The Effect of Lateral/Directional Departure Metrics on the Design of the F-14A

Timothy T. Takahashi[1]
William P. Lorenzo[2]
Arizona State University, Tempe, Arizona, 85281, United States of America

In aircraft design, the underlying flying qualities of the airframe will ultimately determine program success or failure. This research is a technical and historical overview of flying qualities issues experienced with the F-14A Tomcat U.S. Navy fighter and serves as a case study to showcase the evolution of lateral/directional departure design metrics post-WWII. This work concludes that the death of the fictional "Goose," from the 1986 hit movie *Top Gun*, was the result of a flawed flight control system. While the F-14 was largely compliant with the 1960s-era MIL-STD 8785B for flying qualities, that standard did not provide an adequate framework to design high-angle-of-attack capable high-maneuverability aircraft.

I. Nomenclature

α	=	angle-of-attack (deg)
β	=	sideslip-angle (deg)
ϕ	=	roll-angle (deg)
Λ_{LE}	=	Leading-Edge Sweep (deg)
ω_{SP}	=	Short-Period Frequency (rad/sec or Hz)
ζ_{SP}	=	Short-Period Damping Ratio
ω_{DR}	=	Dutch-Roll Frequency (rad/sec or Hz)
ζ_{DR}	=	Dutch-Roll Damping Ratio
AR	=	reference aspect ratio
b	=	reference wingspan
c	=	reference chord
$SREF$	=	reference wing area
Vs	=	stall speed (knots equivalent airspeed)
CL	=	lift coefficient
CD	=	drag coefficient
CY	=	side force coefficient
Cm	=	pitching moment coefficient
Cn	=	yawing moment coefficient
Cl	=	rolling moment coefficient
Cmq	=	pitch damping dynamic derivative
Cnr	=	yaw damping dynamic derivative
Clp	=	roll damping dynamic derivative
Clr	=	roll-due-to-yaw dynamic derivative
Cnp	=	yaw-due-to-roll dynamic derivative
Ixx	=	rolling mass moment of inertia
Iyy	=	pitching mass moment of inertia
Izz	=	yawing mass moment of inertia
Ixz	=	crossproduct of inertia

II. Introduction

TOP GUN was a popular 1986 action film depicting the trials and tribulations of US Navy combat pilots during their training at the Navy Fighter Weapons School.[1] The movie tells the story of the fictional US Navy Lt. Pete "Maverick" Mitchell, as played by the young Tom Cruise. He, along with his Radar Intercept Officer, Lt. JG Nick "Goose" Bradshaw, as played by Anthony Edwards, begin the film as combat Grumman F-14 pilots aboard the USS Enterprise. Due to fortuitous circumstances, the two protagonists are assigned to attend TOPGUN, the Naval Fighter Weapons School at Miramar Naval Air Station (NAS). The drama peaks when "Maverick" inadvertently flies his F-14 through "Iceman's" jet wash and finds himself in an unrecoverable flat spin with both engines stalled. "Maverick" and "Goose" both eject; but "Goose" does not survive the ejection. While the board-of-inquiry cleared "Maverick" of

[1] Professor of Practice (retired), Aerospace Engineering, P.O. Box 876106, Tempe, AZ. Associate Fellow AIAA.
[2] M.S. Graduate, Aerospace Engineering, P.O. Box 876106, Tempe, AZ. Student Member AIAA.

any wrongdoing, the death of his buddy irrevocably changes him. After a single combat mission, “Maverick” changes his career direction becoming a flight instructor and test pilot; all of which set the basis for the 2022 sequel film.

The script to Top Gun was based on a May 1983 *California Magazine* article by Ehud Yonay.[2] This article memorialized the adventures of real F-14 crews who trained at the Fighter Weapons School. Yonay called the F-14, “the U.S. Navy's supreme air war machine - a huge, luxurious monster that could have been designed by the Star Wars special effects crew.” Its variable sweep wings were key to its stunning performance, not only in terms of its ability to fly at low speeds as well as “at more than twice the speed of sound, [while] haul[ing] seven tons of guns and missiles - including the heatseeking Sidewinder, the radar-guided, mid-range Sparrow, and as many as six Phoenix missiles.”

FIGURE 1 – F-14 at Miramar NAS – from the magazine article that inspired the Top Gun film.[2]

Yonay highlighted that the F-14 could be as dangerous to its pilots as to its targets. “In 1976 a deadly streak of accidents swept Miramar, killing four men in one 48-hour period alone.”[2] Fellow Top Gun pilot “Heater” opined that “You think there must have been something [the pilot] could have done and didn't - that if it were you it wouldn't have happened.”[2] However, Yonay highlights the fact that that some of these deaths can’t be chalked up to pilot error: “even accident reports that clearly demonstrate technical failures don't erase the lingering doubt.”[2]

According to B.M. Eastwood, writing in the *National Interest*:

> “After watching both the Tom Cruise movies, you probably thought the F-14 Tomcat had no problem intercepting enemy airplanes and winning supremacy in the skies. … You wouldn’t guess the Tomcat had an issue that almost sank the U.S. Navy’s prized fighter. Its early engines did not work well and even led to the loss of some 40 airplanes. The TF-30 turbofan engine had an inauspicious start. Pratt & Whitney planned that it would be installed on [a civilian transport. … With an] afterburner ... it ended up on the F-111B. …. The F-14 had nine feet of space between each engine. So, if one engine had a stall, it created a flat spin that could be uncontrollable.”[3]

Donald D.D. Smith, Naval Aviator, and test pilot recalls:

> ‘The F-14 hadn’t been in the fleet a year when some handling problems at high AOA (low speed) started to bust out. Good low speed maneuvering is right up there with speed and acceleration in the list of “must haves” for a good fighter plane. It turns out that F-14 jocks were having to cross-control to get the roll performance they

needed during low speed hassles. Being fighter pilots, it didn't bother them much, but it set off alarms all throughout the test community.

> …
>
> The Tomcat, after all, was America's world-class fighter, and was the greatest carrier-based fighter ever built! NAVAIR told Grumman to fix it – and tasked the Naval Air Test Center to test it.
>
> …
>
> We learned from Grumman model tests that the F-14 had a vicious spin mode that was NOT SURVIVABLE BY THE CREW – but not to worry, it was not possible for a pilot to fly the airplane into a spin – they said. The Navy had gone along with this – there was never a spin test program on the F-14."[4]

Co-author Takahashi can corroborate this. One of his early career mentors, William J. "Bill" Evans, who was an Aerodynamics Engineering Manager at Grumman during the F-14 era, told him that the Navy came back to Grumman with concerns about the high-angle-of-attack handling qualities of the aircraft.[5] He recalled that in operational test, pilots were flying the F-14 at angles-of-attack far in excess of the aerodynamic database used for design. This prompted Grumman to conduct additional wind tunnel testing of the completed design.[6]

In AGARD CP-199 Grumman noted that from 1965-1972 (prior to the introduction of the F-14), the USN and USMC suffered 169 hull-losses and 96 fatalities due to spins or departures.[7] In 1989, NASA's W. H. Philips lamented that in "comparison with other disciplines required in airplane design, the subject of flying qualities is in a less satisfactory state."[8]

Donald D.D. Smith, Naval Aviator and test pilot reminisced about his own experience with the Tomcat provoking lateral-directional departures during flight test:

> "Unlike the A-7, the F-14 [was supposed to exhibit] a docile post-departure mode and [recover] easily after half a turn. All this changed on Flight 12 … The show didn't stop – it was just beginning. During departure recovery with the nose well down, the airplane started to develop a yaw rate and within a couple of turns the nose came up to a flat attitude. Within seconds the rotation rate shot up to 180-deg/sec and g forces reached 7.8-g eyeballs-out! My head was slammed against the instrument panel and my mask and eyes filled with blood! Twenty seconds had gone by, and I lost consciousness.
>
> …
>
> There is much more to this story – obviously since I am here to tell it."[4]

Thus, we see that "Maverick's" unrecoverable flat-spin of the F-14 was not just Hollywood drama. It seems that under foreseeable conditions, the F-14 could exceed its maximum lateral-directional stability or control limits with disastrous consequences.

This provides the motivation for this technical paper: were existing military design standards used to design the F-14 adequate or inadequate? If they proved inadequate, what lessons were learned by its operational service history?

III. The Genesis of the F-14A

The F-14 arose from a U.S. Navy requirement for a carrier-based, long-range, high-endurance interceptor; basically, a Fleet Air Defense (FAD) aircraft with considerably more capability than the McDonnell F-4 Phantom II.[9] USN leadership believed that their fighter inventory which relied on the Vought F-8 Crusader and the F-4 Phantom would be obsolescent by the late 1960's because they were 15+ year old designs and lacked the performance of a number of then new Soviet aircraft.[10]

In the early 1960's, the concept of a "Joint-Service" aircraft found great favor under U.S. Secretary of Defense Robert McNamara. [11] One byproduct of this directive was the USAF procuring the McDonnel F-4 Phantom II, originally designed for the Navy. [11][12] Another byproduct was the TFX program, which ultimately led to both the USAF F-111A and the USN F-14. The TFX was to have been a joint-services aircraft with high commonality between USAF and USN airframes.[13][14] In November 1962, the DoD selected the General Dynamics proposal configuration for further development. General Dynamics then turned to team with Grumman for the F-111B Navy variant. Grumman handled Navy specific design aspects and manufactured the aft section for all F-111 customers throughout its production life.[15]

Carl Droste, now retired from General Dynamics, opined that "both the Air Force and Navy knew the F-111 would deal with a wide range of flight conditions and wing configurations" from Mach 1.2 at Sea-Level through Mach 2.5 at Altitude.[16] He continues:

> "This program was steeped in controversy. The Air Force versus Navy. The Air Force and Navy versus Robert McNamara. Boeing [the proposal loser] versus General Dynamics [the proposal winner]. Grumman versus General Dynamics … Congressmen [were] actively supporting all the opposing sides. You never heard the word F-111 mentioned without words like "controversial," "star-crossed," or "troubled."" [16]

The USAF F-111A configuration first flew on December 21, 1964; two years after contract award.[13] The F-111B USN variant first flew on May 18, 1965; two and a half years after contract award.[13] F-111 flight tests revealed many problems with the engine integration, which lead to several revisions to the inlet system.[13] The USAF F-111A and its follow-on derivatives, achieved initial operational capability on April 28, 1968.[13] These aircraft remained in front line service with the USAF through final retirement in 1996.[17]

Shortly after McNamara resigned from his post as Secretary of Defense, the USN formally withdrew from the F-111 program. In its place came the F-14; the USN conceived it as a low-risk development program which re-used technology and much of the F-111's developmental wind tunnel test data. The F-14 shared its Pratt & Whitney TF-30 afterburning turbofan engines with the F-111, albeit with a 2-D variable geometry inlet rather than a quarter-conical external compression inlet.[20][13] Contemporary reports claimed that "Grumman's design work on the F-14A [dated] back to 1966 when the company commenced a series of Navy R & D studies of advanced inlet and ejector designs … for the F-111B."[10]

McIver at the Naval Postgraduate School summarized Grumman's position:

> "Grumman's investigations found no way to improve the F-111B for the Navy. What they presented was an unsolicited proposal for a completely new aircraft. The new aircraft would keep the swept wing concept, missiles, fire control system and engines currently installed in the F-111 and put them in an airframe that was smaller and lighter." [18]

W.H. Mason, Professor Emeritus at Virgina Tech [19] and a former Aerodynamicist at Grumman wrote:

> "The F-14 came about when the Navy decided that the F-111B would never be acceptable. In part it was too big and heavy to be used on a carrier, but there were other problems. The subsequent competition was known as the VFX, and Grumman won. Although the Navy picked Grumman, the DOD always thought that Grumman had deliberately messed up the F-111B (remember they had the subcontract from General Dynamics) so they could come up with their own airplane.
> …
> In Bethpage, Mike Pelehach was the program manager of the VFX study [20]. He had several designers working on the design. He kept telling them that "this isn't it; I'll know it when I see it." One day he was down at the design room where another team was working on the proposal for what would become the F-15 (I don't recall the letters that described that program before it was awarded to McDonnell). That was where Mike saw "it" and came back to the VFX project and got Nathan [Kirschbaum] and took him over to see it and told him to "draw it up." [21] I can't remember the name of the designer, but that was the guy that everybody inside Grumman gave credit for the design.
>
> So that was the genesis of the very early design, the way I heard the story.
>
> Not many papers were written [on the F-14 aerodynamic design]. Rudy [Meyer] wasn't proud of the wing design. It had too much washout, and thus apparently separated flow and had high drag at low lift.[22]
>
> Rudy Meyer said the computational methods weren't [mature enough in 1968] to do good wing design. … They did have the Harris [Wave Drag] code. Ron Hendrickson did the area rule work … under the supervision of Bill Evans." [5][23]

The F-14 program progressed rapidly from its "go-ahead" in 1968.[9] First flight occurred on December 21, 1970.[9] The F-14 entered operational status with a 1974 deployment aboard the USS Enterprise (CVN-65).[9] It successfully replaced bother the F-4 Phantom II and the F-8 Crusader in USN inventory.[9] The F-14 served as the U.S. Navy's primary maritime air superiority fighter, fleet defense interceptor, and tactical aerial reconnaissance platform through 2006.[24]

IV. Relevant Military Flying Qualities Guidelines

The two most pertinent Military Standards controlling the aerodynamic design of the F-14 were MIL-C-5011A for Performance [25] and MIL 8785B for Flying Qualities.[26]

MIL-C-5011A, in effect from 1951, specified that critical combat performance (speed, altitude, etc.) be evaluated at a flight weight with "full internal fuel capacity" of an aircraft like the F-14 which would ordinarily be flown with drop tanks.[25] Fuel consumption data for mission planning will have a 5% added pessimism as a "service tolerance to allow for practicable operation."[25] Performance shall consider a minimum-wind-over-deck requirement so that a catapult launch at maximum take-off-weight will achieve "minimum safe take-off speeds."[25]

MIL-F-8785B, in effect from 1969, governed stability & control issues.[27] It represented a major revision to MIL-F-8785.[27] MIL 8785B provides the aircraft designer with a precise "framework … which permits tailoring each requirement according to 1) the kind of airplane (class), 2) the job being done with the airplane (flight phase categories), and 3) how well the job must be done under various circumstances (levels)."[28][29] The F-14 is thus classified as a "CLASS IV" – high-maneuverability airplane.[26] MIL 8785B standards aircraft flight dynamics were developed in conjunction with the Cooper-Harper subjective pilot rating system.[26][28][30][31]

MIL 8785B introduced the concept of "levels of handling qualities" suitable for various piloted maneuvers.[29] Aircraft are expected to comply with the LEVEL 1 standard with all systems functioning normally as well as with inoperative systems with probable failure states; i.e., engine inoperative within their "operational flight envelope."[29][32] The idea is that military aircraft should have "good flying qualities where the airplane is expected to be used … [and] acceptable flying qualities in … infrequently expected conditions."[32] LEVEL 1 handling qualities in the MIL STD corresponds to a Cooper-Harper rating of 3 or better; i.e., at least "fair" and "satisfactory" subjective flying qualities on a scale from excellent (1) to unflyable (10).[30][31][33] Aircraft are permitted to display LEVEL 2 handling qualities within their theoretical flight envelope; these correspond to a Cooper-Harper rating of 4 to 6, i.e. "unsatisfactory" but not imminently "dangerous" behavior.[32][30][31][33] LEVEL 3 qualities are deemed "poor" or "dangerous;" these are completely unacceptable attributes for military aircraft to possess.[30][31][33]

MIL 8785B implicitly sets lateral-directional control power requirements through a cross-wind and a minimum control airspeed criterion.[17] A CLASS IV high-maneuverability aircraft like the F-14 must be able to take off and land in a 15-knot crosswind (the minimum for LEVEL 3 handling qualities); and ideally demonstrate capabilities in a 30 knot crosswind (for LEVEL 1 or 2 ratings). At 135 knots approach speed flown by an F-14, a 30 knot crosswind represents trim to β~13° which is consistent with the requirement that the re be sufficient rudder and aileron control power to trim out "at least 10 degrees of sideslip."[28] Similarly, during takeoff roll above the "refusal speed," sufficient rudder control power must exist to trim out the asymmetric moments arising from an inoperative engine. Once airborne, it shall be possible to achieve straight (i.e., trimmed) flight with a bank angle not to exceed 5° with 75% of available control power.

