JOURNAL OF AIRCRAFT
Vol. 45, No. 4, July–August 2008

Design Forum

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Effective Design of Highly Maneuverable Tailless Aircraft

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DOI: 10.2514/1.32083

The correct design of unusual aircraft configurations is critical to achieving proper closed-loop flight dynamics. This paper is a guide based on decades of design experience on how the proper incorporation of yaw control power effects (their magnitude and bandwidth) within the iterative conceptual design process is critical to a successful aircraft design from a dynamics standpoint. Also critical is the open-loop level of stability. Factors that must be accounted for in designing tailless aircraft are also covered. These include the use of high-fidelity simulations with higher-order actuator models, time delays, and frequently excluded second-order effects for effective vehicle modeling and simulation.

Nomenclature

C_D	=	drag coefficient
C_L	=	lift coefficient
C_m	=	pitching moment coefficient
C_n	=	yawing moment coefficient
$C_{n\beta}$	=	directional stability derivative
C_l	=	rolling moment coefficient
$C_{y\beta}$	=	side force derivative with respect to β
g	=	Earth's gravitational acceleration, 32.17 ft/s^2
I	=	moment of inertia, slug $\cdot$ ft^2
L, M, N	=	moment component in the x, y, and z directions, lb $\cdot$ ft
m	=	aircraft mass, slug
P, Q, R	=	angular speed components in the x, y, and z direction, rad/s
$\bar{q}$	=	dynamic pressure, lb/s^2
S	=	wing reference area, ft^2
S_{vt}	=	vertical tail reference area, ft^2
T	=	engine thrust, lb
T_g	=	gross engine thrust at the given engine power setting, lb
u, v, w	=	linear airspeed components in the x, y, and z direction, ft/s
V	=	total velocity magnitude, ft/s
W	=	aircraft weight, lb
X, Y, Z	=	force component in the x, y, and z direction, lb
x_t	=	distance from center of gravity to the engine nozzle centroid (parallel to the longitudinal axis), in.
α	=	angle of attack, deg
β	=	sideslip angle, deg
γ	=	climb angle, deg
Δt	=	angular deflection in the XY plane of the thrust vector relative to the aircraft centerline, deg
δ_A	=	aileron deflection, deg
δ_R	=	rudder deflection, deg
δ_E	=	elevator deflection, deg
δ_{th}	=	throttle deflection, deg
θ	=	pitch attitude, deg
ρ	=	air density, slug/ft^3
ϕ	=	bank angle, deg
ψ	=	heading angle, deg

I. Introduction

EVER since the invention of the airplane, there have been attempts to completely remove the vertical tails from aircraft to reduce aerodynamic drag and weight [1]. Until the late 1970s, all of these attempted aircraft designs were failures, due primarily to poor handling qualities and some dangerous flight characteristics [1]. In the mid-1970s, the development of modern Fly-By-Wire (FBW) Flight Control System (FCS) technology finally made it possible to design tailless aircraft that were safe to fly and would have excellent handling qualities. Also in the mid-1970s, the development of stealth technology made the removal of vertical tails even more desirable, because it could reduce the side sector radar cross section [2–4]. Even though these two technologies have been available for over 20 years, only one aircraft without vertical tails has been designed, developed, and put into operational service: the B-2.

Although it is possible to design a tailless aircraft that can perform the subsonic cruise mission, it is still extremely difficult to design a

Presented at the Modeling and Simulation Technologies Conference (this is a highly modified paper based on this paper), Monterey, CA, 5–8 August 2002; received 10 May 2007; revision received 4 January 2008; accepted for publication 13 March 2008.

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highly maneuverable tailless aircraft, such as a tailless air superiority fighter. The primary obstacle that prevents the successful design of such a fighter is the lack of alternative yaw control devices with enough effectiveness and bandwidth to take the place of conventional rudders or all moving vertical fins. This lack of control power means that, if a tailless air superiority fighter is developed, it cannot look like a conventional fighter with its tails removed. As a result, the configuration development of these vehicles is critical, and significant new constraints are imposed on the design team. The entire development process has to be focused on how to shape the vehicle so that it can do the specified mission without the stabilizing effect of the vertical tail under the constraint of limited directional control power availability. The correct dynamic modeling for the analysis and simulation of these configurations very early in the design process is essential to assure that they can meet the mission requirements while simultaneously achieving closed-loop flight dynamics specifications (i.e., handling qualities). High-fidelity simulations with higher-order actuator models, along with the inclusion of time delays and often-excluded second-order effects, are also critical to achieving a successful design.

This paper briefly covers the limitations of existing FBW technology. It emphasizes what must be considered in the earliest conceptual stage of designing a low aspect ratio tailless aircraft under the constraint of limited yaw control power. A brief description of the use of vertical tails on conventional low aspect ratio aircraft is presented. Importantly, an explanation of the design problems that arise when tails are removed is given, and a set of important design goals and procedures are defined.

The primary advantage of vertical tails is that they are a rapid and highly reliable method for providing directional stability and yaw damping. With rudders mounted on the trailing edge, they are also effective directional control effectors. In addition, the directional control power is linear with rudder deflection (over nominal α and β ranges) and the aerodynamic time delay is negligible. When appropriately designed power actuators are used, a bandwidth of 3–5 Hz is easily attainable. The combination of effective rudders and high-bandwidth actuators provide good lateral-directional handling qualities for those aircraft that need stability augmentation.

From a performance point of view, the primary disadvantages of vertical tails are increased aerodynamic drag and weight penalties. Depending on the type of aircraft and the design mission, eliminating these penalties by eliminating the tails reduces aircraft weight and drag as follows [5–8]:

1) A drag reduction of between 5 and 10 % of the total aircraft drag in cruise flight.

2) A weight reduction of between 5 and 10 lb/ft^2 of projected side area.

Reducing the side profile of a military aircraft reduces its detectability. One more publicized detectability improvement is the reduction in the aircraft's radar cross section when vertical tails are eliminated.

From a stability and control point of view, the primary disadvantage of vertical tails is that their directional stability contribution and effective yaw control contribution usually decreases significantly at the higher α range [9,10]. They also decrease significantly at supersonic flight conditions above approximately Mach 1.3. For many conventional aircraft, the maximum usable α is determined by a loss of directional stability, which leads to the loss of yaw axis control. Depending on the configuration, this is characterized by nose wander, wing rock, or abrupt nose slice yaw departures. To prevent this from occurring, some type of warning must be given to the pilot as the critical α is approached. If not, some automatic limiting of the maximum allowable α has to be implemented in the FCS.

To provide a given level of directional stability for supersonic flight, vertical tails have to be made larger to offset losses due to Mach effects and aeroelastic deformations. As shown in Fig. 1, the rigid vertical tail contribution to the directional stability derivative $C_{n\beta}$ decreases above Mach 1.3. It is decreased even further due to aeroelastic deformation at low-altitude, high-speed flight conditions. The combination of these two effects typically determines the maximum allowable safety of flight limits for many supersonic aircraft.

