AUTOPILOT CONTROL STRATEGIES FOR REDUCING WEAPON SEPARATION TRANSIENT MOTION

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Abstract

One of the essential elements in assessing the effectiveness of a weapon system is the determination of the separation characteristics of the weapon system from its release platform. The release platform can impart significant transient motion on the weapon due to flow field interference effects. This, in turn, can result in limitations in the employment envelope of the weapon system. When the AGM-130 air-to-ground missile system was integrated with the F-15E aircraft, this aircraft-induced transient motion was found to be excessive. To increase the employment capability of this weapon system, a program was undertaken to develop new autopilot control strategies. This paper presents a definition of the problem, the proposed solution, and the results of subsequent flight testing.

Nomenclature

ACSTV	Active Control Separation Test Vehicle
EPROM	Erasable Programmable Read-Only Memory
GTV	Guided Test Vehicle
Hz	Hertz
KCAS	Knots Calibrated Airspeed
ms	Milliseconds
N_y, N_{yc}	Lateral Acceleration, Lateral Acceleration Command
t	Time (Seconds)
WSO	Weapon Systems Officer
β	Sideslip Angle (Degrees)
δ_r, δ_q, δ_p	Yaw, Pitch, Roll Flap Deflections (Degrees)
$\ddot{\psi}$	Yaw Angular Acceleration (Degrees/Second^2)
$\dot{\psi}$, r	Yaw Angular Rate (Degrees/Second)

1. Introduction

During the certification of a weapon system on a release platform, various elements of both the weapon system and release platform are assessed. These elements include physical compatibility, electrical and mechanical interfaces, and mission planning. Another essential element is the weapon system's separation characteristics. Whether the weapon is released in a jettison or launch mode, the primary concern is the safety of the release platform. In the launch mode, the recovery from the separation transient motion must also be considered. If the weapon system does not recover from this transient motion, the entire mission's effectiveness is compromised. Thus, investigation of the separation characteristics of a weapon system is warranted during its certification process. Aircraft flow field effects on the separation characteristics can be particularly severe for weapon systems released from high-wing aircraft where the store is carried on the wing pylon close to the aircraft fuselage, such as presented in Figure 1.

2. Background

To enhance the standoff ground attack capability of the F-15E aircraft, a certification program was conducted to include the AGM-130/GBU-15 family of weapon systems. In general, the GBU-15 weapon system consists of a forward guidance section (including four cruciform strakes and a seeker section having either a charge coupled device or an imaging infrared camera), an aft control section (including four cruciform wings and trailing-edge flaps, a flap actuator package, an autopilot, attitude and acceleration sensors, and a data link), and either a MK84 or BLU-109 2000-pound warhead. The AGM-130 weapon system airframe is similar to the GBU-15 airframe but also includes an underslung rocket motor for increased standoff range. During development of these weapon systems, the primary release platforms were the F-4E and F-111F aircraft. Since the initial development, these weapon systems were implemented on the F-15E aircraft. Both the GBU-15 and AGM-130 weapon systems are carried (and subsequently released) on the F-15E aircraft's inboard wing pylon. The certification process involved integrating two independently-developed systems (i.e. the missile and the aircraft) and determining the adequacy of the resulting separation characteristics. While the user goals for carriage, jettison, and employment capabilities were readily met for the GBU-15 weapon system, the AGM-130 weapon system did not initially satisfy the user goal for employment capability.

3. Nature Of The Problem

The limitation of the employment capability was directly related to the severity of the aircraft flow field in the yaw plane. The aircraft flow field intensity is greatest near the carriage location and decreases to zero at approximately 5 to 8 feet below the pylon. During separation, the weapon system translates this distance in 0.5 to 1.0 seconds. Within this time, a large nose-outboard yawing moment acts on the weapon and is presented in Figure 2. The moment was computed from separation flight test data via the methodology presented in Reference 1. The amplitude is given in units of equivalent yaw flap deflection angle. Notice that with the weapon in the carriage position (t = 0), the flow field is equivalent to 8 - 10 degrees of yaw flap. The four control surfaces each have a maximum travel of +/- 20 degrees; however, 5 degrees must be reserved for roll control. Therefore, only 15 degrees is available for yaw control to counter the flow field moment. It takes 50% - 70% of this allocation just to balance the moment at carriage.

If the surfaces start at zero deflection, the actuator bandwidth is 10 Hz, and the maximum fin rate is 200 degrees/second, then at least 100 ms is required for the yaw flap deflection to reach the 10 degrees needed to balance the flow field. During this time, the weapon is yawing nose-outboard, and the angular rate is increasing. As verified by flight testing, recovery of the weapon system from the separation transients is doubtful if the yaw rotation exceeds 8 degrees. This is due to the statically unstable yaw-plane characteristics and the yaw/roll coupling of the airframe.

The flight control system must provide large yaw flap deflections as quickly as possible before the yaw angles and rates of the weapon airframe become excessive. The ability to provide these control deflections is limited by the available sensors (rate gyros and accelerometers) and actuators. The actuator response is in turn limited by finite bandwidth and rate and torque limitations. These limits are functions of aerodynamic loads on the control flaps.

4. Flight Control Strategy

The original autopilot approach for controlling the weapon separation transient consisted of 3 structures, one for each rotational axis. To control the pitch motion, a high bandwidth (3 - 5 Hz) rate loop was employed. The pitch rate command was 10 degrees/second nose-downward which decayed linearly to zero at 0.5 seconds after autopilot control was initiated. This resulted in a 2.5-degree, nose-down pitch attitude (with respect to the carriage position), assuming the aircraft flow field effect in the pitch plane was negligible. This command aided in the translation of the weapon away from the release aircraft.