MIL 8785B also sets a minimum permissible level of static directional stability: "above 1.4 *Vmin*, with asymmetric loss of thrust … the airplane, with rudder pedals free, may be balanced directionally [through sideslipping and banking] in steady straight flight." [28]

MIL 8785B gives guidelines for desirable specifying longitudinal short-period dynamics which impact how the airplane responds to pilot inputs.[28] The standard specified frequencies and damping ratios, but admits that these two parameters are "not sufficient to describe the acceptability of airplane longitudinal dynamics"[33] The sensitivity to elevator inputs, and the pitch-rate response n_Z/α of the aircraft is also important; see FIGURE 2.[33]

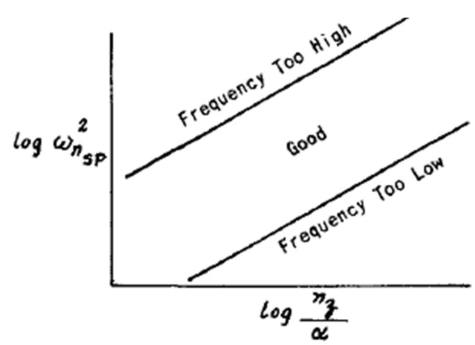


FIGURE 2 - Defining Minimum and Maximum Short Period Frequencies for LEVEL 1 controllability.[28]

This understanding is known as the "Control Anticipation Parameter," CAP. It was devised by W. Bihrle, then a lead engineer at Republic Aircraft on the F-105 "Thunderchief, " after considerable simulation work with line pilots; see FIGURE 3.[28][34][35] CAP represents the "ratio of the instantaneous angular pitching acceleration generated per unit of steady state load factor."[32] "Below the lower limit, the desired … response … cannot be anticipated by the pilot since … [the] angular pitching acceleration [cue] is not present."[32] Whereas "the upper limit may be indicative of a horrendous anticipatory signal for a relatively small change in flight path."[32] Neither basic piloting bandwidth nor rigid-body / structural-dynamics interactions, which also govern acceptable short-period rigid-body dynamics were considered in this original formulation. Ashkenas advocated a need to have a minimal phase lag at 0.5-rad/sec for good flight path regulation.[35] When the rigid-body frequency is too low, pilots often resort to "pumping the controls;" that is intentionally commanding the aircraft knowing that the aircraft will not respond to the input with any reasonable phase lag. Pilots typically employed control pumping during landing approach of both F-105 and F-84 at aft CG locations.[32]

Following Roskam [36], Yechout [37] and Takahashi [38], we may approximate the inherent short-period, rigid-body oscillatory mode of the aircraft (frequency in rad/sec) at a given speed (*M*) and angle of attack (α) as:

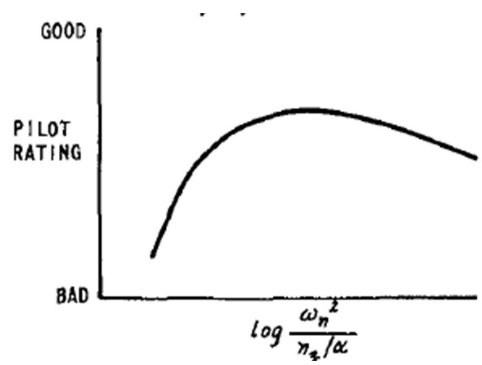


FIGURE 3 - Correlation between Control Anticipation Parameter ($\omega_n^2/(n_Z/\alpha)$) and Pilot Rating.[28]

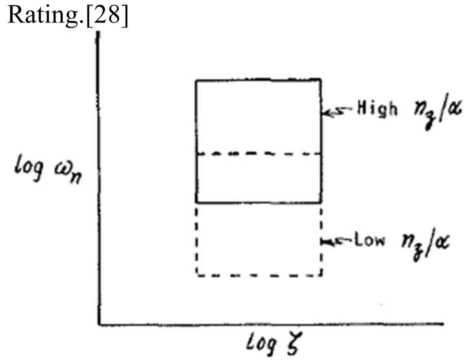


FIGURE 4 - Short-Period Frequency (ω) and Damping (ζ) Ratio Requirements.[28]

$$\omega_{sp} \approx \sqrt{\frac{-57.3\frac{\mathrm{dCm}}{\mathrm{d}\alpha}\bar{q}\, S_{ref}\, \bar{c}}{Iyy}}. \tag{1}$$

Chalk & Wilson found that the available variable stability airplane flight-test data revealed that pilots desire the fastest minimum acceptable short-period frequency for combat, aerial-refueling and/or ground attack missions (flight phase A).[28] Pilots could tolerate a slower Short Period frequency for takeoff and landing (flight phase C) and even slower Short Period frequencies for basic climb / cruise / loiter activities (flight phase B). However, in all cases they found that mission effectiveness requires the short-period frequency to scale with the pitch rate responsiveness:

$$\frac{n_Z}{\alpha} \approx \frac{57.3\frac{dCL}{d\alpha}\bar{q}\, S_{ref}}{W}. \tag{2}$$

MIL 8785B also considers the need for an appropriate level of damping commensurate with the rigid body frequency; see FIGURE 4.[28] If pitch damping is too low, the airplane response to pilot commands overshoots and oscillates; the aircraft also becomes difficult to control in turbulence. Conversely, if the pitch damping is far too strong, the aircraft becomes sluggish to control inputs. Thus, the standard provides both upper and lower bounds to the preferred damping ratio applicable across a wide range of rigid body frequencies and pitch responsiveness. Bihrle noted that the desirable damping ratio range is ζ = .4 to .7, which corresponds to a desire for aircraft to have nearly "deadbeat" non-oscillatory responses to inputs.[32]

We may estimate the short-period damping ratio as:

$$\zeta_{sp} \approx -\frac{M_q + Z_\alpha/U_1}{2\,\omega_{sp}} \tag{3}$$

where the pitch angular acceleration due to pitch rate is:

$$M_q = C_{m_q} \frac{\bar{q}\, S_{ref} \bar{c}^2}{2\, Iyy\, VKTAS \left(\frac{6076}{3600}\right)} \tag{4}$$

and the normalized vertical acceleration due to the angle of attack is:

$$\frac{Z_\alpha}{U_1} \approx -57.3 \frac{\mathrm{dCL}}{\mathrm{d}\alpha} \frac{\bar{q}\, S_{ref}}{\left(\frac{W}{g}\right) VKTAS \left(\frac{6076}{3600}\right)}.[39] \tag{5}$$

and $VKTAS$ is the aircraft velocity in knots of true airspeed.

MIL 8785B contained similar guidelines for the preferred Dutch Roll frequency and damping. The Dutch Roll mode is an oscillatory mode coupling roll and yaw motions [36][37][38]. Its frequency, in rad/sec, may be estimated by:

$$\omega_{dr} \approx \sqrt{\frac{57.3\, C_{n_{\beta dynamic}}\, \bar{q}\, S_{ref}\, b}{Izz}} \quad .[39][40] \tag{6}$$

Where:

$$C_{n_{\beta dynamic}} = \left(C_{n_\beta}\right)_{BODY} cos(\alpha) - \left(C_{l_\beta}\right)_{BODY} sin(\alpha) \left(\frac{Izz}{Ixx}\right).[39][40] \tag{7}$$

Note that the values of directional stability $(dCn/d\beta)$ and dihedral effect $(dCl/d\beta)$ feeding this equation must be given in terms of body axis. So long as $C_{n_{\beta dynamic}}$ is positive, the aircraft will oscillate; if $C_{n_{\beta dynamic}}$ goes negative, the aircraft will depart.

At low angles of attack, $C_{n_{\beta dynamic}}$ is dominated by the static weathercock stability $(dCn/d\beta)$. As the angle of attack increases, the dihedral effect plays an additional stabilizing role ($dCl/d\beta\ <\ 0$ and $Izz/Ixx\ >>\ 1$ for slender, swept configurations) [39].

For an oscillatory system, the Dutch Roll damping ratio is approximated by:

$$\zeta_{dr} \approx -\frac{\left(N_r + \frac{Y_\beta}{U_1}\right)}{2\, \omega_{dr}} \tag{8}$$

where the normalized lateral acceleration due to sideslip is:

$$\frac{Y_\beta}{U_1} = 57.3\, C_{Y_\beta} \frac{\bar{q}\, S_{ref}}{\left(\frac{W}{g}\right) VKTAS \left(\frac{6076}{3600}\right)} \tag{9}$$

and the yaw angular acceleration due to yaw rate is:

$$N_r = C_{n_r} \frac{\bar{q}\, S_{ref}\, b^2}{2\, Izz\, VKTAS \left(\frac{6076}{3600}\right)}. \quad [37][39] \tag{10}$$

MIL 8785B requires a stable Dutch Roll mode with stick fixed for the basic, unaugmented airframe.[26] For LEVEL 1 flying qualities in combat aircraft situations, the F-14 should have a minimum Dutch Roll frequency $\omega_{DR} > 1.0$-rad/sec (0.16-Hz); a minimum damping ratio $\zeta_{DR} > 0.19$ and $\omega_{DR}\, \zeta_{DR} > 0.35$ with relaxed requirements of $\zeta_{DR} > 0.08$ and $\omega_{DR}\, \zeta_{DR} > 0.15$ in cruise, takeoff and landing.[26] The standard specifically warns the designer to ensure that sideslip deviations during combat flying should not exceed +/-3 mils; i.e., $-0.17^{o}<\beta<+0.17^{o}$.[26]

Slender, body heavy aircraft, like the F-14 may express their Dutch Roll behavior in terms of a wing rock (changes in ϕ) rather than a tail wag (oscillations in β), this behavior is determined by the ϕ/β ratio.[37] If $\phi/\beta >> 1$, wing rock predominates; if $\phi/\beta << 1$, the aircraft will wag its tail in yaw. While traditional transport aircraft feature active rudder control, the "yaw damper," to improve Dutch Roll damping, aircraft where $\phi/\beta >> 1$ may also require a "roll damper" to have satisfactory Dutch Roll characteristics.

The need to address adverse-yaw was a hard lesson learned through a number of close-calls, pilot deaths and hull-losses during the X-1A, X-2 and F-100 flight test programs.[40][41][42] By the early 1960's, high-speed aircraft designers began to pay close attention to previously unforeseen lateral/directional stability and control issues. MIL 8785B requires the designer to attend to unintentional causes of sideslip; "if roll or roll control excites sideslip, the flying qualities can be degraded by such motions as an oscillation of the nose on the horizon during a turn, by a lag or initial reversal in yaw rate during a turn entry, or by making it difficult for a pilot to quickly and precisely take up a given heading."[28] When roll control excites sideslip, either adverse (positive β due to positive roll rate) or proverse (negative β due to positive roll rate), "the pilot cannot damp Dutch Roll oscillations through the use of aileron control only."[28] When "the coupling of sideslip with roll and roll rate becomes important, … oscillations in roll rate and ratcheting of bank angle or lateral-directional pilot-induced oscillations" are probable undesirable outcomes.[28]

MIL 8785B stipulates that aircraft "response to roll commands [should] not be oscillatory."[26] Weil & Day realized that inertial coupling occurs when the short-period and Dutch-Roll frequencies grow too close to one another **or** when the pilot commands the aircraft to roll at a rate which excites these frequencies, especially on aircraft with vulnerable mass properties when $\left|\frac{Izz-Iyy}{Izz}\right|$ or $\left|\frac{Izz-Ixx}{Iyy}\right|$ >>0.5).[43] The critical roll rate, p_{crit}, can be found by searching for the lowest magnitude rate as predicted by the following equation:

$$p_{crit} \approx \pm \min\left(\sqrt{\frac{57.3\,\frac{dCn}{d\beta} q\; S\; b}{Iyy-I}}\,,\sqrt{\frac{-5\ \ .3\,\frac{dCm}{d\alpha} q\; S\; \bar{c}}{Izz-Ixx}}\right). \tag{11}$$

If the aircraft rolls at this rate, energy from the commanded roll maneuver will excite yawing and/or pitching motions.[43]

Because it is essential that aircraft do not spin due to lateral stick inputs, MIL 8785B specifies a maximum tolerable adverse yaw level from lateral-stick inputs with rudders fixed.[26] Flight tests revealed that there "is more to coordination … than whether the sideslip is adverse or proverse; the … phasing of the disturbing yawing moment also [affects] the coordination problem."[32] Pilots find it much easier to coordinate adverse yaw with application of rudder than proverse yaw.[32] Following a rudder-fixed maximum roll input, a LEVEL 1 combat aircraft shall not exceed more than 6° of adverse sideslip (right roll leading to right sideslip) or more than 2° of proverse sideslip (right roll leading to left sideslip); these standards are relaxed somewhat in cruise, takeoff and landing but cannot exceed 15° adverse sideslip or 4° of proverse sideslip.[26]

V. Development of the Production F-14A Configuration

A. Prequel: Lessons Learned from NASA Generic Variable Geometry Fighter and the F-111

Cornell, writing in 1970, recites a claim that "Grumman's design work on the F-14A actually dates back to 1966 when the company commenced a series of … product improvement tests for the F-111B."[10] He maintains that the F-14 was the end product of testing "some 900 aerodynamic variations and 540 propulsion inlet and ejector configurations ….[accumulating] an estimated 27,000 hours of wind tunnel tests involving 29 basic model types for aerodynamics, propulsion, structural loads, spins, store separation, etc."[10] These numbers seem fanciful; consider that in any given year a wind tunnel test facility is unlikely to be able to accommodate more than 4,000 hours of testing (two shifts a day of customer occupancy). During the late 1960's, the F-14 program would have competed for tunnel access with the F-111A, F-15, Boeing 2707-SST, B-737, B-747, L-1011, and DC-10 aircraft as well as the Saturn launch vehicle. With the aerodynamic configuration largely frozen by contract award in January 1969, it is likely that the claimed wind tunnel time involved studies of more generic variable sweep configurations; those predating the TFX program.[44][45][46][47][48][49][50][51][52]

While Grumman had relatively little time for extensive experimental validation of multiple F-14 concepts, they were able to draw upon a large collection of NASA wind tunnel tests of variable sweep combat aircraft. Since 1959, NASA had worked on conceptual variable sweep configurations that would lead to a single "airplane combining the characteristics of low-speed efficiency and supersonic "dash" or supersonic cruise ability."[44] They specifically considered an aircraft which would "operate from an aircraft carrier … loiter for long lengths of time and … accelerat[e] to supersonic speeds for the purpose of acquiring a target at some distance from the task force."[44]

The 1959 NASA Study Configuration featured a single vertical tail; see FIGURE 5.[44][46] The tested configuration had a trimmed supersonic (Mach 2.01) *L/D* ratio of 5.5:1 at *CL*~0.2 and α=5° with the CG positioned for 20% static margin.[44] Supersonic directional stability was weakly positive ($dCn/d\beta$ ~ +0.0015) at α=5° but declined sharply to become unstable when α>11°.[44] Differential taileron for roll showed strong roll-yaw coupling (*Cn/Cl* ~ 0.8) but with proverse unintended roll.[44] The wind tunnel data revealed that application of 5° differential taileron without rudder compensation (which would produce *Cl* ~ 0.005 of rolling moment) would drive the aircraft to 2.7° of unintended sideslip. Subsonic (Mach 0.25) testing of the same configuration, but with wings at Λ_{LE} =12.5°, found it neutrally stable in pitch.[46] They noted that having a highly swept inboard fillet largely eliminated the stall break, and allowed the aircraft pitch stability to remain consistently neutral all the way through α~20°.[46] Subsonic directional stability was strongly positive ($dCn/d\beta$> +0.0035) at α=5° but declined sharply to become unstable when α>18°. Roll control from ailerons had reasonable power with the wings unswept, but fell off dramatically if the wings were swept to Λ_{LE} =75°. Thus, the 1959 configuration represented a promising start, but one which would require considerably larger vertical tail volumes to be made controllable.

Following this, the work of Spencer and Polhamus clearly contributed to attributes of the final F-14 design.[47][52]

Polhamus' 1960 report convincingly argues the superiority of a midspan pivot design to reduce aerodynamic center shifts with wing sweep.[52] He too was concerned about the unintended yaw due to roll of various advanced "aileron" concepts; see FIGURE 6. He recommended both differential taileron and asymmetric wing spoilers for roll control, both of which were incorporated on the production F-14.[52]

Similarly, Spencer tested two configurations at low subsonic speeds and found one configuration (which anticipates the production F-14) with a pair of 2-D ramp

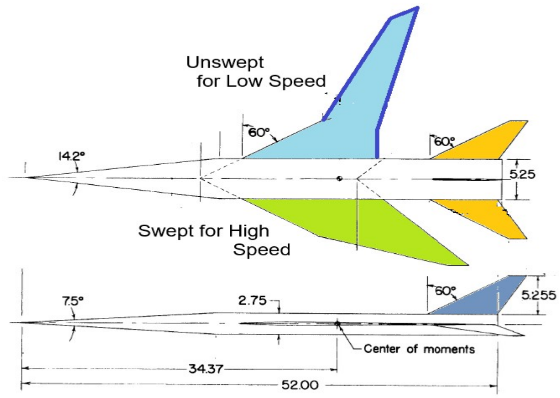
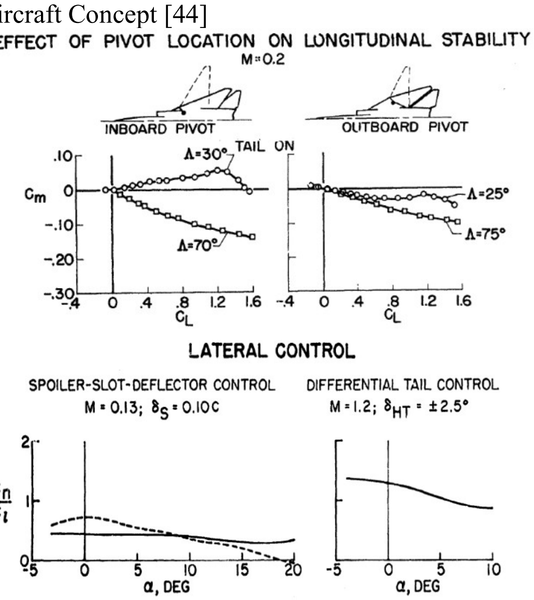


FIGURE 5 - 1959 NASA Variable Sweep Navy Aircraft Concept [44]



a

b

FIGURE 6 - Studies from Polhamus & Hammond's 1960 NASA publication. a) longitudinal stability, b) adverse yaw from candidate roll control surfaces.[52]

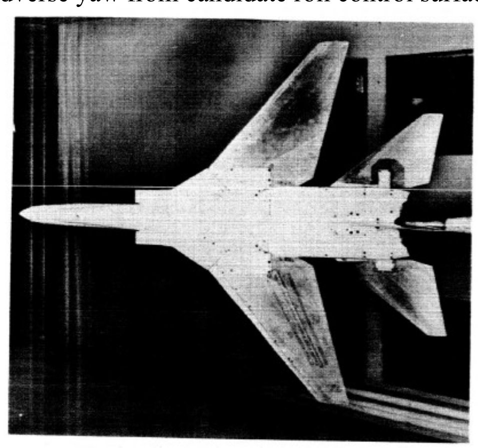


FIGURE 7 – Configuration from Spencer's NASA TM-X 303.[47] Essentially similar configurations tested in TM-X 304 and TM-X 343.[48] [49]

inlets and an outboard pivoting swing wing to have desirable longitudinal (neutral stability for $-2^o<\alpha<24^o$) and directional stability ($dCn/d\beta$~+0.002 for $\alpha<14^o$); see FIGURE 7.[47] Follow on transonic and supersonic tests of similar configurations may be found in Bielat [48] and Foster. [49] They also found satisfactory longitudinal stability, and weakly positive static directional stability ($dCn/d\beta$~+0.001 at Mach 2.01). We suspect that Grumman leveraged this baseline when laying out their initial aerodynamic configuration.