II. Configuration Constraints

Suppose that the vertical tails of a modern fighter such as the F-22 were removed. As shown in Fig. 2, the aircraft would have a relatively large projected area forward of the c.g. By definition, it would be extremely unstable in the directional axis without the stabilizing contribution of the vertical tails. The time to double amplitude in the directional axis would be on the order of 0.1–0.2 s for a tailless F-22 flying at Mach 0.9 at sea level. This means that if a side gust changed the angle-of-sideslip (usually called beta or β) by 1.0 deg, the β angle would double every 0.1– 0.2 s thereafter if no corrective action were taken. Although it may be theoretically possible to stabilize the aircraft at this level of instability, the combination of time delays associated with available FCS components, the limitations on the available directional control power, and the requirement to fly safely in the presence of atmospheric turbulence would make it impractical due to safety of flight considerations. The FCS time delays are the combination of pure transport lags and many of the small first-order time constants that arise in the FCS control law design. An equivalent time delay is

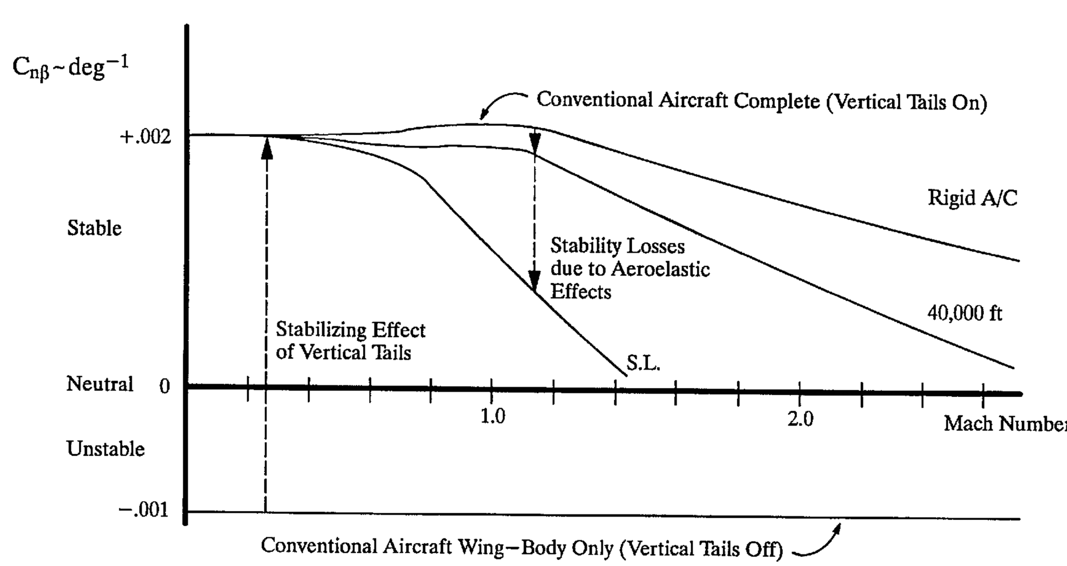

Fig. 1 Factors affecting directional stability.

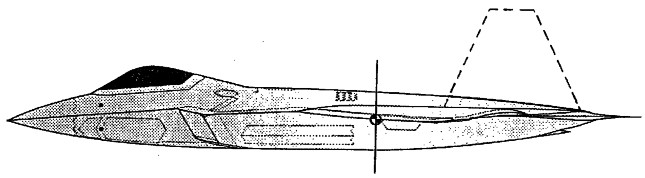

Fig. 2 Tailless F-22.

defined which combines the effects of both the pure transport lags and the small first-order time constants to provide a loop "reaction time" for the FCS. This loop reaction time is a measure of how much time is required to sense an external disturbance (e.g., a side gust) and then generate a corrective response.

Table 1 shows the range of equivalent time delays for the various components used to implement a modern digital FBW FCS. These values represent standard ranges for currently used, state-of-the-art sensors. Accelerometers and yaw rate sensors are used in several military and commercial aircraft. Sideslip sensors are usually found in military aircraft, especially in fighter aircraft. Digital flight control computers are found on many military aircraft and on the latest Boeing and Airbus airliners. The shorter time delays represent the latest production computer systems. Digital to analog (and analog to digital) converters are found in all current FBW systems. Power actuator time delay numbers are based on available numbers for multiple, deployed, military aircraft with traditional hydraulic actuation systems.

The loop reaction time for a FBW FCS using these components approaches 0.10 s, that is, from the time the side gust appears until the FCS begins to generate an effective correction. If these FCS components are used in a system where β is fed back to stabilize a directionally unstable aircraft with limited directional control power, the result is either marginal closed-loop stability due to a lack of phase margin or the actual loss of control due to aircraft departure.

The time histories shown in Fig. 3 show how controllability is affected following the insertion of a 20 ft/s side gust when the time to double amplitude is varied from 0.2 to 0.8 s. It is assumed that the directionally unstable aircraft is flying at Mach 0.9 at sea level, stabilized by a β feedback system with a 0.10 s loop reaction time. When the time to double amplitude is 0.2 s, an aperiodic divergence in β occurs before effective corrective action can be taken because the FCS loop reaction time, combined with effective delay due to actuator rate limits, is approximately the same as the time to double amplitude. When the time to double amplitude is increased to 0.4 s (assuming a configuration with a lesser degree of directional instability), the FCS is able to prevent aperiodic divergence but the closed-loop system is dynamically unstable. When the time to double is increased still further to 0.8 s, the FCS is able to maintain control after the insertion of the side gust.

From this example, it is seen that the FCS loop reaction time, combined with the effect of actuator rate limiting, has to be significantly less than the yaw axis time to double amplitude to guarantee safety of flight in turbulence and provide acceptable handling qualities. As a consequence, an aircraft designed to fly safely without vertical tails must have more inherent directional stability than a conventional aircraft with its vertical tails removed. In other words, additional constraints apply to the configuration layout of the purpose-designed tailless aircraft, so that it can be controlled with the available FCS components and yaw control effectors. Large-nose mounted radomes, forward-mounted bubble canopies, and long slab-sided fuselages forward of the c.g. are no longer permissible. Further discussion on configuration constraints is given in Sec. IV.

Table 1 FCS Component Time Delays

System component	Equivalent time delay, s
Accelerometer	0.005–0.01
Yaw rate sensor	0.005–0.01
Sideslip angle β sensor[a]	0.015–0.3
Flight control computer[b,c,d]	0.012–0.03
Digital to analog converter	0.01
Power actuators	0.02–0.04

[a]Depends on flight condition and sensor design details.
[b]Depends on frame rate, input data selection method, redundancy management, etc.
[c]This range assumes a frame rate of 100/s.
[d]Control law implementation can add additional delays due to structural mode filters, etc.

