EAGLE TALK

not for public release

P.S. 1257
1 MARCH 1985
VOLUME II

MCDONNELL AIRCRAFT COMPANY

MCDONNELL DOUGLAS

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A Crewman's History of the McDonnell...

F-15 EAGLE

Volume II
(Technical)

(REPRINTS FROM MCAIR PRODUCT SUPPORT DIGEST)

MCDONNELL DOUGLAS

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P.S. 1257 - VOLUME II

INTRODUCTION

Welcome to Volume II of F-15 "EAGLE TALK." With this book, MCAIR is continuing its long-standing practice of offering collected copies of articles previously published in our PRODUCT SUPPORT DIGEST. Volume I (published in January 1984) contained aircrew articles of general interest, arranged in a chronological order and covering the period from 1972 to date. Volume II presents the more technical articles from the same period, arranged in a "subject" order, as indicated in the Table of Contents. Some of the articles in this volume were written by other than company test pilots, but all are relevant to "flying the airplane." Also, in addition to the identified technical articles, we have sprinkled some interesting F-15 trivia throughout - like, who were the first 100 Eagle flyers ... who won the 1974 Mackay Trophy and what did they win it for ... what F-15 was the first one to fly 1000 hours ... are the Bitburg Eagles landing or taking off ... ?

As explained with all previous reprint publications of this nature, there has been no attempt made to "up date" any of these articles to reflect the latest systems or possible modifications to the airplane. Each article includes its original publication date - if you read something that sounds obsolete or conflicts with current DASH ONE coverage, the official manuals apply. This book contains good information, but for your information only!

In case you have not seen Volume I of this series, the introduction to it and the table of contents are included on the inside back cover of this volume. Copies of both volumes are available from the MCAIR representative on base, or by direct request to the address printed with the "restriction notices" included herein. Finally, we again apologize for the substandard visual appearance of many of the pages in this book - as with other publications of this type, the original text pages and photographs are no longer available and reproduction from two-color printed copies is necessary.

"Be prepared" - read up on the Eagle. As Pat Henry, MCAIR test pilot and current Director of Flight Operations, notes in an article published back in 1978 and included in this volume of reprints, ...

" ... Until they manage to build the perfect airplane and engine, remember that you don't pick the failures - the emergency finds you. If you take this bird too much for granted, even the world's easiest flying and most forgiving airplane may find you unprepared. ..."

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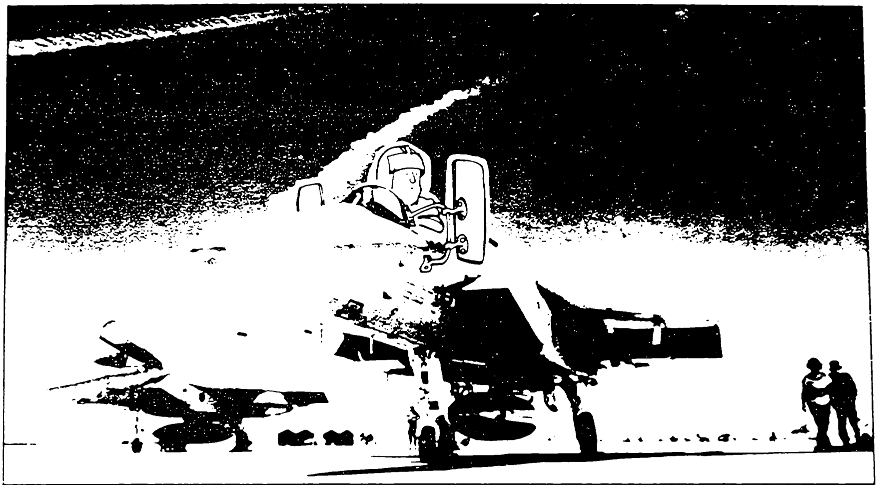
"ALL DONE WITH MIRRORS..." (PUBLISHED 1980)



F-15 PREFLIGHT CONTROL CHECKS

By PAT HENRY/*Chief Experimental Test Pilot*

From time to time the subject of pilot preflight checks comes up, and there seems to be as many techniques as there are contestants. Theoretically, if all goes well on the flight, it was a waste of time anyway, right? Wrong! Depending on the profile actually flown, some control system faults may not even be evident, yet are lurking there ready to bite the unsuspecting pilot who treads too heavily into certain areas of the flight envelope. So what do we recommend for the military pilot?

The *minimum* checks accomplished must be in accordance with established (DASH 1) procedures and local guidance. Beyond that, I can't imagine any heartburn with additional checks, assuming you're just sitting in the chocks and not late for a scheduled mission. To my way of thinking, additional checks are like paid-up insurance - it doesn't cost anything and it's a little hard to get after the terminal illness.

For those of you who like to compare notes and techniques, I'm offering a fairly detailed description of the control checks I do here in St Louis, and what I'm looking for. Granted these checks may be something of an overkill, but as company pilots we're dealing with airplanes fresh off the production line (i.e., unflown), so there's extra incentive to make sure everything plays as advertised. Besides the whole thing only takes three or four minutes, which still leaves me a spare five to run assorted BIT's while waiting for the INS.

The order of accomplishment is really up to you, particularly in the F-15 since it's "all done with mirrors" (i.e., no hand signals required between crew chief and pilot). My recommendation is to develop a set pattern that is logical to you, then any variance from the pattern, either in technique or control response, is immediately obvious. Here's my system (while reading these somewhat lengthy notes, please remember it's a lot quicker for you to do these checks than for me to talk about them).

1. Initial Setup

All CAS on, takeoff trim position, and anti-skid switch on.

2. Stabilator Longitudinal Cycle

Full leading edge down (L.E.D.), and then up (L.E.U.). Stabilators should immediately go to the full authority position, which is easily recognized by the time you've checked out in the aircraft. The inboard leading edge corners of the stabilators should be visible (two to three inches) in the mirrors, just above the wing surface, in the L.E.U. (forward stick) position. This is important because this same visual reference is used in the stick force sensor and PTC (Pitch Trim Compensator) checks a little later on in the procedure.

3. Lateral Authority Check

From takeoff trim position go to full right stick. Verify full R.H. aileron up, and R.H. stabilator matches (large L.E.D. deflection). Rudders will be deflected in the nose right direction. Then, moving the stick full aft while holding full right will do two things - stabilator and rudder deflections should both increase, while aileron remains full up. As you then move the stick full forward and full right, the stabilator (R.H.) and rudder deflections should both reverse while the aileron remains at attention. Repeat the above on the left side.

▶

4. Flaps vs ARI

While holding full left and near neutral longitudinal stick, select flaps down while watching rudder position in the mirror. As the flaps start down, the rudders should move more to the left because of a shift in the ARI schedule. After noting this, there is still time to look inside the cockpit and watch the flap position lights change from amber to green.

5. Roll Ratio Check

Still holding full left stick, select emergency Roll Ratio. Three things should happen: (1) Roll Ratio and Master Caution lights should come on; (2) aileron and stabilator deflections should slowly decrease; and (3) the rudders should immediately go to neutral, verifying that ARI has turned off. When the roll ratio switch is returned to auto, the rudders should return to their original deflected position in less than a second. This is a demonstration of the "Fast ARI Turn-on" portion of the anti-spin control modification. The original, or slow ARI turn-on, takes several seconds.

6. ARI Cutout

While *still* holding full left stick, turn anti-skid off. Verify that this turns off ARI by again watching the rudders go to trail. Check that the anti-skid warning light is on; then if you missed it before you can observe the fast ARI turn-on when the anti-skid switch is returned to normal.

7. Pitch Ratio Check

While *still* holding full left stick, select emergency Pitch Ratio. The ARI should again cut out. Then go to full aft and center stick and watch both stabilators trim out (reduce their L.E.D. deflection) smoothly as the indicated pitch ratio motors down to 0.4. Verify Pitch Ratio and Master Caution warning lights are on.

8. Stick Force Sensor Check

With the Pitch Ratio in the full emergency position, trim the stick full nose up. It takes only a few seconds, and you'll feel the stick bottom out or stop moving when you've reached the limit of trim authority. Now hold the stick full forward and verify you can see the stabilator tips as you did in the initial Stabilator Authority check. (What has been done by this sequence is to put the mechanical input to the stabilators at the nose-up end of the actuator stroke. Then the CAS input from the forward stick force is that extra authority which moves the stabilators the rest of the way and allows the leading edge tips to be seen. Repeat the test without pitch CAS, and you won't see the tips. Or simply by holding the stick full forward by pressing below the control stick grip, the stick force sensor input/output will be zero, and you won't get full stabilator deflection - the CAS authority will be missing).

9. Mechanical Control Check

After turning all CAS off, essentially repeat the control cycle previously done with CAS on. The aft stick (stabilator L.E.D.) should look the same as before. The forward stick check is slightly different. When you first reach the forward stick stop, the stabilator leading edge tips should not be visible above the wing. This is because the PTC is biased to the full nose-up position for maximum stabilator authority during takeoff. As you hold the stick forward, the PTC will run its little legs off in the nose-down direction. It takes several seconds to make the full excursion, at which time you will be able to see the stabilator tips. This check is only possible with pitch CAS off, because the CAS authority masks the movement of the PTC in the full up system.

The Lateral Authority check without CAS is just like the one with CAS on, except that stabilator deflections will be reduced. Remember, roll CAS works through differential stabilator, not the ailerons, so that accounts for your reduced stabilator deflection with no change in ailerons.

10. Rudder Checks

This is one of the simpler steps, yet ironically could be one of the most important due to the higher history of problems with rudders than with other control surfaces. All the CAS is still off from the previous steps, so the first half of this check is with the mechanical system only. Rudder pedal displacement in both directions should produce one half rudder, or 15°. When the pedals are released rapidly from this displacement, there should be no significant residual rudder displacement - they should return rapidly and freely to within 1°-2° of neutral. Next turn all CAS on and repeat. Rudder displacement should now be full or approximately 30°.

11. Other Static Preflight Checks

"Static" as in display - I am still talking about checks done prior to taxi. Many checks are self-explanatory and will not be discussed, such as speed brake, hook, and IFR door cycles. Ditto on other standard preflight checks such as fuel gauge, warning lights, etc.

(a.) Trim Check - Trim the stick and the rudder pedals as far as possible from neutral. You'll know you're there when the stick and rudder pedals stop moving. Holding the T/O trim button, the stick and rudder pedals should return smoothly to neutral in approximately 10 seconds.

(b.) Eddy Current Damper - This little "gee whiz" check can be done with or without electrical and hydraulic power on the bird, since the damping is simply a function of stick movement. The different stick response when displaced and released laterally versus longitudinally is a graphic demonstration of damper effectiveness. To make the comparison, displace the stick laterally, about 1/2 travel or beyond, and release. The stick should return to neutral with little or no overshoot - many will look totally deadbeat, returning smoothly to neutral. Conversely, when the stick is displaced longitudinally and released, it should take two or three rapid cycles to dampen out.

(c.) Emergency Generator - This is not a *control system* check, but since it may not be fully understood or discussed elsewhere, I've included it here. As background info, you should know/remember that with the original design, it was possible to inadvertently leave the emergency generator switch in the ISOLATE position (it was labeled EMERG ON in the early configuration), which meant that when the emergency generator was needed for real, only the emergency essential bus would be powered. With the present improved design, it is possible to step down to this minimum electrical output configuration if desired (e.g., electrical fire), but it has to be a deliberate switch action on your part - not something left over from the previous flight or maintenance.

Here's a quick check for part of that mechanization and system logic. Move the emergency generator switch back swiftly from AUTO to ISOLATE (no pause in MANUAL position), and hold it there. Both the boost system malfunction (BST SYS MAL) and the emergency boost pump output pressure (EMER BST ON) lights should illuminate because you're forcing the system into an illogical mode in that the emergency boost pump is on, but is not receiving its power from the emergency generator. When you release the switch, it should spring back to the MANUAL position, illustrating that you cannot inadvertently leave the switch in ISOLATE. Also, the BOOST SYS MAL light will go out when the switch is released. You can even select ISOLATE from MANUAL and it will remain, showing that you can ISOLATE yourself to the Emergency Essential Bus once the emergency generator is on the line.

12. Taxi Checks

There are two systems worth examining while taxiing.

(a.) Emergency Brakes/Steering - Pull the emergency brakes/steering handle and hold the paddle switch down (to ensure that Utility A system pressure is blocked and, therefore, the nose wheel steering shuttle valve has shifted over to the emergency hydraulic source, the #2 JFS accumulator). Verify both normal ($\pm 15°$) and maneuvering ($\pm 45°$) range steering is available, and that nose wheel steering is *not* cut out with paddle switch actuation.

Also, check the feel of the brakes for familiarization, and make sure nothing's dragging - that could indicate residual pressure at the wheel which could blow a tire if the handle was pulled at high speed. Besides exercising the system, it's just good practice to reach for and pull the handle. P.S. - the handle should reset easily with a light push force.

(b.) Departure Warning Tone - The yaw rate warning tone that's heard when the T/O trim button is held down is generated by a false signal to the roll/yaw computer, telling it the aircraft is yawing to the right at about 40°/sec. So, while taxiing around a corner, or during mild S-turns, hold the T/O trim button and you should hear the interrupt frequency increase when turning right (higher rate spin) and decrease when turning left. This always gives me a little more confidence that the anti-spin mods are installed and working.

13. Post-flight Checks

These additional checks are discussed for your general interest; they are more appropriate for demonstration or full FCF purposes.

(a.) Emergency Generator - After switching generators off individually to ensure that either picks up the entire load, switch them both off to put the essential bus load on the emergency generator. Time it for 10 seconds to ensure it's going to stay on line.

(b.) Hook Cycle and Aileron Schedule - I've lumped these dissimilar checks together because they should be done at about the same time; one is a test, the other is more of a demonstration. With only the emergency generator supplying power, switch to ISOLATE to drop everything except the emergency essential bus. *When you do this, try to limit operation to 10 seconds or less* because the primary heat exchanger doors will close; and the ECS will get quite warm, as indicated by smoke and an oil smell in the cockpit. in ISOLATE, you must be able to drop and raise the tail hook. This needs to be coordinated with the ground crew, of course, for safety and confirmation of at least the hook release - retract is not significant, in my opinion. Also while in ISOLATE, the ailerons will follow their normal inflight schedule, in which they are reduced proportionally to longitudinal stick position, either fore or aft. This is the *only* way to see this very important control logic during pilot checks.

(c.) Fire Warning Light Shutdown - Terminate the above check/demonstration by bringing either the emergency generator out of ISOLATE, or one of the mains back on line. Then, when ready to shut down, depress the fire warning light/button for that engine. Indicated fuel flow should go to zero, followed by a flameout sound (it actually goes "poof") and engine unwind a few seconds later. Some residual fuel (between the firewall shutoff valve and the engine) may try to reignite, so go ahead and chop the throttle shortly after the "flameout." Then, of course, reset the fire warning light so the next guy can get the engine started.

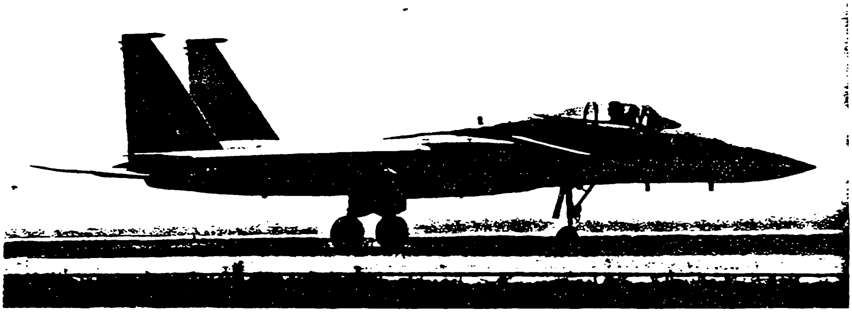
Much of the above is inappropriate for routine flight operations. However, it happens to be our thing, and we do go to extremes to uncover anomalies. If you've found any strange or unexplained aircraft behavior during your checks, please let us hear from you. We have lots of system experts to turn loose on the problems. Furthermore, since most of what we fly are brand new airplanes (it's a rough job, but someone's got to do it), we really don't know all the quirks and problems that age may bring on. ■

Three Famous Fighters

Recent photograph of three McDonnell fighter airplanes taken at Edwards AFB provides picture-lesson in changing aircraft design concepts over the past 25 years. Side-by-side Voodoo, Phantom, and Eagle provide interesting and revealing comparisons in wing, fuselage, inlet, landing gear and tail design concepts. Each fighter was designed for its times and each has carved its own place in the still-unfolding history of military aircraft.