Other NASA tests focused on more specific aerodynamic details. Foster studied the effects of asymmetric spoilers for roll control at supersonic speeds.[50] While these control surfaces produced roll, that moment was accompanied by unintended proverse yaw.[50] In response, Henderson & Hammond looked at slat, flap, symmetric spoiler and asymmetric spoilers for both high-lift and roll control at subsonic speeds.[51] They found that roll commands from asymmetric spoiler deflections at subsonic speeds produced noticeable pitching moment artifacts as well as moderate proverse yaw. Roll control power declined rapidly when $\alpha>10^o$.

NASA continued to mature their concept configuration throughout 1961 with only minor geometric changes. Spearman [53][54] and Bielat [55] tested the next concept (which retained 2-D ramp style inlets and featured an AR~5 wing, unswept) at supersonic, transonic, and supersonic speeds. They found that an aircraft balanced for 4% positive static margin at low speeds with wings unswept would have ~23% positive static margin with wings swept $\Lambda_{LE=}75^o$ at Mach 2.2; see FIGURE 8.[53] This seemed promising, in that the aircraft would not need to have a speed dependent CG management system on board. Static directional stability seemed favorable; $dCn/d\beta$~+0.0035 at subsonic speeds and declined to ~ +0.0025 at Mach 2.2.[53] Supersonic aerodynamic efficiency peaked at *L/D* ~ 6 @ *CL* ~ 0.2 with the wings fully swept aft.[54] Testing noted trends for longitudinal stability as well as directional stability to decline with increasing angle of attack, although the configuration did not display a classic "stall break" with a peak in the *CL* vs *a* trend.[55]

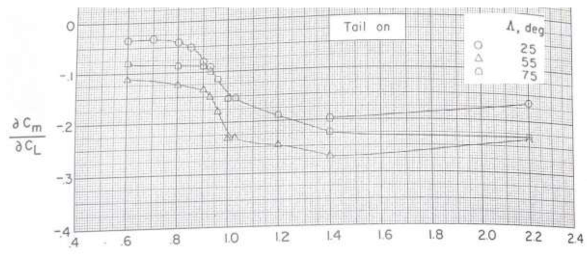


FIGURE 8 - Longitudinal Stability from Spearman's 1961 configuration.[53]

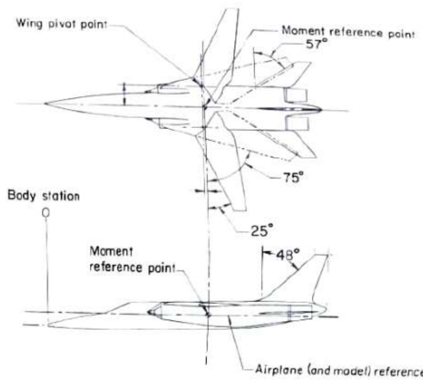


FIGURE 9 – Immature F-111 type configuration from NASA TM X-994.

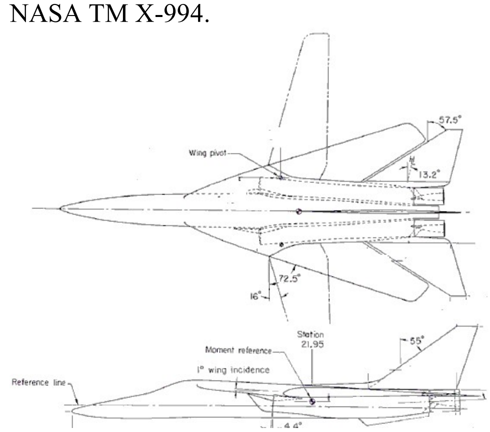


FIGURE 10 – Refining the final F-111 configuration from NASA TM X-1039.[58]

After contract award, the NASA reports concern aerodynamic testing of an aircraft with striking resemblance to the final production F-111.[56][57][58][59][60][61][62][64][l5][65]

The 1964 tests of a notional 1/20th scale airframe featured the basic aerodynamic configuration including quarter conical "spike" inlets; see FIGURE 9. The tested aircraft was somewhat smaller that the production aircraft (~60 vs 73.5-ft fuselage length and ~29 vs 32-ft wing span when swept) with simplified wing airfoils (constant NACA 0008-63 with no washout or dihedral).[56] Parametric studies included six forebody concepts and two wing variations (*AR*~7 and *AR~8* at low sweep).[56] All of the tested configurations offered a subsonic *L/Dmax* ~ 6.5 @ *CL*~0.3 declining to a *L/Dmax* ~ 4.5 @ *CL*~0.4 and *M*~2.[56] Low speed static margin was ~22% stable increasing to ~60% stable at Mach 1.2.

The next tunnel entry series continued with an immature F-111 type configuration with w large single vertical.[57] Low speed data explored longitudinal and lateral/directional stability through α~40^o.[57] Hassell found that the baseline vertical tail did not ensure positive directional stability for α>12^o. Similarly, the test data found 50% **proverse** yaw-due-to-roll from a taileron deflections. While preferred to adverse yaw, these characteristics may still result in unfavorable flying qualities.

Lessons learned from this test clearly influenced the next four tunnel entries.[58][60][61][62] In the next series, the geometry is clearly converging on the final F-111 configuration; see FIGURE 10. The overall vehicle size now closely represents the as-flown aircraft (~72 vs ~73.5-ft long fuselage, ~31.5 vs 32-ft wingspan when swept). This tunnel entry explored three different wings both with and without prominent leading edge droop and twist at Λ_{LE}=16^o, 20^o and 72.5^o. NASA also tested several different forebody configurations (which altered the transonic area distribution). Low speed testing through α~24^o noted no classical stall, but did note strong shifts in longitudinal stability when *CL*>~0.8. Aerodynamic efficiency with the wings unswept was good; *L/Dmax* ~ 12.5 @ *CL*~0.45 and *M*=0.77.

FIGURE 11 – F-111 configuration from Aviation Week [63]

Aviation Week, in its June 1, 1964, magazine, publicly unveiled the configurations for both the USAF F-111A and USN F-111B; see FIGURE 11.[63] In hindsight, this milestone indicated a point in the design process where the major configuration trades had to have concluded. Detail design was under way to produce flyable prototypes for the Air Force and Navy.

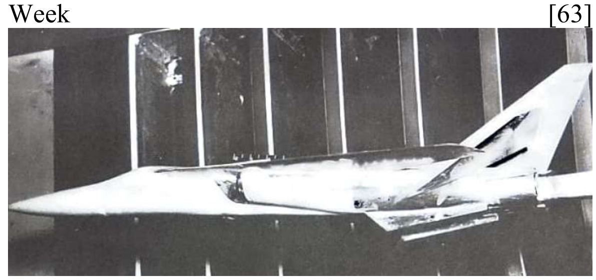


FIGURE 12 – Near Final F-111 Configuration from NASA TM X-1039 [58]

The team continued to worry about the strength of positive static directional stability. In the next series of tests, NASA began by testing three vertical tail sizes and four forebodies at Mach 2.20.[61] They found that all three vertical tails had similar directional stability at zero alpha, but the tallest and least swept vertical fin option had the best directional stability at α~13^o; desirable for the fighter design. Since a tall single vertical tail would have clearance issues when stored on an aircraft carrier, NASA then continued to test further alternative tail configurations.[62] In the next series, NASA tested three vertical tail concepts. They found a medium sized 102.6-ft^2 single vertical tail along with twin ventral strakes fins could develop $dCn/d\beta$ ~ +0.010 through α=8^o at Mach 2.5; see FIGURE 12. That same configuration, with the ventral strakes, had strong positive directional stability at transonic speeds ($dCn/d\beta$ ~ + 0.020) with the wings fully swept.[58]

Although promising, those tests did not guarantee a viable departure resistant aircraft, especially for the USN. NASA openly discussed problems of carrier compatibility limiting the height of the vertical tails.[64] In a further test, with both F-111A (USAF) and F-111B (USN variant) wings tested at Λ_{LE} = 50^o, 65 ^{o} and 72.5^o they found lower body fairings decreased static directional stability, reduced span vertical tail significantly decreased directional stability, and canted ventral strakes improved static directional stability.[64]

In the next series of tests, beginning with the established baseline with twin ventral stakes, NASA explored the stability changes associated with external stores and/or open bomb bay doors.[64] Testing the near final F-111A configuration at Λ_{LE} = 65^o, 70^o and 72.5^o over a range of supersonic speeds from Mach 1.6 through 2.86 and angles of attack up to

28^{o}. They found unacceptably large losses in directional stability with missiles prominently installed on body.[64] Internal carriage did not help, as directional stability was also compromised when the bomb bay doors were opened.[64] In a follow on test, NASA found for the F-111A with wings swept to 72.5 that opening main landing gear doors best "speed brake" without detriment to directional stability.[65] They also found that the furthest outboard wing stations for missiles was the least detrimental option for external stores in terms of directional stability.[65]

All of these trends were discouraging. While the F-111A USAF aircraft could tolerate configuration changes further growing the span of the vertical fin, the F-111B USN variant would need to fold such a fin in order to fit within the hangar space of existing aircraft carriers. If the F-111 program were to choose the second option, the USN variant would clearly exhibit further weight growth as well as experience a significant schedule slip. As the MTOW of the production F-111A was 100,000-lbm, it greatly exceeded the 74,000-lbm elevator weight limit and catapult capability of 1960's USN aircraft carriers such as the CVA-43 Coral Sea.[66] The third option would have the USN accept a configuration with inadequate static directional stability which could not be provisioned or fueled below decks; this option was unacceptable.

At the time, the F-111 and F-14 were designed, the state-of-the-art for aircraft flight control systems was limited. Droste states that "only [discrete] analog hardware was allowed; no integrated circuits … [and that if] a failure was detected … the pilot would [slow and descend] to a flight condition where the ... servos could be turned off."[16] In other words, the electronic flight control system was designed to enhance the flyable envelope of an aircraft which already had a reasonable envelope in the absence of any flight control system. The F-111 lacked a stick pusher to provide "envelope protection."[16][67] In the longitudinal channel, it had an adaptive gain pitch rate feedback system to augment damping implemented with a limited deflection authority symmetrically on both horizontal tail planes.[16][67] It also had a limited authority roll damper implemented through differential deflection of the horizontal tail planes cross coupled to the rudder.[16][67] Its rudder was further controlled by a yaw damper which sensed lateral accelerations, yaw rates, and sideslip, along with an angle-of-attack dependent roll rate sensor.[16] [67]

Droste notes that the F-111 open loop performance was unfavorable.[16] To improve flying qualities, the gains had to be quite high **and** had to vary with speed, altitude, and attitude. Given the analog nature of the available flight control technology, General Dynamics with its avionics partner General Electric, implemented an adaptive gain system (inspired by the MH-96 system on the North American X-15 Hypersonic research aircraft) to achieve a target longitudinal damping ratio $\zeta \sim 0.3$; this would ensure that all step inputs would damp within three zero-crossings.[16][17] It is interesting to note that the target short-period damping ratio was lower (less well damped) than that recommended by Bihrle, but is consistent with the preferred "deadbeat damping" of X-15 pilots on extreme speed and altitude missions.[32][17] Droste indicates that flight in either very calm or turbulent air would drive the system to maximum gain which could lead to the stability augmentation system exciting itself and oscillating.[16] While this system was used in the final production F-111A, its protracted development added risk to the program.

The USN discontinued acquisition of the F-111B before a significant flight test program could produce a production NATOPS. The final NATOPS for this aircraft, dated 16 May 1968, includes basic ground-based takeoff and landing data for operational test and evaluation but lacks any sort of advice concerning maneuvering or aerial combat.[67]

We can therefore see that the major lessons learned from the failed development of the F-111B were that:

- Future USN aircraft must be designed to fit within and not exceed the size and weight constraints of existing aircraft carriers.
- Fixing unintended open-loop flying qualities problems with "electronics" is a high-risk strategy. Because F-111 open loop flying qualities were not particularly good, it relied upon an adaptive gain analog flight control system to achieve reasonable short-period and Dutch Roll dynamic response across much of the flight envelope.
- A single vertical tail was not a viable design choice for a variable-sweep supersonic carrier based aircraft as it could not be large enough to provide necessary static directional stability.
- The large bomb bay doors needed to accommodate substantial internal stores as well as the external stores locations tested tended to be directionally destabilizing, particularly at supersonic speeds.

Thus, Grumman building on lessons learned from the F-111B program could re-use much of the initial aerodynamic configuration work performed at NASA so long as the resulting aircraft could have twin vertical tails and a layout that allowed substantial external stores to be carried in a manner which did not compromise directional stability. An overlay of the top-view of the NASA TM X-710 configuration with the production F-14 configuration shows the direct heritage of the Grumman design; it is clearly an evolution of the pre-contract-award NASA baseline; see FIGURE 13.[14][53]

B. Development of the F-14 Production Configuration

Official sources state that the "F-14 weapon system is … a supersonic, tandem two seat, twin engine, variable wing-sweep, carrier based fighter aircraft capable of performing several missions including: combat air patrol … which includes carrying Sparrow and Sidewinder missiles and an internal gun … [and] fleet air defense using Phoenix missiles … [and] ground attack missions."[68] Kress states that the combat air patrol mission "drove the design of the aircraft from a size and weight point of view" although it had to be large enough to carry six Phoenix missiles for fleet air defense missions. [15] Its unswept configuration was driven by carrier takeoff and landing; its fully swept configuration was optimized for flight as fast as Mach 2.4; intermediate sweep angles were needed for aerial combat (instant and sustained turn capability) without a tendency to stall or spin.[15]

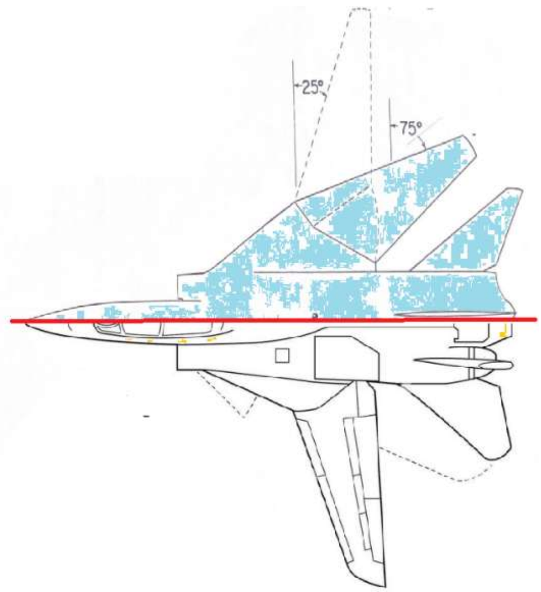


FIGURE 13 – Comparison of NASA TM X-710 planform with production F-14 configuration.[14][53]

Grumman officially maintained that "for a fighter to acquire and maintain air superiority … it must be able to take full advantage of its aerodynamic lifting potential."[69] An aircraft cannot be flown to its full potential "if the aircraft has an inherent instability [as] the fear of stall/spin departure will deter the pilot from exercising the aircraft to its limits."[69] Leveraging all of the NASA study work described previously, Grumman decided to optimize its wing for combat maneuvering in the Mach 0.8 to 0.9 speed range at Λ_{LE}=50°; see FIGURE 14.

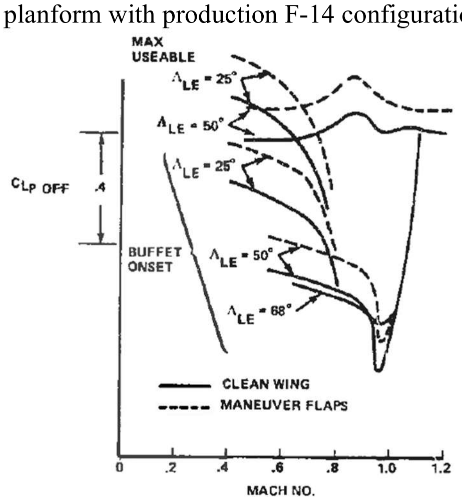


FIGURE 14 – F-14 Transonic Aerodynamic Envelope [69]

Again, the timeline of full-scale development does not support the idea that the F-14 had extensive wind-tunnel data supported design refinement between contract award (in January 1969) and first flight (in December 1970). Grumman appears to have leveraged their internal facilities and the private CALSPAN wind tunnel complex for F-14 developmental work. Honig & DeLuca [69] in a 1974 AIAA conference paper discussing the design of the F-14 largely source Cornell Aeronautical Laboratory reports for their substantiating data. Contemporaneous NASA documents rarely present themselves in the public record; though Honig makes passing references to the F-14 being tested in 1969 at NASA ARC, NASA LaRC 8-ft tunnel and the NASA LaRC vertical spin tunnel. He also mentions that a 1/10th scale "free-flight" remotely controlled model was "flown" at the NASA LaRC 30x60-ft low speed tunnel in "the latter part of 1970." [69]

Pre-contract award NASA wind tunnel tests, documented above, did not permit comprehensive exploration of high angle of attack aerodynamics. Only the low speed data set extended to α~40° .[57] Transonic datasets were restricted to -2º<α<+24º [58] Supersonic datasets were restricted to -4º<α< +28º [59] NASA TM X-1036, see FIGURE 15, revealed some encouraging trends. For a variety of wing sweeps 25º<Λ_{LE}<60º , the basic F-14 planform delayed "stall" until α~30º while longitudinal stability remained broadly positive.