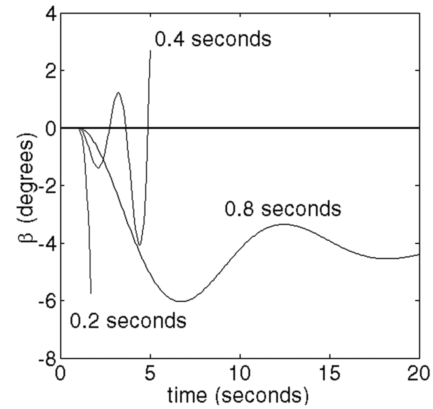

Fig. 3 Sideslip angle β due to $\beta_{\text{gust}} = 20$ fps for three T_2 values.

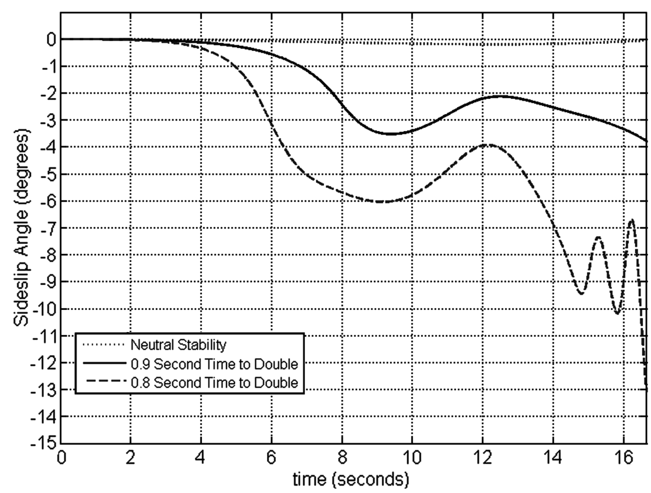

Fig. 4 Sideslip angle β due to rudder raps, open loop.

Another example demonstrating a conventional aircraft with its vertical stabilizers dramatically reduced in size is shown in Figs. 4 and 5. The aircraft simulated is a modified model of the F-15 ACTIVE (Advanced Control Technology for Integrated Vehicles) flight vehicle. The flight condition examined is Mach 0.8 at 20,000 ft. Figure 4 shows the open-loop responses, with the configuration modified 1) to be approximately neutrally stable directionally, 2) to have an approximately 0.9 s directional time to double amplitude, and 3) to have an approximately 0.8 s directional time to double amplitude. These responses were generated using a complete, nonlinear 6-DOF model. None of the control power magnitudes were changed from those for a conventional F-15B, as was used as the basis for the F-15 ACTIVE flight vehicle. Note that, as this is a nonlinear 6-DOF simulation, when the model departs the vehicle achieves very high bank angles, which does affect the open-loop (and, in Fig. 5, the closed-loop) directional response.

Figure 5 shows the closed-loop response of the modified F-15 ACTIVE flight vehicle with 0.9 and 0.8 s directional open-loop times to double amplitude. No changes were made to the conventional flight control system, which is equivalent to that used on the F-15B, despite the configuration change to make the aircraft directionally unstable. The time from the command to the resulting aircraft motion shows a closed-loop time delay of 0.1 s. The F-15 ACTIVE FCS frame length is 0.0125 s, which results from a frame rate of 80 frames/s.

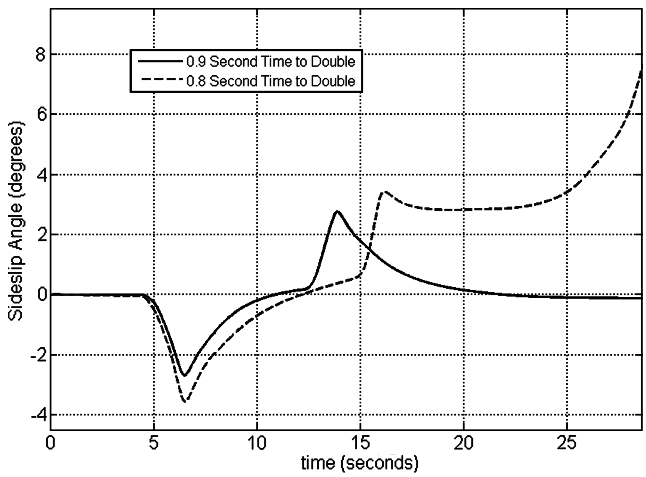

Fig. 5 Sideslip angle β due to rudder raps, closed loop.

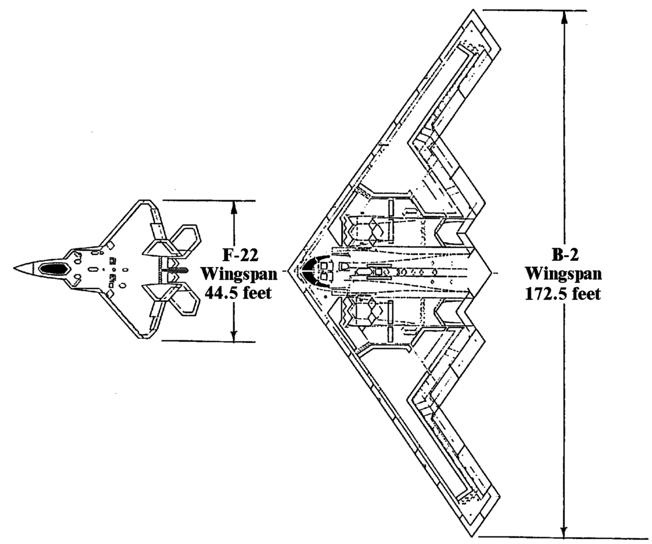

Fig. 7 F-22 vs B-2: top view.

III. Yaw Control Power Requirements

The yaw control power needed for the gentle maneuvers used by a high-altitude subsonic reconnaissance drone will be much less than that required to perform the aerobatic maneuvers used by a supersonic air superiority fighter. For such fighters, the yaw control power required is typically dominated by the requirement to coordinate rolls at high α, as shown in Fig. 6. For a conventional fighter at 20 deg α, the required yawing moment is twice as large as the required rolling moment. This increases to a yawing moment requirement five times greater than the required rolling moment at 40 deg α [9].

Wing tip drag devices are effective yaw controls for high aspect ratio flying wing designs like the B-2. The high aspect ratio wing provides a long moment arm so that the magnitude of the drag needed to produce a specified yawing moment is relatively small. When fully extended, the wing tip device can produce about the same amount of yawing moment as the conventional vertical tail with a deflected rudder. The average drag is less because the wing tip device is not fully extended at all times. One serious disadvantage of wing tip drag devices for multi-engine flying wing aircraft is that it is necessary to add more drag to maintain control of the yaw axis following an engine failure. This can be particularly serious if the engine fails at takeoff when the aircraft is at or near maximum weight and the engines are operating at maximum thrust. To overcome the additional drag, the engines need to have an emergency power rating or be oversized and derated for normal operation. This partially negates one of the flying wing's advantages, namely, that the lower drag of the aircraft produces better fuel economy because lower thrust engines can be used.

