Aircraft Design Optimization with Dynamic Performance Constraints

Stephen J. Morris* and Ilan Kroo†
Stanford University, Stanford, California 94305

This paper describes an integrated optimization procedure for aircraft design in which multidisciplinary criteria are used as performance measures. A composite objective function consisting of a handling qualities parameter and other measures of mission performance is minimized using an unconstrained optimization method. The design variables may include aircraft configuration variables and control system gains. Three example design problems using this methodology are discussed: tail sizing for minimum trimmed drag with longitudinal handling qualities constraints, wing weight minimization with aeroelastic constraints, and oblique wing design for commanded response decoupling. Results show that the integrated design procedure can achieve performance that exceeds that obtained in a sequential design synthesis.

Nomenclature

$[A]$ = system dynamic matrix
AR = aspect ratio
$[B]$ = system control matrix
$[C]$ = feedback gain matrix
C_D = drag coefficient
C_{D0} = nonlift dependent parasite drag coefficient
C_{D2} = lift dependent parasite drag coefficient
C_L = lift coefficient
g_y, g_z = body axis acceleration component
J = composite objective function
J_d = dynamic performance cost
J_{nd} = nondynamic performance cost
J_{pen} = constraint violation penalty
K_α = angle of attack to elevator feedback gain
K_d = handling qualities weighting factor
K_δ = wingtip deflection to elevon feedback gain
p, q, r = roll, pitch, yaw angular velocity, respectively
$[Q]$ = output weighting matrix
$[R]$ = control effect weighting matrix
S = area
t = time
$\boldsymbol{u}$ = control input vector
u, w = body axis velocity component
X = longitudinal position
$\boldsymbol{x}$ = state vector
$\boldsymbol{y}$ = output vector
δ_{bt} = wingtip bending deflection
δ_e = elevator or elevon deflection
$\delta_r, \delta_p, \delta_y$ = roll, pitch, yaw control deflection, respectively
$\boldsymbol{\varepsilon}$ = output error vector
ξ = wing bending mode shape amplitude
ψ, θ, ϕ = yaw, pitch, roll Euler angle, respectively

Subscripts

c = pilot input command
m = model
T = tail
w = wing

Introduction

THE dynamic response of an aircraft is often an important aspect of its performance; yet the analysis of handling qualities and control system design are often performed after the major aerodynamic and structural properties have been established. In many cases, this sequential approach to multidisciplinary design leads to suboptimal results. The method presented here integrates the design of the aircraft and its control systems in order to obtain better mission performance than could be achieved in a sequential design procedure. By minimizing a cost function consisting of both conventional performance criteria and a measure of aircraft handling qualities, a design with maximum performance for a specified level of handling can be achieved. Handling qualities are measured using a quadratic cost function similar to that used in the design of optimal feedback control systems. This function is proportional to the difference between the dynamic response of the aircraft and a model case with dynamics that are considered acceptable. The variables to be optimized may include both aircraft configuration parameters (e.g., span, tail area, skin thickness) and control system feedback gains. The design variables are determined by an unconstrained numerical optimization procedure, using penalty functions to enforce both explicit and implicit constraints. The method is most useful in the simultaneous synthesis of airframe and flight control systems to achieve improved handling or to improve performance with a specified level of handling quality. In certain cases, results obtained by this integrated synthesis procedure are substantially better than those obtained by the usual sequential design methods.

Previous research on the integrated synthesis of dynamic systems and full-state feedback controllers is described in Refs. 1 and 2. Sawaki et al.[1] optimized the wing and tail geometry of an actively controlled aircraft excited by gust disturbances. The synthesis assumed a full-state feedback control architecture and used a random search procedure to solve for the global minimum. In Ref. 2, Zeiler and Weisshaar describe the integrated design of a four degree-of-freedom aeroservoelastic system. The design variables consisted of the elastic axis location and the full-state feedback control gains. A multilevel linear decomposition scheme was used to solve for the optimal system variables. Reference 3 discusses the application of more sophisticated structural models in the optimal design of composite wings with active control.

In the work presented here, an integrated synthesis procedure is extended to handle any linear control system architecture including reduced-order controllers and passive (no controller) designs. The procedure is used to optimize the system commanded response as well as its unforced response.

Received Oct. 11, 1989; revision received and accepted for publication May 7, 1990.

*Doctoral Student, Department of Aeronautics and Astronautics.
†Assistant Professor, Department of Aeronautics and Astronautics. Member AIAA.

Multidisciplinary performance indices are considered, illustrating the trade-offs between handling qualities and other measures of performance.

Three examples are used to demonstrate the method. Reduced-order controllers and multidisciplinary performance indices are first introduced in the synthesis of an aft-tail aircraft. The proposed method is used to find the tail size and static margin that produce acceptable handling qualities with minimum drag. When the control system gains are included as design variables, the procedure automatically synthesizes statically unstable configurations and an appropriate reduced-order control system.

The minimum weight aeroservoelastic design of a free-flying aircraft represents a highly integrated structural and control design. The second example considers this problem by synthesizing a tailless aircraft that experiences a coupled short-period/wing-bending flutter mode. The synthesis procedure determines the spar cap thickness (as a function of span) for minimum wing weight with acceptable handling qualities. In some cases, a reduced-order control system is also designed. This synthesis differs from conventional flutter suppression studies because not only must the flutter mode be stable, but the phugoid and short-period dynamics must meet specified handling quality criterion.

In some cases, handling qualities may be the only performance measure that the designer wishes to improve. This is exemplified by the third example, in which an oblique wing aircraft is designed for improved handling qualities by simultaneously optimizing the wing geometry and control system gains. The integrated synthesis procedure uses configuration variables to enhance the controllability of the closed-loop system. This produces an aircraft with handling qualities superior to those achievable in a sequential design procedure.