Bihrle & Meyer wrote that high angle-of-attack lift generation capabilities of the F-14 were confirmed in an April 1970 wind tunnel test at the NASA/ARC 12-ft pressure tunnel.[70] At angles of attack up to 90º, the "F-14 configuration at Λ_{LE} of 22, 35, and 50 deg [had] … desirable lift and pitching moment characteristics."[70] These same authors also noted that the "NASA vertical spin tunnel results [indicated] that the F-14 had fast flat spin mode … [but] a parallel Grumman … study … [indicated] that although recovery from the flat spin mode could not be effected, spin entry characteristics were difficult to attain."[69]

C. Aerodynamic Design Features of the F-14A

The production F-14 features an *AR*~7.3 wing; this is right in the middle of the NASA variable sweep fighter trade space (which considered wings from *AR*~5 through *AR*~8). [47-65] The wings can be unswept to Λ_{LE}=22º for takeoff and landing or swept to Λ_{LE}=68º for supersonic dash; these sweeps are also within the NASA study range.

Kress stated that "the airfoil sections on the F-14 wing were very straightforward in design. There is nothing particularly unusual about those airfoils. The root and tip sections are both modified 63A series airfoils of 9 percent thickness. The tip section is twisted 5 degrees, leading-edge-down with respect to the root section." [15]

Kress describes the detailed aerodynamic design features of the F-14 which were innovations beyond that found in the NASA reports; see FIGURE 16.[15] For the wing, these include "leading edge slats" and "main flaps" used for takeoff, landing and maneuvering; a set of inboard flaps used only in the unswept configuration for takeoff and landing. Wing mounted spoilers are used for roll control. An extensible "glove vane" is deployed at the highest Mach numbers to reduce excessive static margin.

Kress states that the pivot location for the F-14 wing is further outboard than the F-111.[15] He corroborates that the effect of pivot location on longitudinal stability was passed on prior "NACA" research; i.e. the Polhamus paper.[52] With the F-14's outboard pivot location, "the

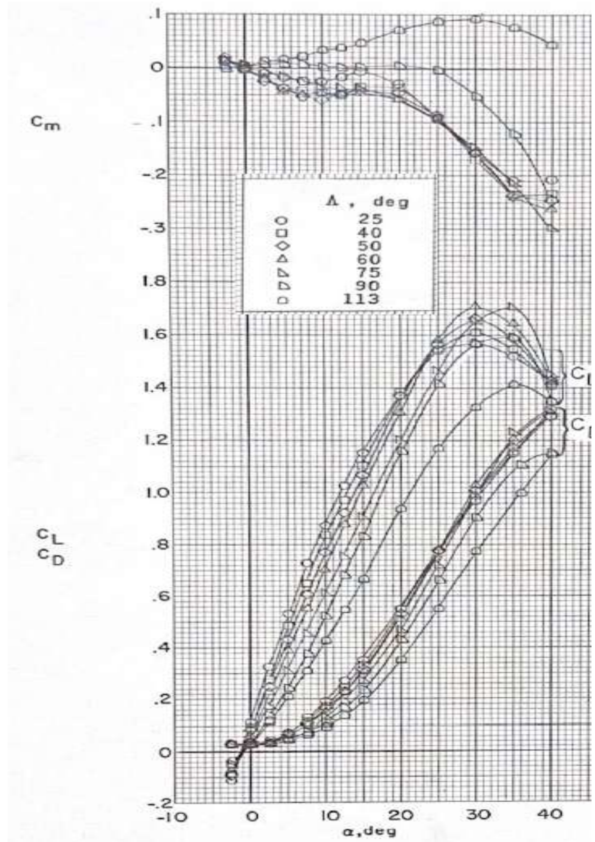


FIGURE 15 – Subsonic Longitudinal Aerodynamics from NASA TM X-1036.[57]

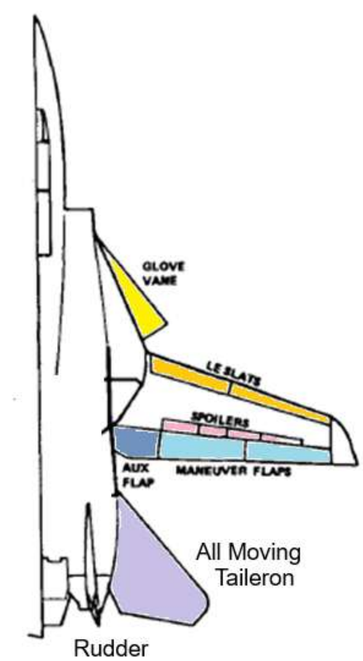


FIGURE 16 – F-14 control surfaces

static margin first increases as the wing is swept back to a peak at around 50º … then begins to move in the other direction."[15] Between the outboard pivot and the deployment of the glove vane, the F-14 can limit its speed dependent change in longitudinal stability to less than 20% of the reference chord.[15] Kress holds that the benefits of the seemingly subtle differences between the F-14 and F-111 are manifest: the F-14 can develop + 8-g's instantaneous load factor instead of less than +2-g's for the F-111.

The all moving tailerons of the F-14 are generally of the same configuration as found with the NASA TM X-710 configuration but have been enlarged somewhat for additional control power; see FIGURE 16. They will be used collectively as the primary pitch and differentially as a primary roll control surface. At subsonic speeds, with the wings at $\Lambda=50^{o}$ or less, the tailerons are augmented by asymmetric spoilers (spoilerons) for roll control.

A side profile view of the F-14 in comparison with the NASA TM X-710 configuration (in grey) shows more marked differences; see FIGURE 17.[14][53] We see that the forebody is considerably deeper with the F-14. Tall single dorsal vertical of the NASA configuration has been replaced with a pair of dorsal and ventral fins located as far aft as practicable; the dorsal fins each have a rudder.

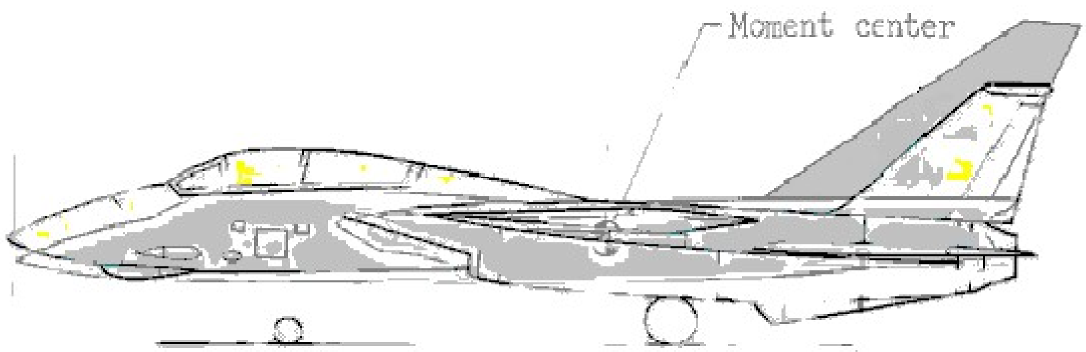


FIGURE 17 – Side View F-14A vs. NASA TM X-710 configuration.[14][53]

Since the Essex and Midway class carriers of the USN had a 17.5-ft tall hanger, it becomes clear that the F-14 was configured with the tallest possible vertical fins that could be accommodated on-board without a folding mechanism as it is officially 16.0-ft high.[71]

The F-14 omitted the aerodynamically troublesome bomb bay of the F-111. Its widely spaced engines gave additional room along the centerline to accommodate external stores that would not otherwise degrade directional stability. We shall see shortly that this attempt to increase static directional stability over the F-111 proved insufficient.

D. Original Flight Control System

A review of the F-14 NATOPS confirms that pitch is controlled by symmetrical deflections of the horizontal fins; notional roll is predominantly controlled by anti-symmetric deflections of the horizontal fins (taileron) and is augmented by unsymmetric wing spoilers for wing sweeps < 62º; and notional yaw is controlled by the dual rudders.[72] Symmetric wing spoiler deflections can be independently controlled by the pilot for use in approach and landing.

The NATOPS indicates that the F-14 also lacks a stick pusher.[72] The stability augmentation system mechanically mixes its commands with the physical stick placement and a trim unit. The flight control system has limited authority (+/- 3º symmetric taileron deflection vs the stick +10º/-33º) and is bandwidth filtered to have noticeably less rate (20º /sec) than would be limited by actuator response (36º/sec).[72]

The NATOPS also provides block diagrams for the lateral control system. Once again, the stability augmentation system mechanically mixes its commands with the physical stick placement and a trim unit. With wings unswept, the differential taileron action is augmented (at moderate to large stick placement) with asymmetric spoiler. The flight control system has limited authority (+/- 5° anti-symmetric taileron deflection vs the stick +/-7°) and is bandwidth filtered to have a slight lower rate (33° /sec) than would be limited by actuator response (36°/sec).[72]

Finally, the NATOPS also provides block diagrams for the yaw control system. Yet again, the stability augmentation system mechanically mixes its commands with the physical stick placement and a trim unit. The flight control system has limited authority (+/- 19° rudder vs the stick +/-30°) and is bandwidth filtered to have a slightly lower rate (80° /sec) than would be limited by actuator response (106°/sec).[72] Rudder authority is mechanically limited to +/-9.5° above ~400-KIAS. High angle-of-attack warning is provided for α>20° by a rudder pedal shaker system.

Superficially, the F-14A stick and rudder system appears to lack any form of fundamental aileron-rudder interconnect. It is our understanding that ARI was first implemented with the F-14D and F-14B digital flight control system retrofits from 1990.

VI. Low-Speed Flying Qualities

A. Low-Speed Estimated Aerodynamic Data

Our investigation of the limited publicly released wind tunnel data [73] in comparison with publicly released flight test data [74] reveals that the in-service CG location was likely to be further forwards than used at the tunnel (16.2% MAC). As the NATOPS states "with the gear and flaps lowered and 20-deg of wing sweep … [and] CG location of 18−percent MAC or greater, the static margin is greatly reduced from normal … **Loss of control is likely.**"[72]

Our aerodynamic data will be prepared about the mass properties described in TABLE 1.

TABLE 1 – F-14A Nominal Mass Properties at the End-of-Mission [74]

Weight	*Ixx*	*Iyy*	*Izz*	*Ixz*	*Sref*	*b*	*c*	*CG*	*XMRP*	*ZMRP*
48,531 lbm	66,120 slug-ft^2	265,681 slug-ft^2	327,689 slug-ft^2	-2,537 slug-ft^2	565 ft^2	64.13 ft	9.8 ft	13.4% MAC	35.0-ft	-0.5-ft

In order to analyze the stability of the F-14A, we will need to develop a comprehensive aerodynamic model including controls-neutral aerodynamic characteristics, control power due to "elevator," "aileron," and "rudder" inputs and the dynamic derivatives (*Cmq, Clp, Cnr*, …). Given the highly fragmental public data set, we utilize VORLAX2024d to estimate the missing terms; see FIGUREs 18 and 19.[75] VORLAX is a potential flow panel-method flow solver code for low and high speed, fixed-wing aircraft.[76][77] With a panel model carefully constructed to best match available wind tunnel data, we may then create a full aerodynamic model so that we may address fundamental rigid-body mode behavior (frequency and damping) as well as control power requirements under stick-fixed and augmented control conditions. Since VORLAX is an inviscid code, we will limit our study range to moderate angles of attack (α<~20°). When we discuss viscously dominated control surface issues, like the spoileron control power, we will restrict ourselves to NASA sourced wind-tunnel data, see FIGURE 20.[51]

VORLAX can also estimate the dynamic derivatives for arbitrary configurations. Pitch damping, *Cmq*~-15.2; roll damping, *Clp*~-0.4; yaw-due-to-rolling, *Cnp*~-0.05; yaw damping, *Cnr*~-0.24; roll-due-to-yawing, *Clr*~+0.3.

Kress confirms that the F-14A flight control system is "not really" dependent upon sophisticated electronics. He states that the "horizontal tails, rudders and wing sweep have direct mechanical linkage control systems … the spoilers are driven by a direct electrical [connection] … [and that] without any … automatic systems operation, the aircraft flying qualities are quite good and will allow mission completion."[15] In other words, absent any automatic systems, Kress indicates that the F-14A should exhibit LEVEL 1 flying qualities over a reasonably large flight envelope. If this statement is true, we should be able to confirm the F-14's compliance with MIL 8785B specifications through an evaluation of its basic aerodynamic database.

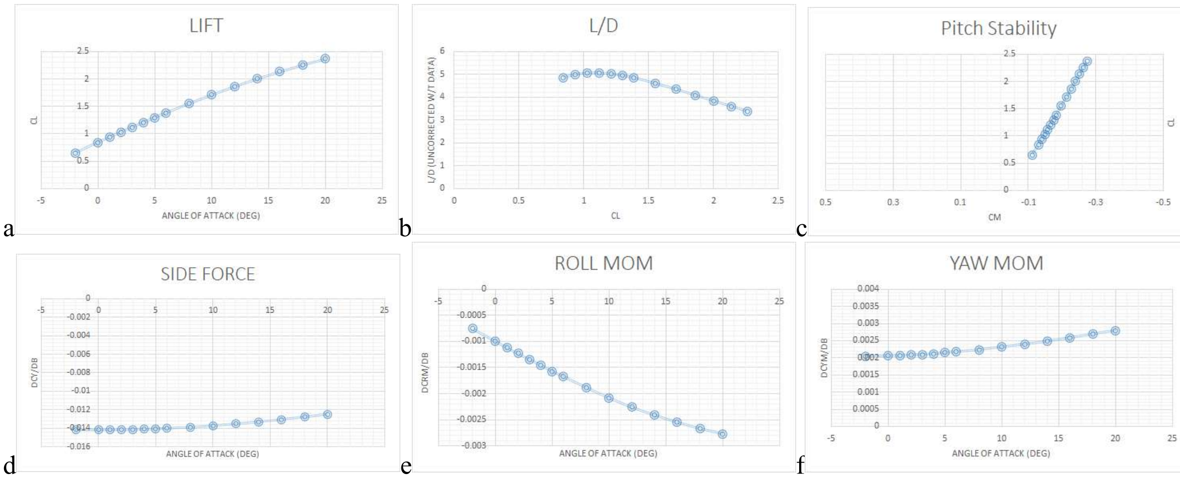


FIGURE 18 – F-14 Low Speed Configuration (Λ=22º) Estimated Controls Neutral Aerodynamic Data. Landing Flaps. Stability axis. a) *CL vs* α, b) *L/D* vs α, c) *CL* vs *Cm*, c) *dCY/d*β vs α, e) *dCl/d*β vs α, f) *dCn/db* vs α.

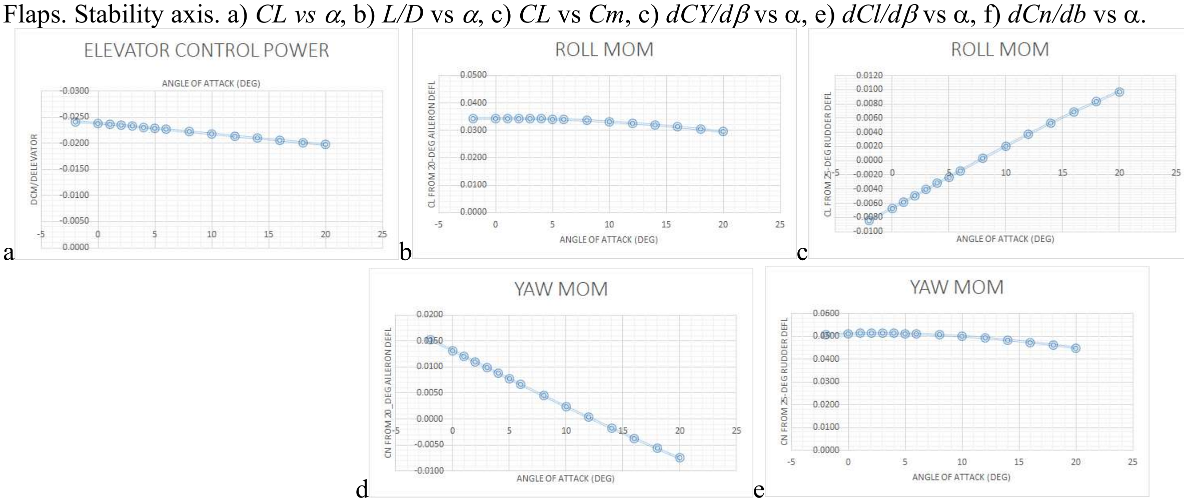


FIGURE 19 – F-14 Low Speed Configuration (Λ=22º) Control Power. Landing Flaps. Stability axis. a) *dCm/delev* vs α, b) *Cl from taileron* vs α, c) *dCl from rud* vs α, d) *dCn from taileron* vs α, d) *Cn from rud* vs α.