High aspect ratio flying wings are not suitable for supersonic flight. Low aspect ratio wings or highly swept delta planforms are required. Wing tip drag devices are much less attractive for these planforms, because the short wings require much higher drag forces at the tip to provide the specified level of yaw control moment. Figure 7 is a top view of the F-22 and B-2 to the same scale. It graphically illustrates the difference between low aspect ratio supersonic fighter and high aspect ratio subsonic bomber planforms. Given that the fighter requires relatively high yaw control power for high α rolling maneuvers, it is clear that other types of yaw control devices are required. Some possible alternatives are listed as follows: 1) wing-mounted spoilers for roll control that provide proverse yawing moments; 2) rotation of the entire outer wing about an axis parallel to the lateral axis of the aircraft to provide both roll and proverse yaw; 3) extendable strakes mounted on the nose (these can be very effective at high α but are typically ineffective at low α); 4) extendable dorsal and ventral fins to increase directional stability, so that other yaw effectors can provide yaw control power for maneuvering; 5) extendable high aspect ratio vanes on the top and bottom of the aircraft (Depending on the effectiveness and the required redundancy levels, it may be necessary to mount another set of similar vanes on the top and bottom of the aft fuselage.); 6) rotation of the entire forebody from side to side about an axis normal to the direction of flight; 7) differential horizontal tail producing a yawing moment due to differential drag on one side vs the other side of the aircraft.

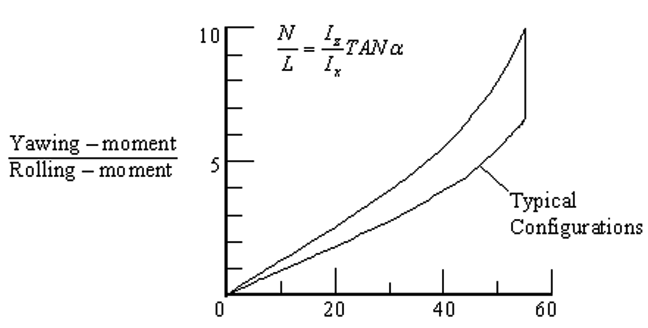

Fig. 6 Yaw control for roll coordination.

All of these devices individually add mechanical complexity and increase both cost and weight. With the possible exception of the aforementioned item 6, none of these devices (when used individually) will have the same effectiveness as that of the conventional rudders. Consequently, several different possible alternate devices may have to be combined to provide the required level of control power over the design α range. This compounds weight and complexity penalties, which could cancel most, if not all, of the cost and weight savings realized by the elimination of the conventional vertical tails.

IV. Yaw Thrust Vectoring

Given the difficulty of finding suitable aerodynamic yaw controls, the designer of low aspect ratio tailless aircraft is usually forced into considering the use of yaw thrust vectoring (YTV) as a yaw control device. The primary advantage of YTV is that it retains its effectiveness in flight regimes where conventional aerodynamic yaw controls are ineffective, for example, at low airspeeds or high α. YTV is also very effective in controlling the yawing moment due to an engine failure on a multi-engine aircraft having the engines mounted close to the centerline. The YTV can force the thrust vector of the remaining engine or engines to pass very close to or through the c.g. This eliminates, or at least minimizes, the yawing moment created by the failed engine.

There are two primary disadvantages of YTV. First, the engine power setting and the limited angular deflection of the thrust vector limits the total control power. Second, except for mechanical panels that are inserted directly into the exhaust jet, methods that have been developed and tested to change the direction of the engine's exhaust typically have a lower bandwidth capability than conventional aerodynamic effectors.

Because aircraft are typically optimized to reduce the aerodynamic drag to a minimum, and the removal of conventional vertical tails reduces drag still further, the thrust required for cruise is also minimized. Combined with the limited angular deflection of the thrust vector, the yaw control power attainable with YTV is sometimes severely limited. In any case, the aircraft must remain under control at any point within the permissible flight envelope for any possible engine power setting. This includes the case of the total loss of engine thrust due to compressor stalls during high-α maneuvers, fuel starvation due to fuel system malfunctions, etc. Control must be maintained for some period of time sufficient for engine restart or the clearance of malfunctions. Consequently, some aerodynamic yaw control device must be available to provide the necessary control power until the YTV can be restored.

Depending on how the thrust vectoring mechanism is implemented, the attainable bandwidth for YTV is typically limited to approximately 1.0–1.5 Hz for deflections in the range of plus or minus 50% of maximum deflection. This is entirely satisfactory for use as steady-state yaw trim devices or for slow, gentle maneuvers. However, it is not fast enough to provide the good lateral-directional handling qualities required for the rapid, large-amplitude maneuvers used by fighter aircraft. Once again, some high-bandwidth aerodynamic yaw control device is required to supplement the YTV.

For fighter aircraft with a high thrust-to-weight ratio, the use of thrust vectoring is beneficial for rapid, large-amplitude maneuvers at speeds up to about 250 kt calibrated airspeed. This corresponds to the upper left-hand corner of the Mach–altitude flight envelope. Fortunately, this is the part of the flight envelope where very high-α maneuvering takes place and where the engine would typically be operating at maximum power settings. In the case of the F-22, the use of thrust vectoring is of most benefit in the pitch axis, where it is used to prevent pitch up during high-α rolls. This unloads the horizontal tails and allows them to be used as both roll and yaw effectors to provide high rolling performance [2–4]. If the aircraft is designed to have neutral aerodynamic stability, the aerodynamic moments to be overcome during maneuvers are reduced to a minimum. Then thrust vectoring can be used for quasi-steady-state low-amplitude maneuvering throughout the flight envelope. For rapid, large-amplitude maneuvering at high airspeeds, aerodynamic control effectors must be used. Their effectiveness continues to increase as the square of the airspeed.

It is seen that YTV can be a useful yaw control for a tailless aircraft, but it can never be the primary or only yaw control device. Some type of aerodynamic control device is still always required, because control must be maintained even if all thrust is lost due to engine failure.

V. Air Data Sensor Requirements

As a minimum, all aircraft need to measure barometric pressure altitude and calibrated airspeed to operate within civil or military controlled airspace. Most high-performance aircraft also measure Mach number and α, because these parameters are used to specify the maximum safe boundaries of the flight envelope. For aircraft without vertical tails, the inherent directional stability is usually deficient at all flight conditions. Augmentation through the FBW FCS is required to provide satisfactory handling qualities. For these aircraft, the measurement of the angle of sideslip β is required to provide a signal for use as a feedback to augment the directional stability derivative $C_{n\beta}$. The reason for this follows.