To control the roll motion, an attitude loop was employed with an inner rate loop for stability. To minimize the effects of sideslip angle (β) on roll, a component of the roll flap command was proportional to lateral acceleration. The roll loop was assured at least 5 degrees of flap and some fin rate capability by limits on the pitch and yaw fin commands.

To control the yaw motion, an acceleration loop was utilized with an inner rate loop as shown in Figure 3. The acceleration loop command was zero to minimize the vehicle sideslip angle. With regard to the goal of minimizing the launch transient by achieving large yaw flap deflections quickly, this concept had several limitations. These included fin rate and position limits of 150 degrees/second and 15 degrees, respectively. As stated previously, roll control was allocated some authority by these limits. In addition, the speed of application of the control is restricted by actuator dynamics (10 Hz), autopilot sampling (100 Hz) and computational delays, and sensor dynamics (15 Hz). Further delay was involved due to sensing angular rate, where the disturbance was a torque which generates angular acceleration.

The elements of the new autopilot control strategy are:

(1) Nonlinear yaw flap commands based on angular acceleration
(2) Preset yaw flap before release based on release aircraft speed and altitude
(3) Pitch attitude control with nose-down command

The rationale for this strategy is to counter the aircraft-induced yawing moment acting on the weapon airframe. Since the moment is known to be weapon-nose-outboard, the control algorithms take advantage of this knowledge. The weapon/aircraft umbilical provides the autopilot with a logic signal to determine if the weapon is on the right or left aircraft wing. Since the duration of the disturbance is less than 1.0 second after release and the intensity decays with time, the elements of the new autopilot control strategy involving the yaw flap command are restricted from 0.4 to 0.5 seconds, and authority will be diminished with time. The first two elements, which affect only yaw control, are shown in Figure 3.

The first element of the yaw fin command based on angular acceleration is designed to balance the disturbance. The angular acceleration is proportional to the net yawing moment acting on the weapon airframe. This includes the aircraft flow field disturbance and the weapon fin control. Use of acceleration introduces noise and is potentially destabilizing. However, these effects can be reduced by appropriate right/left wing logic, thresholds, and time variable gains. This component also bypasses the rate loop compensator fin rate limiters of 150 degrees/second.

The second element of the new autopilot control strategy is a preset yaw flap prior to release but only after Intent-to-Launch by the Weapon Systems Officer (WSO). By commanding a yaw flap approximately 400 milliseconds prior to release, the bandwidth and rate limitations of the actuator can be partially overcome.

The introduction of pitch attitude control element was to ensure weapon velocities away from the release aircraft. The shortcoming of the "rate loop only" approach was that the aircraft flow field pitching moment can be "captured" as a pitch up attitude. The pitch attitude control loop returns the weapon to a 2.5 degrees nose-down orientation consistent with the rate command.

5. Design & Implementation Considerations

Since the AGM-130 employs a digital autopilot, the new strategies were implemented entirely in software. The primary changes occur in the yaw control algorithms, involving a nonlinear fin control concept plus commands to preset fins prior to release. These changes effectively give the yaw channel additional priority over pitch and roll for the first half second after release. The combination of the two elements allows the use of a smaller preset. This avoids overcorrections which rotate the weapon nose inboard toward the aircraft fuselage. This property yields greater tolerance for errors associated with presetting the flaps as a function of release speed and altitude.

The nonlinear fin commands involve a set of 3 logic equations (Figure 4) with 5 inputs (aircraft wing I.D.(left/right), time, angular acceleration ($\ddot{\psi}$), angular rate ($\dot{\psi}$), and logic mode) and a single output (yaw fin command). This can be considered a "fuzzy logic-like" concept. The wing I.D. signal is used to decide if additional control is needed above the "prepositioned" level. The additional yaw control is applied only if nose-outboard yaw rotational motion is exhibited in a 100-millisecond window from sensed weapon release. No control will be applied unless an outboard angular acceleration ($\ddot{\psi}$) exceeds a threshold (100 degrees/second^2) within this window. The additional yaw control is decayed linearly to zero at 0.5 seconds. After the angular acceleration ($\ddot{\psi}$) falls below the threshold, the fin command is decayed to zero proportional to angular rate ($\dot{\psi}$) to provide continuous control. The combination of modes # 2 and # 3 prevent oscillations in the output.

A typical time history of the control fin output is shown in Figure 5 along with the flow field represented as a equivalent fin deflection. Notice that after a 50-millisecond delay for the signal to exceed the threshold, the shape of the control roughly follows the flow field. Also shown is the effect of mode # 2 and # 3 in eliminating oscillations. From a classical stability perspective, the component is destabilizing. The mode control logic prevents more than one half cycle of oscillation. The linear gain decay provides a smooth transition to the original system. Selection of the control gains (K_o, K_1) is based on maximum control (15 degrees) and maximum yaw rate (30 degrees/second) and balancing the $\ddot{\psi}$ due to flow fields, respectively. Constant gain values have been found to be acceptable.

The magnitude of the preset yaw flap deflection is computed from WSO-entered values of planned release speed (Mach number) and altitude (h). The equation for the preset has the form:

$$\delta_{r(\text{preset})} = (12 \text{ degrees} / 0.20) * (\text{Mach} - 0.70) * [\,1.0 - (h / h_1)\,(\,1.0 - k_{20})\,]$$

where parameters h_1 and k_{20} are functions of Mach.