SOMETHING OLD, SOMETHING NEW...

(PUBLISHED 1975)



Something borrowed, Something (Air Superiority) blue!

By DON STUCK/*Advanced Design Project Engineer*

Don Stuck has been keeping DIGEST readers up to date on F-15 status in the Cat I flight test program and other interesting aspects of the Eagle. Now that the airplane is into Cat II and squadron service, he's turned his attention in this issue to the future through a look at the past. We hope that Don's discussion here of F-15 ground handling characteristics with reference to the F-4 will be the first of several such "pilot pointers from a past pilot."

Being an old, bold, pilot (old bald pilot is probably more appropriate), I can't help hanging around the Fleigenwerke; and some of the F-15 conversation I overhear is mighty interesting (matter of fact, *all* of it is interesting, but much is either not printable or not pertinent to what I want to discuss here!). What particularly strikes me in a lot of this hangar flying is the similarity to and continued relevance of a lot of lessons learned in earlier jet programs. Since I was pretty involved in some of those old programs - the F-101 and the F-4 for example - I thought I'd take a trip back through a lot of personal Voodoo and Phantom flight hours and DIGEST "Ready Room" articles to see if there might be anything worth restating for the Eagle.

Sure, there are scads of brand-new tips, techniques, and tricks of the '15 flying trade - and you'll be learning about all of them in upcoming pages of this magazine, from the guys who are out there doing the actual pole work. But my research has also revealed some things which seem to be just as true and appropriate today as they were two airplanes ago. And since I believe that the good fighter pilot is (to borrow another adage much older than any of us)" . . . *among the first to adopt the new and the last to discard the old . .*" let me share my research into some of the "old" to see if it might be of interest to you.

Even after 25 years of driving various types of airplanes, I haven't changed my basic philosophy, which is that a pilot should have a good understanding of the general laws affecting how all airplanes "do their thing," supported by an equally good understanding of the specific systems of his particular aircraft. From this point it is merely a matter of making the airplane do what you want it to do. I do not mean to imply that all airplanes handle alike - just that the same rules of "law & order" apply to all of them. Again philosophically, I wonder how much *really* new learning is required for a new aircraft? There is nothing new or magic about controlling any new airplane - just relearning, understanding, and proper application. For an example, let's look at ground control techniques, as analyzed for the Phantom back when we both were younger.

"Wheels, Brakes, & Tires . . . or Tired Wheels Break" was an article I wrote for the DIGEST about a hundred years ago and concerned itself with taxi techniques for the F-4A/B aircraft as they affected wheels, brakes, bearings, and tires. It brought up the shortcomings of required test criteria (which incidentally haven't improved one little bit since way back then), and it outlined the basic laws of physics as they applied to taxing aeroplanes. Now a few years have rolled by; the aircraft is a different size, weight, and color but that article could just as well have been written for the Eagle as for the Phantom. (It could also just as well have been written for the Hawk in 1939, if anybody besides me goes back that far, because the basic ground handling peculiarities associated with the narrow landing gear on the P-40 can certainly be applied to any other narrow-tread aircraft such as . . . you guessed it . . . the F-15.) Energy is still (and forever) a function of mass and velocity; brakes still convert kinetic energy into heat; side loads during cornering still vary as an inverse function of changing turn radius and as a square of changing speed. Therefore, taxiing/turning/stopping experience gained in the F-101, F-4 (and other vintages) certainly applies in some degree to your new Blue Beauty.

The aluminum wheel is still affected by heat in the same way, and as far as I can determine the same unrealistic MIL-Spec applies to wheel manufacture and testing. The F-15 brake works on the same basic principle as its predecessors - converting kinetic energy to heat energy. Therefore, abnormal heating of the wheel through brake heat dissipation will adversely affect the F-15 wheel just as it did the F-4. While the F-15 brake has greatly improved energy absorption and heat dissipation characteristics compared to other aircraft, everything has its limits so good basic handling practices still apply to preclude abusing the hardware.

Permanent change in wheel strength value will occur if heating is applied for extended periods of time - here are some numbers from that old article which are still valid: An aluminum wheel loses 33% of its allowable design stress and 60% of its potential fatigue life under operating conditions of 400°F; the same wheel loses 85% of its operating stress when heated to 600°F.

Short periods of heating to low levels do not permanently affect the strength factor. However, heating to the higher levels for prolonged periods adversely affects strength even after the wheel has cooled. A wheel heated to 300° loses 18% of its allowable stress *at that temperature* but is back to normal strength when returned to normal temperature. The same wheel heated to 600°F for 30 minutes will suffer a *permanent* 55% loss of allowable stress after being returned to normal temperature.

These test conditions aren't too far from real life when considering that the fusible tire plugs on the F-4/F-15 are both set for over 400°F, and that after heavy braking the wheel and tire reach their max temperatures approximately 15 minutes after the braking application. I don't mean to say you'll suddenly find yourself sitting in a pile of melted rubber and wheel aluminum if your ground control actions induce abnormally high temperatures; but keep in mind that the less heat you put into the wheel, the more longevity you'll get from it. Therefore, while the following common sense items were laid out for the F-4, they should be applicable with minor modifications to the F-15:

- Don't fly an aircraft you suspect has a dragging brake
- Don't taxi faster than necessary (requires more than normal use of brakes).
- Don't pump the brakes - use steadily increasing pressure to the amount needed for as long as needed, then get off the pedals.
- Use steering for directional control.
- Plan ahead. Make as few stops as possible.

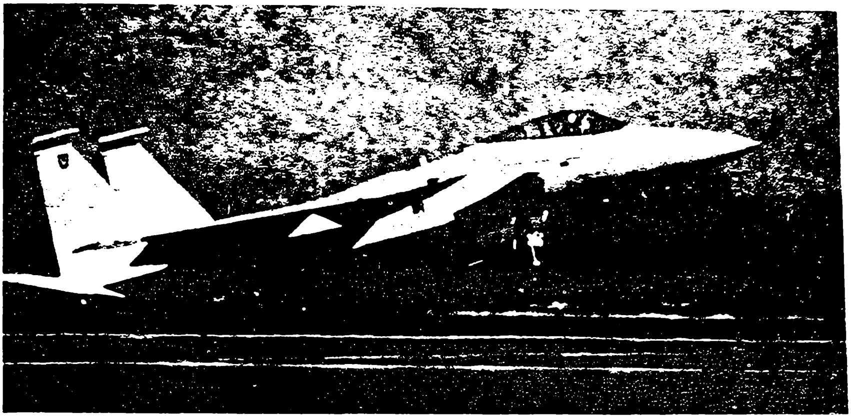
Landing gear side loads were also discussed in that old article and several points are relevant today. An obvious effect of excessive side loads is tire wear during taxi. A more insidious (and more hazardous) impact is wheel and bearing fatigue. It's obvious we can't stop making turns, but we do have control over how we make them. Therefore, for a given number of turns required through a given taxi course the pilot and the pilot alone dictates the amount of side load applied to the aircraft. Again we go back to basics, since it is centrifugal force which gets into the act to cause side loads around the corners. Actually the narrow tread on the F-15 should help hold down cornering speeds since I would think there'd be an inherent fear of falling over and "dragging a wing tip" if you cornered too fast.

Since side load is directly affected as an inverse proportion to radius and as a square of speed increase, it is obvious that the wider and, more importantly, the slower we corner the better. Weight also gets into the act, so it follows that taxi out at heavy gross weights is more critical than taxi back.

I stated in that old F-4 article that by my estimate, pilots of F-4's and F-101's taxied at about twice the ground speed they thought they were using. In talking with Irv Burrows and his F-15 pilots I get the feeling that this estimate is even greater in the F-15 since the high idle thrust, better visibility, and long straight taxi distances encourage even higher taxi speeds. Bearings, wheels, struts, brakes, and structure all feel the side loads of cornering and the heat of braking. They're built to take it, but you can help by applying the same good common sense rules you used in past aircraft.

Take it easy on the ground - save that Eagle's speed for after you get the gear up! ■

Six Hundred words on... (PUBLISHED 1976)



F/TF-15A Takeoff Trim

By PERRY HOFFMAN/*Senior Engineer, Flight Control Section, Avionics Engineering Laboratories*

We've gotten reports, following production acceptance flights of the F/TF-15, that the Eagle has a tendency to roll right wing down immediately after liftoff with takeoff trim selected. Though the roll rates are reported as being in the order of 5 to 8 degrees per second, pilots have indicated that an untrimmed aircraft is uncomfortable at this point of a flight.

Normally it takes only two short "beeps" to correct the situation, well within the trim authority provided for this purpose. The "rub" comes when pilots observe ailerons deflected with a clean aircraft. Experience with previous aircraft tells us that "if differential aileron is required to maintain straight and level flight, do not perform high angle-of-attack maneuvers."

But the story is somewhat different with the F-15. In the words of McDonnell Chief Test Pilot Jack Krings (the Eagle spin pilot): "The F-15 doesn't want to depart at maximum angle-of-attack, even when full lateral stick inputs are applied." This characteristic is obtained through a blend of good aerodynamic design plus a generous amount of help from a sophisticated control system.

However, getting back to our original concern: why would the F-15 want to roll on takeoff, and particularly, why does the aircraft always roll *right* wing down? Our aerodynamics group indicates that the unbalanced weight of a gun mounted in the right wing root is the cause. According to their calculations, it would require about one-half a degree of differential aileron or stabilator to counteract this slight off-center gun weight.

Now what? Should we add 1200 pounds to the left wing root as well? That's no more practical than removal of the gun. We find ourselves, therefore, faced with two alternatives. First, we could use the twenty percent of manual trim assigned to this sort of task. Second, we could bias the differential stabilators sufficiently to yield as near a trimmed airframe as possible with takeoff trim selected.

The second choice has more appeal to U.S. Air Force acceptance pilots as well as pilots within operational units. In addition, the small amount of bias does not affect aircraft performance.

Changes to the applicable technical orders are now in work. In an effort to minimize errors, the method selected to set stabilator bias is to vary the neutral setting by 0.4 degrees. Maintenance personnel will find new F/TF-15's already have the bias installed, plus a variation in stabilator position when takeoff trim is selected. The neutral stabilator position should be verified prior to individual stabilator power cylinder replacement. (To do this, pin the Pitch PRCA and place the Pitch Ratio switch to EMERGENCY.) Then set the new power cylinder to the old neutral point.

The use of differential stabilator bias settings will minimize the Eagle's unruly tendencies in takeoff trim, but will not insure a perfectly-neutral flying aircraft every time. Takeoff trim was designed to be used as a method of getting surface controls to some preselected starting point, thus removing any trim retained within the airframe because of corrections for things like asymmetric loading due to external stores or trapped internal fuel. Takeoff trim was not designed to be used in flight except in an emergency where pilot disorientation makes normal trimming *seem* impossible.

We hope this short article answers some of the questions that have been asked about F/TF-15 takeoff trim, and clarifies the solution to the problem. Operational pilots won't notice any change if the adjustments have been properly performed, but in the event discrepancies *are* noted, you'll have some sort of an idea of what might cause them. ▪

(3RD QUARTER 1971) (PUBLISHED 1971.)

Takeoff Abort Speeds

*By IRV BURROWS/ *Chief Experimental Test Pilot*

Another letter - this one concerning some seeming contradictions in the Air Force Dash One-Dash One data on abort speeds:

". . . In T.O. 1F-4C-1-1, would you please compare Figure B2-3 MAXIMUM ABORT SPEED/Maximum Thrust/Without Drag Chute with Figure B2-4 MAXIMUM ABORT SPEED/Military Thrust/ Without Drag Chute? And when you do, please see if you generate the same fire drill among the jocks in St. Louis as occurred here! The gross weight curves plotted on B2-3 indicate that the higher the gross weight of the aircraft, the higher the maximum abort speed can be!?

"I'm sure these figures are plotted wrong as the gross weight figures should be just the opposite or reversed with the lower gross weights plotted on the curve which would allow higher maximum abort speeds. I need information confirming same, or if the chart is correct, can you please explain why?. . ."

Here are the subject USAF curves, but we invite our other service readers to come along also, because if the USN/UK curves for MAXIMUM REFUSAL SPEED were superimposed on each other instead of being separately plotted for gross weight, you'd see the same phenomena as our Air Force friends went into fire drill about -

The question is - "Why does a 60,000 pound aircraft have a higher abort speed than a 40,000 pound airplane?

Your first reaction is that the curves on Flight Manual Figure B2-3 are erroneously labeled - anyone knows that a light airplane can successfully abort at higher speeds than a heavy! The hooker, though, is pilot decision and reaction time. If aborts were accomplished truly instantaneously, the above theory would hold water. But now let's add some time for decision (three seconds), and for getting engines to idle and brakes on and effective (five seconds). During this eight second period, the light airplane accelerates more than the heavy one. So the increment of velocity (ΔV_L in Figure 1) that the light aircraft picks up during that time is considerably more than that generated by the loaded plane (ΔV_H). As you can see, the ΔV is created partially by acceleration due to MIL (or MAX) thrust, and partially by "residual" acceleration during the period of time required for thrust to decay to idle and brakes to be applied and effective.

If we plot velocity against runway distance we'll have something like Figure 2. (Obviously, these are generalized curves with the sole intent to display the relationship of speeds involved.) Here the ***chart*** abort speed for the light aircraft (V_1) is seen to be lower than that for the heavy (V_2), but by adding the eight-second ΔV's an ***actual*** abort speed for the light bird that's higher is produced. That should put the whole situation in proper perspective.

Now that you're convinced (?) that what looks wrong is actually right, you'll ask - "OK smart guy, I just looked in the book at the portions of the curves that weren't included with this article, and how come the lines cross in Figure B2-3 MAX THRUST with drag chute, and B2-4 MIL THRUST without drag chute, but look 'normal' in B2-4 MIL THRUST with drag chute?" Again, these charts are generated from accel/decel curves like Figure 1, including the effect of three second and five second time increments. Different thrust levels, plus or minus the drag chute, will alter the shapes of these curves so that V_1 will be above V_2 at times and below at other times.

You may question the eight second delay, as I did, because it seems like excessive time. But there is some finite time involved and the eight second delay is simply an arbitrary, conservative, mutually (us and the customer) agreed-upon figure which is used to generate the charts.

Seems like a lot of words and music, but I hope it has cleared up this question. Incidentally, the information currently presented in the charts has not changed - only the format (Air Force charts used to be plotted in same manner as Navy). Just goes to show that format changes don't always eliminate questions - sometimes generate them!

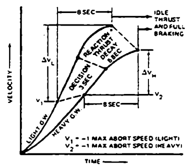


FIGURE 1

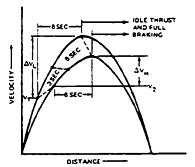
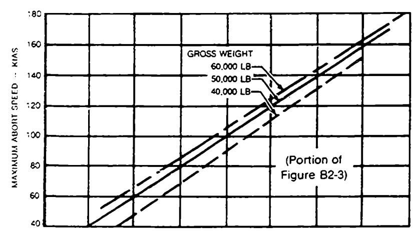


FIGURE 2



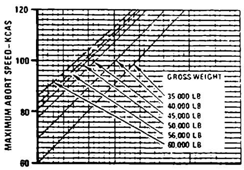
(Portion of Figure B2-3)

* Editor's Note: We have chosen to reprint this article exactly as it originally appeared in 1971 because both the general subject and the specific point with respect to decision/response times are still valid today. However, don't try to look up some of the references in the discussion! T.O. 1F-4C-1-1 (Performance Data Manual) doesn't exist today; information therein was absorbed into the F-4 Dash One and the -1-1 discontinued back in 1975. Also, there are references to "maximum refusal speed" and different arrangements of data with respect to USN/(UK) flight manuals; the F-4J/S NATOPS abort charts have since been condensed to the same format used by USAF. And finally, Irv Burrows, the author of this 1971 article, is no longer Chief Experimental Test Pilot. His name now appears at the top of the masthead of this magazine, as Vice President of Product Support, but if you continue reading, you will note that his successor in the cockpit also has something to say on this topic.