Before about 1960, the vertical tails on conventional aircraft had always been made large enough to provide the required level of directional stability at the most critical flight conditions. Thus, no safety of flight augmentation was necessary. As aircraft performance increased up to the vicinity of Mach 2, the decrease in the directional stability contribution of the vertical tails required either larger vertical tails or artificial augmentation of the directional stability. This was done through the FCS to maintain safety of flight at those conditions. By using the latter approach, the designers avoided having to put on even larger vertical tails to provide this additional directional stability for safe flight. For these aircraft, the measurement of lateral acceleration provided a substitute for direct β measurements. This is possible as the lateral accelerometer signal is proportional to the side forces acting on the vehicle. With such configurations, the side force derivative with respect to β ($C_{y\beta}$) is typically quite large. In contrast, the $C_{y\beta}$ for aircraft without vertical tails is usually much smaller. It is essentially zero for flying wing aircraft like the B-2. As a consequence, the use of lateral accelerometer feedback for directional stability augmentation is usually not workable for tailless aircraft. Even where the $C_{y\beta}$ is not zero, the feedback gains have to be so high in a tailless aircraft that there is a strong possibility of structural mode coupling.

VI. Design Rules For Tailless Aircraft

From the preceding problem description, the design of a tailless aircraft requires consideration of a number of factors that are of no concern to the designer of a conventional aircraft with vertical tails. Every aspect of configuration development of tailless aircraft must be focused on what must be done to allow the elimination of the vertical tails, while simultaneously providing the yaw axis control power needed to achieve good handling qualities through FCS augmentation.

A. Fundamental Design Goal

The primary goal for the designer of an aircraft without vertical tails is to arrange the configuration to have neutral or very slightly positive directional stability at the cruise flight condition. If this is done, the directional control power required to achieve the desired handling qualities will be reduced to that required to overcome the moment of inertia around the yaw axis and the aerodynamic yaw rate damping. The total amount of control power required is still relatively large, because the moment of inertia around the yaw axis is always larger than the pitch or roll axis moments of inertia. Fortunately, the aerodynamic yaw damping is usually negligible due to the absence of vertical tail(s). Some of the things that can be done to achieve this goal are discussed next.

B. Requirements for Neutral Directional Stability

Neutral or slightly positive directional stability is easily achieved for high aspect ratio subsonic flying wing configurations by sweeping the wing back 25–35 deg. This amount of sweep makes the outboard trailing-edge control surfaces effective as pitch axis controls and provides a good compromise for both the pitch and yaw axes. It is not surprising that the Horton brothers and John Northrop arrived at this configuration independently for their flying wing designs.

For low aspect ratio tailless fighter configurations intended for transonic and supersonic flight, all directionally destabilizing features must be reduced or eliminated. Conversely, any features that add to directional stability should be retained or enhanced, provided it does not violate other configuration constraints, such as low radar cross section. The wing planform, thickness, and area will usually be dictated by performance requirements. It is best to choose features that produce the greatest contribution to directional stability over the α range required for the mission. Large amounts of pitch axis control power are required for basic fighter maneuvers. This control power is also required to prevent pitch up during high-α rolls. Canards, horizontal tails, and pitch thrust vectoring are possible candidates and should be investigated. They offer the possibility for synergistic use as roll or yaw effectors at high α. For the rest of the configuration, the following guidelines are suggested.

1) Make the fuselage cross section forward of the c.g. into a flattened oval and add chines. Flattening reduces the projected side area, and the chines improve $C_{n\beta}$.

2) Make the forward fuselage as short as possible by eliminating any nose-mounted radome.

3) For a piloted aircraft, the canopy must be smaller than usual or eliminated entirely.

4) Engine inlets should be located as far aft as possible to minimize destabilizing mass flow effects $\dot{m}V$. Engine inlets can generate forces which are directionally destabilizing if they are located forward of the c.g. The further forward the inlets are, the greater the destabilizing moment is. Locating the inlets aft of the c.g. is ideal for long slender configurations with highly swept delta wings.

5) For multi-engine aircraft, locate the engines as close to the centerline as possible. For a twin-engine aircraft, stacking the engines vertically on the centerline is preferable to having them side by side, because the side area aft of the c.g. will be greater and the yawing disturbance after an engine failure would be eliminated.

6) After the landing gear is extended, the nose gear doors should be closed to minimize the side area forward of the c.g. Conversely, the main gear doors should be left open to maximize the side area aft of the c.g. In addition, the gear should retract forward so that the c.g. of the clean configuration is as far forward as possible.

7) Avoid external stores with directional aerodynamic centers forward of the c.g., as these are aerodynamically destabilizing in sideslips.

It is recognized that some of these suggestions are controversial. They are listed here to indicate the kind of configuration details that must be seriously considered to make a tailless fighter aircraft feasible.

C. Case of Directional Instability

Even if the aircraft has neutral directional stability at the cruise condition, there may be other flight conditions where it is directionally unstable. This could be due to the effect of open weapon bay doors at high α, extension of nose-mounted mission sensors, fuel system failures causing an abnormal aft c.g. condition, etc. Assuming that the total available directional control power at these critical flight conditions is limited both in absolute terms and in the attainable control rates, the yaw axis time to double amplitude should always be greater than the following minimum allowable values.

For piloted aircraft, the worst case time to double amplitude should be greater than 0.50 s. This is recommended to ensure good lateral-directional handling qualities using available FBW FCS components to manipulate the limited directional control power available. With a directional time to double amplitude of less than 0.50 s, it is not reasonable to expect level 1 lateral-directional handling qualities throughout the flight envelope. For an unmanned air vehicle (UAV), the worst case time to double can be as low as 0.30 s because a UAV is typically controlled through a heading command or a way point steering autopilot mode. Inner-loop handling quality deficiencies caused by directional control power limits that would be unacceptable for a piloted aircraft may be tolerable in a UAV.

D. Redundant Aerodynamic Yaw Effectors

For aircraft that rely on a multiredundant FBW FCS to provide good handling qualities and insure safety of flight, redundancy has to be extended to the individual aerodynamic yaw effectors so that no single failure in the effector or its actuator will cause a loss of control. As mentioned previously, a tailless air superiority fighter will probably require several different types of aerodynamic control devices located at different places on the aircraft. This provides the necessary redundancy and reduces the vulnerability of the aircraft to combat damage in terms of exposed side area.

E. Sideslip Angle β Sensor Location

Redundant measurements of β are required because β feedback to the FCS is safety-of-flight critical. Dispersed locations for β sensor installations must be identified and reserved early in the configuration development phase. These locations must be chosen so that good measurements of β are possible. Also, β measurement errors due to gear extension, changes in engine power settings, etc., need to be minimized.

VII. Systems Analysis and Simulation

The stability and control and flight control system concepts are developed according to the design process given in Fig. 8. Note the iterative nature of the configuration development process. The responsibilities for this process include various disciplines not limited to designers, aerodynamicists, flight control engineers, structural dynamists, systems engineering, and weights. The objectives of this analysis are the identification of critical items (especially long lead items), the identification of potential tradeoffs, and the demonstration of feasibility of the design concept. The basic architecture is defined by the vehicle requirements. It is validated and refined using various linear analysis methodologies. The full-envelope design is further refined and nonlinearities included using 6 degrees-of-freedom simulations. The validation of this synthesis and analysis approach is continued using real-time simulation, bench and subsystem testing, system testing of the hardware/software combination, iron-bird testing, and through aircraft ground testing. This approach minimizes risk before in-flight testing of the entire system. Synthesis and analysis of system reconfiguration is also conducted as a part of this approach. This work results in a fully integrated system design combining navigation, guidance, and control functions, as in Fig. 9.