The preset is shown graphically in Figure 6 for Mach = 0.85. Also shown are the +/- 2 sigma error bands about the desired preset and the allowable extremes (max / min presets).

6. Flight Test Results With Autopilot Modifications

Flight testing involved test vehicles that were not all-up weapon systems but were Active Control Separation Test Vehicles (ACSTVs). ACSTVs differ from Guided Test Vehicles (GTVs) in that the ACSTV did not contain an active seeker, a live warhead, or a live propulsion unit (since the propulsion unit does not activate in the vicinity of the aircraft). The mass properties of these elements of the AGM-130 were accounted for by ballast and/or inert systems, which resulted in reduced cost when compared to a GTV. The active components of the AGM-130 which were essential to the separation phase of flight and therefore included were the actuator package, the autopilot assembly, the sensor package consisting of three-axis accelerometers as well as pitch/yaw and roll gyros, and a telemetry package. The actual release conditions were recorded on the aircraft and utilized in subsequent analyses. In addition to telemetry data, film coverage of the release was acquired via onboard and chase aircraft cameras. This allowed for qualitative verification of the telemetry information as well as verification of safe separation from the release aircraft.

After the EPROM chips on the autopilot hardware board were modified to incorporate the autopilot software modifications, the autopilot function was verified through hardware-in-the-loop testing. Once completed, the autopilot assemblies were re-installed in the ACSTVs for flight test. Figure 7 presents a comparison of before and after the autopilot modifications for the 0.85 Mach/500 KCAS release. As seen, the separation transients were reduced significantly. Previously, the ACSTV had rolled lugs-outboard to an amplitude of 63 degrees. With the modifications, the roll amplitude was reduced to 6 degrees. Figure 8 presents the flap deflection time histories, which indicated that the desired preset flap deflection was implemented correctly. Since reduction in sideslip angle is essential to recovery from the separation transient motions, the sideslip angles were derived via the method in Reference 1 (since no direct determination was available). Figure 9 displays the results of an effective reduction in motion by comparing the sideslip angle time histories. Figure 10 presents a comparison of before and after the autopilot modifications for the 0.85 Mach/550 KCAS release. Whereas the previous ACSTV did not recover from the launch transient motions, the autopilot modifications resulted in the transient motions being significantly damped. Figure 11 presents the flap deflection time histories, which indicated that the desired preset flap deflection was implemented correctly. Figure 12 displays the results of an effective reduction in motion for this release condition as well by comparing the sideslip angle time histories.

7. Conclusions

Future weapon systems will employ nonstandard airframe configurations for carriage and/or stealth considerations. These airframes may introduce difficulties when separating from a release platform. To address this, autopilot capabilities will require robust algorithms to address these challenges brought about the aircraft flow field environment. For the AGM-130 weapon system, a capability has been developed to enhance the weapon system's effectiveness on a new release platform while not compromising its established effectiveness from other release platforms. This type of robust approach to solving the separation element has application for the weapon systems of the future.

Appreciation is expressed to Col. Frank DeLuca and all members of the AGM-130 System Program Office for the dedicated support of this program, thereby allowing for enhanced employment of the AGM-130 weapon system on the F-15E aircraft.

References

1. "Determination of Aircraft Flow Fields from Flight Test Data of Separating Stores", AIAA-96 -0069, R.D. Ehrich & M.N. Lamb, 1996 AIAA Aerospace Sciences Meeting.