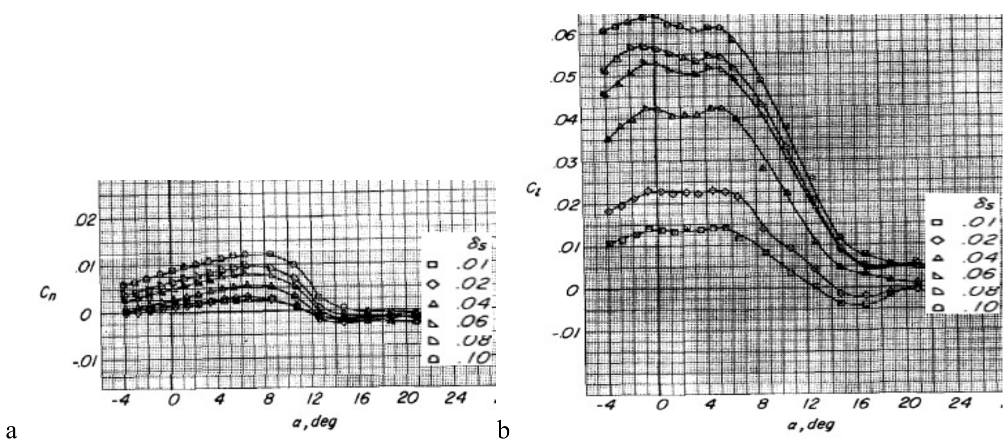


FIGURE 20 – Generic Variable Sweep Wing Configuration wind tunnel data from NASA TM X-542 to estimate Spoileron Control Power. a) dCn/dspoiler, b) dCl/dspoiler [51]

B. Basic MIL 8785B Compliance at Low-Speeds

Table 2 (overleaf) shows estimated parameters for a typical landing approach scenario of the F-14A.

The fact that the Short Period and Dutch Roll Frequencies are close together ($\omega_{sp} \approx \omega_{dr}$) at the evaluated CG position is worrisome, as energy can exchange between the two motions and inertia couple. At approach and landing conditions, the critical roll rate which controls inertia coupling is controlled by the roll rate interacting with the longitudinal mode, $p_{crit} \approx 45^o$/sec. As the CG moves aft and $dCm/d\alpha$ grows less negative, the critical roll rate will decline further. Fortunately, the Short Period mode is inherently well damped.

At the time that the F-14 was designed, spin resistance was largely thought of as being dominated by the strength of the fundamental Dutch Roll mode as represented by $C_{n_{\beta dynamic}}$. Under 1960's era metrics, the F-14 should have "acceptable stall characteristics with some wing rock, so long as sideslip is kept within bounds, no yaw departure tendency so long as stall is not prolonged."[78]

At the analyzed forward CG position, FIGURE 21a shows that there is ample longitudinal control power to trim through α~20^o; only ½ the total deflection is needed. Thus, full back stick can command angles of attack far in excess of 20^o. FIGURE 21b shows that the asymmetric horizontal tail (tailerons) alone is insufficient to trim out the required β=+/-13^o sideslip from crosswinds. It also shows that taileron augmented by the spoilerons has generous control power to trim sideslip through at least α~13^o. At high angles of attack, roll control power declines substantially.

The quality of the available lateral/directional control power at low speeds may be assessed first by considering the amount of rudder needed to trim out the unintended yawing moment of the lateral control effectors. Recall from FIGURES 19b and 19d that for α<12^o the rolling moment (*Cl*) and yawing moment (*Cn*) from differential taileron have the same sign, the yaw-due-to-roll is proverse. Similarly, FIGURES 20a and 20b shown that the yaw-due-to-roll from the asymmetric spoiler (spoileron) is similarly proverse. The amount of rudder needed to trim out the unintended yawing moments are reasonable, no more than 1/3 of available control power is needed to counteract the unintended roll (δ_{rudder}=10^o) if full taileron and full spoileron are simultaneously deflected; see FIGURE 22a.

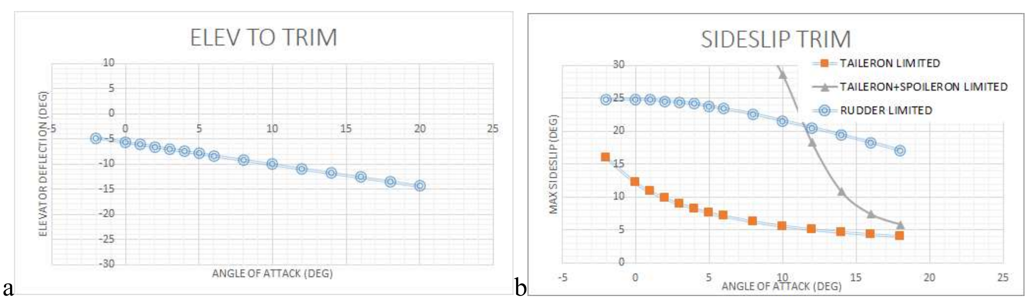


FIGURE 21 – Quantity of Control Power – F-14A Landing Configuration. a) collective taileron (i.e., elevator) needed to trim, b) maximum trimmable sideslip angle in yaw from rudder and in roll from tailerons and spoilerons.

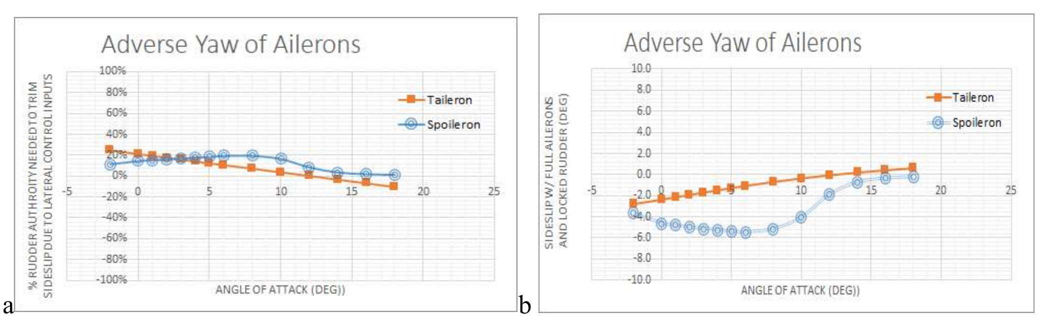


FIGURE 22 – Quantity of Control Power – F-14A Landing Configuration. a) % rudder authority needed to trim out unintended sideslip due to maximum taileron (+/-7^o) or spoileron deflection, b) sideslip angle where static directional stability balances the unintended yawing moment due to maximum taileron or spoileron deflection.

TABLE 2 – F-14A Landing Approach Characteristics [72]

	Parameter	Value	Rationale
GENERAL CHARACTERISTICS	$VKEAS$ (knots)	135	F-14A weighing 48,531-lbm is to approach at 135 knots
	q (lbf/ft^2)	61	The landing $VKEAS$ corresponds to a reasonable dynamic pressure
	Izz/Ixx	~5	F-14A is quite "body heavy" and sensitive to disturbances in roll
	$\left\lvert\frac{Izz-Ixx}{Iyy}\right\rvert$	~0.98	Strong tendency to inertia couple
	$C_L = \left(\frac{W}{S_{ref}}\right)/q$	~1.4	F-14A can fly in excess of C_L ~ 2.0 without buffeting/stalling from wind tunnel experiments in a typical landing condition
	α (deg)	~6	Estimated from FIGURE 19a, flown at flaps down landing condition
	$dC_m/d\alpha$ (1/deg)	-0.0077	Estimated from FIGURE 19c
	dC_m/dC_L	-0.092	F-14A is ~9.2% stable; estimated from FIGURE 19c
	L/D	~4.9	Estimated from FIGURE 19b; will provide sufficient sink rates for landing
SHORT-PERIOD	n/α (g/rad)	~3.5	Short-period flying qualities are firmly within Level I boundaries set by MIL 8785B for "Category C" flight phases (i.e., takeoff or landing)
	ω_{sp} (rad/s)	~0.93	
	$CAP = \frac{\omega_{sp}^2}{n/\alpha}$	0.17	
	ζ_{sp}	0.60	Damping ratio is well above 0.35-Level I requirement from MIL 8785B
DUTCH ROLL	$C_{n_{\beta,dynamic}}$	~0.0023	Dutch roll stability is present and stable; both $dC_l/d\beta$ and $dC_n/d\beta$ are stable
	ω_{dr} (rad/s)	~1	Within Level I region set by MIL 8785B, but slow
	$dC_Y/d\beta$	-0.014	Side force coefficient is reasonable
	ζ_{dr}	0.17	Dutch roll damping ratio is well within "Category C" Level I requirements
SPIRAL/ ROLL	$s \approx \frac{L_\beta N_r - N_\beta L_r}{L_\beta + N_\beta\left(\frac{Ixz}{Ixx}\right)}$ (1/s)	0.15	Spiral mode time constant is stable and within Level I requirements
	$\tau_r \approx -\frac{1}{L_p}$ (s)	0.5	Roll mode time constant is excellent for roll sensitivity; well within Level I requirements

Recall that MIL 8785B specifies a maximum tolerable adverse yaw level from aerodynamic roll; a LEVEL 1 aircraft cannot exceed 15º adverse sideslip or 4º of proverse sideslip from a maximum uncoordinated lateral stick input during takeoff and landing. Turning to FIGURE 22b, we estimate that the F-14A is marginal here our analysis indicates that the strong proverse yaw-due-to-roll in combination with only moderate static directional stability has a quasi-static moment balance $\frac{dCn}{d\beta}\,\beta \equiv -\frac{dCn}{dail}\,\delta ail_{max}$ where β=~5º proverse.

MIL 8785B requires the manufacturer to demonstrate satisfactory spiral stability and roll response. The spiral mode manifests itself as a tendency for an aircraft to roll into an ever-tightening spiraling turn when experiencing moderate sideslip (unstable) or roll out of a turn back to wings-level flight (stable). Its counterpart, the roll mode, represents the time to achieve 63% of the aircraft's peak roll rate with the lateral stick fully deflected. TABLE 2 finds F-14 compliant with necessary requirements.

C. Low-Speed Flying Qualities Issues Noted During Initial Acceptance Testing and Operational Service Led to the Understanding that MIL 8785B was incomplete

In theory, compliance with MIL 8785B should produce a superior flying aircraft. As Woodcock & Weissman stated in 1976, "initial F-14 … flight results show that substantial improvement can be secured by concentrated design attention."[79] As shown above, with some careful reverse-engineering to establish the aerodynamic characteristics of the F-14A, our initial assessment is that the Grumman engineers should have engineered the F-14 to have satisfactory low speed flying qualities even in the absence of synthetic pitch, roll or yaw damping.

Such was not the case.

The Navy wrote that:

> "After many years of operational experience with the F-14A aircraft … major deficiencies in the handling qualities have been identified.
> …
> In the power approach configuration (landing gear and flaps down), the F-14 generates significantly large adverse sideslip in response to lateral stick inputs. This adverse sideslip, coupled with the airplane's strong positive dihedral effect tends to excite the Dutch-roll mode. This characteristic significantly degrades the pilot's ability to make accurate lateral line-up corrections during the terminal phases of a carrier approach. The pilot is constantly required to coordinate lateral stick inputs with rudder during the carrier approach phase, detracting from his overall situational awareness during this critical flight phase."[80]

Kelley & Enevoldsen at NASA report that pilot comments found:

> "Handling-qualities problems in the landing configuration [which] are primarily due to adverse yaw following lateral stick inputs and a lightly damped Dutch Roll mode. These effects are apparent both in visual and instrument tasks. Unless the pilot applies a generous amount of coordinating rudder, there is substantial **adverse yaw** both rolling into and out of turns. The adverse yaw and resultant heading excursions cause difficulty both in holding a specific heading and in making precise lineup corrections during an approach. The result is that pilot workload, already heavy because of the difficulty of a carrier approach, increases considerably, because of the adverse control and damping characteristics."[74]

The NATOPS describes the F-14 Dutch Roll as a "wallowing, snaky motion of the nose that severely degrades heading and/or lineup control."[72] While meeting MIL 8785B LEVEL 1 standards, during final approach "the period of this motion is quite long [~6-sec -*T*] and has the unfortunate result that the pilot perceives a heading error when referenced to centerline, when in fact the flightpath is correct."[72]

Our analysis in the preceding section indicates that, absent a closed-loop synthetic Dutch Roll damper, the Dutch Roll damping was marginal. However, the F-14A did have an analog stability-augmentation system with pitch, roll and yaw damping authority which should have improved its open-loop flying qualities when functioning correctly.

But what of the pilot reports of having adverse yaw? The wind tunnel and computational data all indicate that the F-14A should have had proverse yaw. Even the NATOPS says that:

> "Since roll control is provided by wing−mounted spoilers and differential stabilators, the aircraft exhibits proverse yaw throughout the flight envelope (yaw in the direction of the lateral stick input)."[72]

However, a retired F-14 fleet pilot disagrees. From his perspective, under the canopy:

> "I never experienced an overall proverse yaw in the F-14 **ever**.
> …
> In the landing pattern the adverse yaw was large but manageable. Some pilots used the rudders to control it with varying success. Many (including me) just learned to compensate for the adverse yaw by managing our roll inputs and left the rudders alone. The [Lt.] Kara Hultgreen accident was the result of large rudder input to control adverse yaw [during a carrier approach] and resulted in an engine stall due to intake blanking in an F-14A with the TF30 engines."[81]

Was the wind tunnel incorrect? No, as there is an underlying factor in the aircraft dynamics.

Here we see how a dynamic interaction causes the F-14A to exhibit an adverse yaw response despite having static proverse yawing moments.

First, we need to understand how much roll control power is actually used to execute a roll maneuver. Since the roll time constant is quite short ($\tau_r \approx 0.5$ sec), the aircraft will rapidly attain the roll rate implied by a quasi-static analysis. If the pilot wants to achieve a sustained roll rate of *p*=30°/sec (0.52 rad/sec) [slower than the critical roll rate discussed above], the rolling moments from the aileron will need to balance to opposite rolling moments due to the roll damping derivative, *Clp*. $\delta Cl_{aileron} = -C_{lp}\left(\frac{p\,b}{2\,VKTAS\left(\frac{6076}{3600}\right)}\right) \approx -0.03.$

To promote this roll rate at low speeds (135 knots) requires full authority on antisymmetric taileron (+/- 5°) and about 1/4 action on the spoilerons; see FIGUREs 20b, 20d, 21a and 21b. Thus, the aerodynamic database at α=6° solves for the case when $\delta Cl_{aileron} \approx 0.03$; $\delta Cn_{aileron} \approx 0.003$. Roughly 10% static **proverse** yaw results from the uncoordinated roll command.

However, because the yaw-due-to-roll rate derivative, *Cnp*, is negative, we see that the adverse yaw of a rolling motion can overpower the proverse yaw of the ailerons: $\delta C_n = -C_{np}\left(\frac{p\,b}{2\,VKTAS\left(\frac{6076}{3600}\right)}\right) \approx +0.0042$ which is greater than $\delta Cn_{aileron} \approx 0.003$. Therefore, we see that the F-14A exhibits a dynamic situation where a static proverse yawing moment is masked by a dynamic adverse moment once the aircraft begins to roll. When the pilot neutralizes lateral stick, the proverse moments vanish and all the aircraft "feels" are the adverse moments from the developed roll rate. If the pilot attempts to reverse the stick to arrest the rolling moment, the proverse yaw from the command to roll out of the bank will compound the adverse yawing moment.

The ϕ/β ratio represents the tendency for an aircraft to express its Dutch Roll behavior as a wing rock (ϕ/β>1) or as a tail wag (ϕ/β<1). For the F-14A, in the landing configuration we may estimate the $\left|\frac{\Phi}{\beta}\right| \approx \frac{\frac{dCl}{d\beta}}{\frac{dCn}{d\beta}}\frac{Izz}{Ixx}\frac{1}{\rho\,VKTAS} \approx 12$; the F-14A will express its Dutch Roll as a wing rock at low speeds.

To improve the Dutch Roll damping the stability augmentation system will need to make the either roll-damping-derivative larger (i.e., increase the negative magnitude of *Clp* by scheduling a negative rolling moment from the control surfaces due to sensed roll rate) or increase the negative magnitude of *Cnr* by scheduling a negative yawing-moment from the rudder motion with sensed non-accelerated yaw rate. Indeed, the F-14A had an analog "roll damper" which could provide inputs to differential taileron along with very limited control of the inboard spoilerons all as a function of sensed roll rate **and** an analog "yaw damper" which provided rudder commands as a function sensed yaw rate.

Since the action of both taileron and spoileron surfaces are "proverse" at typical approach speeds in the landing configuration, we see that any attempt to improve the roll damping would drive the yaw-due-to-roll dynamic derivative, *Cnp*, more negative as well. This means that the "roll damper" system would exacerbate pilot perceived adverse yaw characteristics due to lateral stick input. The basic roll damper on the F-14A was scheduled to command -5° aileron at a +135°/sec yaw rate; *p*=2.35 rad/sec.[74] $Clp_{synthetic} \approx 2\,VKTAS\left(\frac{6076}{3600}\right)\left(\frac{\delta Cl_{aileron}}{p\,b}\right) \approx -0.5$; $Cnp_{synthetic} \approx 2\,VKTAS\left(\frac{6076}{3600}\right)\left(\frac{\delta C\ \ aileron}{p\,b}\right) \approx -0.05$. Adding these values to the bare airframe estimates gets us an effective $Clp_{augmented}$ ~ -0.9 and $Cnp_{augmented}$ ~ -0.1. The augmented *Clp* further quickens the already fast roll time constant but decreases the sustained roll rate for a given lateral stick input. At the same time, the augmented *Cnp* doubles the dynamic adverse yaw effect. The only way to increase the magnitude of *Clp* while reducing *Cnp* to near zero requires the simultaneous application of rudder with aileron – basically introducing a feedforward ARI into the roll damper feedback loop. This functionality would have to await the DFCS of the F-14B and D many years in the future.

