a longitudinal mis-trim condition experienced during YB-49 cruise flight testing[27,28]. For this particular cruise-point, the flight test crew selected a trim flap setting commanding nose up trim beyond the trim point, which had to be compensated by deflecting the elevons nose down to arrive at the steady-state trim point. The presented methodology, in contrast, estimates the correct trim flap deflection resulting in a smaller trim flap deflection angle and zero elevon deflection angle, thereby avoiding the performance-adverse mis-trim condition experienced during flight testing. The low-speed flight discrepancy observed, see Fig. 12(b), can be attributed to the fact that the particular YB-49 vortex-lattice method (VLM) model does not yet model the wing slot at YB-49's outer wing panel, see Fig. 12(c). Incorporating the wing slot as an anti-stall device into the YB-49 VLM model will produce a larger nose down pitching moment. Consequently, a larger nose up trim flap deflection will be required to trim, ultimately reducing the delta observed. These discrepancies aside, the results demonstrate physical correctness for this DCFC.

To explore the effects of relaxing longitudinal static stability, the c.g. location was varied from 21% to 30% MAC for the cruise, low-speed flight, and approach mission segments. All three mission segments utilise the trim flap as the primary LoCE to trim. The required trim flap deflections are shown in Fig. 13. The additional control power required due to SAS is shown for

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standard maximum disturbances of $\Delta\alpha = \pm 5°$ and $\Delta q = \pm 10°$deg/s, see Roskam[30]. The gain settings selected meet MIL-Spec(29) Level 1 longitudinal dynamic stability requirements, resulting in additional CE deflections indicated with the doted lines in Fig. 13. The generated results demonstrate that the YB-49 does possess adequate longitudinal control power for all three mission segments while the aircraft is flown statically stable in the longitudinal plane (i.e. the c.g. is ahead of the neutral point). However, control saturation can occur due to additional control power demands for stability augmentation when flying the YB-49 statically unstable, a fact especially critical during the approach mission segment, see Fig. 13(c). In conclusion, The methodology proves to generate a valid representation of the full-scale YB-49. This DCFC demonstrates no particular need for additional trim flap control power while relaxing longitudinal static stability for the cruise and low-speed mission segments considered. The critical DCFC for sizing the trim flaps is the approach mission segment when positioning the c.g. at and beyond 31% MAC.

5.0 CONCLUSION

The incentive for developing a generic stability and control method has been discussed. The literature survey clearly indicates that no other known party is working on a stability and control methodology of generic character. Without touching the preparatory activities required to develop a generic process logic, the generic stability and control methodology concept has been introduced. The overall aim, to derive a generic method for conceptual design application with focus on full functionality for the most advanced fixed-wing aircraft, being the oblique flying-wing (OFWC), has been achieved while striving for minimum complexity as demanded by the conceptual design phase.

The methodology presented in this paper is developed, from the outset, to operate in a multidisciplinary synthesis environment. Applicable design guidelines and certification requirements have been taken into account. The aerodynamic tools employed are primarily of generic and non-linear character since CEs are usually sized in the non-linear corners of the flight envelope. A common set of DCFCs, valid for the full range of fixed-wing flight vehicle configurations and concepts, is used to restrict the evaluation to only design critical areas in the flight envelope. The DCFCs represent the link between conceptual design and flight test. A flight control system (FCS) design-logic is integrated to take the design variables longitudinal and lateral-directional relaxed static stability into account. The primary parameter of concern in the context of analyzing and designing CEs is 'control power'. The control power available and required for the range of aerodynamic CEs is quantified for each DCFC, by specifying the volume coefficient (geometry), stability derivative coefficients (aerodynamics), and the CE deflection angle (operation) required. Note that in modern aircraft design, adequate stability-, control-, and trim characteristics can be entirely ensured with provision of sufficient control power!

All design decision-making is based on the evaluation of the trimmed aircraft in six degrees-of-freedom. The fully coupled steady state equations of motion have been implemented and results have been presented for three selected DCFCs, demonstrating application and overall correctness of the methodology for the YB-49 flying wing design case study. A decoupled implementation of the small perturbation equations of motion has been utilised in the present context for the analysis of the YB-49 dynamic characteristics. A more complete design analysis of the YB-49 has been beyond the present context and can be found elsewhere. Work is underway to integrate the methodology into a generic aerospace flight vehicle synthesis system.

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