Description of the Method

Overview

The basic approach is outlined in Fig. 1. The design method is an unconstrained optimization procedure that minimizes a composite objective function J consisting of three terms: the nondynamic performance measure J_{nd}, the weighted dynamic performance J_d, and the constraint violation penalty function, J_{pen}:

$$J = K_d J_d + J_{nd} + J_{pen} \tag{1}$$

The nondynamic performance measure describes the mission performance that is not directly related to handling qualities.

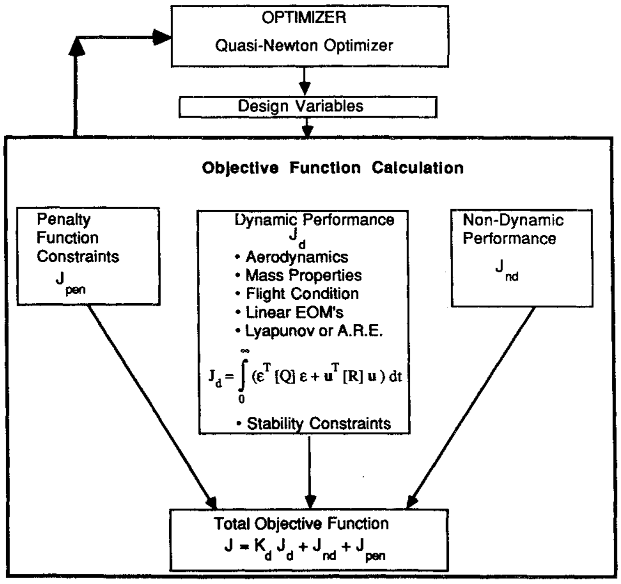

Fig. 1 Method flowchart.

Typically, this function is the structural weight, drag, or direct operating cost, etc., of the aircraft. The content of this term is problem specific and will be discussed further within each design example, but for cases in which only the handling qualities are to be improved, this term is excluded.

The dynamic performance J_d provides a measure of the aircraft's handling qualities. It is calculated from a quadratic cost function identical to that used in the design of optimal control systems, but the control system architecture is not restricted to full-state feedback. Dynamic performance is related to the difference between the aircraft's response and a model case chosen for its desirable handling qualities. In general, the control gains are included as design variables in the synthesis. This permits the designer to choose any linear control system structure: reduced order, full-state feedback, or a passive design with no controller at all. Control systems can be synthesized that tailor the forced and/or unforced response of the aircraft. The design examples clarify how these cases are synthesized. In the composite objective function, J_d is weighted by a constant, K_d, which is used to specify the relative importance of handling qualities in the design synthesis. As K_d is varied from zero to a very large number, the optimal solution moves from one in which dynamics are not considered, to one that achieves acceptable handling qualities at the expense of all other types of performance. Configuration design variables affect the handling qualities (and, therefore, J_d) by altering the aerodynamic and mass properties of the design being considered. Particular designs may require a complete aerodynamic analysis of an unusual aircraft configuration whose geometry is changed during the synthesis. An aerodynamic analysis method that can predict the aircraft's forces and moments with little CPU time is desired. The vortex lattice method described in Ref. 4 fulfills these requirements for designs in which a thin lifting surface model is appropriate. For the examples presented here, in which neither shock waves nor separation is expected at the design condition, this method provides a useful, if approximate, result.

The third term of the objective function, J_{pen}, consists of penalty functions that are used to enforce any explicit and implicit constraints in the synthesis.

A numerical optimizer based on the quasi-Newton or variable metric gradient method is used to solve for the design variables that minimize J. This optimizer searches for a minimum along descent directions that improve in accuracy (based on a second-order model of the objective function) as the optimization progresses.[5] This optimization procedure was chosen because it represents a satisfactory compromise between the high computation costs of second-order methods and the poor convergence speed of first-order schemes. Other optimizers may be better suited for particular problems. For example, if the nondynamic performance index is known to have local minima, a more robust optimizer may be required. Results presented here were checked by starting the optimization at several points in the design space to reduce the possibility that the converged solutions represented local minima. The independence of the solutions on the initial designs suggests that the present objective function formulation, in contrast with those based on eigenvalue distances, for example, avoids local minima.

Calculation of the Dynamic Performance Index

A key aspect of this approach is the calculation of the dynamic performance index J_d. J_d is a scalar equal to the weighted integral over time of the difference between the state vectors of the aircraft being synthesized and a model system whose dynamic response is considered ideal. An additional term representing the control surface activity is also included.

Formulation of J_d

At each evaluation of the objective function, the dynamics and control matrices $[A]$ and $[B]$, respectively, representing the linearized equations of motion of the aircraft being synthesized, are formed. In general, $[A]$ and $[B]$ are nonlinear functions of the configuration design variables that affect the dynamic performance:

$$\mathrm{d}\boldsymbol{x}/\mathrm{d}t = [A]\boldsymbol{x} + [B]\boldsymbol{u} \tag{2}$$

$$\boldsymbol{u} = [C]\boldsymbol{x} \tag{3}$$

The designer may specify any linear feedback control architecture by choosing appropriate definitions for $[C]$, $\boldsymbol{x}$, and $\boldsymbol{u}$. For example, the control gain matrix $[C]$ may be defined with terms equal to zero to reduce the order of the controller. Similarly, the state vector $\boldsymbol{x}$ can be augmented to add states to the feedback controller.