A yaw damper increases the negative magnitude of *Cnr*. However, we must consider the proverse or adverse rolling moments developed by the rudder; see FIGURE 20c and e. At a typical approach angle of attack, the unintended roll from the rudder is quite small. As the angle of attack rises, the unintended roll due to yaw becomes increasingly adverse. Thus, at high alphas the yaw damper will reduce the positive magnitude of *Clr* which could potentially impact spiral stability. The basic yaw damper on the F-14A was scheduled to command -19º rudder at a +50º/sec yaw rate; *r*=0.87 rad/sec.[74] Since +19º rudder produces *dCn* ~ +0.04, we can see that at 135 knots approach speed $Cnr_{synthetic} \approx 2\,VKTAS\left(\frac{6076}{3600}\right)\left(\frac{\delta Cn_{rudder}}{r\,b}\right) \approx -0.31$. Adding this to the bare airframe *Cnr*, results in an effective $Cnr_{augmented}$ ~ -0.55 which increases the damping to $\zeta_{dr} = 0.33$ and $\omega_{dr}\,\zeta_{dr} = 0.30$; see FIGURE 23.

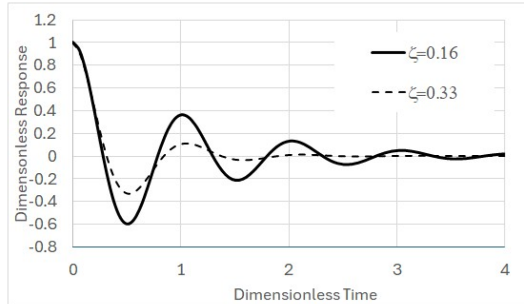


FIGURE 23 – Dutch Roll Damping response to a step input. a) ζ=0.16 without yaw damper, b) ζ=0.33 with yaw damper.

With the implemented control laws, the operational F-14A had a "lightly damped" Dutch Roll mode with sideslip excursions in excess of 10º being seen in operational use due to the dynamic response to uncoordinated roll inputs.[74] The roll damper needed to help damp the Dutch Roll mode only exacerbates the problem.

As a result of F-14A development and operational experience, MIL 8785C introduced a requirement limiting the amount of permissible sideslip following "a yaw-control-free step roll command."[82] The early F-14A clearly could not comply.

VII. Prediction of High-Speed Flying Qualities

A. High-Speed Estimated Aerodynamic Data

At high subsonic speeds, the F-14A will be flown with wings swept aft 50º; common convention will hold the reference area (Sref), span (b) and chord (c) the same as the low speed configuration.

As before, because of limited publicly released wind tunnel data, we will estimate the controls-neutral aerodynamic characteristics, control power due to "elevator," "aileron," and "rudder" inputs and the dynamic derivatives (*Cmq, Clp, Cnr*, …) using VORLAX2024d; see FIGUREs 24 and 25. We continue to limit our study range to moderate angles of attack (α<~15º). FIGURE 26 has wind tunnel data from swept wing asymmetric spoiler deployment.

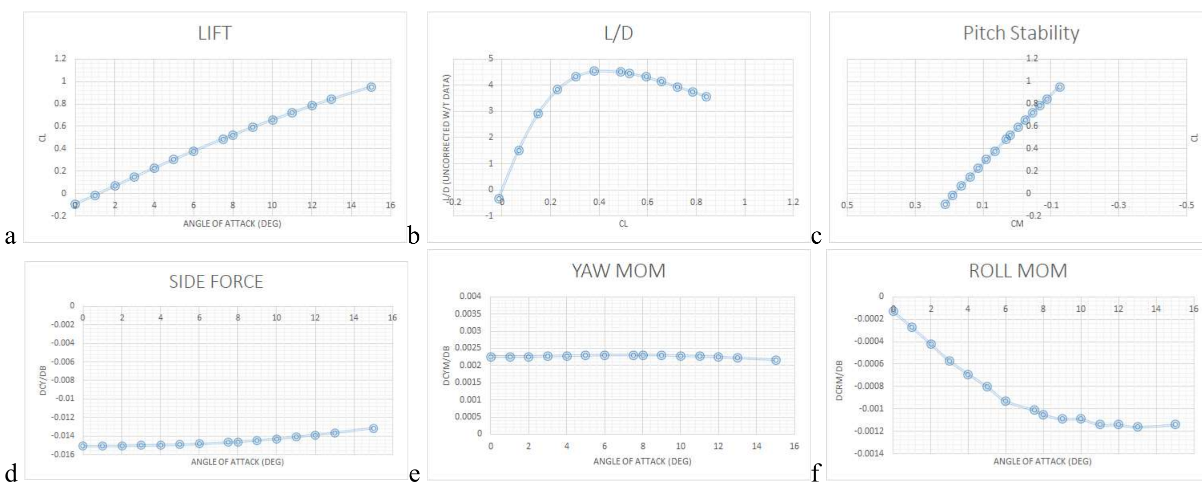


FIGURE 24 – F-14 High-Subsonic Configuration (Λ=50º) Estimated Controls Neutral Aerodynamic Data. Landing Flaps. VORLAX stability axis. a) *CL vs α*, b) *L/D* vs *α*, c) *CL* vs *Cm*, c) *dCY/dβ* vs α, e) *dCl/dβ* vs α, f) *dCn/db* vs α.

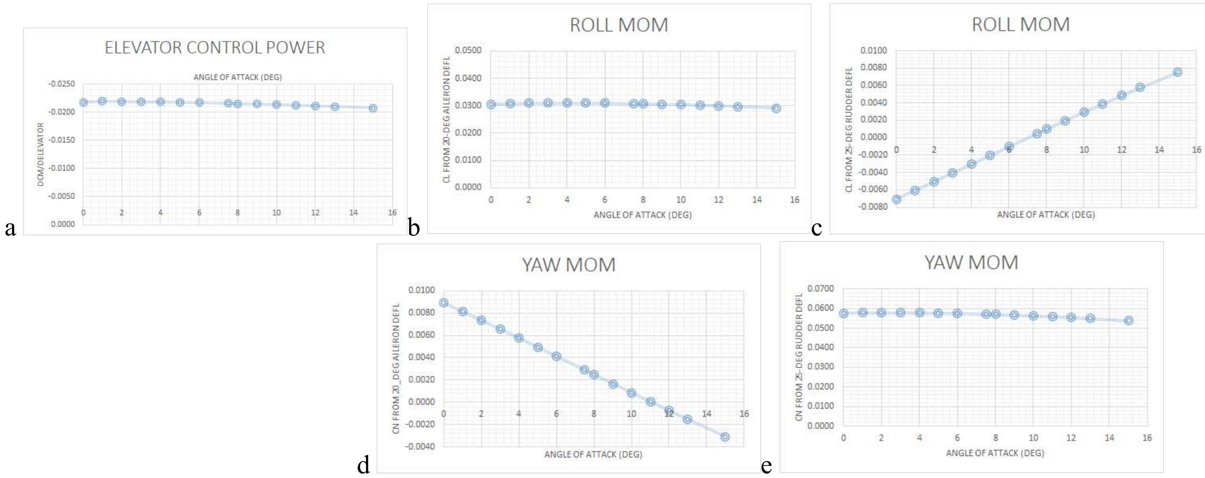


FIGURE 25 – F-14 High-Subsonic Configuration (Λ=50º) Control Power. VORLAX stability axis. a) *dCm/delev* vs *α*, b) *Cl from taileron* vs *α*, c) *dCl from rud* vs *α*, d) *dCn from taileron* vs *α*, d) *Cn from rud* vs *α.*

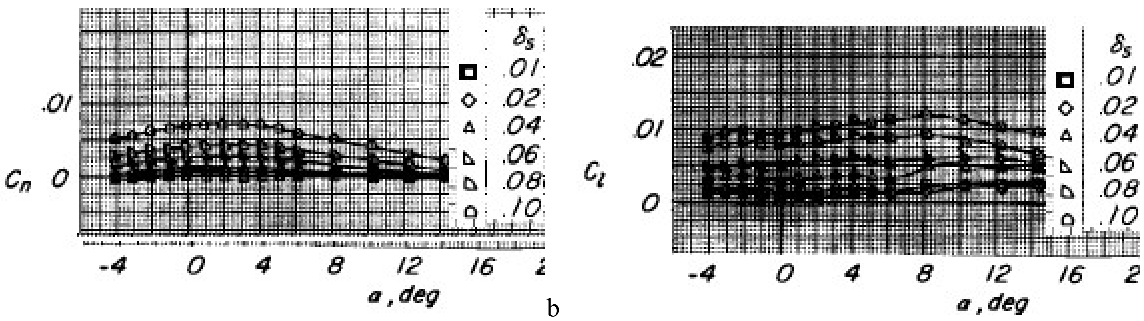


FIGURE 26 – Generic Variable Sweep Wing Configuration wind tunnel data from NASA TM X-542 to estimate Spoileron Control Power. a) dCn/dspoiler, b) dCl/dspoiler

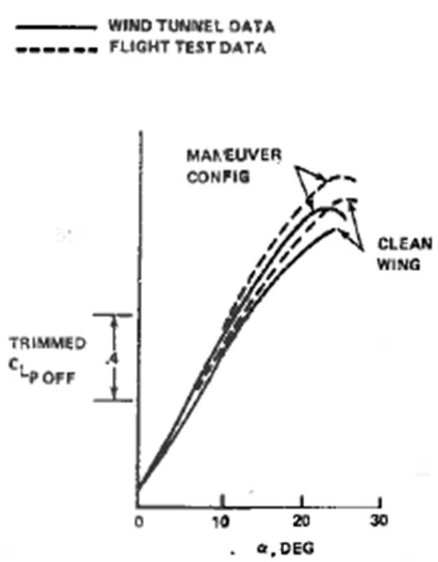


FIGURE 27 - F-14 High-Subsonic Configuration (Λ=50º) *CL* vs α data from Honig [69].

With wings swept, VORLAX estimates stick-fixed pitch damping, *Cmq*~-17; roll damping, *Clp*~-0.15; yaw-due-to-rolling, *Cnp*~+0.001; yaw damping, *Cnr*~-0.14; roll-due-to-yawing, *Clr*~+0.01.

Honig [69] stated that Grumman was aware of the F-14's good maneuvering capability at high subsonic speeds as the wind tunnel data indicated :

- *CL* vs α was relatively linear through α ~22º; see FIGURE 27 and consistent with VORLAX FIGURE 24a.

- Static longitudinal stability (dCm/dα < 0); consistent with VORLAX FIGURE 24c.

- Static directional stability (dCn/dβ > 0) to α=16º; consistent with VORLAX FIGURE 24f.

- Dynamic directional stability (Cnβdynamic > 0) to α=18º.

B. Basic 8785B Compliance at High-Subsonic Speeds

Now let us consider a training flight maneuver at W=48,531-lbm, Mach 0.8 (530 KTAS) and ALT=15,000-ft in a sustained Φ=70º bank turn to replicate the conditions found in *Top Gun* at the moment "Maverick" encounters "Iceman's" jetwash and loses control. This represents flight at a dynamic pressure, q=~535-lbf/ft^2; i.e., 400 KEAS.

TABLE 3 – F-14A 3-*g* Transonic Bank Maneuver Characteristics from *Top Gun* [72]

	Parameter	Value	Rationale
GENERAL CHARACTERISTICS	$VKEAS$ (knots)	400	Near the upper-end of maneuverable equivalent velocity in the F-14A
	q (lbf/ft^2)	535	This corresponds to a relatively high dynamic pressure, but is within reasonable material and structural failure limitations
	Izz/Ixx	~5	F-14A is quite "body heavy" and sensitive to disturbances in roll
	$\left\lvert\frac{Izz - Ixx}{Iyy}\right\rvert$	~0.98	Strong tendency to inertia couple
	$C_L = \left(\frac{W}{S_{ref}}\right)/q$	~0.48	The lift coefficient for an estimated 3-*g* 70º bank turn is sufficient for sustained speed and altitude, and is far from buffet/stall limitations
	α (deg)	~7.5	Estimated from FIGURE 27a, flown at flaps down landing condition
	$dC_m/d\alpha$ (1/deg)	-0.027	Estimated from FIGURE 27c
	dC_m/dC_L	-0.32	F-14A is ~32% stable; estimated from FIGURE 27c
	L/D	~4.5	Estimated from FIGURE 19b; typical for a transonic maneuver
SHORT-PERIOD	n/α (g/rad)	~31	Short-period flying qualities are firmly within Level I boundaries set by MIL 8785B for "Category A" flight phases (i.e., combat maneuvering)
	ω_{sp} (rad/s)	~4.2	
	$CAP = \frac{\omega_{sp}^2}{n/\alpha}$	0.56	
	ζ_{sp}	0.25	Damping ratio is Level II following "Category A" requirements from MIL 8785B
	$\zeta_{sp_{(synthetic)}}$	~0.80	The F-14A's basic pitch damper enhanced short-period damping; firmly in Level I with expected synthetic damping
DUTCH ROLL	$C_{n_{\beta,dynamic}}$	~0.0031	Dutch roll stability is present and stable; both $dC_l/d\beta$ and $dC_n/d\beta$ are stable
	ω_{dr} (rad/s)	~2.8	Well within Level I region set by MIL 8785B; comfortable frequency
	$dC_Y/d\beta$	-0.015	Side force coefficient is reasonable
	ζ_{dr}	0.07	Level II; insufficient damping for precise control, but a manageable workload
SPIRAL/ ROLL	$s \approx \frac{L_\beta N_r - N_\beta L_r}{L_\beta + N_\beta\left(\frac{Ixz}{Ixx}\right)}$ (1/s)	-0.27	Spiral mode is beyond the Level III minimum requirement
	$\tau_r \approx -\frac{1}{L_p}$ (s)	0.63	Roll mode time constant is excellent for roll sensitivity; well within Level I requirements

The basic pitch damper on the F-14A was scheduled to command 1º collective taileron at a 3º/sec pitch rate; q=0.052 rad/sec; $dCm/delev$ ~ -0.022. Thus, $C_{mq_{syntheti}} = \frac{2\,VKTAS\left(\frac{6076}{3600}\right)dC_m/delev}{q\,c}$=-77; adding this to the bare airframe damping results in $Cm_{qaugmented}$ ~94 which increases Mq to ~-5.8. Thus, with the pitch da mper engaged, we estimate $\zeta_{sp} \approx$ ~0.8 which is LEVEL 1 under combat conditions; see TABLE 3.

We estimate $\left|\frac{\Phi}{\beta}\right| \approx \frac{\frac{dCl}{d\beta}}{\frac{dCn}{d\beta}} \frac{Izz}{Ixx} \frac{1}{\rho\,VKTAS} \approx 3$; the Dutch Roll will appear as a wing rock with some tail wag.

At combat speeds, the Short Period and Dutch Roll Frequencies do not align. However, the aircraft may still inertia couple due to pilot roll inputs. Under these conditions, we estimate that critical roll rate to trigger inertia coupling is controlled by the roll rate interacting with the yaw mode, $p_{crit} \approx$ 200º/sec; an excellent limiting rate especially in light of the weak bare-airframe Dutch Roll damping.

The spiral mode presents an issue with pilot workload and controllability if it is unstable AND if general flying qualities follow suit such that a poor spiral mode pushes pilot workload to extreme levels. The F-14A's inherent aerodynamics make for a spiral mode during a 3-g, 70º bank turn that exhibits a strong tendency to increase the bank angle beyond controllable levels. However, this would have largely been alleviated with precise aileron and rudder control. The quick roll mode supports flight path control immensely.

Furthermore, the F-14A analog roll damper gains were not scheduled with sweep or sweep.[65] Under 3-g maneuvering conditions, the combination of differential taileron and asymmetric spoiler continues to demonstrate strong proverse yaw; roughly 50% unintended yawing moment per unit rolling moment. Thus, we see a more pronounced version of what we observed at landing:

$$Clp_{synthetic} \approx 2\,VKTAS\left(\frac{6076}{3600}\right)\left(\frac{\delta Cl_{aileron}}{p\,b}\right) \approx -0.6;$$
$$Cnp_{synthetic} \approx 2\,VKTAS\left(\frac{6076}{3600}\right)\left(\frac{\delta Cn_{aileron}}{p\,b}\right) \approx -0.4.$$

Adding these values to the bare airframe estimates gets us an effective $Clp_{augmented}$ ~ -0.75 and $Cnp_{augmented}$ ~ -0.4.

A yaw damper increases the negative magnitude of Cnr. The basic yaw damper on the F-14A was scheduled to command -19º rudder at a +50º/sec yaw rate; r=0.87 rad/sec. [74] Since +19º rudder produces dCn ~ +0.042, at 500 KTAS :

$$Cnr_{synthetic} \approx 2\,VKTAS\left(\frac{6076}{3600}\right)\left(\frac{\delta Cn_{rudder}}{r\,b}\right) \approx -1.3.$$

Adding this to the bare airframe Cnr, results in an effective $Cnr_{augmented}$ ~ -1.41 which increases the damping to ζ_{dr} = 0.56 and $\omega_{dr}\,\zeta_{dr}$ = 1.6 which are LEVEL 1 compliant for combat situations; see FIGURE 28 to compare unaugmented to augmented responses.

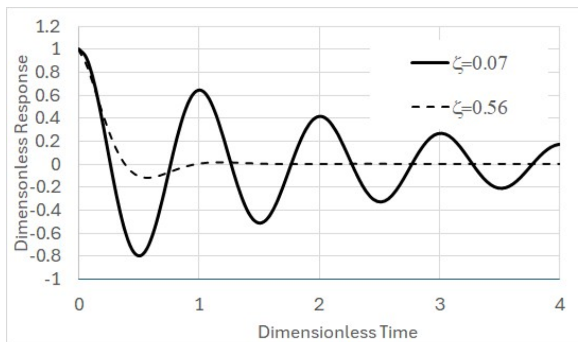


FIGURE 28 – Dutch Roll Damping response to a step input. a) ζ=0.07 without yaw damper, b) ζ=0.56 with yaw damper.