The Abort Speed charts in the Takeoff Performance section of the pilot's manual can be quite confusing. At least once in the life of every fighter aircraft, we can count on receiving a letter taking our flight handbook writers to task for misprinting, misplotting, or otherwise messing up the "Maximum Abort Speed" graphs.

In essence, the letters all say the same thing — the gross weight curves on these charts "must" be wrong because they show a heavily loaded airplane able to abort at a higher speed than a more lightly loaded one. Can't be. . . no way! Well, there is a way, and it results from a factor inserted in the mathematics used to create abort speed charts that is pretty obscure to most of us. That factor is "time" — the number of seconds allotted in the calculations for the pilot to recognize and react to an abort situation and for the aircraft to respond to his inputs. We first discussed abort speed charts and reaction times back in 1971, in the Product Support Digest article reprinted at left. That twelve-year old analysis concerning the F-4 Phantom is just as appropriate today in response to our latest "no way" letter, but this time on the F-15 Eagle, which shows the same apparent upside-down abort speed-to-gross weight relationship (see chart portion at right) as the F-4.

As long as we're taking another look at abort speed charts, how about our other airplanes? We don't expect any



letters on the Hornet charts since the speed/weight relationships look "normal" on them. While data for the F-4, F-15, and F-18 are all based upon the same three-second pilot recognition/reaction time, the Hornet charts assume just two seconds for aircraft response, for a five-second total period in which a lighter aircraft will not build up any significant increment of velocity over a heavier one. As for the AV-8, the Harrier pilot's manual contains no abort speed charts at all – if you can take off "straight up," who needs an abort speed!

Since this topic seems to get discussed only once every airplane generation, we'd like to conclude with just a little more on . . .

Takeoff Abort Speeds

By PAT HENRY/*Chief Experimental Test Pilot*

(PUBLISHED 1983)

At the risk of offering more about takeoff abort speeds than you care to know, I'd like to take the discussions at left just a step or two further, with a couple of ideas from my own experiences in disconnecting airplanes from runways. To me, an abort speed chart is either a good friend or a waste of time, depending on circumstances as presented below.

It is my suspicion that most high performance jocks don't spend a lot of time studying the takeoff abort charts, and I can't fault them for that. We don't exactly have an airline type operation in a fighter cockpit – can you picture yourself calling out "V_1; V_R – Rotate?" Not exactly applicable.

Every so often, however, we find renewed interest in these Dash 1 charts, either as a function of slippery field operations, new jocks, or both. We welcome the questions that then result; it gives us an opportunity to dust off our cobwebs and perhaps philosophize a little.

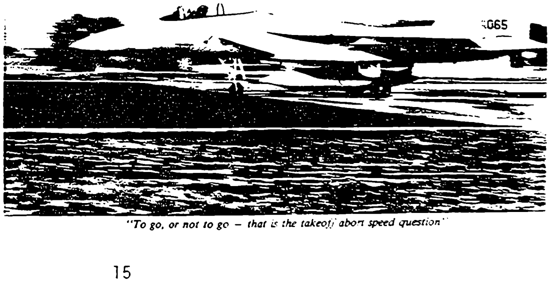
In my opinion, the abort charts are part of an overall body of engineering knowledge that should be examined and compared against actual experience, particularly when first transitioning into an aircraft. Another time that takeoff and abort charts are of more interest is when operating conditions change significantly, either via higher gross weights or shorter runways. Normally, however, rather than computing "V_1" (decision/abort speed) and splitting your attention looking for it to rapidly come and go, I see more appropriate uses for your attention. With the F-15 for an example, let's take them from the beginning:

• Pre-taxi control check. If your bird passed a thorough check of flight controls, including pitch ratio scheduling, PTC trimming, and stick force sensor output, there's a high probability you're going to have beaucoup stab authority for rotation. One last check of stabilator travel before takeoff is quick, cheap life insurance, of course.

• A/B verification. If at all possible, an A/B check prior to takeoff roll is highly desirable – wash filters have been known to clog, or ignitors fail, and that additional thrust just might not be available and waiting. If you're making a max power takeoff, select A/B immediately at brake release, and get the good (or bad) news early. After selecting A/B, watch or recheck the nozzle for some pre-open with throttle movement and at least two segments worth of consecutive opening after A/B light. One failure that is popular this semester – clogged wash filters – will cause the nozzle to twitch with each segment's quick fill, looking a little like a good A/B light at first glance.

• Pitch ratio. One last glance at the pitch ratio indicator is a good habit pattern to form. Of all the things that could prolong your stay on the concrete, a minimum pitch ratio is probably the most dramatic.

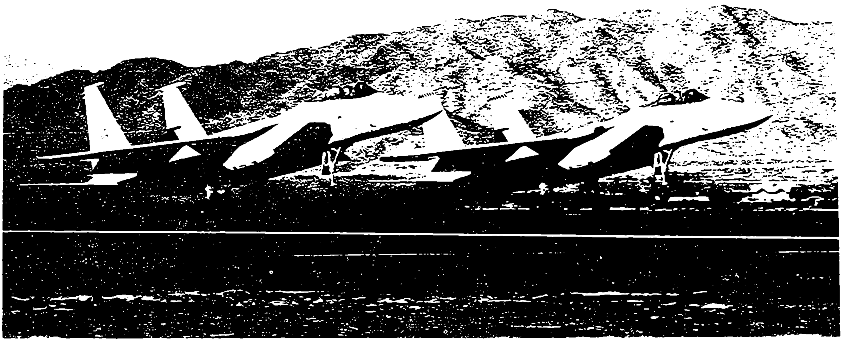
In summary, if you have stab authority and thrust, you'll have a hard time not flying, even with a poorly serviced nose strut. I personally like the idea of getting airborne, then reducing gross weight while sorting out options. A massive failure during takeoff, such as engine stalls, fire, or flight control jamming are obvious candidates for a change of plans; and it would be awfully comforting to have an abort arresting gear waiting just in case. As with most philosophical discussions, no decisions are made for you, so the monkey is still on your back to handle any given (soggy) situation. That's the responsibility that accompanies the pride of professional flying. ■



"To go, or not to go – that is the takeoff abort speed question"

(PUBLISHED 1978)

nose wheel liftoff, or...



LAST OF THE LATE ROTATERS!

During the past year, there have been many questions concerning a phenomenon that came to be known as F-15 "late rotation," i.e., nose wheel liftoff speeds some 10-15 knots higher than normal. Using a service-supplied aircraft that exhibited these peculiar characteristics, MCAIR investigated the problem, discovered the cause, and recommended the corrective action. This article is presented to report on that problem and also to provide both Eagle and Phantom pilots with a good general refresher course on "getting the airplane off the ground."

By PETE PILCHER/ *Experimental Test Pilot*

Q — What does it take to get the Phantom or Eagle nose wheel off the runway on takeoff?

A — The aircraft must rotate about the main landing gear. To do this the moment created by the stabilator must overcome the moment that is the result of the weight of the aircraft acting through the center of gravity, which is ahead of the main landing gear. The stabilator will create a down force to provide this moment when it is deflected (aft stick) and the speed of the machine is sufficient.

WAIT A MINUTE!

If you have a hunch that all sounds too simple, your hunch is correct. Actually quite a few variables affect the nose wheel liftoff speed. Some of these are:

- Aircraft gross weight
- Aircraft CG location
- Stabilator position
- Flap position
- Aircraft speed at aft stick initiation
- Plus one other very interesting item that we'll call to your attention a little later!

Since the effects of these variables may not be that obvious to all of us, let's expound a bit.

WEIGHT AND CG

As gross weight increases, the tail-down force required to move this load must also increase. You engineers in the crowd can sum the moments about the main gear to prove this fact.

This same sum of moments about the big tires also shows that if the CG moves forward, the tail load required to rotate will increase and vice versa. If the CG were above the wheels, the airplane would rotate at any speed, even in the chocks. That wouldn't be good because it would require a wheel or strut at the back of the airplane, but we sure wouldn't have a nose wheel liftoff problem.

Seriously, center of gravity is one of the most important variables. It is also constantly changing as internal fuel is used. The CG typically moves forward about 1% MAC (mean aerodynamic chord) for every thousand pounds of JP consumed in the F-4 (the F-15 CG moves 1/4% *aft* in the same situation). If the F-4's motors run on the ground for much more than 30 minutes, the CG can move forward enough to cause a five knot increase in the nose wheel liftoff speed. Unusually long ground run time before commencing takeoff has caused several takeoff aborts in the Phantom. I also suspect that the CG at engine start is not known accurately enough to predict a nose wheel liftoff speed within a few knots anyway.

FLIGHT CONTROLS

Air flow over the tail causes the tail to lift downward, provided the angle of attack of the horizontal tail is correct, i.e., leading edge down (aft stick). This makes it easy for us pilots since the same natural action and response occurs inflight; i.e., pull the stick back to make the nose come up and cars and houses get smaller. This applies to both the F-4 and F-15.

Flap position affects the speed at which the nose rotates. Flaps down inflight causes a nose-down pitching moment that is generally trimmed out with little thought or effort in the F-4. It was this moment that necessitated the slotted stabilator on some models of the Phantom (not enough horizontal tail to trim the aircraft at extreme forward CG conditions during full flap landings in ground effect). The pitching moment with flaps in the F-15 goes essentially unnoticed by the pilot. The flap contribution to nose wheel liftoff is significant in both fighters. Flaps down tends to reduce stabilator effectiveness, which actually causes an eight knot increase in nose wheel liftoff speed in the Eagle and a 12 to 14 knot increase in the Phantom. Also, flaps up during takeoff in the Phantom makes the nose rotation rate snappy — one of the reasons the handbook says to use flaps.

STICK BACK!

Another variable that can cause a change in nose wheel liftoff is the aircraft speed at aft stick initiation. A couple of profound statements can be made regarding this matter. An F-4 in normal takeoff configuration would probably *never* rotate and fly off the runway without pilot-supplied aft stick. The F-15, on the other hand, flies off the runway at about 165 knots without any pilot action other than selecting takeoff trim before takeoff roll. The nose wheel liftoff speed will be increased as much as 12 knots by pulling the stick aft at 130 KCAS vs 100 KCAS in an F-15 at Mil power equipped with a full centerline tank. This basically means that we are probably delaying nose wheel liftoff by 5 or 6 KCAS by waiting until the handbook number of 120 shows up on the airspeed indicator.

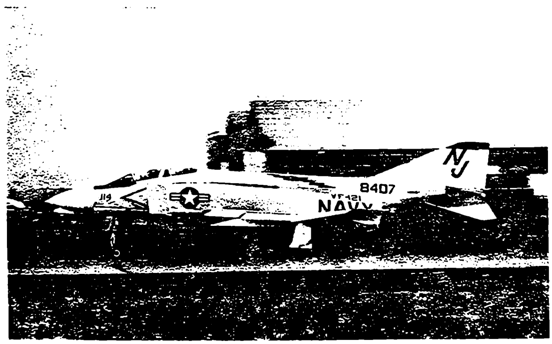
WOULD YOU BELIEVE?

Now for the little item that we thought might interest (and surprise) you — another of the variables that can affect nose wheel liftoff speed is — believe it or not — the nose strut servicing. Nose strut servicing can have a significant impact on nose wheel liftoff in the F-15 and may have a similar effect in the F-4. If the nose landing gear strut is properly serviced, the nose strut continues to push the nose of the aircraft up during strut stroke to maximum extension. This aids in the nose up rotation of the aircraft. If the nose strut is loaded with too much oil, and therefore not enough air, it does not provide this reactive force throughout the strut extension and is less of an aid to the rotation of the aircraft. The degree of strut mis-service is another variable, of course.

Mr. Clarence Mongold, of our F-15 aerodynamics group, claims that a mis-serviced strut can cause a rotation speed increase of 10 to 15 knots. He supports his claim with the results of both simulation modeling and flight tests of a "late rotater" F-15 that we borrowed from Luke AFB this summer.

During the late rotater tests, we conducted routine checks of the gross weight, center of gravity, airspeed system, and flight control system, and found everything in order. The airplane was first flown as-received and gave a very rough ride on the ground. It felt as though the front spring was very stiff. On takeoff the nose wheel came off the runway approximately 10 knots late. Upon inspection, the nose strut was discovered to be considerably overserviced with oil. Doesn't sound important, and the aircraft had the proper attitude on the ground, but a load-stroke test on the nose strut showed that the strut gave up most of its energy in the first half-inch of travel and did not help lift the nose after that point. The F-15 nose must come up about a foot before the nose wheel clears the runway, so the strut was not much help in rotating the airplane after the first half-inch of travel.

It appears from the load-stroke curves of the F-4 that the same phenomena applies; the strut aids in rotation throughout the two foot stroke of the nose strut.



THOUGHTS ABOUT NOSE WHEEL LIFTOFF SPEEDS

Nose wheel liftoff speeds have been a sporadic problem in the F-4 and F-15. About once per year per aircraft, one of these machines is aborted at high speed and high gross weight because the nose did not come up at the precomputed airspeed. In nearly every case, the brakes, wheels, and tires are consumed in the stopping effort. When a nose wheel liftoff problem is reported, the standard variables of CG, flight control system, and airspeed system are checked. Rarely is a discrepancy found with any of these variables. The airplane is then flight checked and found to be flightworthy.

Since stopping a heavier than normal fighter from a higher than normal speed on the ground is not my idea of a fun time, I believe in and follow the following guidelines —

- Check the stabilator before takeoff in either the F-4 or F-15 for freedom from restrictions, for correctness of direction (stick aft = leading edge down), and for full travel in both directions. If the stabilators pass this test and the motors work, the airplane *will* rotate and fly. Maybe not at the nominal published speed, but it will in fact rotate unless the laws of aerodynamics are repealed or the stabilators suddenly fall off, neither of which is likely.
- Take the nose wheel liftoff curves with a grain of salt, knowing that there are many variables involved; variables that can change between the briefing room and the runway and can account for as much as 20 KCAS.
- Bring the stick back early in the takeoff roll, except as a section leader on a formation go. The F-15 Dash One recommends 120 knots for aft stick movement for all takeoffs. As we indicated earlier, this may cost 5 to 6 knots in nose wheel liftoff speed in a clean machine. The F-4 Dash One allows more pilot judgment in this area. For most F-4 takeoffs, the pilot is merely reminded to pull aft stick well below the nose wheel liftoff speed.
- Make sure the nose gear strut is OK, OK? Looks can be deceiving. If the ride's not right, write it up. ■

(not the) "Last of the Late Rotators"

By GALEN STANLEY/*Lead Engineer, Systems Safety Engineering* (PUBLISHED 1982)

I had the opportunity recently to address a meeting of F-15 Flight Safety Officers on the subject of late rotating aircraft and I could tell that the story I was putting out was *not* one they had heard before. We were discussing a July 1981 incident at Bitburg AB, Germany, in relation to interpretation of two statements made in an article titled "Last of the Late Rotators," written by MCAIR test pilot Pete Pilcher and published in Issue 5/78 of the DIGEST. The statements, taken out of context, were being incorrectly interpreted by some to mean that the pilot of the Bitburg aircraft "caused" the late rotation by delaying aft stick until 135 knots. The key to understanding the apparent discrepancy is to look at the predicted nose wheel liftoff (NWLO) speeds for the configurations quoted in the DIGEST article vs the Bitburg aircraft.

- Statement 1 — *"Nose wheel liftoff speed will increase by as much as 12 knots by pulling the stick aft at 130 KCAS vs 100 KCAS."*

This statement from the article was referring to the centerline tank configuration of the aircraft we had borrowed from Luke AFB for flight tests. The minimum NWLO speed of 122 KCAS was achieved by pulling aft stick at 100 KCAS. If the stick was pulled aft at 130 KCAS, the nose wheel would be off the ground in approximately one second at 136 KCAS; exactly the kind of performance you would expect. The 12-14 knot increase is from the minimum nose wheel liftoff speed of 122 KCAS; not from the 130 KCAS stick aft speed.

- Statement 2 — *"We are probably delaying nose wheel liftoff by 5 or 6 KCAS by waiting until the handbook number of 120 shows up on the airspeed indicator."*

This statement from the article is referring to a "clean machine" with a nominal CG. This gives us a computed minimum nose wheel liftoff speed of 118 KCAS. For that clean machine delaying stick input to 120 knots is obviously going to cause the nose wheel to be on the ground a little longer than if it leaped off smartly at 118. For the Luke aircraft with centerline tank (a normal training configuration), nose wheel liftoff should occur at 129 KCAS if full aft stick were applied at 120 KCAS. In this case, we would increase NWLO speed by seven knots above the minimum NWLO speed of 122 KCAS.