Consider a dynamic system with the same states and the same initial conditions as the aircraft being designed. This second system is referred to as a model case because its dynamic response is chosen as the design goal for the aircraft being synthesized:

$$\mathrm{d}\boldsymbol{x}_m/\mathrm{d}t = [A_m]\boldsymbol{x}_m + [B_m]\boldsymbol{u}_m \tag{4}$$

The state error between the model system and the actual aircraft is $\boldsymbol{\varepsilon}$, with

$$\boldsymbol{\varepsilon} = \boldsymbol{x} - \boldsymbol{x}_m \tag{5}$$

The minimization of J_d represents the reduction of $\boldsymbol{\varepsilon}$ for all time and specified initial disturbances. If this is to be done with minimum control effort, the following integral defines J_d:

$$J_d = \int_0^\infty (\boldsymbol{\varepsilon}^T[Q]\boldsymbol{\varepsilon} + \boldsymbol{u}^T[R]\boldsymbol{u})\,\mathrm{d}t \tag{6}$$

where $[Q]$ and $[R]$ are positive definite weighting matrices on state error and control effort. In the examples discussed here, both gust and pilot control inputs are used to excite the system and the weighting matrices are chosen according to Bryson's rules.[6]

The closed-loop unforced response to the aircraft is given by the eigensystem of $[A + BC]$. If $[A + BC]$ is asymptotically stable (e.g., the real part of all eigenvalues of $[A + BC]$ are less than zero) then the integral that represents J_d may be evaluated numerically by solving three Lyapunov algebraic matrix equations:

$$[A + BC]^T[P1] + [P1][A + BC] + [Q + C^TRC] = 0 \tag{7}$$

$$[A + BC]^T[P2] + [P2][A_m] + [Q] = 0 \tag{8}$$

$$[A_m]^T[P3] + [P3][A_m] + [Q] = 0 \tag{9}$$

Equations (7–9) may each be solved numerically for $[P1]$, $[P2]$, and $[P3]$ with the methods presented in Ref. 7:

$$[P] = [P1 - P2 - P2^T + P3] \tag{10}$$

$$J_d = \boldsymbol{x}_0{}^T[P]\boldsymbol{x}_0 \tag{11}$$

For special cases in which the control system architecture involves full-state feedback, the control gains $[C]$ that minimize J_d may be found by solving a single algebraic Riccati equation (ARE):

$$J_d = \int_0^\infty (\boldsymbol{x}^T[Q]\boldsymbol{x} + \boldsymbol{u}^T[R]\boldsymbol{u})\,\mathrm{d}t \tag{12}$$

If $\boldsymbol{u} = [C]\boldsymbol{x}$ with no restrictions on $[C]$, then

$$[C] = -[R]^{-1}[B]^T[P] \tag{13}$$

where $[P]$ is obtained from the solution to the following ARE, assuming $[A]$, $[B]$ controllable:

$$[P][A] + [A]^T[P] - [P][B][R]^{-1}[B]^T[P] + [Q] = 0 \tag{14}$$

$$J_d = \boldsymbol{x}_0{}^T[P]\boldsymbol{x}_0 \tag{15}$$

The full-state feedback design has the advantage that the control gains that minimize J_d can be calculated by solving a single ARE at each objective function calculation. This reduces the dimension of the variable space in which the optimizer must search to minimize J because all of the control gains are now calculated inside the optimization loop. As a result, the time required for a given synthesis may be substantially reduced. The main disadvantage of the full-state feedback approach is that the designer is now restricted to a specific control system structure that may be unrealistic for implementation. This restriction can affect the type of system performance obtainable in a synthesis. For example, it is not possible to synthesize a full-state feedback controller that will cause the error between the plant and model unforced response to approach zero unless the model states are included in the feedback path. Using a reduced-order design technique, the designer can synthesize a controller that feeds back only the plant states and minimizes the model following error. The reduced-order technique can achieve these results because the designer is free to specify the control architecture arbitrarily.

An alternate and perhaps better method for calculating J_d is suggested in Ref. 8. J_d is evaluated from a finite time integral of the dynamic cost integrand [Eq. (6)] rather than the infinite time integral assumed in the Lyapunov and full-state feeback formulations. This has the advantage that stability constraints on $[A + BC]$ are no longer required and that the initial guess for the design variables need not produce a stable system. Furthermore, Ref. 8 shows how the gradients of J_d with respect to the control gains $[C]$ may be expressed analytically for both the finite and infinite time integral cases. Implementation of this technique is reserved for future work.

Interpretation of J_d

J_d is a scalar measure of the aircraft's handling qualities. Because there is no way to relate this number directly to a Cooper-Harper rating or a Mil. Spec. 8785C classification, additional analysis of the synthesized design (simulations, frequency response, etc.) must be performed to determine the adequacy of a design's handling qualities. The formulation of J_d does, however, guarantee that for fixed $[Q]$ and $[R]$ matrices the handling qualities will improve as J_d decreases. It is this fact that enables this synthesis method to improve the handling qualities in each optimization iteration and allows the designer a means of trading dynamic performance for nondynamic performance.

The dynamic performance index calculation requires that the linearized equations of motion for the aircraft be created as a function of the design variables at each objective function evaluation. This portion of the synthesis can be the most costly in terms of CPU time, particularly if the aerodynamic stability derivatives must be re-evaluated. The overall utility of this method relies on the careful choice of the analysis routines that evaluate the $[A]$ and $[B]$ matrices. Methods that capture the essential physical phenomena and minimize computation time are desired.