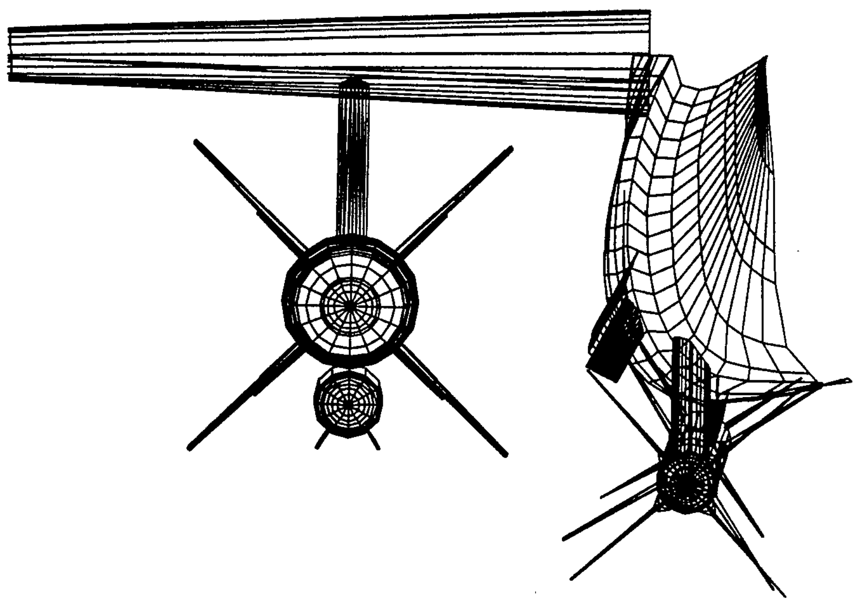

Figure 1 Typical Store Carriage On Release Aircraft

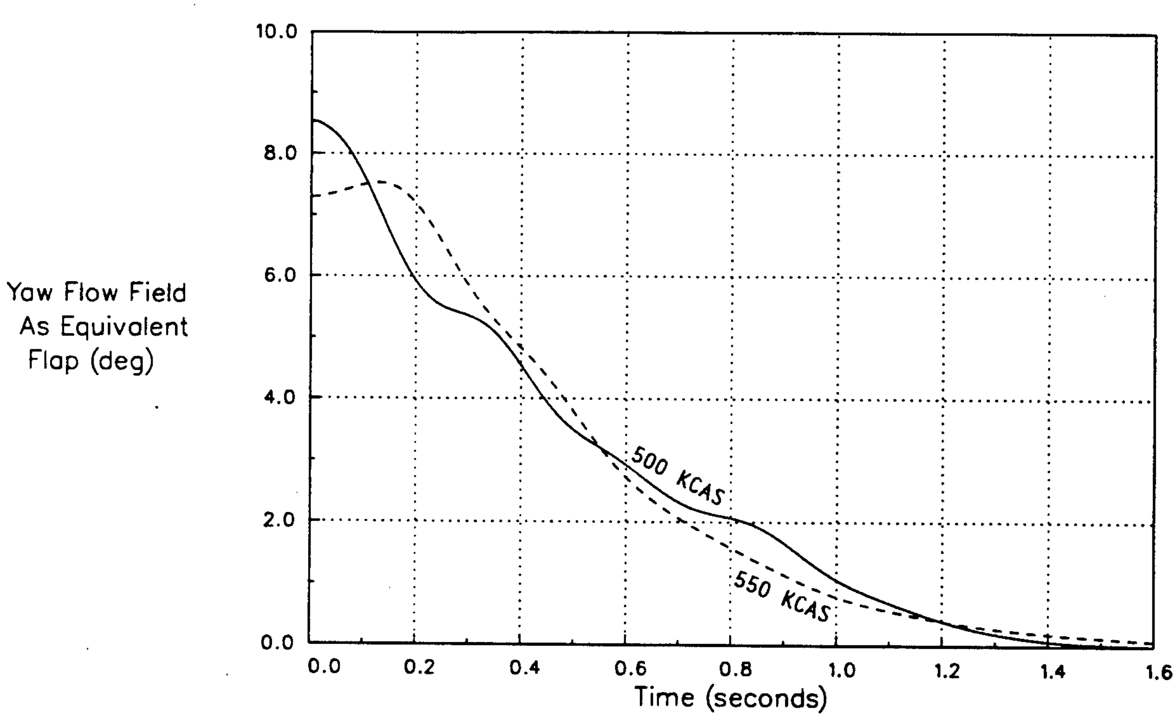

Figure 2 Yaw Flow Field Magnitude As An Equivalent Flap Deflection (Mach = 0.85)

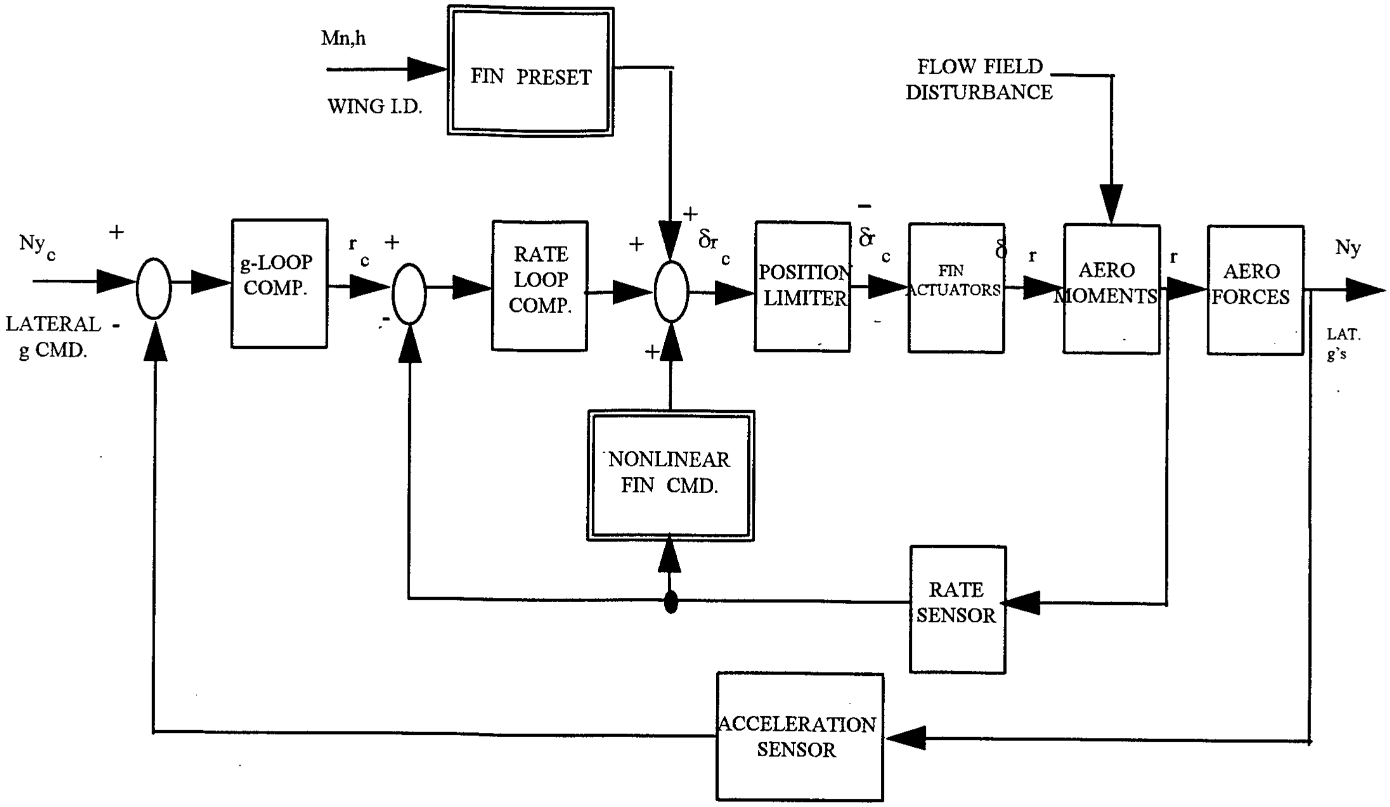

Figure 3 Block Diagram of Yaw Separation Control Concept

BASIC MODES (RIGHT WING)