NOSEWHEEL LIFTOFF SPEED

Let's jump back to Bitburg for a moment and review the incident where

Nose wheel liftoff - moment of mastery . . . and mystery. Until this moment occurs, you are really doing nothing more than (an extremely rapid) taxi. Misunderstanding continues concerning relationship between stick input speeds and nose wheel liftoff speeds. This article takes another look at the variables involved.

the aircraft didn't rotate when aft stick was initiated at 135 knots during a wing formation takeoff. The computed nose wheel liftoff speed was 135 knots but the nose still wasn't off at 150 knots when the abort was initiated. Now there are really only four factors that affect nose wheel liftoff: gross weight (GW), airspeed, stabilator deflection, and the upward force exerted by the landing gear struts. We know the gross weight; the airspeed was OK; the flight control system checked OK; but the strut servicing was incorrect. After reservicing the struts, the jet rotated on cue.

Flight tests of the "late rotator" from Luke in 1978 demonstrated that rotation speed can be delayed 10 to 15 knots by mis-servicing the nose strut. This matches well with both simulation modeling and field experience; if the nose strut is mis-serviced, it's going to take more airspeed or more stabilator deflection to rotate the aircraft (nose strut servicing has much greater effect than main strut servicing). Unfortunately, neither the static strut deflection nor the pressure is a valid indication of proper fluid level servicing, so it is difficult to tell if the struts are properly serviced during preflight. MCAIR has just completed evaluation of an ultrasonic inspection procedure with very favorable results, and this procedure is now available in the field.*

Another factor that deserves consideration is the accuracy of computed nose wheel liftoff speed. Well, the charts are pretty good, but they:

- Are based upon full stabilator deflection (nearly full aft stick).
- Assume that actual takeoff gross weight is accurately known.
- Assume that actual CG location is accurately known.

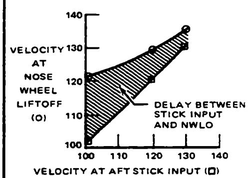
If you are very far off on any one or more of these factors, you could be 10-20 knots off on predicted nose wheel liftoff speed. Some pilots have voiced concern that pulling aft early can reduce aircraft acceleration because of the drag of the deflected stabilator. This effect is negligible, particularly when compared with the benefits of ensuring the stabilators are properly deflected when minimum nose wheel liftoff speed is reached.

One picture is worth a thousand words. There is nothing mysterious about F-15 takeoff performance, and everything we have discussed so far can be summarized in Figure 1. As expected, the higher the airspeed at stick input, the quicker the aircraft response (time between input and nose wheel liftoff). However, the minimum nose wheel liftoff speed is always achieved by applying full aft stick at 100 KCAS; delaying aft stick input always increases NWLO speed.

TAKEOFF TRIM

At this point in the meeting, a revelation occurred when one of the FSO's volunteered that some pilots take a couple clicks of nose down trim just before takeoff to avoid having to do it immediately after gear and flaps up.

FIGURE 1
F-15 TAKEOFF PERFORMANCE



Boy - what we have here is a classic example of the right hand (pilot) not knowing what the left hand (designer) is doing! Let's take a quick look at this part of the Eagle flight control system.

The pitch trim compensator (PTC) is part of the mechanization in the mechanical control system that minimizes trim changes in flight as the aircraft speeds up or slows down. It is an energetic device with 7.5° of stabilator authority. On the ground with the stick centered (trimmed for 1G), the PTC could be either adding 2° to the stabilator position commanded by the mechanical system or be subtracting 5.5°; but it *won't* be neutral! The F-15 design compensates for this by establishing the takeoff trim position 0.4 inches aft of the centered stick position (i.e., commanding slightly more than 1G). This causes the PTC to be in the full aircraft nose up position during takeoff. With takeoff trim selected and the stick at half of full aft stick travel, you would normally get 20° of stabilator leading edge down deflection on takeoff (full stabilator deflection is 26°).

If you defeat the takeoff trim design feature by either trimming the stick forward or by holding a small amount of forward stick during the takeoff roll, you may drive the PTC to the full nose down position. In this case, you would initially get only 12.5° of stabilator for the same half stick input and it would take five seconds to get back to the full 20° deflection normally commanded by a 50% stick input. Five seconds at 150 knots would seem like an eternity and I guarantee that if your PTC is full nose down when lead rotates, you'll have a difficult job compensating quickly enough to fly off anywhere near him.

ABORT OR CONTINUE?

The decision to abort or continue the takeoff following a failure to rotate is never simple or easy. If the failure to rotate is due to improper nose strut servicing or an improper estimate of minimum nose wheel liftoff speed, an additional 10-20 knots should produce a satisfactory rotation rate. If the problem is due to a flight control system malfunction (would require at least two major faults), addition of 10-20 knots may or may not provide rotation. The best way to ensure that the flight control system will not cause you a problem is to accomplish a thorough pre-flight check of the flight controls, including a stick force sensor check. (I suggest you re-read MCAIR Chief Experimental Test Pilot Pat Henry's article on this subject in DIGEST Issue 3/80). Then verify that CAS is on, pitch ratio is 1.0, and T.O. trim is set prior to takeoff. If, however, you are 20 knots or more above computed minimum nose wheel liftoff speed with *full* aft stick and the aircraft has not started rotating, your best bet is probably to abort.

You might ask, "Is the Bitburg aircraft finally 'The Last of the Late Rotators'?" I doubt it. The problem is likely to occur occasionally. The problem, by the way, is exactly the same for C and D Eagles as for A and B except that the nose wheel liftoff speed on the C/D is noticeably higher due to higher gross weight. Pilots transitioning from F-15A/B to F-15C/D aircraft should remember that.

In conclusion, I hope this article has provided some additional understanding in how pilot procedures do and do not affect nose wheel liftoff speed and on the effects of improper nose strut servicing. *The bottom line is that the technique of applying approximately half stick input prior to 120 KCAS is a good idea for all takeoffs, and especially during wing formation takeoffs.* This will ease the problem of matching lead's takeoff attitude and minimize the effects of all other aircraft variables, including improper strut servicing. ■

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*Note: Procedure is contained in Interim Operational Supplement 1F-15[ ]-36S-1 released in May 1981, and will be included in next revision of TO 1F-15[ ]-36 (Non-Destructive Inspection)

(PUBLISHED 1979)

Optimizing the Low Altitude Dive Recovery Maneuver

By CLARENCE MONGOLD/*Section Chief, Technology*

The outstanding performance of the F-15 permits maneuvering in the vertical plane for tactical advantage. Air Combat Maneuvers have been developed to exploit this performance advantage, and much time is devoted to training pilots in the performance of these maneuvers. The combat may be engaged at low altitudes or may descend from higher altitudes during the maneuvering. It should therefore be expected that at one time or another the pilot will find himself in a steep dive with the ground rapidly approaching. He must judge quickly the altitude needed for recovery and, in some situations, immediately initiate the optimum pre-planned recovery maneuver. TO 1F-15A-1 contains three dive recovery charts to inform the pilot of altitude requirements, and little need be added to this information. Therefore, in this brief presentation, I'd like to discuss dive recovery *techniques*.

AIRSPEED-LOAD FACTOR TRADEOFF

Pull-out radius, which determines altitude lost, is dependent simply on airspeed and load factor. High load factor combined with low airspeed provides a tight turn radius. This generic relationship, applicable to all air vehicles, is shown by Figure 1, which also shows the F-15 operating line for full aft stick pull-outs.

The data represents low altitude operation with 10,000 pounds of fuel, a centerline tank, and one AIM-9L missile — a loading often used for ACM training missions. Variations in loading will affect the pull-out radius but have only secondary effect on the optimum airspeed. For the F-15, the minimum pull-out radius and therefore the least altitude loss occurs at 300 KCAS.

This appears simple and straightforward, but before you decide to make all pull-outs at 300 KCAS, there are some other factors to consider. Figure 1 assumes a constant airspeed during the maneuver, which is difficult to achieve. Also, your airspeed at the start of the pull-out may be higher or lower than 300 KCAS. So, some relevant questions are —

- Should you delay the pull-out until you have adjusted your speed?
- Should you change engine thrust?
- What about the speedbrake?

Figure 2 has been prepared to answer these questions, using the same conditions as Figure 1. Note that here we are presenting the airspeed prior to the pull-out and the altitude loss during the pull-out.

DON'T WAIT

The first thing to note from Figure 2 is that a delay in initiating the pull-up in order to allow airspeed to build up is very unwise — too much altitude is lost during the wait.

Compare the upper set of three curves with the lower set. For the upper set, the 70 degree dive was continued to allow the airspeed to increase by 50 KCAS over the initial speed. For the lower set, the pull-up began immediately, resulting in significant altitude saving. Altitude loss in both cases is measured from the initial airspeed condition.

"... at one time or another, the pilot will find himself in a steep dive with the ground rapidly approaching. ..."

LOW AIRSPEED DIVE RECOVERY

The second observation from Figure 2 is that doing a hard pull from a low airspeed without sufficient thrust can get you in trouble — note the steep upturn at low initial airspeed of the four curves using idle or military power. The cause of this is reduction in load factor capability due to excessive speed bleed-off. When the load factor drops below 1.0, the reduction in flight path angle ceases and recovery is precluded.

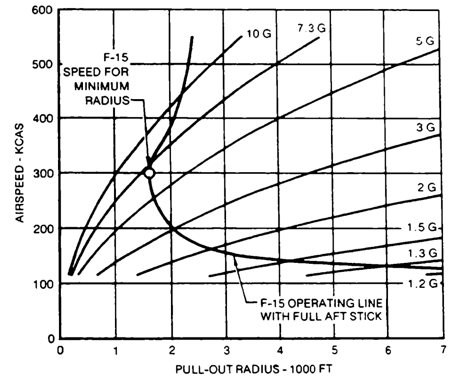
This is not the result of abrupt aerodynamic changes since the F-15 high angle of attack characteristics are smooth and continuous. Nevertheless it is a situation which should be avoided by using maximum power. With maximum power used in the pull-up initiated at 50 KCAS, the airspeed increased to 150 KCAS even though the stick was held full aft. The load factor built up to about 2 G before dropping back to 1.4 G at completion of the recovery, at which time the airspeed was 130 KCAS. The high thrust level provided this increased airspeed but it is essential that maximum afterburner be selected at the time the stick is moved full aft. Incidentally, these conditions are in the envelope of highest afterburner light reliability.

Of course, airspeeds and load factors are greater during the higher initial airspeed recoveries. A load factor of 2 G is generally not considered significant, but at low airspeeds you have something else going for you — the low speed itself, and a small load factor here will work wonders. This effect is shown on Figure 3, which compares a low airspeed - low G recovery with one made at high load factor and a higher speed. The high load factor in this situation is comforting initially but is no guarantee of recovery.

HIGH AIRSPEED DIVE RECOVERY

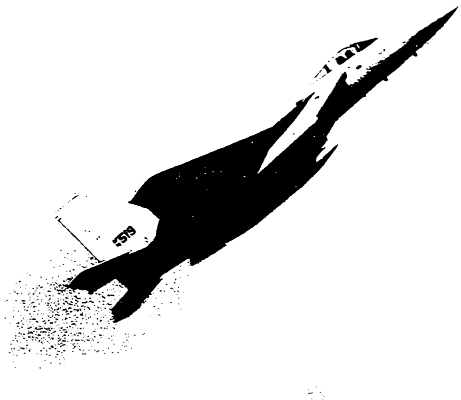
High airspeed dive recovery presents a different situation, a problem of high velocity. A look back at Figure 1 shows that although the aircraft load factor capability increases rapidly at high airspeeds, the turn radius also increases at speeds above 300 KCAS. However, the higher load factors still produce the tightest turns. TO 1F-15A-1 recommends load factors up to 10 G

FIGURE 1 — AIRSPEED-LOAD FACTOR TRADEOFF IN LOW ALTITUDE DIVE RECOVERY



(at a gross weight of 36,000 pounds) for emergency dive recovery. Obviously a rapid deceleration toward the minimum radius airspeed would be desirable, but again the laws of physics work in a contrary manner.

The structural load factor limitations restrict the angle of attack and the induced drag coefficient, limiting the aircraft drag increase in the ▸



pull-up. Because of this and the low profile drag of the F-15, more airspeed than desired is retained in a high airspeed dive recovery. The airspeed can best be reduced by immediately chopping the throttle to idle and pulling high load factor. Figure 2 shows the altitude loss resulting from such maneuvers and the adverse effects of military or maximum power. Figure 3 illustrates the greater altitude lost in high airspeed recoveries although a much greater load factor is available at high airspeeds. The pilot is cautioned to consider this carefully.

SPEEDBRAKE OPERATION

The speedbrake has very little influence on dive pull-out since its drag is small compared to the high angle-of-attack induced drag of the aircraft. Also, it will automatically begin to retract when the AOA exceeds about 25 cockpit units. However, if the switch is left in the aft detent the speedbrake will again extend when the AOA is reduced at completion of recovery. Therefore, retracted speedbrake should be selected for low airspeed recoveries. For high airspeed recoveries, good piloting technique dictates that advantage be taken of the small favorable effect of the extended speedbrake.

EASILY REMEMBERED RULES

The best recovery technique for other dive angles was found to be the same as that for the 70 degree dive, although with proportionally varying altitude loss. Based on this analysis, we have developed some "easily remembered rules" for low altitude dive recovery. The techniques vary between low speed and high speed primarily in the thrust used. The decision speed is biased to the high side to insure that pilots select maximum power for the critical low airspeed recoveries.

RULE NO. 1 — DON'T WAIT

- At airspeeds below 350 KCAS, select full afterburner immediately, while at the same time applying full aft stick (or the recommended maximum load factor).
- At airspeeds above 350 KCAS, select idle power immediately, while at the same time applying the recommended maximum load factor.

RULE NO. 2 — DON'T FORGET RULE NO. 1

FIGURE 2 — ALTITUDE LOSS DURING PULL-OUT
Initial Dive Angle — 70°

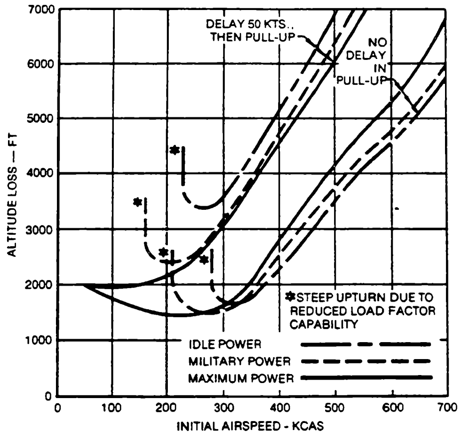
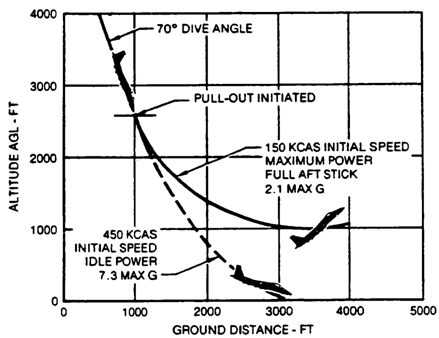


FIGURE 3 — F-15 DIVE RECOVERY



Pilot to Pilot...

(PUBLISHED 1976)

Some F-15 Observations

After recent air show demonstration flight in F-15, Chief Experimental Test Pilot Pat Henry signs autographs for kids in audience.

By PAT HENRY/*Chief Experimental Test Pilot*

Recently promoted to the position of company Chief Experimental Test Pilot, Pat Henry now spends about equal time at the controls of neat new airplanes and big steel desks. Somehow, he has found time at the latter to write about experiences in the former; and here is the first of what we hope will be a long-running series on a McDonnell airplane some of you may have heard about recently...the F-15 Eagle.

Mr. Henry took the Eagle to Farnborough a couple months ago; has more than 560 hours in the F-15 as one of the principal flight evaluators (primarily in the engine test program); and probably knows as much about flying this new machine as any pilot in the rapidly growing list of "Eagle Drivers" (on which list he is Driver #4). In this initial "pilot-to-pilot" presentation, Pat talks first about some new boxes and bits effective with Blocks 12 and 13 airplanes, and then moves on to a few random thoughts and observations about some specific systems.