Example Applications

Aft-Tail Design for Minimum Trimmed Drag

Three examples are presented to illustrate the use of this method. The first of these, and the simplest, is the design of a wing and tail system. The configuration is required to trim at a selected lift coefficient while minimizing drag and retaining adequate longitudinal handling qualities and control authority. The design variables include horizontal tail area and wing

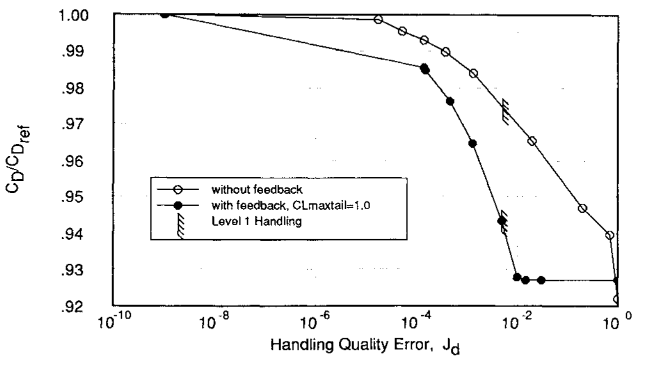

Fig. 2 Trim drag vs handling, aft-tail results.

location; in some of the designs a reduced-order controller (consisting of angle-of-attack feedback to the elevator) is also synthesized.

Dynamic Peformance

The dynamic performance J_d is computed based on a model case with wing location and tail area sized to provide Mil. Spec. 8785C level 1 response ("clearly adequate" flight qualities) in the short-period and phugoid dynamics. The longitudinal dynamics are modeled using axial velocity u, plunge velocity w, pitch rate q, and angle θ as states. The design variables are horizontal tail area S_t, wing position (relative to the c.g.) X_w, and a feedback gain K_α from sensed angle of attack to elevator deflection for the cases with a reduced-order control system. J_d is calculated from the difference between the model and subject aircraft's dynamic response to the same initial disturbances. This guarantees that, as J_d is minimized, the short-period and phugoid dynamics of the aircraft will approach those of the model case.

Nondynamic Performance

The size of the horizontal tail and static margin required for acceptable handling qualities may result in a design with excessive trimmed drag. The drag coefficient is used as the nondynamic performance measure so that trade-offs between handling qualities and drag may be studied. The drag coefficient is calculated analytically assuming elliptic loading on the wing and tail:

$$J_{\rm nd} = C_D = \{[1/(\pi \mathcal{R}_w)] + C_{D2w}\}C_{Lw}^2 + C_{D0w} + \{[1/(\pi \mathcal{R}_T)] + C_{D2T}\}C_{LT}^2 + C_{D0T} + 2\{[1/(\pi \mathcal{R}_w)]\}C_{Lw}C_{LT} \tag{16}$$

This formulation accounts for parasite drag, life-dependent viscous drag, and the vortex drag associated with the interfering lifting surfaces (under the assumption that they are coplanar).

Constraints

Three constraints are enforced during the design synthesis: 1) pitch trim at a specified lift coefficient, 2) trim at maximum lift without tail stall, and 3) dynamic stability. Dynamic stability implies that the largest real part of any eigenvalue must be less than zero. This constraint must be enforced explicitly because the Lyapunov equation solution for J_d is only valid if the system is dynamically stable.

Aft-Tail Design Results

Figure 2 shows the trade-off between trimmed drag and the handling quality parameter for the aft-tail design synthesis. Curves are shown for designs with and without a feedback control system. Each point on the curves represents a unique design that is optimal for a fixed weighting of handling qualities. As the weigthing on handling is increased, J_d decreases and the dynamic response of the aircraft approaches that of the model case. Note also that the trimmed drag increases with improved handling quality. This occurs because the optimal tail size and static margin increase as J_d decreases, with a subsequent increase in parasite and trim drag. Designs with feedback control show reduced trimmed drag for a fixed level of handling compared to designs without control systems. The synthesis method has recognized that relaxed static stability and smaller tail size can reduce trimmed drag, while feedback control can ensure adequate handling qualities by providing artificial stability. As a result, designs that are statically unstable and have optimally designed reduced-order controllers to provide stability are automatically synthesized. The values of J_d for which the longitudinal dynamics meet the Mil. Spec. 8785C level 1 handling quality requirements are marked on each curve.

Because the selected control system does not provide rate feedback, adequate damping requires some tail area; thus, it is not possible to eliminate the tail completely. Even if large values of J_d (poor handling) are accepted, trim constraints still yield a nonzero tail area when the wing pitching moment at zero lift is not zero. This leads to the flat part of the curve with feedback at higher values of J_d.

Figure 3 shows the eigenvalues associated with the longitudinal dynamics of each optimal design as the handling qualities weighting factor K_d is increased. When the weighting is large, the eigenvalues associated with both the short-period and phugoid modes are driven to those of the model case. This eigenvalue analysis is one means of assigning a physical interpretation to the handling qualities parameter J_d.

Tailless Aircraft Flutter Suppression

The second example deals with a tailless aircraft designed for minimum wing weight and an acceptable level of handling quality. Tailless aircraft with swept wings may exhibit a unique flutter mode, characterized by a coupling of the short-period dynamics with the wing-bending modes. The frequency of this flutter mode is slow (2 Hz) and corresponds to the wing-bending and short-period frequencies. Reference 9 decribes a high-performance tailless sailplane for which this flutter mode was the critical factor in the spar box design.