<u>MODE #1</u> LOGIC: $[(t \leq 100\text{ms})$ AND $(\dot{\varphi} > 0)$ AND $(\Delta\dot{\varphi} > 1°/\text{s})]$
CONTROL: $\delta' rc = [K_1\, \Delta\dot{\varphi} + K_o\, \dot{\varphi}]\, U(t)$

<u>MODE #2</u> LOGIC: [(EXIT FROM MODE #1) AND $(\dot{\varphi} > 0)$]
CONTROL: $\delta' rc = K_o\, \dot{\varphi}\, U(t)$

<u>MODE #3</u> LOGIC: [(EXIT FROM MODE #2) AND $(\dot{\varphi} \leq 0)$]
CONTROL: $\delta' rc = 0$

$$\text{WHERE } U(t) = \begin{cases} [1.0 - (t/t^*)] & \text{FOR } t < t^* \\ 0.0 & \text{FOR } t \geq t^* \end{cases} \qquad t^* = 0.5 \text{ sec}$$

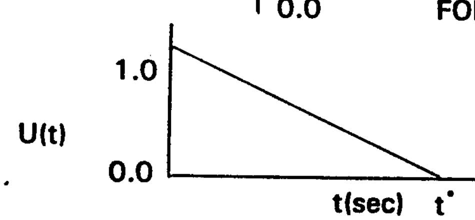

Figure 4 Logic Equations For Nonlinear Fin Commands (Based on Angular Accelerations)