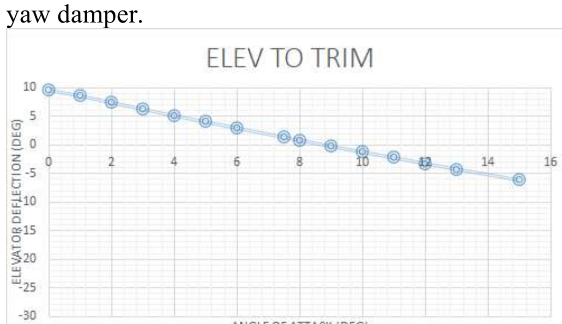


FIGURE 29 – Quantity of Control Power – F-14A High-Subsonic Configuration. Collective taileron (i.e. elevator) needed to trim.

FIGURE 29 shows that there is ample longitudinal control power to trim through α~15º; only ¼ the total deflection is needed. Thus, full back stick can command very high angles of attack far in excess of 20º. Kress confirms the broad trends of the VORLAX data: "with full leading edge down stabilizer one can fly subsonically steady-state at angles of attack between 40º and 45º … of course, in this situation the wing outer panel is completely stalled."[15]

The quality of the available lateral/directional control power at low speeds may be assessed first by considering the amount of rudder needed to trim out the unintended yawing moment of the lateral control effectors; see FIGURE 26. Recall from FIGURES 25b and 25d that for $\alpha<12^o$ the rolling moment (Cl) and yawing moment (Cn) from differential taileron have the same sign, the yaw-due-to-roll is proverse. The amount of rudder needed to trim out the unintended yawing moments are reasonable; no more than 15% available rudder control power is needed to counteract the unintended roll of full differential taileron or full spoileron; see FIGURE 30a.

The F-14A may exceed MIL 8785B tolerable roll induced sideslip; a LEVEL 1 aircraft cannot exceed 6° adverse sideslip or 2° of proverse sideslip from a maximum uncoordinated lateral stick input. Turning to FIGURE 30b, we estimate that the F-14A has undesirable proverse yaw-due-to-roll which achieves a moment balance $\frac{dCn}{d\beta}\ \beta \equiv -\frac{dCn}{dail}\ \delta ail_{max}$ where β=~3.5° proverse.

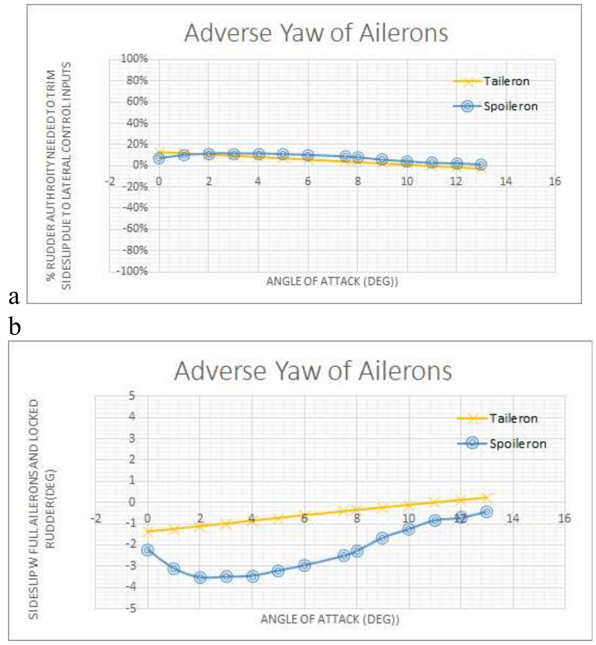


FIGURE 30 – Quantity of Control Power – F-14A High-Subsonic Configuration. a) % rudder authority needed to trim out unintended yaw due to maximum taileron (+/-7°) or spoileron, b) Sideslip to balance unintended yaw due to maximum taileron or spoileron.

VII. High-Subsonic Flying Qualities Issues Noted in Operational Service Led to the Identification of New Handling Qualities Metrics

As shown above, a carefully reverse-engineered aerodynamic model of the F-14A predicts good flying qualities provided the pitch and Dutch Roll damping augmentation systems function as expected. Once again, service experience demonstrates that the tracked qualities are necessary but not sufficient to categorize an aircraft.

The NATOPS admits:

> "The F−14, like all high−performance aircraft capable of producing high−rate, multiple−axis motion, is susceptible to coupling. High−rate, multiple−axis motions, particularly at high AOA, can produce violent coupled departures. In [F-14A experience] a guns−defense/collision−avoidance maneuver using full rudder followed by full coordinated lateral and aft stick produced violent coupled departures with up to 66° per second yaw rate in less than 2 seconds."[72]

The Navy wrote that the F-14A has some flying qualities deficiencies especially in:

> "Maneuvering flight [at transonic speeds] requires the pilot to avoid large lateral stick or rudder inputs since they may result in violent departures from controlled flight. If the departure inducing controls are held in long enough the aircraft will progress towards a stabilized flat spin with angle of attack constant around 80-85°, yaw rate of approximately 180°/sec, and pitch and roll rates essentially zero. Under these conditions, the pilot is experiencing positive 6-g's in the X-body axis direction (eyeballs out) and is virtually incapacitated."[80]

Honig from Grumman notes that departures were encountered during initial flight test and that they "occurred at high subsonic Mach numbers (M>=0.8) in response to moderate lateral stick inputs."[69] Loss of control was the result of adverse yaw characteristics of the differential [taileron coupling] with a relatively low level of static directional stability and a high level of aircraft dihedral effect."[69]

Let us begin by revisiting the dynamic adverse yaw problem noted at approach speeds; do the numbers show that this is a non-problem in basic, non high α, combat?

Let us consider two roll rates attainable at altitude: first a gentle roll rate of p=30°/sec (0.52 rad/sec), the second an aggressive roll rate p=100°/sec (1.74 rad/sec). At these roll rates at 15,000-ft and 530 KTAS, the adverse yaw-due-to-roll arising from *Cnp* with the roll damper engaged is $\delta C_n = C_{np}\left(\frac{p\,b}{2\,VKTAS(\frac{6076}{3600})}\right)$ ≈ -0.0075 at 30°/sec and -0.025 at 100°/sec. At the high roll rates, the dynamic adverse yaw will be balanced by the static directional stability ($dCn/d\beta$~+0.0023) only when $\beta>11°$. Thus, we see how aggressive rolling at combat speeds can invoke a high sideslip departure and eventual spin.

Developmental experience with new, high-performance aircraft across NATO led to a bonanza of scholarly work being commissioned and published. In 1975, AGARD coordinated the first of many sessions where industrial and government engineers tried to devise better analytical and experimental design methods and criteria to understand departure and spin resistance.[78]

The USAF summed it up well:

> "Stall/spin problems, as old as the history of airplane flight, still plague aircraft designers.
> …
> Simple prediction methods are not always enough, and data inadequacies impair the usefulness of more sophisticated analysis.
> …
> The military specification for flying qualities defines good high-α characteristics in terms that are qualitative rather than quantitative.
> …
> We shall need quantitative requirements that will be of more use in the design stage for all classes of airplanes.
> …
> Initial F-14 and F-15 flight results show that substantial improvement can be secured by concentrated design attention. … [But] there remains the need to establish definitive requirements and develop a greatly expanded basis for aerodynamic and flight control design. To that end, improved design methods and criteria are being sought for the high-a characteristics of present and future aircraft."[79]

As seen above lateral-directional control-coupling occurs when commanded lateral rolling moments ($dCl/dail$) produce unintentional yawing moments either directly ($dCn/dail$) or indirectly (Cnp). If the innate static dihedral effect ($dCl/d\beta < 0$) is strong enough at the developed sideslip angle ($\beta > 0$) due to unintended yawing moments, it may overwhelm the static directional stability of the airframe and lead to a spin.

Willam Bihrle had moved from Republic to Grumman for development of the F-14.[83] He identified the following observations regarding the loss of lateral/directional control of fighter aircraft:

- That the unintended yawing moment due to a roll (i.e., aileron input) has "an appreciable influence on the spin."[83] The worse the adverse yaw ($dCn_{ail}/dCl_{ail} < 0$), the more rapidly the aircraft will attain a flat spin. The F-14A does not suffer from such issues at moderate angles of attack, however the roll control surfaces do become adverse when $\alpha>20°$.[cite Bihrle high alpha]

- The stronger the dihedral effect ($-dCl/d\beta$), the longer the vehicle can maintain a flat spin. The F-14A does have a fairly strong dihedral effect with wings swept; see FIGURE 24.

- As the aircraft becomes more body heavy ($Ixx << Iyy$) the aircraft becomes more likely to enter a steep spin than a flat spin. For the F-14A, *Ixx* is roughly ¼ the magnitude of *Iyy*; see TABLE 1.

- The roll damping derivative (*Clp*) impacts the tendency to remain in a flat spin. As *Clp* decreases, a flat spin will transition to a steep, oscillatory spin; this is considered easier to recover from. The F-14A bare airframe had insufficient *Clp* which led to the inclusion of an active roll damper stability augmentation system.

The USAF at Wright Laboratories leveraged earlier work by Weissman.[84][85] For several years he had worked "to establish a correlation between lateral-directional static stability and spin susceptibility, preferably in the form of criteria utilizing static aerodynamic data, since [such] data [can be] obtained early during the initial design stages."[84] His hope was that "if spin susceptibility [was] investigated early in the design and development stage, design changes [could] be made which will produce satisfactory … lateral-directional stability characteristics and thus minimize spin susceptibility" in the finished aircraft.[84]

Woodcock & Weissman spent considerable flight simulator time with F-4 and T-38 models working to "verify or extend existing stability criteria, and possibly develop new criteria which can be used with a reasonable degree of confidence for a first-cut prediction of full-scale aircraft departure characteristics and spin susceptibility as early as possible in the design stage."[79] They identified a total of five parameters as being important:

1. The Dutch Roll stability coefficient: *Cnβdynamic* > 0
2. The Lateral Control Departure Parameter, *LCDP*, unaugmented: $LCDP = C_{n_\beta} - C_{l_\beta}\frac{(\frac{dC_n}{dail})}{(\frac{dC_l}{dail})}$
3. The Lateral Control Departure Parameter, *LCDP*, including aileron-rudder interconnect:
   $LCDP = C_{n_\beta} - C_{l_\beta}\frac{(\frac{dC_n}{dail}+\mathrm{K_{ARI}}\frac{dC_n}{drud})}{(\frac{dC_l}{dail}+\mathrm{K_{ARI}}\frac{dC_n}{drud})}$

   If $LCDP > 0$, adverse yaw-due-to-roll is unlikely to lead to dynamic instability.
   If $LCDP < 0$, roll command inputs are likely to lead to a spin
4. The β axis stability indicator: $\alpha_{-\beta} = \alpha - \arctan\left(\frac{C_{n_\beta}}{C_{l_\beta}}\frac{Ixx}{Izz}\right)$
5. The δ axis stability indicator: $\alpha_{\delta} = \alpha - \arctan\left(\frac{\frac{dC_n}{dail}}{\frac{dC_l}{dail}}\frac{Ixx}{Izz}\right)$

   Where $\alpha_{-\beta} > \alpha_\delta$ and $\alpha_{-\beta} > 0$ to ensure spin resistance.

Weissman concluded that **Cnβdynamic** and aileron-alone, i.*e., unaugmented, LCDP metrics were the most powerful indicators to predict spin susceptibility. But noted, that while promising, were "not conclusive proof of the criteria."[84]*

For the F-14A, we see that it satisfies the β and δ axis indicators: $\alpha_{-\beta} = \alpha - \arctan\left(\frac{C_{n_\beta}}{C_{l_\beta}}\frac{Ixx}{Izz}\right)$ = ~+42 for1-g, ~+32 for 3-g and $\alpha_{\delta} = \alpha - \arctan\left(\frac{\frac{dC_n}{dail}}{\frac{dC_l}{dail}}\frac{Ixx}{Izz}\right)$ = ~+0.6 for 1-g, ~+6.4 for 3-g.

Weissman's work, in conjunction with Bihrle's earlier observations, led to the development of the Bihrle-Weissman plot.[39] While the Bihrle-Weissman criteria are neither formally part of MIL 8785C nor of MIL 1797, they have been used in the flight dynamics community for many years to assess the departure risk inherent to an aerodynamic design.

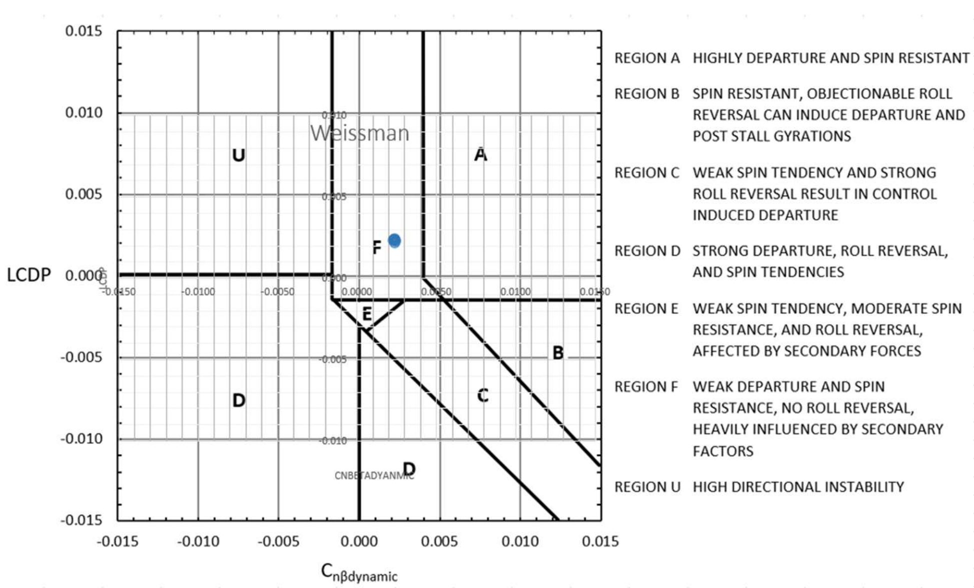


FIGURE 31 - F-14A Bihrle-Weissman Chart – Mach 0.8 0º<α<16º

For the F-14A, lacking aileron-rudder-interconnect, we must use the bare airframe roll and yaw response to feed these equations. We see over a range of angles of attack that this aircraft falls into the "F" region – one with weak departure resistance that is "heavily influenced by secondary factors;" see FIGURE 31.

Today, MIL STD 1797A represents the latest publicly accessible compendium of experience relating to aircraft Flying Qualities and Handling Qualities.[5] MIL 1797 requires that an aircraft be designed not to exceed a critical sideslip angle in response to step roll control inputs without active rudder coordination; see TABLE 4. These values are not appreciably more stringent than those specified in MIL 8785B. Its goal is that the flight control system makes it easy to coordinate flight with "minimum yaw coupling in roll entries and exits."[5] The "the real objective is to keep sideslip near zero so that the aircraft tracks bank angle … [with a minimum] yaw control crossfeed necessary to maintain zero sideslip."[5]

TABLE 4 – MIL-1797A maximum sideslip developed due to uncoordinated "large" lateral stick inputs.[5]

LEVEL	FLIGHT PHASE CATEGORY	ADVERSE SIDESLIP (RIGHT ROLL COMMAND CAUSES RIGHT SIDESLIP)	PROVERSE SIDESLIP (LEFT ROLL COMMAND CAUSES RIGHT SIDESLIP)
1	A	6 degrees	2 degrees
	B and C	10 degrees	3 degrees
2	All	15 degrees	4 degrees

After all of our retrospective analysis, we conclude that the obvious candidate for F-14A lateral/directional departures at moderate angles of attack arise from a byproduct of its simple roll damper system, which forces the airframe to develop high adverse sideslip to roll as a byproduct of its Dutch Roll suppression characteristics.

VIII. Making the F-14 More "Spin Resistant" Without Enlarging Its Tails

In the early 1970's, Grumman promised the USN that "spin prevention is an attainable goal."[7] To solve this problem, Grumman designed a "aileron-rudder-interconnect which limited adverse yaw characteristics during rolls by phasing out the differential [taileron] … and simultaneously phasing in the rudder for roll control."[69] The flight test program spanned "one and one-half years" 71 flights and over 1200 maneuvers beyond the nominal stall angle of attack. Grumman hoped to deliver the first Navy fighter with an unrestricted angle of attack capability by revising the flight control laws to include aileron-to-rudder interconnects.[69] According to Renfro at the USN, "**Due to funding constraints, [revisions to] control laws were never incorporated into the analog flight control system."**[80]

Retrofits and new-build F-14's with a more complex flight control system did not reach the fleet until the 1990s. With the advent of the F-14D "Super Tomcat," and a "digital flight control stability augmentation system" the F-14 fight control laws finally included ARI. Both ARI and roll damping control laws are scheduled with speed, wing sweep and angle-of-attack. "At low speeds, clean, [DFCS] introduced a "wing rock suppression" function (*Clr* enhancement) above $\alpha>\sim15^o$. [72] The DFCS "gradually applies coordinating rudder with lateral stick" to provide the pilot with "the desired roll response with lateral stick input alone at elevated AOA."[72] It also includes roll rate inputs into the yaw (i.e. rudder) channel and yaw rate inputs into the roll SAS channel.[74] Together, the DFCS "reduces the effects of kinematic coupling and results in less adverse sideslip with lateral stick deflection … and reduces the tendency for lateral control induced departures."[72]

Even with the DFCS, the NATOPS recommends that pilots keep $\alpha<28^o$ as a hard limit at low speeds. At combat speeds, with the wings swept, it clearly calls out that "flight operations [are] prohibited] above $\alpha>20^o$. Similarly, "high AOA handling qualities are significantly improved by keeping ROLL SAS on. But in all cases *restrict rolls to less than full lateral stick deflection and to not more than 180º of bank angle change at one time.*" [72]

The F-14D also has revised F110 engines which are much less likely to exhibit compressor stalls at high sideslip angles. If an engine does fail, the F-14D has an "Automatic Thrust Limiting System (ATLS) exists to reduce the power on the remaining engine in the event of a single engine failure." [72] This scheduling allows sufficient aerodynamic control to fly under asymmetric thrust conditions. However, the NATOPS clearly states the "rudder is the primary control to offset yawing moment from asymmetric thrust. … [and that] t***he use of lateral stick to offset the uncommanded roll caused by yaw from asymmetric thrust at high AOA will generate adverse yaw and aggravate the yaw caused by asymmetric thrust. The result may be a yawing, rolling departure."*** **[72]**

IX. Summary & Conclusions

To conclude, the existing military design standards in place at the beginning of the F-14 program (MIL 8785 from 1954) were inadequate. The revisions incorporated into MIL 8785B were a good step in the right direction, but it does not appear that the F-14A could fully comply with those standards with its original flight control system.