Using the design procedure presented in this paper, a swept-wing tailless aircraft, operating in a flight regime where flutter is critical, was designed for minimum structural weight with specified longitudinal handling quality. The spar cap thickness at various stations along the span were used as design variables. Some designs included a feedback control system with elevons deflected in proportion to the wingtip deflection to help control the flutter. The design synthesis involves solving for the skin thickness distribution (and elevon feedback gain for cases with active control) that yield a stable aircraft with distinct short-period and phugoid modes while using the least amount of structural material in the wings. The integrated design procedure improves the handling by simultaneously suppressing the flutter and driving the unrestrained dynamic modes to be most like that of a rigid

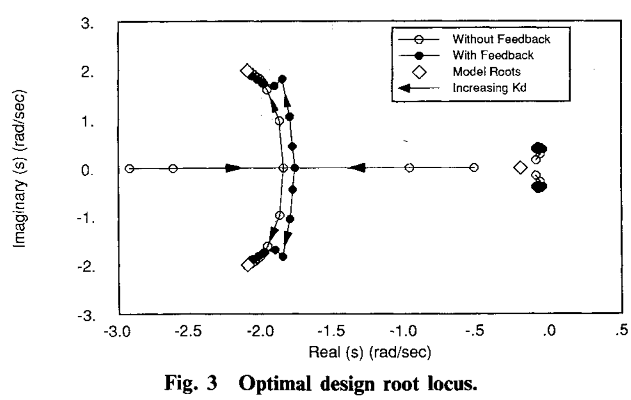

Fig. 3 Optimal design root locus.

aircraft. The flutter boundary for each design is not computed explicitly in this case: since the model system does not flutter and the synthesized design is penalized by the extent to which it departs from the model dynamics, the solution is flutter free. This differs from conventional flutter suppression techniques that only guarantee flutter stability and do not attempt to restore acceptable handling qualities to the unrestained modes of the free flying aircraft.

Figure 4 shows the layout of the swept-wing tailless aircraft whose structure (and possibly control system) are to be synthesized. The design flight speed is chosen so that a wing sized to support the static structural loads will not be stiff enough to avoid flutter. This choice of trimmed flight condition ensures that the dynamic response will be dominated by aeroelastic phenomenon.

Dynamic Performance

A flutter problem of this type is unusual because acceptable handling qualities require not only that the flutter be suppressed, but also that the short-period and phugoid dynamics be acceptable. To achieve these results, a rigid aircraft with longitudinal dynamics meeting Mil. Spec. 8785C level 1 requirements is used as the model for the design synthesis. As J_d is minimized, the tailless aircraft's response will approach that of the rigid model case, ensuring a suppressed flutter mode and acceptable short-period and phugoid response.

In Ref. 2, it is suggested that the present definition of J_d is not a sufficient measure of handling qualities for the flutter suppression problem. If the design synthesis considers only one flight condition, it is possible that the optimized design will be unstable at velocities below the design value. This situation can be avoided, however, by including in the dynamic performance index the sum of J_d evaluated at several flight velocities, up to and including the design flight condition. Designs synthesized using this composite J_d will be dynamically stable *and* have acceptable handling qualities over the entire flight envelope. In this example problem, only one flight condition will be considered.

The linearized equations of motion (EOM) for an unrestrained elastic flight vehicle must be formed to evaluate J_d.

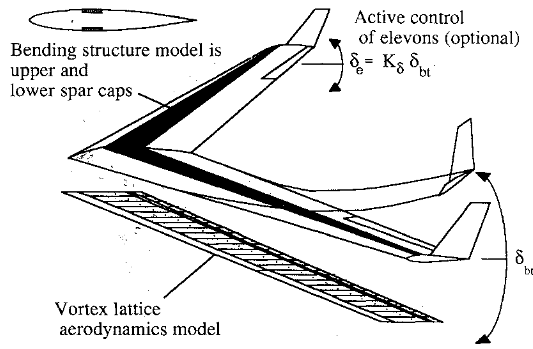

Fig. 4 Tailless flutter analysis.

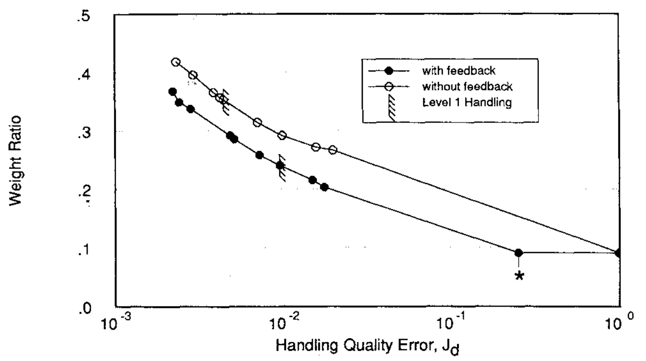

Fig. 5 Wing weight vs handling quality.

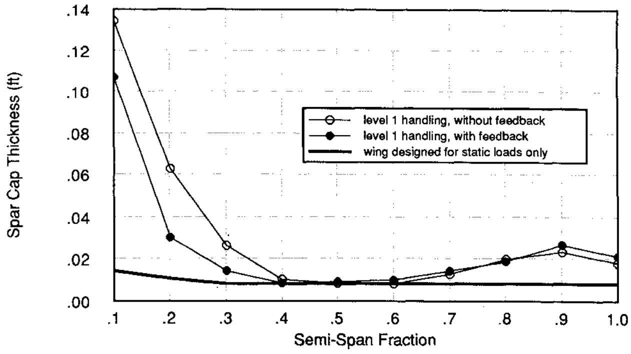

Fig. 6 Optimal spar cap thickness distribution.

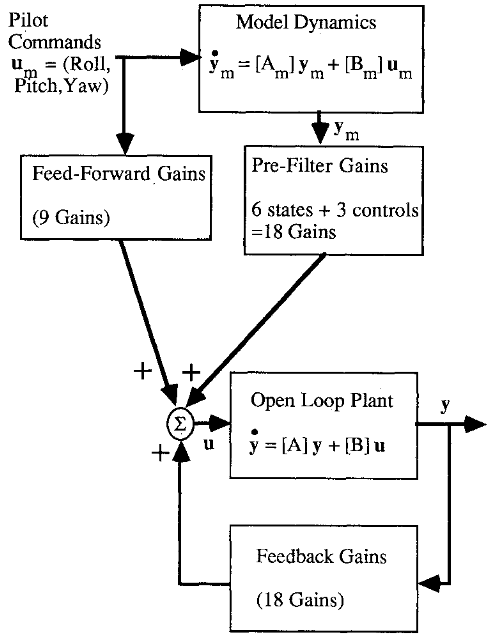

Fig. 7 Explicit model-following controller.