Figure 9 shows the variables which are often used in the flight control laws. The navigation and guidance systems use Earth-referenced Global Positioning System/Inertial Navigation System (GPS/INS) sensors to provide current map positions and the geometric altitude, which are compared with the mission plan to determine current navigational errors. Measured aircraft attitudes and velocities are used by the guidance system to determine attitude and velocity errors. Outer-loop control laws are implemented to eliminate the current navigational errors by working with the inner-loop control laws to generate the desired aircraft attitudes and velocities. Note the use of common sensors for these subsystems whenever feasible. The inner-loop control laws use measured rates, attitudes, and air data values to provide stability augmentation and desired flight dynamics and flying qualities. These include the angular speed components P, Q, R, the aircraft attitudes θ, ϕ, and ψ, the air data angles α, β, and the aircraft's airspeed and dynamic pressure V, $\bar{q}$.

The flight control analysis tools interplay in the design process, as shown in Fig. 10 [11,12]. They cover the full range of linear and nonlinear flight control analysis techniques, and are used in the design process as outlined in the previous figure. The full aircraft is modeled non-real-time using ACSL or, more commonly, MATLAB, Simulink, and Stateflow. A full aerodynamic database from wind-tunnel testing is included in this simulation. One important factor to determine in generating these databases is if the principle of superposition holds for the configuration being studied. Superposition of the effects of two or more aerodynamic control effectors may not be valid if, for example, a nose-mounted effector causes flow separation. In cases where superposition does not hold, more wind-tunnel testing and more complex mathematical models are required to properly account for each possible combination of such effectors.

Multiple approaches exist for determining the static and dynamic derivatives for the aircraft. These span a range from simple analytic estimates to the use of parameter identification methods in flight testing. Normally, greater accuracy is achieved by using more costly and more labor-intensive methods. Early in the design process, all or some of the stability and control aerodynamic data is estimated using DATCOM or equivalent area ruled methods. Vorlax or other vortex lattice methods are often used early in the design process to generate first cut aerodynamic stability and control data. Computational fluid dynamics (CFD) methods can also be used to generate these data, although the process to accomplish this is more time consuming than when using the simpler estimation tools. A complete propulsion deck provided by the vendor is also eventually included in this simulation. Aerodynamics from wind-tunnel tests are also included in the simulation. These data are often trim point linearized for incorporation into the block diagram model constructed using Simulink and Stateflow. The classical design and analysis tools within MATLAB

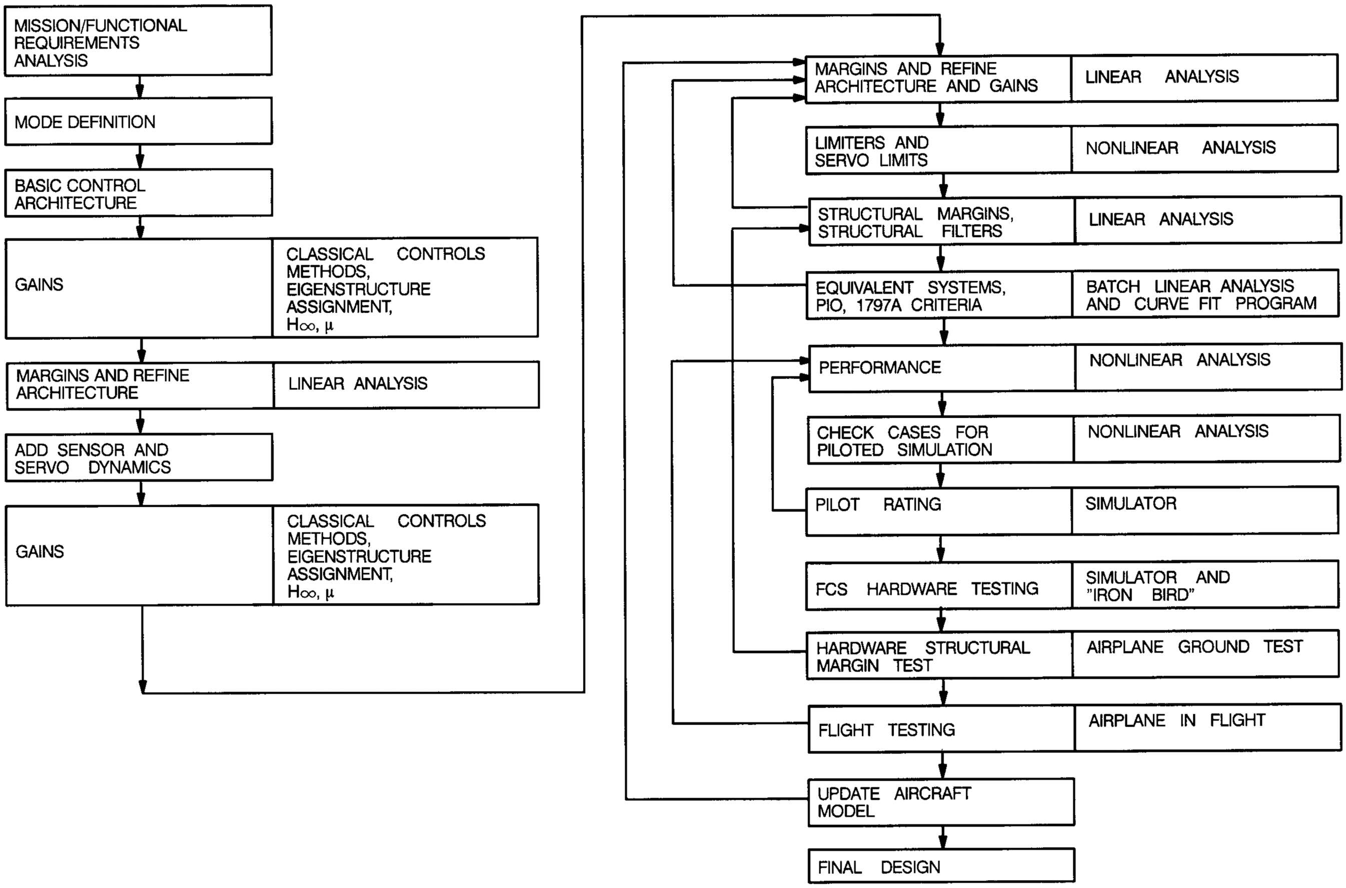

Fig. 8 Iterative design process.