The fact that spin departure design criteria evolved greatly after the 1968 F-14 contract award were an indirect result of the production F-14 (among other aircraft) experiencing controllability issues during operational test and fleet use.

The F-14's vertical tails were as big as practicable, but not big enough. We can see from our reverse engineering that it needed much more innate yaw damping.

The maximum sideslip attained due to uncoordinated lateral stick inputs would be compliant with 8785B and later standards only if the static directional stability were stronger (the tails were larger).

Our most interesting discovery made here is that the F-14A spin tendency at moderate angles of attack was due to unintended interactions between the bare airframe aerodynamics and the "delivered to the fleet" analog flight control system control laws.

We note that revised control laws were developed during the early 1970s but not installed in new build or retrofit to existing F-14A in any timely manner. Evolved versions of these control laws were finally implemented with the F-14D DFCS upgrade many years after Top Gun was filmed. Thus, we see that had the USN retrofit the F-14A with a more sophisticated flight control system (aileron-rudder interconnect on both lateral-stick and on the Dutch Roll damping system), its spin resistance at moderate angles of attack would have been greatly improved.

Acknowledgments

The preparation of the work was supported through several funding mechanisms. Professor Takahashi was Visiting Faculty U.S. Air Force Institute of Technology under the SysPlus, Inc. managed Air Force Summer Faculty Fellowship program. Both Professor Takahashi and Mr. Lorenzo were also supported under the Postgraduate Research Participation Program at the U.S. Air Force Institute of Technology, administered by the Oak Ridge Institute for Science and Education through an interagency agreement between the U.S. Department of Energy and AFIT. Professor Takahashi would also like to thank his AFIT collaborators Prof. Ramana V. Grandhi and Prof. José Camberos for inspiration and encouragement to undertake this study. He would also like to thank the staff at the D'Azzo Research Library at the AFIT for their help and access to their large collection of old NACA and NASA reports.

References

[1] https://www.imdb.com/title/tt0092099/?ref_=fn_al_tt_1
[2] Yonay, Ehud. "Top Guns". California Magazine. Beverly Hills, CA, May 1983, p. 94. Retrieved June 8, 2022.
[3] Eastwood, B.M., "The F-14 Tomcat had 1 flaw that the U.S. Navy Had to Fix," https://nationalinterest.org/blog/buzz/f-14-tomcat-had-1-flaw-us-navy-had-fix-207849. Retrieved October 1, 2024.
[4] Leone, D., "US Navy Test Pilot tells the story of the F-14 Tomcat Fatal Flat Spin Mode That Was Never Supposed to Happen," Aviation Geek Club.com, Mar. 5, 2023.
[5] https://www.legacy.com/us/obituaries/atlanta/name/william-evans-obituary?id=29897222. Retrieved October 1, 2024.
[6] Eckert, W.T. and Maki, R.L., "Low-Speed Wind Tunnel Investigation of the Lateral-Directional Characteristics of a Large-Scale Variable Wing-Sweep Fighter Model in the High-Lift Configuration," NASA TM X-62,306, October 1973.
[7] Sewell, C.A. and Whipple, R.D., "F-14A Stall and Spin Prevention System Flight Tests," AGARD CP-199, 1976.
[8] Philips, W.H., "Flying Qualities for Early Airplanes to the Space Shuttle," *J. Guidance*, Vol. 12, No. 4, 1989, pp. 449-459.
[9] Miller, J., *Grumman F-14A/B*, Aerofax Minigraph 3, Aerofax, Arlington, TX, 1984.
[10] Cornell, D. "Grumman Prepares the F-14A for its first flight," Interavia, 1970, p. 1384-1387.
[11] https://www.thisdayinaviation.com/tag/mcdonnell-f-110a-spectre/ [retrieved Oct. 21, 2024]

[12] McGovern, T., *McDonnell F-4E Phantom II,* Aerofax. Arlington, TX, 1987.
[13] Miller, J., *General Dynamics F-111 "Aardvark,"* Aero Publishers, Fallbrook, CA, 1982. ISBN 0-8168-0606-3.
[14] Taylor, W.R., *Jane's All the World Aircraft: 1973-4*, Jane's Yearbooks, London, 1974.
[15] Kress, R.W., "Variable Sweep Wing Design," AIAA 80-3043, 1980.
[16] Droste, C.S., "The F-111 Adaptive Flight Control System," Automatic Control and Guidance Systems Committee Meeting 110, October 11, 2012.
[17] https://www.nationalmuseum.af.mil/Visit/Museum-Exhibits/Fact-Sheets/Display/Article/196049/ Retrieved October 1, 2024.
[18] McIver, J.E. "The F-14 Contract: A Case Study in Major Weapons Systems Acquisition and Program Management," Naval Postgraduate School Thesis, June 1993. Summarizing Stevenson, J.P. "Tomcat" *A Grumman Publication*, Aero Series 25.
[19] https://news.vt.edu/articles/2019/04/news-williammason.html Retrieved October 1, 2024.
[20] https://www.chicagotribune.com/2002/10/31/michael-pelehach-80/ Retrieved October 1, 2024.
[21] https://www.legacy.com/obituaries/name/nathan-kirschbaum-obituary?pid=18111243 Retrieved October 1, 2024.
[22] https://www.cookfamilyfuneralhome.com/obituaries/rudolph-c-meyer Retrieved October 1, 2024.
[23] Personal Notes of Professor WH Mason, Nov. 2011.
[24] https://theaviationist.com/2013/10/04/last-tomcat-flght/ Retrieved October 1, 2024.
[25] Anon., MIL C-5011A, "Military Specification – Charts; Standard Aircraft Characteristics and Performance, Piloted Aircraft," 5 Nov 1951.
[26] Anon., MIL-F-8785B, "Military Specification – Flying Qualities of Piloted Airplanes," 7 Aug. 1969.
[27] Anon. MIL-F-8785(ASG), "Military Specification – Flying Qualities of Piloted Airplanes," 17 Apr. 1959.
[28] Chalk, C.R., and Wilson, R.K., "Airplane Flying Qualities Specification Revisions," *J. Aircraft*, Vol. 6, No. 3, May-June 1969, p. 232-239.
[29] Chalk, C.R., and Wilson, R.K., "Airplane Flying Qualities Specification Revision," AIAA 68-2456, 1968.
[30] Cooper, G. E. and Harper, R. P., "The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities," AGARD REPORT 567, April 1969.
[31] Harper, R.P. and Cooper, G. E., "A Revised Pilot Rating Scale for the Evaluation of Handling Qualities," AGARD CP 17, 1966.
[32] Bihrle, W., Jr., "A Handling Qualities Theory for Precise Flight Path Control," AFFDL-TR-65-198, 1966.
[33] Chalk, C.R, Neal, T.P., Harris, T.M., Pritchard, L.L. and Woodcock, R.J., "Background Information and User Guide for MIL-F-8785B(ASG) "Military Specification – Flying Qualities of Piloted Airplanes," AFFDL TR 69-72, 1969.
[34] https://bihrle.com/william-bihrle-jr/ (obituary)
[35] Ashkenas, I.L. "Twenty-Five Years of Handling Qualities Research," J Aircraft, Vol 21, No. 5, 1984, pp. 289-301.
[36] Roskam, J., *Airplane Flight Dynamics and Automatic Flight Controls, Parts I & II.*, DAR Corporation, 1995.
[37] Yechout, T.R., *Introduction to Aircraft Flight Mechanics*, Second Edition, AIAA, 2014
[38] Takahashi, T.T., *Aircraft Performance & Sizing, Vol. I & II*, Momentum Press, New York, NY, 2016.
[39] Takahashi, T.T., Griffin, J.A., and Grandhi, R.V., "High-Speed Aircraft Stability and Control Metrics," MDPI Aerospace, Vol. 12, 2025. https://doi.org/10.3390/aerospace12010012
[40] Day, R.E., "Coupling Dynamics in Aircraft Design: A Historical Perspective," NASA SP-532, 1997.
[41] Anon., "Flight Experience with Two High-Speed Airplanes Having Violent Lateral-Longitudinal Coupling in Aileron Rolls," NACA RM H55A13, 1955.
[42] Takahashi, T.T., "Ezra Kotcher: the Father of the Bell X-1 and X-2," RaeS Journal of Aerospace History, 2024/4, Oct. 2024.
[43] Weil, J., and Day, R.E., "An Analog Study of the Relative Importance of Various Factors Affecting Roll Coupling," NACA RM H56A06, 1956.
[44] Spearman, M.L. and Foster, G.V., "Stability and Control Characteristics at a Mach Number of 2.01 of a Variable-Wing-Sweep Configuration with Outboard Wing Panels Swept Back 75°," NASA TM X-32, 1959.
[45] Alford, W.J., Jr. and Henderson, W.P., "An Exploratory Investigation of the Low-Speed Aerodynamic Characteristics of Variable-Wing-Sweep Airplane Configurations," NASA TM X-142, 1959.
[46] Alford, W.J., Jr., Luoma, A.A. and Henderson, W.P., "Wind Tunnel Studies at Subsonic and Transonic Speeds of a Multiple-Mission Variable-Wing-Sweep Airplane Configuration," NASA TM X-206, 1959.
[47] Spencer, B., Jr., "Stability and Control Characteristics at Low Subsonic Speeds of an Airplane Configuration Having Two Types of Variable Sweep Wings," NASA TM X-303, 1960.
[48] Bielat, R.P., Warner Robins, A., and Alford, W.J., Jr., "The Transonic Aerodynamic Characteristics of Two Variable Sweep Configurations Capable of Low-Level Supersonic Attack," NASA TM X-304, 1960.
[49] Foster, G.V. and Morris, O.A., "Stability and Control Characteristics at a Mach Number of 1.97 of an Airplane Configuration Having Two Types of Variable Sweep Wings," NASA TM X-343, 1960.
[50] Foster, G.V, "Effects of Spoiler-Slot-Deflector Control on the Aerodynamic Characteristics at the Mach Number of 2.01 of a Variable-Wing-Sweep Configuration with the Outer Wing Panels Swept Back 75°," NASA TM X-273, 1960.
[51] Henderson, W.P. and Hammond, A.D., "Lateral Control Devises on a Semispan Variable-Sweep Wing Having an Outboard Pivot Location," NASA TM X-542, 1961.
[52] Polhamus, E.C. and Hammond, A.D., "Aerodynamic Research Relative to Variable-Sweep Multimission Aircraft," NASA TM X-420, 1960.

[53] Spearman, M.L. “Longitudinal and Lateral Aerodynamic Characteristics at Mach Numbers from 0.60 to 2.20 of a Variable-Sweep Fighter Model with Wing Sweep Angles from 25° to 75°,” NASA TM X-710, 1962.
[54] Spearman, M.L. “Longitudinal and Lateral Aerodynamic Characteristics at Mach Numbers from 1.41 and 2.20 of a Variable-Sweep Fighter Model with Wing Sweep Angles from 25° to 75°,” NASA TM X-759, 1963.
[55] Bielat, R.P. “Longitudinal and Lateral Aerodynamic Characteristics at Mach Numbers from 0.60 to 2.20 of a Variable-Sweep Fighter Model with Wing Sweep Angles from 25° to 75°,” NASA TM X-886, 1963.
[56] Louma, A.A., “Transonic Longitudinal Aerodynamic Characteristics of an STOL fighter-bomber model with a Variable Sweep Wing,” NASA TM X-994, 1964.
[57] Hassell, J.L., Jr., “Low-Speed Flight Characteristics of a Variable-Wing-Sweep Fighter Model,” NASA TM X-1036, 1965.
[58] Ayers, T.G., “Transonic Aerodynamic Characteristics of a Variable-Wing-Sweep Tactical Fighter Model – Phase 1,” NASA TM X-1039, 1964.
[59] Shaw, D.S., and Campbell, J.F., “Supersonic Investigation of the Static Stability, Performance, and Control of a Variable-Sweep Tactical Fighter Model – Phase 3," NASA TM X-1142, 1965.
[60] Ayers, T.G., “Transonic Aerodynamic Characteristics of a Variable-Wing-Sweep Tactical Fighter Model – Phase 2,” NASA TM X-1040, 1964.
[61] Driver, C., and Spearman, M.L., “Aerodynamic Characteristics at a Mach Number of 2.20 of a Variable-Sweep Fighter Model with Wing Sweeps of 50°, 60° and 75°,” NASA TM X-1041, 1964.
[62] Shaw, D.S., “Supersonic Investigation of the Static Stability, Performance, and Control of a Variable-Sweep Tactical Fighter Model – Phase 1," NASA TM X-1045, 1965.
[63] Anon., “Configurations of the F-111A, B Unveiled,” Aviation Week, Vol. 82, No. 22, June 1, 1964, p. 21-22.
[64] Shaw, D.S. and Wassum, D.L., “Supersonic Investigation of the Static Stability, Performance, and Control of a Variable-Sweep Tactical Fighter Model – Phase 2," NASA TM X-1046, 1965.
[65] Shaw, D.S. and Babb, C.D., “Supersonic Investigation of the Static Stability, Performance, and Control of a Variable-Sweep Tactical Fighter Model – Phase 4," NASA TM X-1143, 1965.
[66] Ferris, J., “Chronology of Outstanding Events for 1968,” CVA43:17, United States Navy, March 4, 1969. (see: https://www.history.navy.mil/content/dam/nhhc/research/archives/command-operation-reports/vietnam/Coral%20Sea%201968.pdf)
[67] Preliminary NATOPS Flight Manual Navy Model F-111B Aircraft, NAVAIR 01-10FAB-1, 15 March 1968.
[68] “U.S. General Accounting Office Staff Study on the F-14/Phoenix Aircraft Program Department of Navy,” April 1973.
[69] Honig, S.B. and DeLuca, J., “Wind Tunnel and Flight Development of the F-14 Subsonic/Transonic Maneuvering Configuration,” AIAA 74-618, 1974.
[70] Bihrle, W. Jr. and Meyer, R.C., “F-14A High-Angle-of-Attack Characteristics,” AIAA J. Aircraft Vol. 13, No. 8, pp. 576-583, 1976.
[71] https://www.armouredcarriers.com/were-the-armoured-carriers-worthwhile (accessed December 27, 2024)
[72] NATOPS Flight Manual Navy Model F-14D Aircraft, 2002.
[73] Eckert, W.T. and Maki, R.L., “Low Speed Wind Tunnel Investigation of the Longitudinal Characteristics of a Large-Scale Variable Wing Sweep Fighter Model in the High-Lift Configuration,” NASA TM X-62244, 1973.
[74] Kelley, W.W. and Enevoldson, E.K., “Limited Evaluation of an F-14A Airplane Utilizing an Aileron Rudder Interconnect Control System in the Landing Configuration,” NASA TM-81972, 1981.
[75] Takahashi, T.T., Griffin, J.A., Gaydusek, B.S. and Lorenzo, W.P., “VORLAX 2024: Further Upgrades to a Legacy Potential Flow Solver,” AIAA 2025-0848, 2025.
[76] Miranda, L. R., Elliot, R. D., and Baker, W. M., “A Generalized Vortex Lattice Method for Subsonic and Supersonic Flow Applications.” NASA CR 2865, 1977.
[77] Souders, T. J., & Takahashi, T. T., “VORLAX 2020: Making a Potential Flow Solver Great Again.” AIAA 2021-2458, 2021.
[78] Titiriga, A., Jr, Ackerman, J.S. and Skow, A.M., “Design Technology for Departure Resistance of Fighter Aircraft,” AGARD CP-199, 1975.
[79] Woodcock, R.J. and Weissman, R., “The Stall/Spin Problem,” AGARD CP-199, 1976.
[80] Renfrow, J., Liebler, S., Denham, J., “F-14 Flight Control Law Design, Verification and Validation Using Computer Aided Engineering Tools,” US Navy, 1995.
[81] anonymous retired USN F-14 pilot interview w/ Prof. Takahashi
[82] MIL-F-8785C, Military Specification: Flying Qualities of Piloted Airplanes, 1980
[83] Bihrle, W., Jr. “Influence of the Static and Dynamic Aerodynamic Characteristics on the Spinning Motion of Aircraft,” J Aircraft, Vol. 8, No. 10, Oct. 1971, pp. 764-768.
[84] Weissman, R. “Criteria for Predicting Spin Susceptibility of Fighter-Type Aircraft,” ASD-TR-72-48, 1972.
[85] Weissman, R., “Development of Design Criteria for Predicting Departure Characteristics and Spin Susceptibility of Fighter-Type Aircraft,” AIAA 72-984, 1972.