Lagrange's method is used to derive the EOM assuming two wing-bending modes ($\xi 1$, $\xi 2$) and all longitudinal degrees of freedom (u,w,q,θ). Linearization with a finite state model is possible because the wing bending is modeled using two assumed modes and the aerodynamics model is quasisteady. This formulation of the EOM has the advantage that the rigid aircraft dynamics are separable from aeroelastic effects, permitting the designer to see the influence of structural flexibility explicitly in the EOM. In some cases, a reduced-order controller is also designed with feedback of the sensed wingtip deflection to elevon deflection. Aerodynamic stability and control derivatives are predicted by a quasisteady vortex lattice method (Fig. 4), appropriate because of the low reduced frequncy of this flutter mode. Wing mass and stiffness properties are recomputed as the skin thickness is varied during the optimization.

Nondynamic Performance

In theory, design problems associated with aircraft flexibility and flutter can always be solved by adding more material to the structure to increase its stiffness. The designer's goal is to add the material intelligently so that the aircraft's structural weight will remain a minimum. By choosing wing-bending weight as the nondynamic performance in this example, designs are synthesized with minimum structural weight consistent with a specific level of handling quality. Wing-bending weight is calculated by integrating the weight of the spar cap material at 20 spanwise stations.

Constraints

Penalty function constraints are imposed to limit the maximum stress in the spar caps at a 3-g load factor; a minimum skin gauge is also imposed. Maximum skin stresses are calculated using a static aeroelastic analysis that accounts for inertia relief and the effect of wing deformation on the spanwise loading. A third penalty function is used to ensure dynamic stability of the system. This is identical to the constraint implemented in the aft-tail design synthesis. The combination of these three contraints ensures that all designs will be strong enough to meet the static load requirements and be aeroelastically stable.

Tailless Flutter Design Results

Results for the tailless aircraft design example are shown in Fig. 5. As in the previous case, each point on the curves represents an optimal design with a specific value of handling quality weighting. Smaller values of J_d indicate improved handling, and increased values of weight ratio correspond to increased wing weight. The regions in which handling quality becomes acceptable lie to the left of the level 1 limiting marks. Designs with feedback control show reduced wing weight since elevon deflection can provide artificial stiffness without additional material in the skins.

In a sequential design procedure, the wing structure is first sized for minimum weight based on static aeroelastic loading and minimum gauge requirements. The reduced-order control system is then designed for the best handling quality with a fixed wing design. The resulting sequentially designed aircraft has a stable flutter mode but its short-period and phugoid dynamics are still highly coupled to the wing-bending mode, giving poor handling qualities. This design is represented by the point marked with an asterisk in Fig. 5.

By contrast, the integrated design procedure achieves a stable flutter mode with acceptable rigid body dynamics and does so with the least penalty in wing weight. The eigenvalues and eigenvectors approach those of the model case as the handling quality weighting is increased. This is important to note because the handling qualities for this example only become acceptable when the short-period, phugoid, and wing-bending modes are distinct and properly damped.

The optimal solutions for the skin thickness as a function of semispan are shown in Fig. 6. Aeroelastic stability requires that the skin thickness be increased at the wing root with greater thickness required for cases without a feedback controller. Interestingly, the results also show increased skin thickness at the wingtip. The presence of this additional mass further separates the frequencies of wing-bending and short-period dynamic modes and is a significant factor in achieving flutter stability.

Dynamic Decoupling of Oblique Wing Aircraft

The oblique wing configuration exhibits reduced transonic and supersonic drag in a lightweight variable sweep configuration. At high sweep angles (greater than 25 deg), asymmetries in the design produce a strong coupling between the lateral and longitudinal dynamics, resulting in poor handling qualities. Previous efforts to decouple the response with a stability augmentation system (SAS) have produced less than acceptable results because of insufficient control authority and lack of controllability in certain modes.[10] This design synthesis seeks to improve the oblique wing's handling qualities by including aircraft geometry variables in the SAS design synthesis. These configuration design variables provide additional degrees of freedom to more thoroughly decouple the aircraft's dynamic response.

This example involves four geometric variables and the SAS control gains that enable the optimized design to approach the handling qualities of a level 1 model aircraft. The geometric design variables include the longitudinal position of the wing pivot along the fuselage and wing root chord, the wing bank angle relative to the fuselage, and the wing dihedral. Because these variables have a minimal impact on the aircraft's trimmed drag, the design synthesis seeks only to improve handling qualities within the specified limits placed on the geometry variables. Although aerodynamic nonlinearities would require consideration of multiple design points, this example illustrates the use of the method at a Mach number of 0.8 at which the wing is swept 45 deg.

Explicit Model-Following Controller

To achieve dynamic decoupling, the closed-loop oblique wing aircraft should respond to pilot stick commands in the same manner as a conventional symmetric aircraft. An explicit model-following controller attempts to achieve this type of performance by implementing a command generator within the SAS that feeds forward the desired motion to the stabilized plant. Figure 7 shows a block diagram of the explicit model-following controller. The feedback path serves to attenuate noise (gust) disturbances and to increase the plant's response bandwidth so that it can adequately track commands. The feed-forward path consists of a prefilter containing the model dynamics and associated feed-forward gains. Pilot commands excite the desired outputs within the prefilter model. These signals are then fed forward as inputs that cause the plant to track the model outputs. During the design process, initial conditions are placed on the model command input channels to mimic control inputs from the pilot.