Blocks 12 & 13 Operational Changes

Some of you may remember "way back in the F-4 days" when we started reporting in the DIGEST on significant changes to the Phantom with respect to the "block" of airplanes in which the changes were first incorporated. That seemed to be pretty helpful information, so it's about time we took a similar approach to Eagle mods. A quick look at page vii of your Dash One will tell you that airplanes in Blocks 12 and 13 are the latest #'s in the field. While most of Block 12/13 changes are maintenance and reliability oriented, a few have operational implications and I'll touch on some of them briefly here.

Yaw Rate Warning — The most significant change is the Departure Warning and Control modification. You'll know it's in your airplane if, when holding the takeoff trim button down, you hear a 900 Hz tone interrupted at approximately 6 cps when trim position is reached. This is the "departure warning tone" that we hope you never hear inflight. It's programmed to come on when yaw rate reaches 30°/second. The interruption rate starts at one cps and increases progressively to ten cps at a yaw rate of 60°/second. Flight tests indicate that if the controls are neutralized at the 30°/second yaw rate warning, the aircraft will recover from the departure. If you, and/or the aircraft, don't change your ways, yaw rate can increase to beyond 60°/second. At this point, full lateral control becomes available, regardless of longitudinal stick position, thereby enabling you to fly out of the worst possible out-of-control conditions — even flat spins.

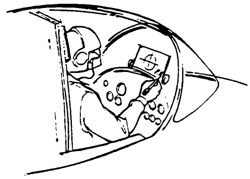
By the way, with this yaw rate warning installed, there is no longer an angle-of-attack warning in the gear-up configuration. With the gear handle down, an aural warning is still present, starting at approximately 28.4 cockpit units AOA. In order to demonstrate this landing configuration AOA warning without lowering the gear, or to test it on a functional check flight, all you have to do is pull the landing gear circuit breaker and pull into 28-29 units. Electrically, you have told the warning tone logic that the gear handle is down (by removing the gear handle "up" signal, right?). If you then reset the circuit breaker the tone will be terminated. If you prefer having an AOA tone, say during ACM to give you an indication of approaching 30 units, then pulling the landing gear circuit breaker will do the trick.

Some Other Warnings — A couple of other new warnings are now available to caution the pilot about degraded control authority; conditions which could easily go unnoticed during certain types of missions, but bite you in other parts of the envelope. At high supersonic Mach numbers, roll rate should be automatically limited to approximately 120°/second. If the roll authority is not reduced as programmed when supersonic, the Roll Ratio warning light will so advise the pilot if he is above 1.5 Mach.

Similarly, the Pitch Ratio warning light is now being used to warn the pilot when the pitch ratio versus airspeed is significantly off schedule. Two basic situations or conditions will trigger this warning. The first one is pressure altitude below 20K, airspeed above 330 KIAS, and pitch ratio above 0.9. Obviously, this is a pretty gross error, thereby suggesting a major pitch ratio malfunction. The second case addresses the other end of the envelope — landing gear down (nose gear) and pitch ratio less than 0.9. It is possible to get a brief PITCH RATIO warning with a completely normal system if the landing gear is extended above approximately 260 KCAS. In this situation, the warning merely invites your attention to the pitch ratio indicator, which should show the ratio steadily increasing as you decel for landing. The caution is advisory only — pilot response is the same as at present.

Obviously less critical is the Auto PLT caution light. It merely advises you that the Attitude Hold and/or Altitude Hold modes of the AFCS have dropped off the line (along with Heading Hold which was dropped off about a year before first flight). The light is reset through the Master Caution light button.

HUD — For those of you who want to take credit for masterful inflight movies, the HUD film titler mod now allows you to enter your own personal pilot code, along with other mundane data, on the HUD. If you're carrying



multiple reels of HUD film, you might want to consider titling them all briefly in advance, unless you're an inflight whiz at the INS keyboard.

Radar — Lastly, the radar software program has been changed to provide improved operational and BIT capability. The new logic should help prevent JAM/HOJ indications during track at long ranges, during target maneuvers, and at the gimbal limits with a simulated missile in flight. The JAM indication in search has been eliminated. Also incorporated are minor changes to track logic for ECM conditions. Most noticeable are the BIT changes.

The BIT matrix can now be read out both in the air and on the ground. Airborne-initiated BIT is nearly identical to ground-initiated BIT; only those tests which would logically fail in an airborne environment (such as antenna drift tests and some portions of scan and roll rate tests) are bypassed. During BIT matrix readout on the ground, G-TEST will be displayed for approximately three seconds immediately upon selecting standby-initiated BIT. This display can also be used for weight-on-wheels verification. Three new codes will be seen frequently: (1) 12–D indicates ground BIT was initiated; (2) 12–B indicates an airborne BIT was initiated; and (3) 12–F shows Maintenance that BIT was initiated in both situations.

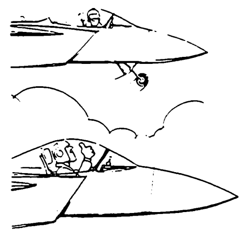
Emergency Landing Gear System

We have a lot of confidence in the emergency gear extension system here in St. Louis; and to verify system performance, we exercise it on every production acceptance test cycle. Our procedure is to extend the gear by the emergency system with the normal handle in the gear-up position and the circuit breaker in. This demonstrates system capability to overcome normal hydraulic and electrical gear-up commands.

Extension times average about five seconds; anything over 30 seconds is indicative of excessive system friction. Before resetting the emergency system and raising the gear through the normal system, it is advisable to check for a JFS Low warning light. If it comes on and stays, indicating the JFS accumulator is not recharging, you wouldn't want to then bring the gear up due to lack of hydraulic muscle for a later emergency extension if needed.

The emergency extension system is one in which previous aircraft experience, such as in F-4's, could be detrimental. As you probably know, the response to an unsafe gear indication in the F-4 is to use the emergency system as a back-up. If this logic is applied directly to the F-15, you could actually be hurting your cause. For example, assume you've attempted lowering the gear normally and that one gear does not indicate down and locked. Your wingman reports that all three appear down, but to be on the safe side, you go ahead and pull the emergency handle as a back-up. What you're actually doing is relieving all the hydraulic pressure that is holding the gear in the down direction, and, therefore, increasing your chances of a gear collapsing during landing. Only the downsprings are left to hold the overcenter mechanism in place. That is why, in fact, in the F-15 we recommend using the normal gear system and utility pressure to back up an emergency extension.

(If any of you know of other situations wherein seemingly similar system designs from different aircraft could lead to incorrect or dangerous procedures when habit patterns are carried over, please write or phone in. We'd welcome the opportunity to look into them and report findings via the DIGEST for the benefit of all.)

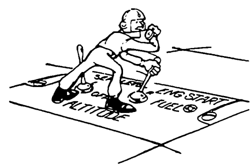


Engine Start Fuel

Let me now call your attention to the "infamous" Engine Start Fuel panel on the aft right-hand console. As promised in an earlier DIGEST article (Vol. 23, No. 1), the labeling of this panel will be changed to a more logical "High," "Auto," and "Low," referenced to engine fuel flow. The incorporation is now scheduled for F-160/TF-26 and up, plus retrofit for earlier aircraft.

In the meantime, it looks like your chances for getting some practice with the existing panel are increasing. We are now seeing that with wear, the Unified Fuel Control (UFC) tends to shift its starting fuel schedule to the lean side. The good news is that this is in a direction away from hot starts. The bad news is that if allowed to proceed too far, the very lean starting schedule might seriously extend air-start spool-up times or even preclude getting a light-off altogether.

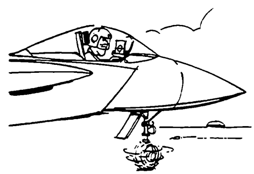
Interim Operational Supplement 1F-15A-13-32 gives the pilot authority



to override the automatic starting fuel derichment system to clear a hung ground start. This is a very practical interim solution, but McDonnell and Pratt & Whitney agree on one limitation to the practice. To help judge just how lean a fuel schedule you can live with, we recommend limiting the use of the "Sea Level" position to 15 seconds or until 60% RPM is obtained, whichever comes first. Greater than 15 seconds worth of enrichment to reach 60% represents a dangerously lean fuel schedule and an unacceptable impact on air-starting. Data to substantiate and quantify this is being collected, and this recommendation will probably be modified in the near future.

Emergency Nose Gear Steering

People tend to believe the first side they hear of a given story, and perhaps that explains the tenacity of certain erroneous information. A case in point is the F-15 emergency nose gear steering (NGS). The idea that the emergency NGS selection locks you into the maneuver mode (±45°) keeps popping up and trying to wriggle its way into the Dash One. When the Emergency Brake/Steering handle is pulled, JFS accumulator pressure is ported to one side of the NGS shuttle valve. As the valve moves, it first blocks off the Utility A flow to the nose gear steering, then opens a path for JFS accumulator flow to drive the NGS. The NGS mode, be it normal (low authority) or maneuver (high authority), is selected electrically and has nothing to do with the hydraulic source.

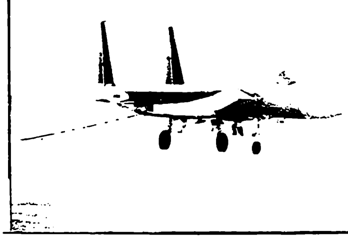


If emergency NGS is selected with good Utility A pressure, there is no way for the pilot to know if the shuttle valve has moved and, therefore, which hydraulic source is powering the NGS. By holding the paddle switch down in this situation, you electrically shut off Utility A pressure to the shuttle valve, thereby eliminating any command conflict between utility system pressure and the JFS accumulator. This assures full travel of the shuttle valve to the emergency side, and is the basis for the handbook recommendation to depress the paddle switch if NGS is not regained after pulling the Emergency Brake/Steering handle.

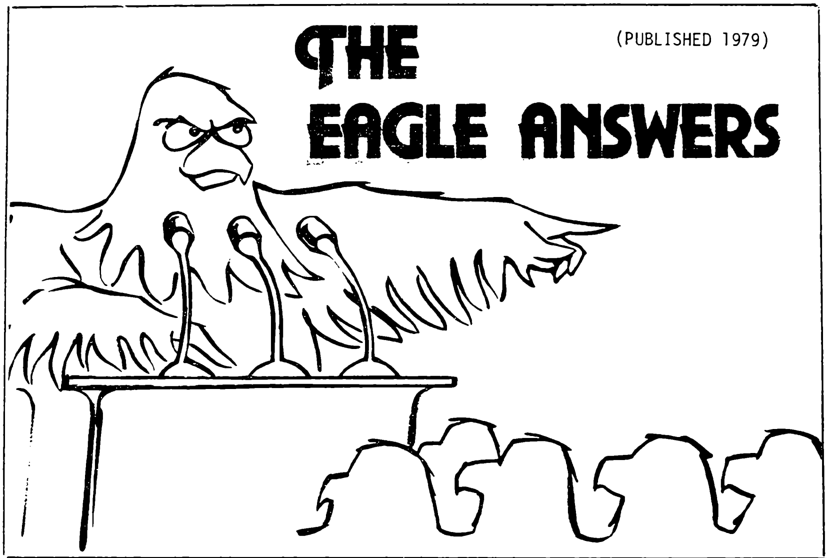
Many pilots like to get much deeper into system design and operation than is possible through the Dash One alone. In closing, here is a list of pilot orientation manuals that can offer significant depth into particular systems. These manuals are available through your local McDonnell Field Service representative.

If other systems are of interest, or if you have particular questions about any given system, please let us hear from you. Hopefully, we can offer clarification in later DIGEST articles.

PS 921 — F/TF-15A Radar System
PS 926 — F/TF-15A Inertial Navigation System
PS 929 — F/TF-15A Head-Up Display System
PS 930 — F/TF-15A Armament Control Set
PS 931 — F/TF-15A Flight Control System
PS 941 — F/TF-15A Navigation Systems (ADI Mode)



EAGLE. Eight world-class Time-to-Climb records in just six flights. Five of the records set in a single day. Three of the records set in a single flight. Previous records beaten by as much as 28 percent. Such was the performance of the USAF F-15 at Grand Forks Air Force Base, North Dakota early in 1975. The Eagle broke five lower altitude records held by the Phantom and the 20, 25, and 30,000 meter marks set in 1973 by the MiG-25 Foxbat. Majors Roger Smith and Willard Macfarlane each set three records and Major David Peterson two. The 30,000 meter climb (98,425 feet) by Major Smith required just 207.80 seconds, after acceleration to Mach 1.1 in 56 seconds! "PROJECT STREAK EAGLE" - a perfect blend of man and machine in demonstration of air superiority.



(PUBLISHED 1979)

By PAT HENRY/*Chief Experimental Test Pilot*

C/D FUEL SYSTEM CHANGES

The external tank pressurization signal for the F-15 C/Ds is gear-handle-up versus weight-off-wheels in the F-15 A/Bs. Why was this change made, and doesn't it create some possible operational problems such as inability to transfer or dump fuel with the gear down?

The C/D fuel system provides two major changes for improved safety. The first is a change to the dump and vent system; the second is the use of the gear handle to initiate/terminate external tank pressurization.

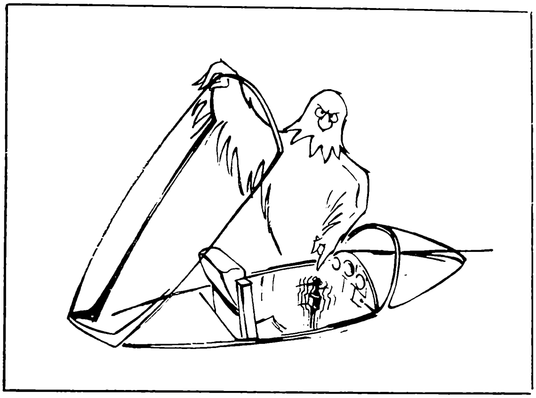
The fuel dump mod now allows fuel to be dumped from the R/H wing vent only, thereby reserving the L/H wing vent for relief of any fuel cell overpressurization. The dual vent dumping (F-15 A/B system) can get you in trouble in a hurry if, for example, you're cruising along with a stuck-open pressure regulator (which you wouldn't know) and then decide to dump fuel. The exiting fuel will almost immediately fill up the dump/vent lines that were being used to relieve the overpressure, and thus the internal cell pressure will start to climb faster than a lightweight Eagle. The result could be a ruptured fuel cell, as has happened once already; obviously a potentially catastrophic event. You can readily see why the decision was unanimous to retrofit F-15 A/Bs to this configuration. The modest reduction in fuel dump rate is a small price to pay for this added safety. Single side dumping is approximately 900 pounds/minute - still not too shabby.

The other change is also the product of safety reviews, but the failure mode impact is not as dramatic; and accordingly, there are no plans for F-15 A/B retrofit at present. The reason for switching to the landing gear handle is to allow depressurization to commence well before touchdown, thus ensuring external tank pressure is completely vented at touchdown. This has obvious advantages for the approach-end engagement or unsafe landing gear scenario, because tank venting is far from instantaneous. However, nothing comes totally without strings. The cost of this change (as you no doubt suspect from the wording of your question) is some increased operational complexity under certain circumstances.

Without pressure to the external tanks, neither transfer nor dump is possible. Therefore, if you're grinding around with the gear down (for example, reported blown tire on takeoff, landing gear retraction problems, etc.), you'll have to trick the system to accomplish either transfer or dump. Our recommended procedure is to pull the emergency landing gear handle, which removes hydraulic pressure to the landing gear and doors. The normal gear handle can then be raised, satisfying the electrical logic to pressurize the tanks, and allowing you to transfer/dump as required.

Once you've achieved the desired external fuel status, the monkey is on your back to *first* return the normal handle to the down position, and then reset the emergency handle. Pulling the landing gear circuit breaker and then raising the gear handle will *not* do the pressurization trick, because that also removes power from the gear handle switch.

A closing reminder: in the event that you have to, or choose to, land with fuel remaining in the external tanks, fuel slosh can significantly affect handling qualities. There are no baffles in the external tanks; so as the aircraft is flown to progressively higher pitch attitudes (for landing or aero braking), the fuel migrating aft can cause up to a 2% shift of the c.g. position - great for increased pitch rate, but not recommended for rock-steady approaches and landings. For most situations, the CAS will probably mask the potentially varying pitch response, but a pitch CAS dropoff will probably demand significantly increased pilot attention to avoid AOA overshoots and/or tail scraping. Worst fuel loading for adverse c.g. shift due to fuel migration is about half fuel in the drops. The worst combination of aft c.g. position plus c.g. shift occurs with about half fuel in the drops and minimum internal fuel, so burning down to the very last drop is not going to help your cause.