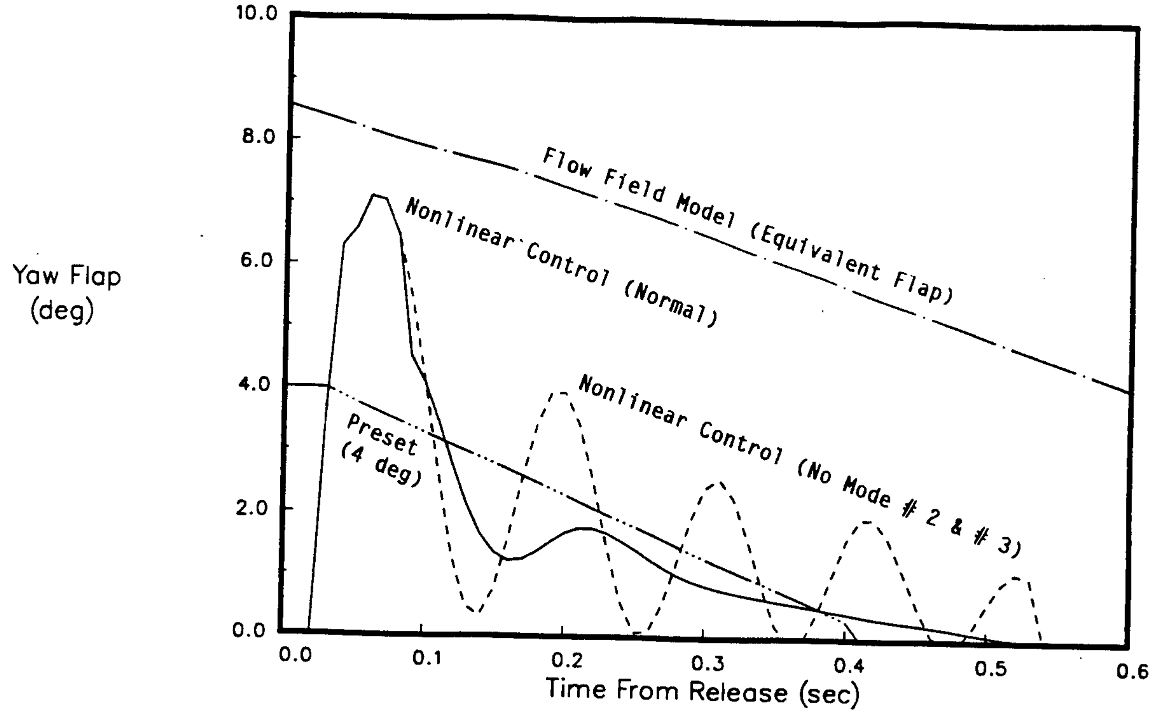

Figure 5 Simulation-Generated Outputs From Nonlinear Fin Control Concept

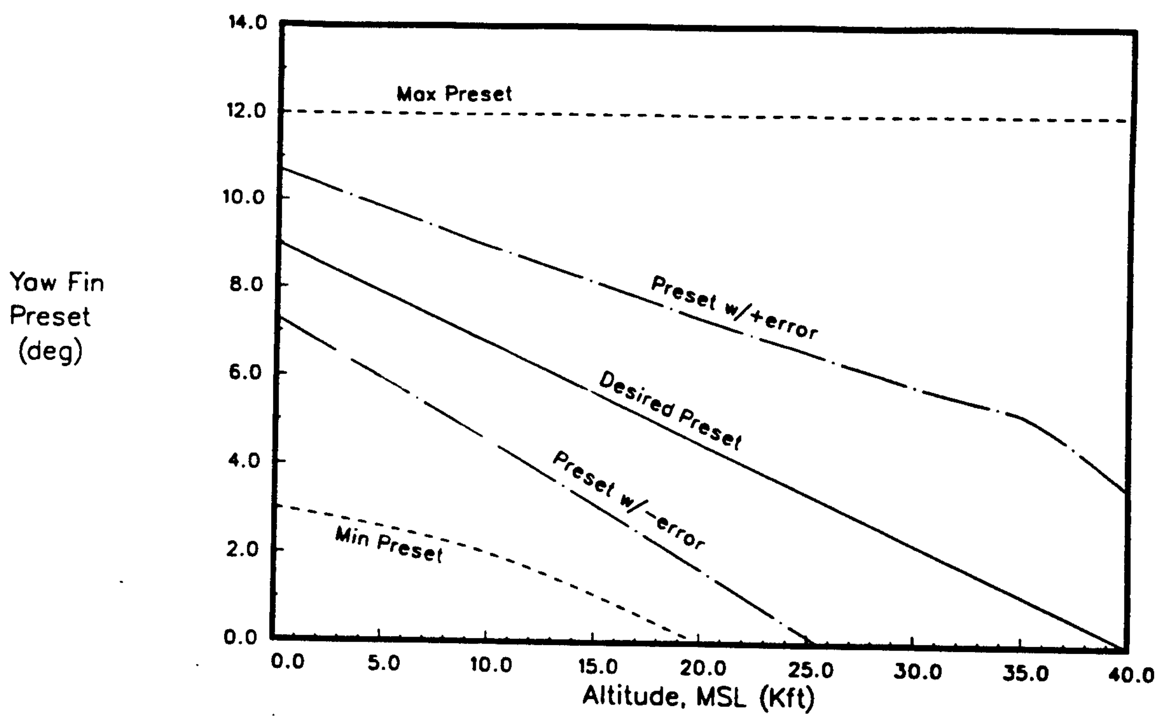

Figure 6 Allowable Range Of Preset Values Compared To Expected +/- 2σ Band Of Errors (Mach = 0.85)