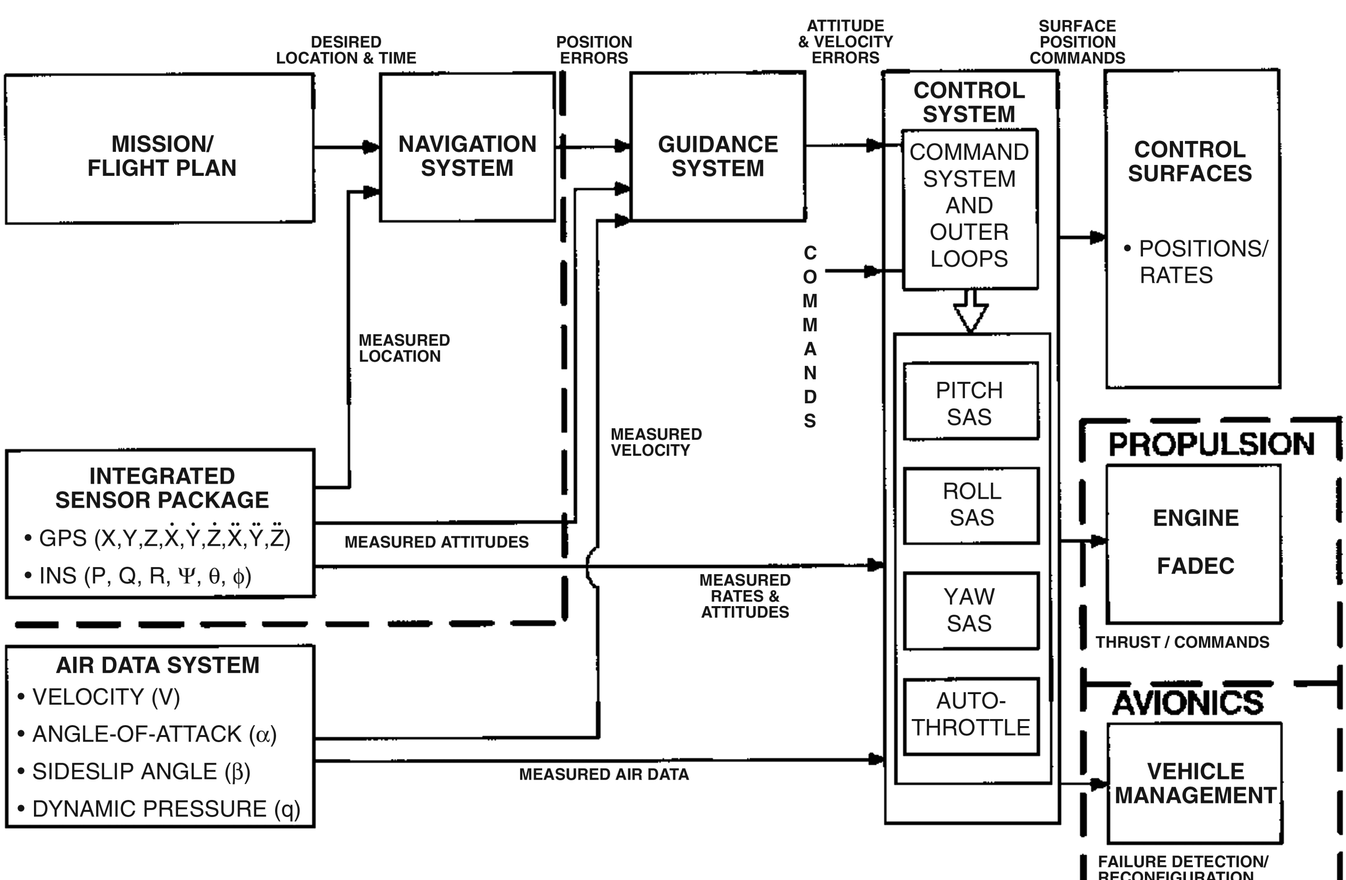

Fig. 9 Navigation, guidance, and control system.

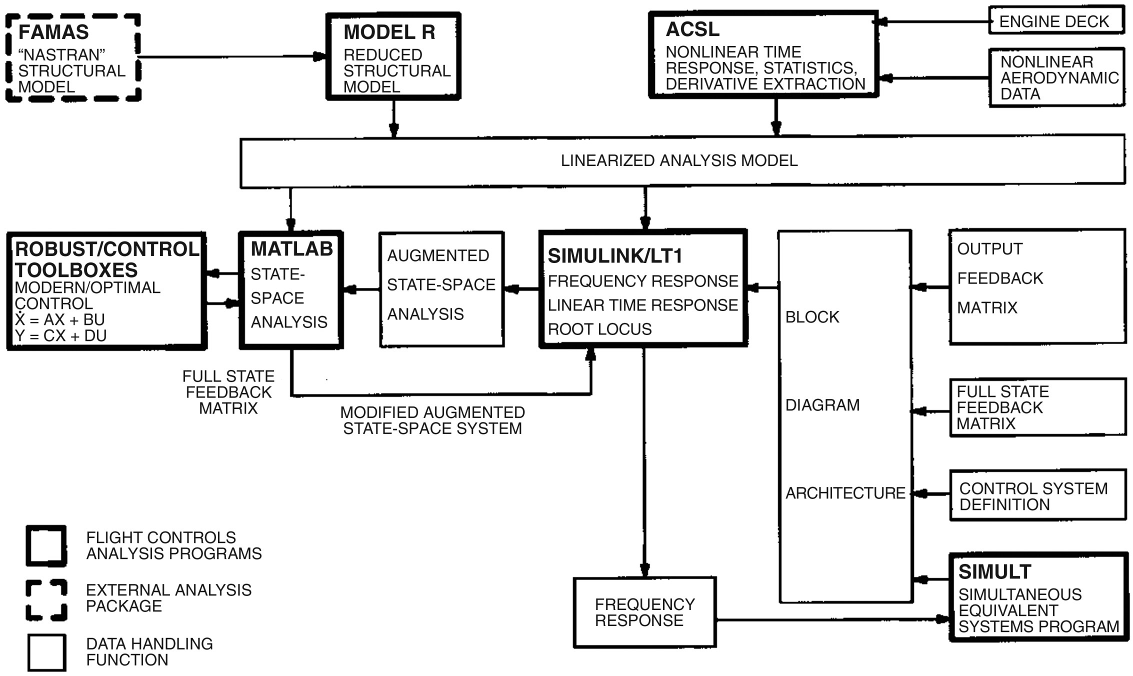

Fig. 10 FCS analysis tools.

are used to facilitate the development of the control laws. The design is validated using equivalent system techniques and real-time simulation. Configuration control becomes essential as the design progresses from the conceptual to the preliminary to the final design, and greater numbers of engineers are involved in the design and modeling process.

Database management and configuration control are critical as other facilities become involved in the design process. Additional facilities are required to provide, for example, a motion base simulator with visual system. Such expensive, special purpose facilities are essential to allow pilots to evaluate handling qualities, potential failure modes, etc., as early as possible in the design process. A variety of documents are generated to document key milestones in this process, which include the Preliminary Design Review, the Critical Design Review, Flight Safety Review, etc. With the greatly improved automated code generation capabilities available today, many more of these reports and specifications are being computer generated and are kept under automated configuration control.