CONTROL STICK OSCILLATIONS

During preflight checks, I have noticed that the stick will oscillate about center for several cycles when released from near full aft or forward. Why is this, and what is considered normal with regard to magnitude and duration of oscillations?

There is no damping mechanism in the longitudinal (pitch) axis of the mechanical control system. An eddy current damper was added to the lateral axis as a result of flight tests that uncovered a tendency for sustained lateral stick oscillations at some flight conditions, particularly in two-seater Eagles. Therefore, the control stick will oscillate for several cycles (approximately 2-3 seconds) if displaced and released in the pitch axis, but should damp out with only one small overshoot in the lateral case. This inherent characteristic of the control system can be demonstrated just as readily inflight; but normally, this sort of stick rap input is only done during lateral and longitudinal stick-free dynamic stability testing.

STABILATOR CHATTER

From time to time, I have noticed vibrations through the airplane when doing preflight control checks. Is this the "stabilator chatter" I have heard about, and if so, what are considered the safe and acceptable limits?

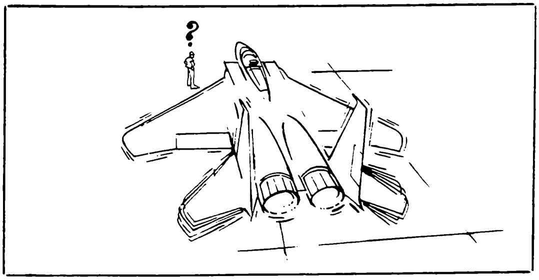
"Stabilator Chatter" is a phenomenon that is inherent to the F-15 and its control system. An understanding of its characteristics and causes will enable Eagle Drivers to correctly distinguish between "chatter" and actual control loop malfunctions, thereby eliminating many unnecessary mission aborts and aircraft squawks.

While the degree of chatter may vary widely between aircraft and even from day to day on a certain aircraft, the frequency of the stabilator oscillations remains approximately 13 hz and feels similar to some forms of inflight high-AOA buffet. The big difference is that in a correctly functioning control system, chatter can occur only on the ground. Therefore, any abnormal inflight vibrations should be reported and fully investigated.

Stabilator chatter develops when the control stick is moved fore and aft at a certain rate, which, coincidentally, is just about the same as the drive rate of the longitudinal trim motor. System friction resulting from this movement generates vibrations which are transmitted through the airframe and picked up by the CAS pitch and roll rate sensors. The CAS, in turn, amplifies the vibrations and tries to drive the control system at that frequency, thus forming a complete, closed-loop cycle.

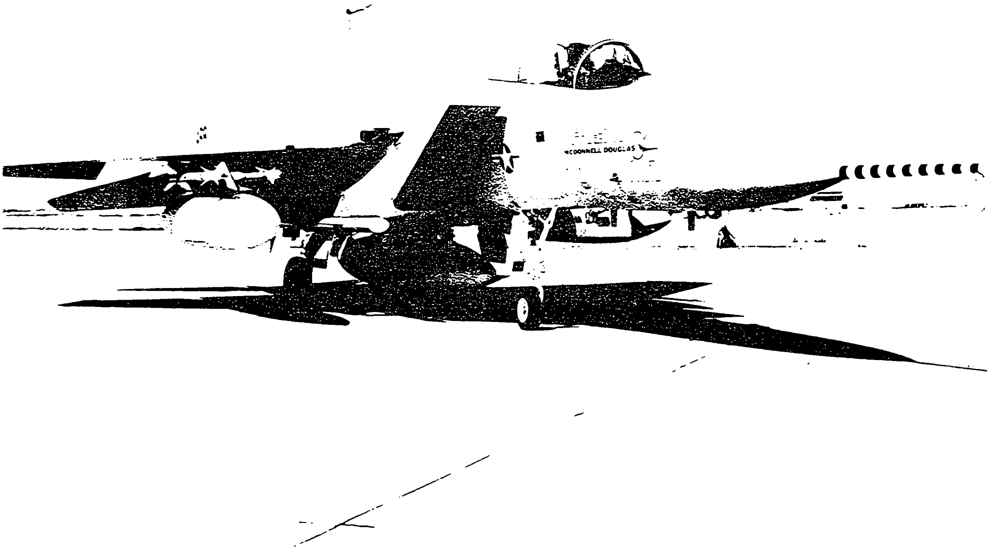
Since the control system vibrations are picked up by both the pitch and roll channels of the CAS, either axis alone is sufficient to support this phenomenon. However, with only one channel on the line, the magnitude is only about a quarter of that produced by both channels together. In view of this, MCAIR was able to demonstrate an acceptable (not total) fix by installing a 10 hz filter in the roll CAS computer. Although the Air Force has not accepted this fix, a renewed interest in it has recently been shown.

As with any closed-loop cycle, if you break the loop, the cycle will stop. With this in mind, you can stop the chatter by one of two methods: either stop the control stick input or turn off the CAS. With either method, the vibrations should die out within the acceptable limits of 3 to 4 seconds. If the vibrations persist and are self-sustaining, then something is definitely amiss and the system should be fully investigated before the machine is flown. ■



F-15C/D Operational Changes

(PUBLISHED 1979)



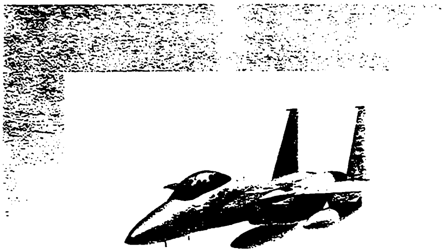
The Eagles currently rolling off the MCAIR assembly line here in St. Louis are all C or D models with two basic differences from the previous A/B models. The changes provide a 2000 pound internal fuel increase and a beefed-up landing gear to handle the 68,000 pound airplane (new max weight with full conformal tanks). Early in April we began the Category 1 test program, which progressed rapidly with no major surprises. We completed the program in May, having verified the designs, so let us pass on what we found with "C Number 1" (USAF 78-0468) during the two months at Edwards AFB.

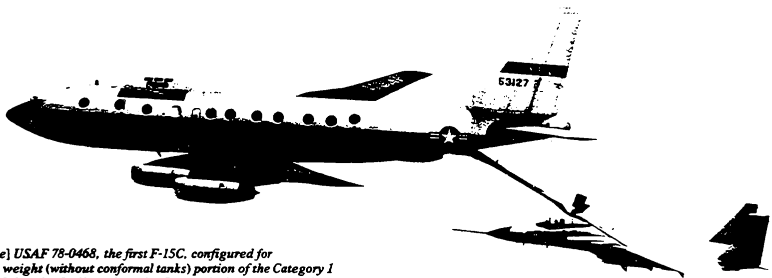
The new fuel arrangement in the C/D can really give you long legs. With internal fuel only, you should see 13,400 pounds on the gage at engine start. And when I dropped off the tanker in C-1 with three externals, I was looking at 25,300 pounds on the gage! Add another 10,000 pounds of conformal tank fuel, and you can fly the Atlantic unrefueled, as was demonstrated in 1974. (Provisions for the conformal tanks — wiring, plumbing, quantity gage, fuel panel switches — are all in the airplane now, although the Air Force has not yet contracted for manufacture or delivery of those tanks.) In case you missed it, the Block 21 story in the last DIGEST described how the engineers squeezed all this extra fuel into the new Eagles.

Two other changes in the fuel system are (a) dump now comes out of the right mast only, and (b) improved dead-banded fuel shut-off valves in the feed tanks should keep internal wing tanks within 200 pounds of each other.

The use of dead-banded fuel shut-off valves will also help the external wing tanks to feed more evenly, although some mismatch will still remain. The dump system change is to

By DEE FRANCIS/*Chief Production Test Pilot* and FRED CHANA/*Lead Engineer, Flight Test*





[previous page] USAF 78-0468, the first F-15C, configured for the maximum weight (without conformal tanks) portion of the Category 1 test program. [above] "C-1" conducted inflight refueling to test the fuel system operation (transfer rates, fill-up times, sequencing, etc); and to "top off the tanks" prior to commencing the fuel transfer vs high fuel flow rate test. [inset] F-15B 71-291, shown during its 1974 unrefueled, nonstop flight from Loring AFB, Maine to Bentwaters AB, England. In this configuration, F-15C/Ds would show 35,300 pounds on the gage!

provide the capability for the tanks to vent through the left mast while dumping through the right mast, thus reducing the posibility of overpressurizing the internal tanks in the event of a pressure regulator failure. Any dumping, not just venting, from the left mast should be a signal to terminate dumping.

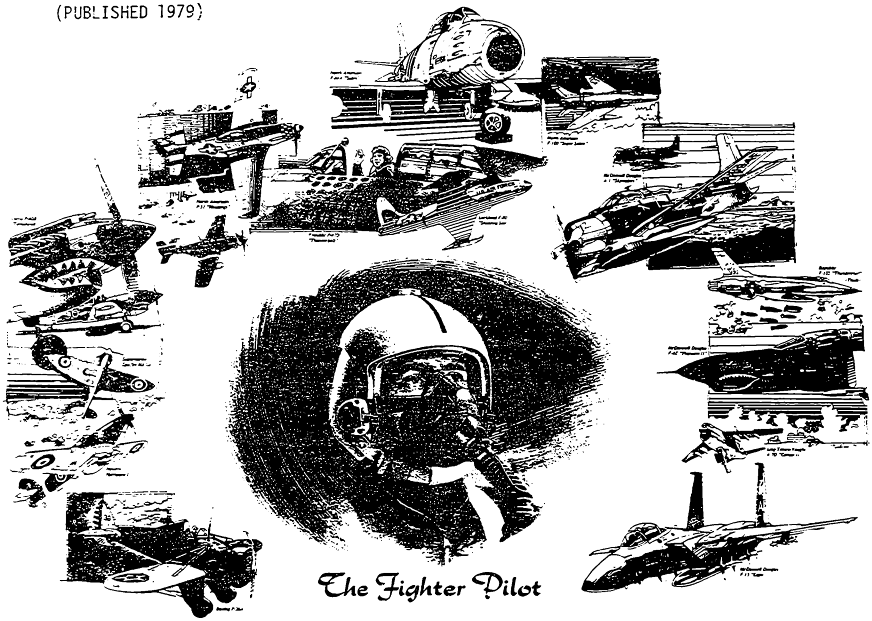
One operational change to emphasize is that external fuel tanks are now pressurized with the gear handle up, not with the weight-off wheels switch as in the A/B models. For the pilot that can't get the gear up and puts the handle back down, he's now in Section 3 of the Dash One with the "external-tank-fails-to-transfer" situation. External fuel will eventually transfer when the fuel low light comes on, but Steps 5 through 8 still apply.

The new brakes, tires and wheels will give you a few ground handling characteristics different from those you were familiar with in the A's and B's. The main difference is a longer break-in period until they are "burned in" and an apparent reduction in the static (engine run-up) holding ability, which has caused a few flight aborts and pilot squawks with the first 15 to 20 C's and D's. The stopping distance, however, is very similar to the A's and B's once the brakes are broken in, both for wet and dry runways.

One last point about C.G. and handling qualities. More fuel in Tank 1 should provide you a slightly more forward C.G. (by only 0.2%) when Tank 1 is full. But since the flight controls can give you the G you command by stick position, you may not even detect the change.

All in all, we doubt if you'll see any real differences when moving from an A/B to a C/D cockpit, but that great big plus of 2000 more pounds of fuel ought to impress you considerably. ■

(PUBLISHED 1979)



The Fighter Pilot

Say what you will about him arrogant, cocky, boisterous,
and a fun-loving fool to boot he has earned his place in the sun.
Across the span of 50 years he has given this country some of its proudest moments
and most cherished military traditions. But fame is short-lived and little the world remembers.
Almost forgotten are the 1,400 fighter pilots who stood alone against the might of Hitler's Germany
during the dark summer of 1940 and in the words of Sir Winston Churchill, gave England "its finest hour."
Gone from the hardstands of Duxford are the 51's, with their checkerboard noses,
that terrorized the finest fighter squadrons the Luftwaffe had. Dimly remembered the 4th Fighter Group
that gave Americans some of their few proud moments in the skies over Korea.
How fresh in recall are the air commandos who valiantly struck the VC with their aging "Skyraiders"
in the rain-and blood-soaked valley called A Shau?
And how long will be remembered the "thuds" over "route pack six" and the flak-filled skies above Hanoi?
So here's a "nickel on the grass" to you, my friend, for your spirit, enthusiasm, sacrifice, and courage
but most of all to your friendship.
Yours is a dying breed and when you are gone,
the world will be a lesser place.

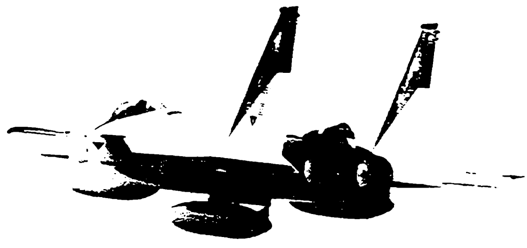
Friar Tuck

Raymond B. Tucker, Lt. Colonel
United States Air Force

Poster by Boeing Aerospace Company/Seattle, Washington; Flight Crewman Illustration by Chuck Wood, McDonnell Aircraft Company

AIRCRAFT PERFORMANCE ►

(PUBLISHED 1975)



F-15 Spin Tests

By JACK KRINGS/*Project Experimental Pilot*

For quite some time, McDonnell test pilot Jack Krings has been involved in an exhaustive study of stall and spin potentials of the F-15 Eagle. Jack presented the results of those studies in a paper to the Society of Experimental Test Pilots [SETP] at their 19th Annual Symposium in Los Angeles in September of this year, and portions of his paper are being included here in the DIGEST. Because of the length of Jack's presentation, his material is being published in two parts, the first covering the program plans, preparations, and test guidelines. In our next issue, he will offer results and conclusions on this major aircraft evaluation program.

The goal of the F-15 High Angle-of-Attack Flight Test Program was to explore, understand, and recover from any and all out-of-control conditions anticipated during service use of the airplane. The plan was to progress from a 1 g stall to wherever the airplane behavior took us in logical, conservative steps in which operational use and recoverability were primary considerations. We made 141 test flights and conducted some 811 stall/post-stall maneuvers in determining that, in its primary role configuration, the F-15 has no angle-of-attack limits. The recoverability of the Eagle allowed us to explore high AOA flight from stall to the steady flat spin systematically, logically . . . and safely. I hope you'll be interested in my brief account of how it all went together.

PROGRAM PLAN

A number of years ago, we started formulating the philosophy for the F-15 high angle-of-attack tests. I had the unique opportunity to be personally involved in all phases of planning and testing. Our aero prediction was no spins; however, contractually, we were committed to specific spin recovery criteria in the event spins were encountered.

The general philosophy was that we would progressively investigate the flight characteristics from approach to stall, through sustained critical aggravated control inputs for up to 15 seconds. The airplane behavior would determine subsequent maneuvers, configurations, and recovery requirements. Considerable flexibility was incorporated to allow exploration of parallel areas of interest, should a limiting characteristic be encountered. Exploration of variables in flight conditions, maneuvers, loadings, and airplane configuration was carefully planned. Each test point was subject to change, addition, or deletion as testing progressed and better planning information became available.

With dominant control to be retained by McDonnell, no military participation flights were included in the Category I phase. Category II (Air Force) tests were to be conducted during a six week period after contractor testing was complete. A backup contractor pilot and an Air Force project pilot were designated and each received comparable preprogram training. Simulation was a large part of the plan. The tests were to be conducted at Edwards AFB using elaborate instrumentation and tracking.

Incidentally, during the flying qualities tests, F-15 Number 1 entered an unintentional spin from a full aft stick transonic windup turn with speed brake extended. Both engines stagnated, aerodynamic recovery was successful, and the "new" goals of the spin program were obvious. Denny Behm, the back-up spin pilot, was flying the test, and the training and simulator paid off.