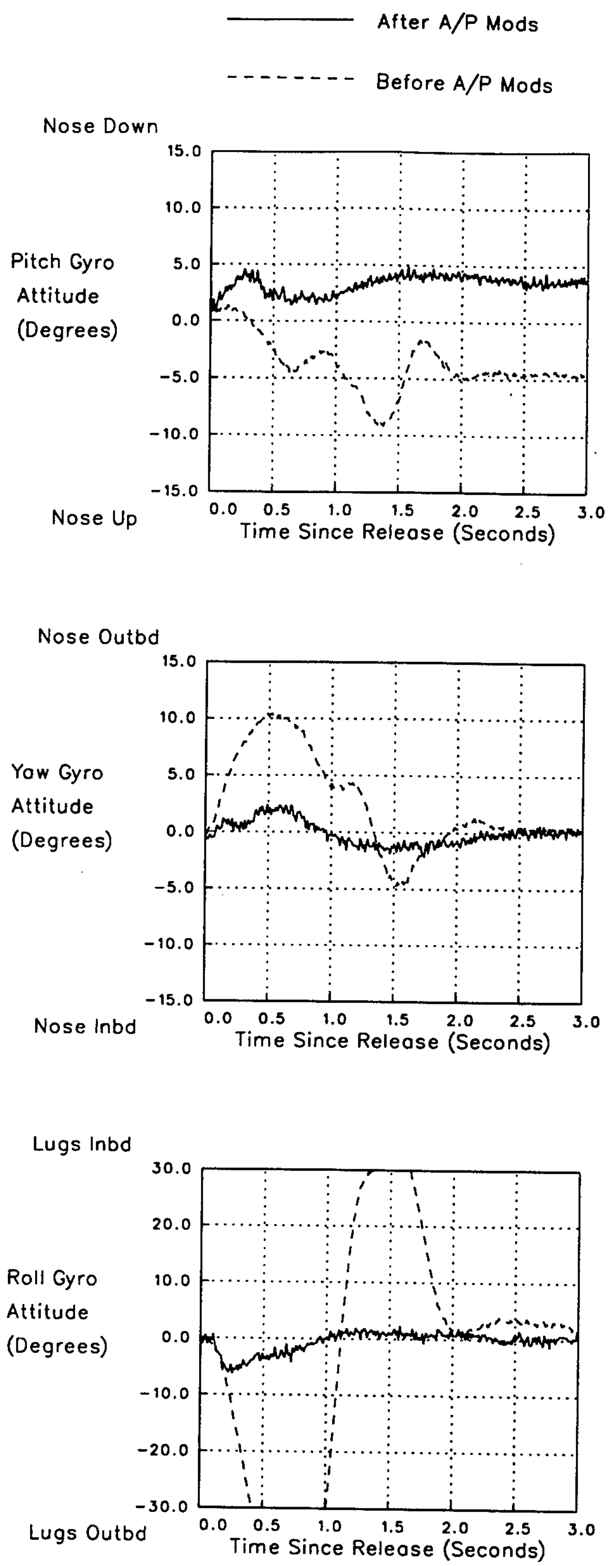

Figure 7 Flight Test Gyro Attitudes -
0.85 Mach/500 KCAS

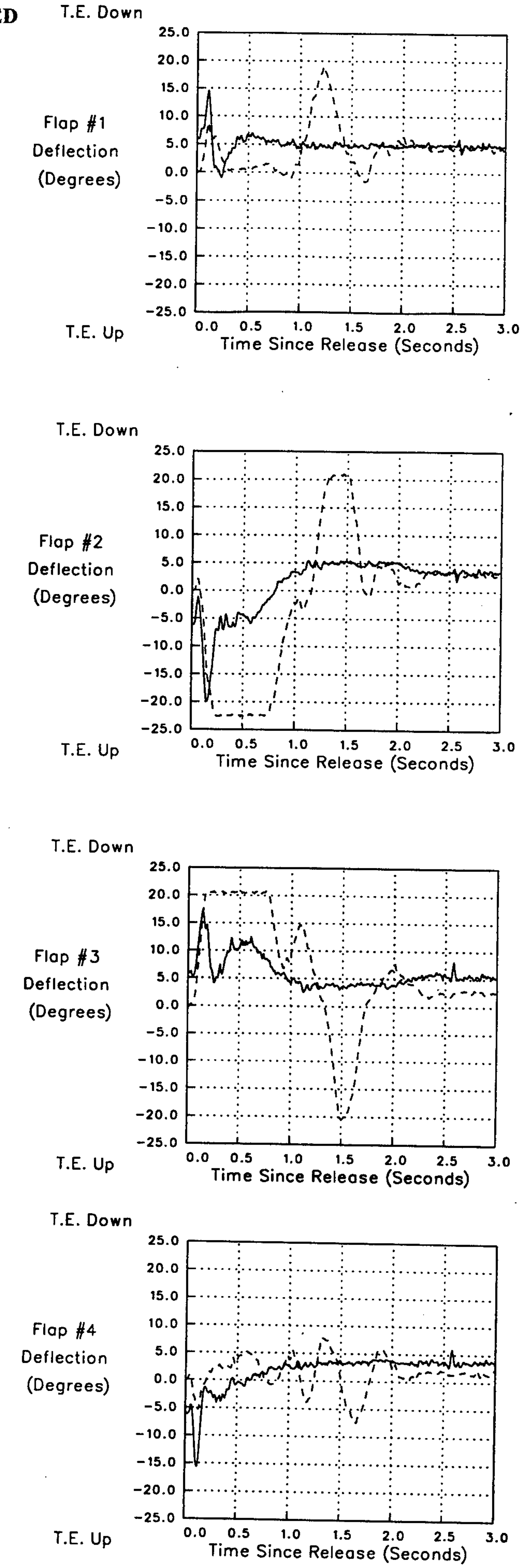

Figure 8 Flight Test Flap Deflections -
0.85 Mach/500 KCAS

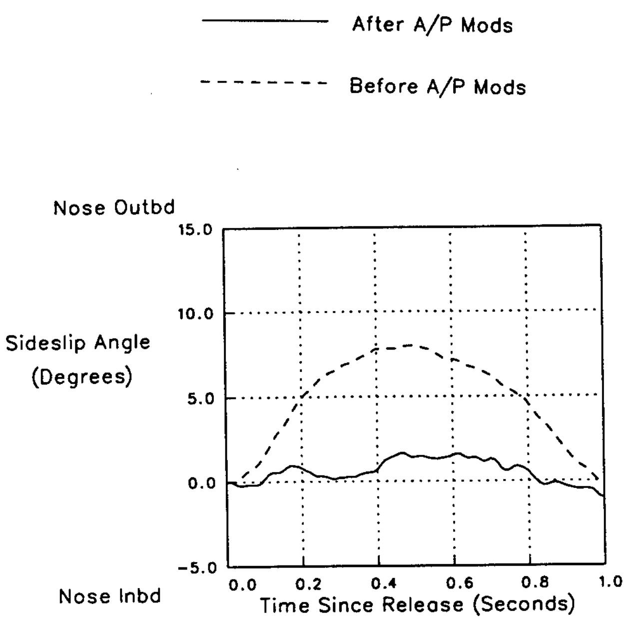

Figure 9 Derived Sideslip Angles - 0.85 Mach/500 KCAS

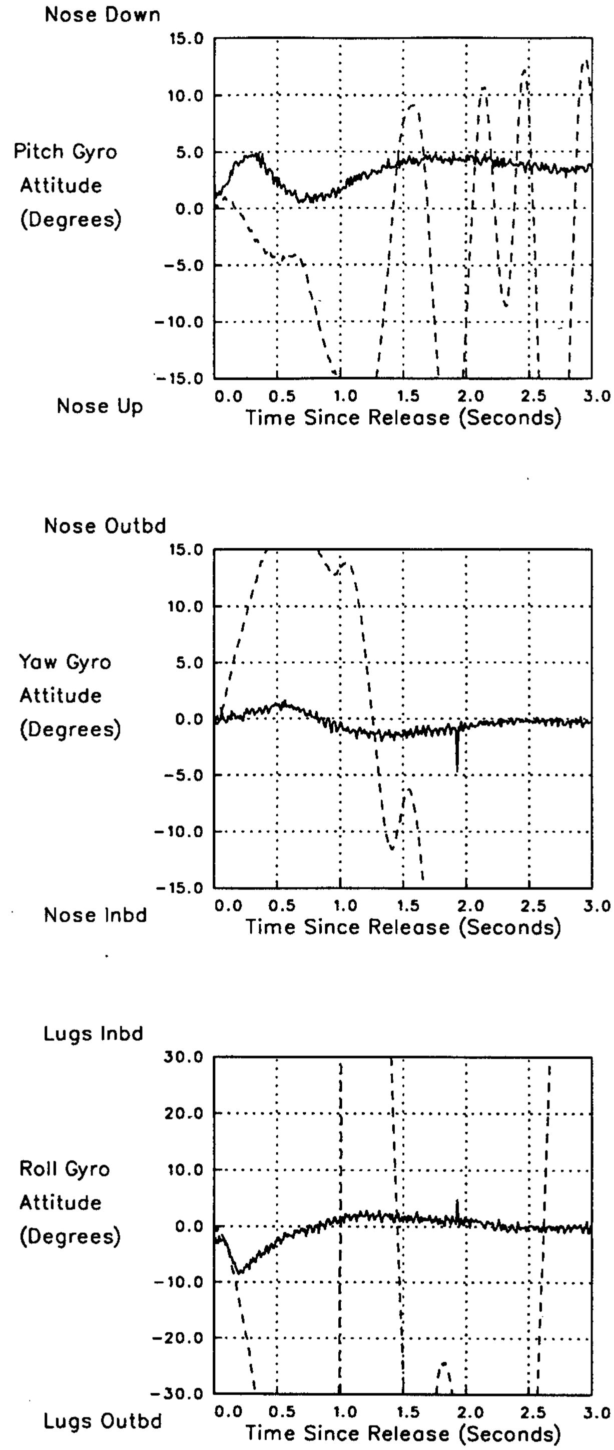