An explicit model-following controller can be synthesized by solving a single ARE if full-state feedback of the model and plant states is assumed. It would be possible to implement a reduced-order controller of arbitrary architecture, but the focus of this problem is to find the best oblique wing configuration for a representative model-following control scheme, and the added computation costs of synthesizing the reduced-order controller (using the optimizer) are not justified at this preliminary design stage.

Design Approach

The aircraft considered in this synthesis is the NASA-Rockwell F-8 Oblique Wing Research Aircraft (OWRA).[11] Early studies performed at the NASA Ames Vertical Motion Simulator revealed serious handling quality deficiencies in the original aircraft (even with a SAS). The goal of this synthesis was to improve the closed-loop handling qualities of the F-8

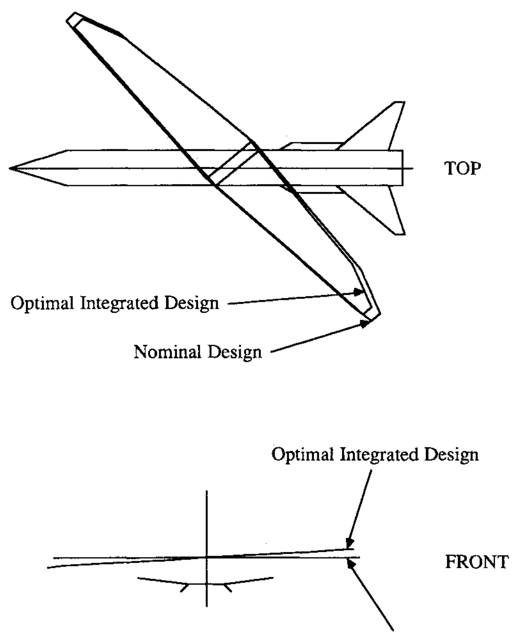

Fig. 8 Optimal OWRA configuration.

OWRA by the integrating the design of the aircraft's geometry and control system.

The model dynamics are represented by a linear system whose commanded response in roll, pitch, and yaw meets the Mill. Spec. 8785C level 1, class 4, category A requirements. The output vectors for both the model and the plant contain the states g_y, g_z, p, q, r, ϕ, δ_r, δ_p, and δ_y. The states ψ, θ and u are omitted to remove the spiral and phugoid modes from the dynamic cost function. These slow modes are easily controlled with simple feedback loops (or by the pilot) and do not significantly influence the higher bandwidth commanded response dynamics. It is important to omit these slow modes from the synthesis because their dynamic error persists over the longest time and can dominate the solution. Three control surfaces are used to maneuver the aircraft: rudder, left elevator, and right elevator. The horizontal tail is full flying and can be deflected asymmetrically to generate rolling moments. Ailerons are less effective at high sweep angles and are not employed in this example.

Because the aircraft's geometry is changed continuously in the design process, the stability derivatives and inertia tensor must be updated at each objective function evaluation. The vortex lattice method[4] is particularly efficient in this capacity because it predicts, with sufficient accuracy, the aircraft's forces and moments with minimal CPU time. In a typical objective function calculation, the mass properties and aerodynamic terms are calculated first, then the linearized EOM are formed and an explicit model-following controller is synthesized. J_d is evaluated from the equation:

$$J_d = y_0^T[P]y_0 \tag{17}$$

In a conventional full-state feedback controller synthesis, the type of initial conditions (y_0) that excite the plant do not affect the optimal feedback gain solution. In the integrated design procedure, this is not the case because design variables other than the controller gains are being considered. Initial condition disturbances are placed on the δ_{rc}, δ_{pc}, δ_{yc}, g_y, and g_z states to account for step commands issued by the pilot and gust disturbances acting on the aircraft. The choice of y_0 determines the relative importance of commanded response (δ_{rc}, δ_{pc}, δ_{yc}) and unforced response (g_y, g_z) in the design process.

Feasible designs are required to trim in 1-g flight at the prescribed flight condition. A linearized model for the forces and moments is used to solve for the trimmed control surface deflections and penalties are added to J for any control deflections that exceed specified limits.

Oblique Wing Design Results

Piloted simulation studies of the nominal configuration[10] resulted in poor handling quality rating due to excessive lateral accelerations and roll motion during abrupt pitch maneuvers. The weighting matrix $[Q]$ was chosen to emphasize reduced lateral acceleration and roll motion for all of the cases studied. For the nominal OWRA configuration (with an SAS), it is not possible to minimize rolling motion and lateral acceleration simultaneously when reasonable limits are placed on control deflections. Because the wing is unbanked relative to the fuselage, the increased leading-edge suction along the wing during pitch-up maneuvers will cause large lateral accelerations. If the control system forces the rolling motions to be small, the lateral accelerations will be greatest because rolling would allow gravity forces to cancel the aerodynamic sideforce. (The sideforce component is too large to be countered by rudder deflection and sideslip motion.) Therefore, minimizing lateral acceleration requires large rolling motions in the nominal design. The integrated design results show how the optimization procedure can reconfigure the aircraft, improving its controllability and closed-loop handling qualities.

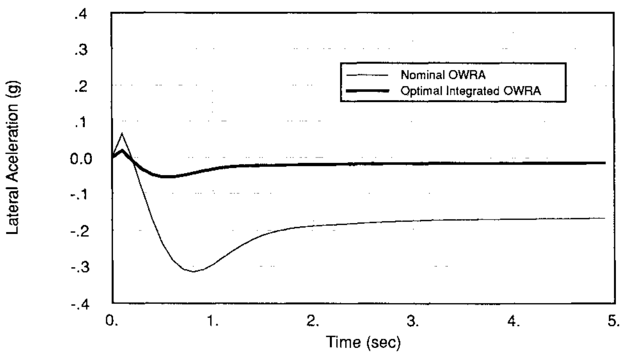

Fig. 9 Lateral acceleration vs time.