VIII. Design Example

The aerodynamic yawing moments, which have to be overcome to maintain directional control of a tailless aircraft, increase with the square of the airspeed. Using standard nomenclature, the aerodynamic yawing moment due to β is defined as follows and is positive for airplane nose-right motion. The aerodynamic yawing moment is given by the following equation:

$$N = \{1/2(\rho)\mathrm{V}^2\}(S)(b)[C_{n\beta}(\alpha, \mathrm{Mach})](\beta)$$

The term in the brackets is the dynamic pressure, usually called $\bar{q}$. When the true airspeed V is doubled, $\bar{q}$ (and N if $C_{n\beta}$ is unchanged) is increased by a factor of 4. For a directionally unstable aircraft, $C_{n\beta}$ (α, Mach) is negative. If we make realistic assumptions for a single engine tailless fighter aircraft flying at Mach 0.9 at approximately sea level with a wing span of 45 ft, a wing area of 700 ft^2, a $C_{n\beta}(\alpha, \mathrm{Mach}) = -0.001$ per degree of β, the aerodynamic yawing moment generated by β is

$$N/\beta = (1200)(700)(45)(-0.001) = -37,800 \quad (\mathrm{ft} \cdot \mathrm{lb}/\deg \text{ of } \beta)$$

For this design example and flight condition, a step side gust of 20 ft/s (approximating a wind shear) generates a β of approximately 1.14 deg. If the gust comes from the right side (positive β), an aerodynamic yawing moment of about $-43,100$ ft · lb would begin to accelerate the aircraft in the airplane nose-left direction, which acts to make β larger. In contrast, the following equation gives the yawing moment available from YTV at a given engine power setting:

$$Nt = Tg[x_t \sin(\Delta t)]$$

If we assume that the engine in our tailless aircraft generates 22,000 lb of gross thrust at the examined flight condition, and that x_t is 20 ft, the corrective yawing moment available for each degree of thrust vector deflection is

$$Nt/(\Delta t) = (22,000)(20)(0.0175) = 7700(\mathrm{ft} \cdot \mathrm{lb}/\deg \text{ of YTV})$$

Approximately 5.6 deg of YTV would be required to just balance the initial 43, 100 ft · lb of yawing moment created by the 20 ft/s side gust computed as before. Assuming that the FCS has a loop reaction time of about 0.1 s, and the maximum attainable rate of thrust vectoring mechanizations is about 20 deg/s, approximately 0.38 s is required to achieve the 5.6 deg of thrust vector deflection. However, because the yaw axis time to double amplitude is about 0.2 s for $C_{n\beta}(\alpha, \mathrm{Mach}) = -0.001/\deg$, the aircraft will already have departed controlled flight by the time 0.8 s have elapsed.

From this simple example, we see that if the YTV is limited in both control power and rate, the FCS cannot generate the corrective yaw moments required to maintain control when $C_{n\beta}(\alpha, \mathrm{Mach}) = -0.001$ per degree. However, if we require that the aircraft have near neutral directional stability [$C_{n\beta}$ (α, Mach) approximately zero or slightly positive], the FCS can maintain directional control using YTV when disturbed by side gusts.

IX. Summary and Conclusions

The design of a supersonic air superiority fighter without vertical tails is possible. To accomplish this, some traditional fighter aircraft features will almost certainly have to be abandoned. New technologies will also need to be substituted. For example, the conventional forward-mounted bubble canopy with its large forward side area must be made significantly smaller or eliminated entirely.

An F-15B-based simulation was used to demonstrate the destabilizing effects of these features without traditional vertical stabilizers. A "virtual reality" cockpit, incorporating sensor fusion to provide situational awareness for the pilot, could replace the bubble canopy. Many other novel features must be incorporated into the airframe to make it safe to fly and yet have good mission capability, including nontraditional control effectors. A number of new aerodynamic yaw effectors have to be investigated, developed, and incorporated into the airframe design so that the FBW FCS can provide good handling qualities. These will need to be evaluated using CFD codes, wind-tunnel tests, and motion-based handling qualities simulators. Eventually, these will need to be verified in flight.

The combination of so many new features increases the development risk. It requires a thorough analysis and a detailed simulation program to be initiated very early in the conceptual design phase. The cost of correcting errors at this stage is about 1% of the cost of fixing them in the fligh-test stage. This justifies the up-front costs associated with the wind-tunnel testing required to provide a good aerodynamic database. The mathematical models used in the simulation must be rigorous. They must include second-order effects that can be critical to the understanding of the potential interactions between the many novel design concepts. Uncertainties in the aerodynamic characteristics must be systematically studied to be certain that there is an adequate design margin before proceeding into detail design. The most important failure modes should be simulated and system redundancy requirements defined. Finally, an evaluation of concept feasibility must be made. The ultimate objective of the concept definition phase is the accurate identification of the real penalties in weight, complexity, and costs associated with the removal of the vertical tails. This is necessary to avoid modifications to the design concept in later phases of the program, resulting in cost overruns and schedule delays.

Acknowledgments

The authors would like to thank Lockheed Martin, NASA, and the University of Kansas for their support of the work documented in this paper.

References

[1] Wooldridge, E. T., *Winged Wonders, The Story of the Flying Wings*, Smithsonian Inst. Press, Washington, D.C., 1985.
[2] Abrams, R., "YF-22A Prototype Advanced Tactical Fighter DEM/VAL Flight Test Program Overview," *Society of Flight Test Engineers Symposium*, Society of Flight Test Engineers, Lancaster, CA, Aug. 1991, pp. 1–23.
[3] Clark, C. K., and Bernans, M. R., "High Angle-of-Attack Characteristics of the YF-22," AIAA Paper 91-3194, Sept. 1991.
[4] Mullin, S. N., "Evolution of the F-22 Advanced Tactical Fighter," *1992 Wright Brothers Lecture*, AIAA Paper 92-4188, Aug. 1992.
[5] Roskam, J., *Part 1: Preliminary Sizing of Airplanes*, Design, Analysis and Research Corp., Lawrence, KS, 2003.
[6] Roskam, J., *Part 2: Preliminary Configuration Design and Integration of the Propulsion System*, Design, Analysis and Research Corp., Lawrence, KS, 1997.
[7] Raymer, D. P., *Aircraft Design: A Conceptual Approach, Fourth Edition*, AIAA Education Series, AIAA, Reston, VA, 2006.
[8] Nicolai, L. M., *Fundamentals of Aircraft Design*, METS, Inc., San Jose, CA, 1984.
[9] Nguyen, L. T., "Flight Dynamics Research for Highly Agile Aircraft," Society of Automotive Engineers TP 89-2235, Sept. 1989.
[10] Brandon, J., and Nguyen, L., "Experimental Effects of Forebody Geometry on High AOA Static and Dynamic Stability," AIAA Paper 86-0311, 1986.
[11] "Getting Started with MATLAB 7," The Math Works, Natick, MA, March 2007.
[12] Colgren, R., "Basic MATLAB, Simulink, and Stateflow," AIAA Education Series, AIAA, Reston, VA, 2007.