Figure 10 Flight Test Gyro Attitudes - 0.85 Mach/550 KCAS

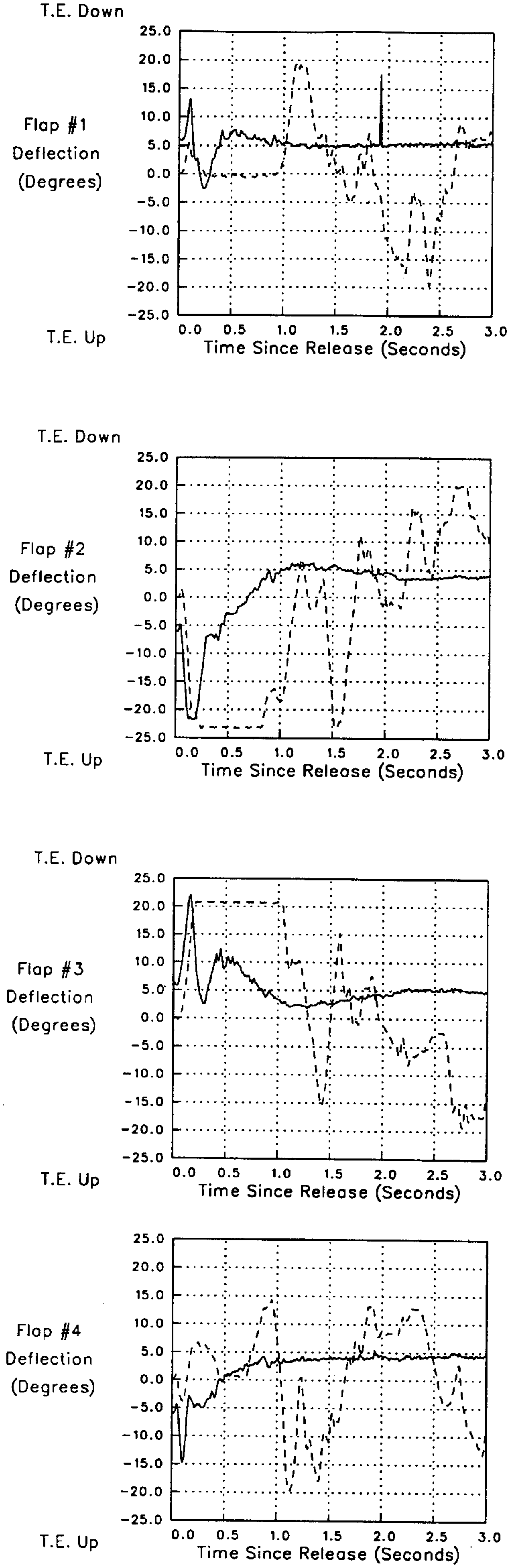

Figure 11 Flight Test Flap Deflections - 0.85 Mach/550 KCAS

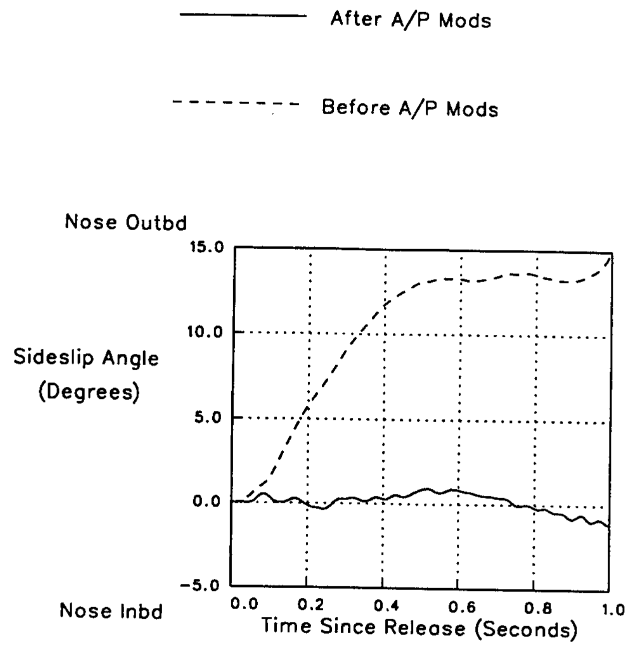

Figure 12 Derived Sideslip Angles - 0.85 Mach/550 KCAS

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