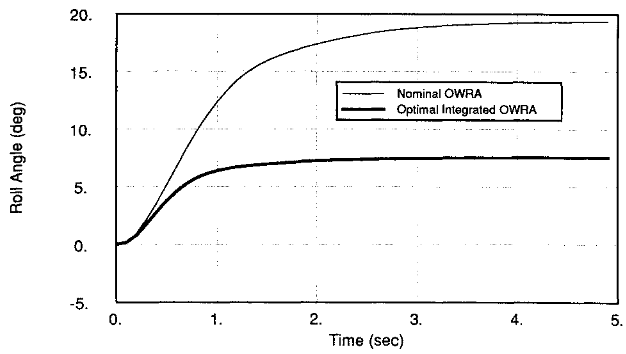

Fig. 10 Roll angle vs time.

The vortex lattice geometries for the nominal (initial guess) and optimal solution are shown in Fig. 8. The optimal solution shows the wing banked 3.5 deg (forward wing low) and the wing displaced slightly to the right. The time history of the aircraft's response to a 4-g pitch-up command is shown in Figs. 9 and 10. The optimized results show significantly reduced lateral acceleration g_y and roll angle ϕ during the 4-g pitch-up. Peak lateral accelerations are reduced from 0.31 to 0.04 g and peak roll angles from 19 to 7.5 deg. The small changes in wing position and bank have significantly improved the aircraft's commanded response.

Designers had assumed previously that the best way to ensure dynamic decoupling of an oblique wing aircraft was to minimize the aerodynamic coupling. The optimized design shows otherwise because the coupling terms Cl_α and Cy_α are not zero. In the case of the open-loop aircraft, dynamic decoupling requires that aerodynamic and inertial coupling terms cancel each other. This implies that a decoupled oblique wing aircraft (without an SAS) would, in general, have nonzero aerodynamic coupling terms. For an aircraft with an SAS, it is possible that dynamic performance may benefit more from a design change that improves controllability than one that improves dynamic decoupling of the open-loop aircraft. This type of integrated approach to the design of the aerodynamic configuration and the control system is required to obtain an oblique wing design with acceptable handling qualities.

Summary and Conclusions

The method described here for aircraft design optimization with dynamic response considerations provides an inexpensive means of integrating dynamics into aircraft preliminary design. By defining a dynamic performance index that can be added to a conventional objective function, a designer can investigate the trade-off between performance and handling. The procedure is formulated to permit the use of control system gains as design variables but does not require full-state feedback. The examples discussed here show how such an approach can lead to significant improvements in the design when compared with the more common sequential design of system and control law.

This integrated design procedure is most useful for studying problems in which handling qualities and other types of mission performance are highly coupled. Many problems of current interest fall into this category: 1) aeroservoelastic design of flexible aircraft, 2) aeroelastic tailoring of aircraft with composite structures, and 3) design of unstable aircraft for minimum trimmed drag. Problems in which only the handling qualities are to be improved are also readily solved by this method. One example is the design of unstable aircraft for supermaneuverability.

In future work, this method may be improved by implementing finite time integrals to evaluate the dynamic cost and using analytic expressions to calculate the gradient of J_d with respect to the control gains.

References

[1]Sawaki, E., Kobayakawa, M., and Imai, H., "A Design Method of an Aircraft with ACT by Nonlinear Optimization," *Translation of Japan Society of Aerospace Sciences*, Vol. 29, No. 85, 1986, pp. 142–162.

[2]Zeiler, T., and Weisshaar, T., "Integrated Aeroservoelastic Tailoring of Lifting Surfaces," *Journal of Aircraft*, Vol. 25, No. 1, 1988, pp. 76–83.

[3]Livne, E., Schmit, L. A., and Friedmann, P. P., "An Integrated Approach to the Optimum Design of Actively Controlled Composite Wings," AIAA Paper 89-1268, April 1988.

[4]Kroo, I., "LinAir—A Discrete Vortex Weissinger Method for Rapid Analysis of Lifting Surfaces," Desktop Aeronautics, 1987.

[5]Gill, P., Murray, W., and Pitfield, R., "The Implementation of Two Revised Quasi-Newton Algorithms for Unconstrained Optimization," National Physical Laboratory Rept. NAC 11, April 1972.

[6]Bryson, A. E., and Ho, Y. C., *Applied Optimal Control,* Halsted, Washington, DC, 1975.

[7]Golub, G., Nash, S., and Van Loan, C., "A Hessunburg-Schur Method for the Problem AX + XB = C," *IEEE Transactions on Automatic Control*, Vol. AC-24, No. 6, 1979, pp. 909–913.

[8]Ly, U., "A Design Algorithm for Robust Low-Order Controllers," Stanford University, Stanford, CA, SU-DAAR 536, Nov. 1982.

[9]Schweiger, J., Sensburg, O., and Berns, H., "Aeroelastic Problems and Structural Design of a Tailless CFC Sailplane," 2nd International Symposium on Aeroelasticity in Structural Dynamics, 1985.

[10]Kempel, R., McNeill, W., Gilyard, G., and Maine, T., "A Piloted Evaluation of an Oblique Wing Research Aircraft Motion Simulation With Decoupling Control Laws," NASA TP 2874, Nov. 1988.

[11]Kroo, I., "The Aerodynamic Design of Oblique Wing Aircraft," AIAA Paper 86-2624, Oct. 1986.