PILOT CONSIDERATIONS

For a program like this, we knew we needed to develop and practice various test procedures. Therefore, I reacquainted myself with out-of-control and spin maneuvers; we surveyed other companies with recent high AOA test programs; and a non-MCAIR committee was formed to review our test plans and approach. Some of the major considerations we developed were as follows:

- Predicted out-of-control and spin recovery procedures were established analytically and were practiced in the MCAIR fixed-base simulator, which was configured with a cockpit as nearly identical to the flight test airplane as possible.
- Low airspeed engine air start procedures and envelopes were defined by flight test on the propulsion system test airplane.
- Engines-out glide procedures and the flame-out approach profiles were developed in simulated tests with Idle power on the test airplane and also on the MCAIR simulator.
- Spin recovery parachute deployment criteria were established based on wind tunnel studies of out-of-control flight. Criteria used throughout the program were: aircraft out-of-control passing through 22,000 feet - DEPLOY CHUTE.
- Pilot ejection criteria were established based on recognized safe minimum ejection altitude. Criteria used throughout the test program were: aircraft out-of-control passing through 15,000 feet with no recovery trend apparent - EJECT.

SCALE MODEL TESTS

NASA tested a 3/8 scale F-15 in the high angle-of-attack regime (this Remotely Piloted Research Vehicle was discussed in detail in the 3rd Quarter 1974 DIGEST). Launched from a B-52, the scale model was remotely flown by a NASA test pilot (usually Einar Enevoldson), and was put through a series of high angle-of-attack maneuvers, including flat spins. We followed the program very closely, and significant information was transferred in both directions. The model program was very valuable and was well-conducted.

SPIN TEST AIRPLANE

The eighth F-15 was designated as the spin airplane, and significant modifications were incorporated during manufacture.

Spin Recovery Chute - Though the F-15 is not normally equipped with a drag chute, our test aircraft was provided with an emergency mortar-fired spin-recovery parachute. Based on previous spin programs, I had three goals in addition to the obvious rquirements:

- The chute was to be permanently attached to the airplane.
- A control was to be designed that could not be improperly operated.
- The number of mechanical parts was to be minimized.

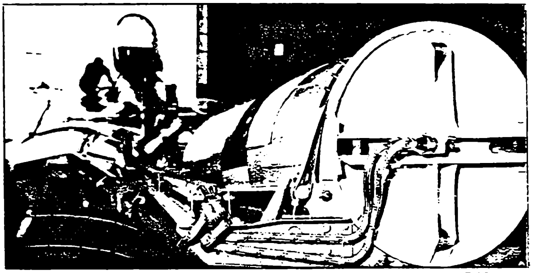
We came up with a design that had no moving parts except for the control handle; it was a large "T" handle with IN (jettison) and OUT (deploy) positions. With the handle in, the jettison circuit was armed; pulling the "chute handle" deployed the chute pyrotechnically. This circuit provided for chute jettison should the chute container move without the pilot selecting OUT, thus solving the accidental deployment problem during takeoffs and landings.

Redundant systems and power supplies ensured reliability, and a mortar-deployed "yankee" tractor rocket extracted the chute. The chute was jettisoned by rotating the chute handle 90 degrees and pushing it back in. It appears that we came up with a chute control that was impossible to operate improperly; it worked during the ground and inflight proof tests, and once in anger.

Emergency Power - Pessimistically forecasting dual engine losses, we installed batteries for instrumentation power and a monofuel hydrazine power unit for hydraulics and aircraft electrical power. The hydrazine unit was replaced by a battery system after it was found to be more hazardous than the spin tests. The battery system performed well throughout the program, though it was never actually needed for emergency use. The emergency power was designed to come on automatically when the second engine (generator) was lost.

Cockpit Displays and Controls - I wanted as little cockpit instrumentation as possible, to more nearly approximate an operational situation. Along with the previously - described "chute handle," we added:

- Control surface position indicators
- Angle-of-attack, sideslip, and yaw rate indicators
- Spin direction lights
- Spin telelight panel - used as a checklist (the green lights on the telepanel constituted the pre-maneuver checklist; any significant failure or emergency system operation would illuminate an amber or red telepanel light.)
- Emergency Power switch
- Instrumentation controls
- Emergency Jettison switch for the recovery chute.



A mortar-deployed spin-recovery chute was specially installed in the eighth F-15.

Restraints - The program began with use of the F-15 production restraint system, an F-4 torso harness (a marginally acceptable restraint system at best). The following restraints were added later in the program:

- A torso strap (eyeballs out g restraint).
- A crotch strap (negative g restraint).

TESTS

After successful ground and inflight recovery chute deployment tests, and replacement of the monofuel hydrazine power unit with a battery system, we got underway with the actual spin tests.

Stalls - Early 1 g stalls produced divergent dutch roll. During the battery layup, we modified the wing tips to ▶

match the production configuration which unexpectedly eliminated the divergence.

Stall was arbitrarily defined as trim angle-of-attack at full aft stick. Variations in rate, control configuration, and altitude were systematically explored. Adverse control inputs at stall were added and full pro-spin controls were held for nearly a minute.

Accelerated stalls (+7 to -2.5g) were systematically explored in the same manner. Vertical stalls were explored. Full pro-spin controls maintained in backward flight (180°α 180°β) produced some rather sensational maneuvers. I didn't get quite enough backward flight time to learn to "fly backwards." The airplane wants to recover, and I was unable to overcome the natural airplane stability.

Power Approach (PA) stalls were explored including 1 g and accelerated stalls with prolonged pro-spin controls.

Departures - These were pursued in all configurations with every entry technique we could envision except intentional coupled entries.

Spins - Though we achieved an unintentional spin, we were able to identify the causes of the incident, prove their effects, and reduce the potential of operational spins due to these factors. We found that the airplane was very difficult to spin.

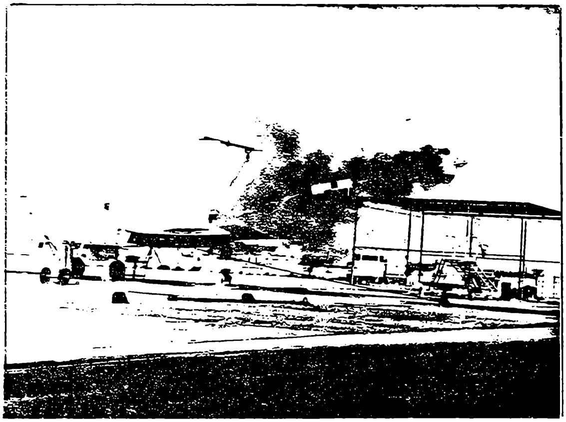
Our greatest success in developing a spin came through use of a technique that was occasionally successful in the simulator and on the NASA model. While performing a "split-S" maneuver at high altitude, full rudder was introduced. This was followed by full aft stick to stall, then the stick was nimbly positioned for full aileron deflection.

Classical pro-spin control will not produce the necessary aileron deflection since full aileron is only available at near-neutral longitudinal stick positions. This is an inherent design in the control system to prevent aileron deflection at high angle of attack.

Once I became adept at control manipulation (defeating the control system anti-spin design), fifty percent of the attempts would produce a spin at susceptible flight conditions. Later on, other entry techniques were perfected. Four pilots have attempted intentional spins and each took practice, coaching, and several tries to get a spin. Maximum RPM spins were attained and spin RPM was controllable with aileron deflection. Effects of individual, multiple, and sequential control application during entries, spins, and recoveries were explored. Spins were unattainable in any symmetrical combat configuration below 35,000 feet even using the most probable spin entry technique. Spins from rudder rolls with prolonged F-15 pro-spin unique controls were developed.

Spins were unsuccessfully attempted from vertical stalls, 1 g stalls, and abrupt symmetrical pullups.

Inverted spins were repeatable with rudder deflection (rudder pedal or lateral stick deflection) at full forward stick stalls. A PA stall produced a PA spin during prolonged pro-spin control application.


The Spin Recovery Chute System was ground fired three times to verify proper operation of each sequence in parachute deployment and jettison.

Auto-Rolls - Sustained rolling motion was attainable at angles-of-attack below stall. A reasonably elusive technique was evolved to produce sustained auto-rolls.

Configurations - The aircraft was tested in a variety of configurations.

- Symmetrical external missiles and tanks were tested in stalls, departures, and spins. Symmetrical external stores to maximum allowable gross weight were tested through departure.
- Asymmetric configurations proved to be an interesting part of the program. A small difference in internal wing fuel will determine which way the airplane tends to yaw at stall. Lateral CG was most important in evaluating high angle of attack flight characteristics. I don't think we ever calculated lateral CG previously except for very large external asymmetric configurations. Control of internal wing fuel allowed a thorough evaluation of asymmetry. Asymmetric configurations up to about 8,000 foot-pounds (two AIM-7's and two AIM-9's on one side) were stalled, departed, and spun. Stall and departure susceptibility was evaluated up to one full external wing tank (40,000 foot-pounds).
- Power approach configuration stalls, departures, and spins were evaluated. The only use of the emergency recovery chute occurred when a PA spin would not recover within the altitude limits we placed on the program. Subsequent tests explained this problem.

Variables - Extensive evaluation was made of departure or spin susceptibility relative to speed brake, engines, control system, CG, and flight conditions. Here is what was accomplished:

- A simple, automatic speed brake retraction system was designed, developed, and proven.
- Stalls, departures, and spins were performed at Idle power, Military power, and maximum A/B. Our pessimistic approach to engine operation was unwarranted; the engines performed exceptionally well, including upright and inverted spins.
- Exhaustive combinations of control augmentation configurations were tested. Center of Gravity locations beyond allowable and static margin approaching zero were tested. Flight conditions included spin attempts from -50 knots to 1.2 IMN and 25,000 to 50,000 feet. Stalls were performed down to 5,000 feet. ■

(PUBLISHED 1975)

Spin Test Data Program

By DEREK WALKER/*Group Engineer - Flight Test*

A month before Jack Krings told SETP in Los Angeles about the operational aspects of the F-15 spin test program, Mr. Derek Walker appeared before SFTE (Society of Flight Test Engineers) in St. Louis to discuss the engineering aspects, including the plan for development of program data. Therefore, as with Jack's paper to SETP, we have extracted portions of Derek's SFTE presentation to give you a feel for another side of the spin test picture with respect to the accumulation, reduction, and analysis of flight test data.

The area utilized for the F-15 spin test program was located five miles north of Rogers Dry Lake, which afforded direct acces to an excellent emergency landing area. The AFFTC Space Positioning Branch (SPORT) provided radar vectoring into the spin area and also provided television monitoring of the test vehicle. SPORT was briefed to provide key minimum altitude calls during test maneuvers and this service was routinely checked on each flight during climbout after takeoff. High angle-of-attack tests were photographed using ground tracker 35mm cine cameras with long lenses.

The test airplane was equipped with a comprehensive onboard instrumentation system which provided analog and digital recording capabilities on a 14-track magnetic tape recorder. A PCM time division multiplex system (TDMS) and a constant bandwidth frequency division multiplex system (FDMS) were the primary data acquisition systems. The instrumentation system as it was configured for the spin program included 203 measurands which documented all functions of the flight control system as well as providing data relative to airplane motions and flight conditions.

The instrumentation system was controlled from a panel located on the main instrument panel in the cockpit. Over-the-shoulder movie cameras were also utilized to monitor cockpit displays and to provide pilot's-eye footage for use in subsequent training films. The aircraft was fitted with a long flight test noseboom which incorporated expanded range angle-of-attack and sideslip sensors as well as airspeed and altitude pressure sources.

Data that described airplane performance during selected maneuvers was produced overnight using the MCAIR F-15 integrated data system which featured data pre-processing at EAFB, transmission of raw data to St. Louis for computer processing, and retransmission of final data to the test site for printing simultaneous with availability of the data to technical groups in St. Louis. For the most part, data was in the form of time history presentations using formats designed to group significant parameters for ready analysis.

Test flights were also monitored in real time using telemetry. All parameters were transmitted continuously from the test airplane to a ground station manned by the engineering test team. Pilot's voice was also transmitted to provide a continuous mission narative. All flights were chased by AFFTC T-38 aircraft flown by a designated group of pilots identified with the F-15 spin program. A photographer occupied the rear seat on most missions. A MCAIR portable telemetry receiver was designed for use in the chase airplanes to permit the chase pilot to continuously monitor telemetry voice.

Tests were monitored by an engineering team that included a test director, aerodynamics engineer, flight test engineer, data engineer, and instrumentation engineer. Other engineering personnel were present as rquired for special tests, including propulsion and flight control system tests. The following displays were available to the test team in the mission control rooms:

- Four 6-channel strip chart recorders.
- Bargraph display of all PCM parameters.
- X-Y plotter displaying longitudinal vs lateral control stick position.
- Status light panel displaying information regarding control augmentation, instrumentation, and emergency power systems.
- Aural/light alarm system monitoring aircraft and engine over-temperature circuits.
- Dual video display providing a visual presentation of aircraft motions and a simultaneous display of aircraft geographical position and altitude on an adjacent screen. Aircraft motion display, with telemetry voice dubbed in, was recorded on a portable video tape recorder for playback in the post-flight debriefings.

McDonnell Douglas flight test engineers Gary Trippensee (left) and Don Warren at the test director's position in the ground telemetry monitor station. All stations in the control room were linked via hot-mike intercom.

(PUBLISHED 1976)

". . . The One G stall typically exhibits classical buffet and non-divergent Dutch roll stabilizing at 90 to 100 knots, approximately 40° angle of attack, ±15° roll, and ±5° sideslip . . ."

F-15 Spin Tests Results and Conclusions

By JACK KRINGS/*Project Experimental Pilot*

The last issue of the DIGEST featured an introduction to the recent stall, departure, and spin tests of the F-15 at Edwards AFB, California. Having reviewed the total plan and tests, let's take a look now at the results and conclusions.

DEFINITIONS & CHARACTERISTICS

A total of 811 high angle-of-attack maneuvers were conducted during 141 flights in this test program. Testing progressed from an initial evaluation of one g stall characteristics to the performance of simulated air combat maneuvers, without losing our airplane or our composure. Because we were specifically trying to get into spin situations rather than stay out of them (as any normal self-respecting pilot would do), there were a few physically uncomfortable moments and we did resort to the spin recovery chute in one safety-first situation.

The program produced approximately 70 developed spins of several types as described below, but many of our spin attempts were unsuccessful, which is a happy thought now that I look back at it! Holding pro-spin controls after accelerated stalls at high energy produced violent motions which would subside immediately when controls were neutralized. Early unsuccessful spin attempts also produced continuous rolls at less than stall angle of attack. Usually, three to four full rolls occurred, which would subside with neutral controls. When first seen, this was initially interpreted as a spin. Often, a rolling motion accompanies recovery from departures or spins.

Prolonged unsuccessful spin attempts on two occasions resulted in unintentional coupled entries. A -5 g couple to an inverted spin resulted in the only dual engine stagnation of the program. Sequential shutdown and relight precluded emergency system operation. The lake bed never looked so beautiful! A +9 g coupled maneuver also resulted from a 15 second aggravated pro-spin control attempt when the airplane really didn't want to spin.

Since one of the purposes of our program was to provide the pilot with an understanding of the high angle of attack flight capabilities of the F-15, here is a set of definitions regarding stall, departure, and spin characteristics.

Stall-Maximum obtainable angle of attack at full longitudinal stick displacement.

The 1 g stall typically exhibits classical buffet and non-divergent dutch roll stabilizing at 90 to 100 knots,

approximately 40 degrees angle of attack, and $\pm$ 15 degrees roll and $\pm$ 5 degrees sideslip. Instant full aft stick abrupt stalls overshoot at 60 degrees angle of attack and 50 knots. Accelerated stalls ultimately typify 1 g stalls, but higher energy enhances the accelerations produced by the dutch roll. Divergence in yaw rate is noticeable with lateral asymmetry. Inverted stalls were stable at 120 knots and -20 degrees angle of attack. Accelerated inverted stalls can reach -30 degrees angle of attack.

Departure - Uncommanded motion at high angle of attack. Pick your own numbers.

Dynamic (less than one second to the aft stop) accelerated stalls would produce yaw and roll rates we termed departures. The effects of lateral asymmetry were very dominant. Departure always occurred opposite to the asymmetry. We chose to define departures as uncommanded maneuvers up to 30 degrees per second yaw rate. Yaw rate alone will maintain angle of attack above stall. Required yaw rates can only be generated by aileron deflection or asymmetric load. The yaw rate is the key parameter in the progression from stall through spin.

Spin - Uncommanded motion with a sustained direction of yaw having a yaw rate average in excess of 60 degrees per second.

• **Oscillatory** spins are defined as spins with pitch oscillation over approximately 10 degrees. These spins were more violent with significant yaw rate hesitations. They were all self-recoverable when controls were neutralized. There is an academic line somewhere between departures and oscillatory spins. Yaw rates spike occasionally to 100 degrees per second and angle of attack can oscillate to 70 to 80 degrees.

• **Non-oscillatory**, steady spins were developed from 65 to 140 degrees per second yaw rates. Precise timing of entry controls was required when laterally symmetrical. Lateral control (aileron) was effective to increase or decrease yaw rate. This capability allowed a step-by-step progress in spins to essentially maximum RPM with recoverability at each increment of spin rate. At lower yaw rates, the spin recovery trend with anti-spin aileron was sometimes barely discernible, but was ultimately effective. This slow-rate recovery was first encountered during the power approach (PA) spin when the recovery chute was deployed. Subsequent tests reproduced this type spin and successful aerodynamic recovery. The loss of altitude was only 1,000 feet per turn, so, at 35 or 40 thousand feet, there was no immediate concern.

Incremental increases in spin RPM allowed evaluation of spin/recovery characteristics, effects of controls and the tendency to recover or increase RPM with neutral controls. The high RPM spins (above 100 degrees per second) produced from 2 to 3 g's (eyeballs out). These spins were obviously uncomfortable. The torso harness was installed after the first few 2+ g flights. Prolonged pro-spin controls repeatedly produced flat 90+ degrees angle of attack, 120+ degrees per second yaw rate non-oscillatory spins. At least a dozen "flat" spins were performed, all of which recovered positively and repeatedly with full anti-spin aileron. No other control or combination of control deflections enhanced recovery from any spins. Minor variation in spin and recovery characteristics were seen with symmetrical tanks/missiles, CG location, engine power setting, and entry technique variables.

• **Inverted** spins are still distasteful. The inverted spin was easily attained and could be progressively explored. It was found to ultimately stabilize at -35 to -45 degrees angle of attack and 50 to 55 degrees per second yaw rate. The airplane is self-recoverable with neutral controls from inverted spins.

• **Power Approach** spin tests were saved until last, as a result of the earlier PA spin chute recovery. A valiant attempt by some of the faint hearted to retroactively eliminate PA spins from the contract specification was denied. The slow recovery in the first PA spin was reproduced in the clean configuration. The PA spin was revisited successfully.

SUSCEPTIBILITY

Spin susceptibility is extremely low since stall departure susceptibility is low and self-recovery probability is extremely high (all cases tested), even with lateral asymmetry. Our tests did show however, that lateral asymmetry definitely increases departure and spin susceptibility. One AIM-9 and one AIM-7 on the same side will cause departures from accelerated stalls with full aft stick only. These departures are self-recoverable with neutral controls.

Greater asymmetry will produce spins with prolonged full aft stick after abrupt accelerated stalls. In previous spin programs, we were mystified as to why it went different ways on different ▸

days. We never had an airplane as repeatable, controllable, and recoverable as this one to explore and define this sensitive parameter.

The speed brake destabilizes the airplane and increases departure/ spin susceptibility. It doesn't affect spin character or recovery. The Control Augmentation System has little or no effect on out-of-control susceptibility; and it turns off at 40°/second so is out of the picture in spins and during recovery.

The cause of the unintentional spin encountered in No. 1 F-15 became obvious during the program - it had 1000 pounds of internal wing fuel asymmetry and the speed brake extended.

MODIFICATIONS

Four airplane modifications were recommended. One of them was something I've been trying to sell for years - a "Spin Warning Cue" which tells the pilot that he has already departed but is still in the "self-recovery phase." In other words, "Let go, and it will recover itself!" Departure prevention is great if departures are really bad, but let's face it, how do you prevent a departure going straight up at zero degrees angle of attack which instantly changes to 180 degrees angle of attack? The audio warning in the F-15 tells the pilot that all this flopping around the airplane is doing can lead to a spin if he doesn't quit. The audio spin warning starts beeping slowly at 30 degrees per second yaw rate; the interrupt rate increases with yaw rate; and when it is steady (at 60 degrees per second), you are in Spin City. Now you have to figure out which way it is going and put in full aileron to ensure recovery.

Modification Number 2, the "Spin Recovery Aid," tells the control system (at 60 degrees/second yaw rate) to give you full aileron at any longitudinal stick position (we do the same thing with the gear down for better approach handling), essentially removing the anti-spin design of the flight control system. You have somehow out-foxed the anti-spin design (since you are spinning!), so let's make recovery easier. We satisfied ourselves that this mod will not affect spin susceptibility.

The third modification affects the fuel system and keeps wing fuel symmetrical; and the final change auto-retracts the speed brake above 15 degrees angle of attack.

For an airplane that is highly resistant to departures and spins, we seem to be proposing a fair number of out-of-control oriented modifications. However, we must keep in mind that the historic ability of early flight tests to predict future operational talent to depart airplanes has been notoriously poor.

RECOVERABILITY

With the modifications installed, stalls and departures up to the spin warning tone are permissible in the air-to-air configuration. Tests have shown airplane self-recoverability in any air-to-air configuration if controls are neutralized when the spin warning tone comes on.

When that guy comes along with the trick we couldn't find and manages to spin this airplane, the tone will stay on steady; now he must determine direction and apply appropriate aileron. When the aileron has done its thing and the spin breaks, the audio stops; now the controls can be neutralized and welcome to the club! Fairly simple, I'd say.

CATEGORY II

The USAF Category II program focused on things we didn't do in Cat I and three areas were explored: auto-rolls, large lateral asymmetry, and the centerline-tank-only configuration. Non-self-recoverable auto-rolls were generated, and recovery with opposite rudder was repeatedly successful. Pete Winters flew the Category II Air Force program, with Don Wilson directing the tests. Pete chased me a lot and had a unique talent for "creating" chase airplanes when none were available.

Neutral control self-recoverability was demonstrated with maximum lateral asymmetry in the air-to-air configuration. Stall characteristics were evaluated with as much as one full (600 gallon) external wing tank. Investigation during Category II of the centerline tank effects indicated that it significantly increased departure susceptibility but did not affect neutral control self-recovery up to spin warning tone yaw rates.

OPERATIONAL CONFIGURATION

One flight was flown with the production system; all flight test indicators were masked; and all mods were installed, approximating the production configuration. Multiple stalls, departures, and spins were performed and recovered. I would personally have no reluctance to repeat such a flight without the emergency equipment.

CONCLUSIONS

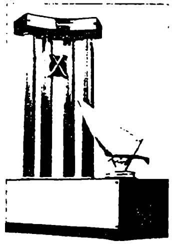
Let's start with a prayer of thanksgiving - the airplane is still intact, and no irrecoverable mode was found.

Here are the key discoveries that were made:

- The airplane is essentially unrestricted.
- The operational pilot has a cue to identify how best to recover and confidence that it will.
- All angles of attack and sideslip (0 to 180 degrees) were achieved without incident.
- The engines performed beyond expectations.
- I think we fully explored all operationally achievable out-of-control maneuvers.
- We recognize our poor track record in forecasting spin losses and tried to do something about it.

Let's also end with a prayer - that your next spin program is as successful and enjoyable as mine was in the F-15.

SETP Award to Jack Krings



For the second year in a row, a McDonnell Douglas test pilot has won the coveted Iven C. Kincheloe Award for outstanding professional accomplishment in the conduct of flight testing. The 1975 Society of Experimental Test Pilots (SETP) Award was presented to McDonnell Project Experimental Pilot Jack Krings for successful completion of the F-15 spin test program. Jack is also highly regarded for his work in the F-4 stall, departure, and spin test programs, and has a large number of PRODUCT SUPPORT DIGEST articles to his credit.

The 1974 Kincheloe Award was presented to McDonnell Chief Test Pilot Irv Burrows for his contributions to the F-15 flight test program. In 1962, McDonnell pilot Don McCracken received the award for work in the F-4 high mach and pre-compressor cooling investigations.

(PUBLISHED 1979)

Whifferdills, Divergences, and Other Roll Coupling Phenomena

By LARRY WALKER/*Experimental Test Pilot*

Nearly every fighter pilot has, at one time or another, done consecutive aileron rolls out of the sheer exuberance of flying, yet suffered no severe consequences. And don't flight demonstration teams regularly do multiple aileron rolls? So, you ask, why are full deflection rolls beyond 360 degrees normally prohibited? Or often, full deflection rolls normally restricted at less than +1.0g? Let's take a look at some roll-coupling problems which make these restrictions necessary and some of the underlying principles which cause them.

The origins of roll coupling always seemed mysterious to me — after all, aren't bullets spin stabilized? If so, then why can't an airplane roll safely at maximum rate for as long as the pilot may desire? The answer is that it is possible, theoretically, but only if the roll rate exceeds a certain minimum value. But, unfortunately, just below this minimum value exists a critical roll rate which reinforces the airplane aerodynamic modes of motion and can cause divergence and possible structural disintegration. Therefore, even if we could roll faster than this minimum value, we would first have to accelerate through the critical rate, making the maneuver extremely hazardous. Fortunately, in most cases and flight conditions, the maximum attainable roll rates are less than critical.

THE COUPLING PHENOMENON

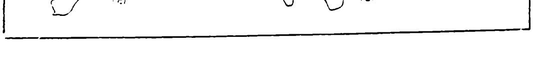
Coupling, by definition, occurs when a disturbance in one axis causes a disturbance in another axis. To illustrate, a longitudinal stick input excites only the pitch axis, producing a single-axis, non-coupled response. A rudder input, on the other hand, excites both the yaw and roll axes, producing a two-axis, coupled response. In this case, the coupling mechanism is aerodynamic — rudder yaws the airplane and dihedral effect rolls it. However, the coupling mechanism can also be due to inertia. For example, inertial forces at high roll rates acting on the airplane can disturb its aerodynamic balance, and in extreme cases, completely overpower its natural stability, sometimes with catastrophic results. However, it is an oversimplification to blame inertial coupling only for roll-coupling problems because in reality roll coupling is composed of three interrelated (and inseparable) coupling mechanisms — kinematic coupling; inertial coupling; and angle of incidence effects.

The roll-coupling mechanisms have been with aviation from the very first, but have only become a problem with the advent of high speeds and jet aircraft; not because of characteristics of the power plants but because of planforms and mass distributions. In order to achieve the necessary high speeds, fuselages have become long and slender and wings small, with a low aspect ratio. This mass distribution is ideally suited for high performance and rapid roll capabilities, but has serious coupling problems at high roll rates. Since none of the contributing mechanisms can be isolated in-flight, I'll try to lay them and their interrelationships out for you.

KINEMATIC COUPLING

Kinematic coupling, as shown in Figure 1, is the simplest contributor to roll coupling.

FIGURE 1
KINEMATIC COUPLING

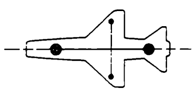


As the airplane is rolled about its longitudinal axis from an initial positive angle of attack (α), the AOA is transformed into sideslip (β) after a quarter roll. As the roll continues, the sideslip is transformed into negative AOA at the inverted position, then into negative sideslip at the three-quarter point, and finally back to positive AOA after 360 degrees of rotation. As the roll continues, sideslip and AOA vary periodically with roll angle. This kinematic effect assumes that the airplane rolls around its longitudinal axis and neglects pitch and yaw stability moments which try to align the airplane with its flight path.

INERTIAL COUPLING

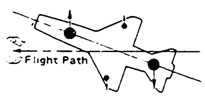
Inertial coupling may best be understood by first simplifying the airplane mass distribution into four equivalent masses — two large masses representing the fuselage and two smaller masses representing the wing (Figure 2).

FIGURE 2
EQUIVALENT MASSES



For any given roll rate about the flight path, the fuselage masses are acted upon by centrifugal force and tend to pull away from the roll axis (flight path in this case). These forces are depicted in Figure 3.

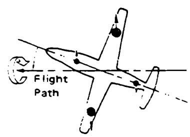
FIGURE 3
DESTABILIZING YAW FORCES



The magnitude of this force couple increases with the square of the roll rate and is highly destabilizing. The wing masses similarly form an opposite stabilizing force couple, but are relatively weak in proportion to the destabilizing fuselage-mass force couple of our long, slender airplane. Although it sounds as if our example airplane in unsafe to fly, fortunately, both longitudinal (pitch) stability and directional (yaw) stability, which are normally quite high, act upon the airplane by trying to keep it heading into the relative wind. It is only with high roll rates that the destabilizing forces can overpower the normal aerodynamic stability and cause a roll-coupling yaw divergence.

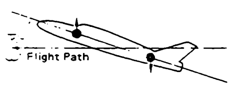
Now that roll coupling is becoming clearer, the astute reader may wonder if coupling can be eliminated by making the wing mass effect (roll inertia) greater than fuselage mass effect (pitch inertia), as shown in Figure 4. The wing-mass force couple now can overpower the smaller fuselage-mass force couple and prevent the nose from yawing away from the flight path.

FIGURE 4
STABILIZING YAW FORCES



This approach does indeed eliminate the tendency to diverge in yaw, but unfortunately, no mass couple exists above and below the airplane which would oppose a similar divergence in pitch (Figure 5).

FIGURE 5
DESTABILIZING PITCH FORCE



Actually, airplanes which have higher roll inertia than pitch inertia have long straight wings (high aspect ratio) and a relatively low roll rate capability. Therefore, even though their mass distribution precludes yaw divergence, their roll rate capability is so low that pitch divergence never becomes a problem.

". . . Even though rolling limitations may sometimes seem unnecessary, they do have a very firm grounding based on some very real problems . . ."

To place the whole mass distribution issue in perspective, a clean slatted F-4 has approximately six times more pitching inertia than rolling inertia. Even with full external wing tanks and three 500-pound bombs on each inboard wing station, the pitch inertia is still three times greater. The numbers for the Eagle are only slightly less; a clean F-15 has approximately five times more pitching inertia than rolling inertia. Even when loaded with three full external tanks, four Sidewinders and four Sparrows, the ratio is still three times as great. Therefore, inertial coupling can be a problem no matter what the loading.

What of our original example — the bullet? The bullet is spun well above the critical roll rate so that it is spin stabilized, rolling about its "fuselage masses." The inertial gyroscopic forces are highly predominant and stabilize its attitude, similar to the very rapidly rolling, hypothetically spin-stabilized airplane of Figure 6.

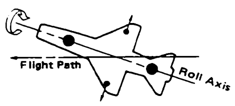
Occasionally projectiles have been known to tumble; this occurs when the roll rate decreases to the critical rate which reinforces aerodynamic modes of motion until divergence occurs. This divergence is identical to the roll coupling divergence of a long slender rolling airplane except for the direction of approach to the critical roll rate.

PUTTING IT ALL TOGETHER

Now that roll coupling is almost understandable, how can I, as fighter pilot extraordinaire, do consecutive 360 degree aileron rolls safely? Our above discussion suggests that we can if we roll at zero g, keeping the fuselage masses centered on the roll axis and their destabilizing force couple at zero. Wrong — for three reasons!

First, zero g does not insure that these fuselage masses are on the flight path. (This is the angle-of-incidence effect, the last of the three contributors mentioned earlier.) Depending upon the angle of attack required for zero g and upon the fuselage mass ▶

FIGURE 6
SPIN STABILIZED



distribution, the angle of incidence of these fuselage masses can be above or below the rolling axis, as seen in Figure 7. This may be most easily visualized in an airplane with a high vertical tail, possibly a high tail-mounted engine, and a low nose. Worse, with this hypothetical airplane, add in low angle of attack at zero (or negative) g and at high speed which further aggravates a negative angle of incidence below the flight path. These are some of the most conducive design and flight circumstances for a catastrophic roll-coupling departure!

Second, even if we could keep the fuselage masses on the roll axis longitudinally, it is impossible to keep the fuselage aligned directionally with the roll axis. Aerodynamic cross-coupling effects such as yaw due to aileron, yaw due to roll rate, yaw due to rudder, and about a dozen other minor lateral-directional aerodynamic effects combine to generate some sideslip. Further, as the airplane rolls, this sideslip becomes angle of attack, becomes opposite sideslip, becomes opposite angle of attack, etc., as these kinematic effects magnify and transform these small disturbances. No longer are the fuselage masses aligned with the roll axis, but are diverging from the axis, increasing the size of their destabilizing force couple.

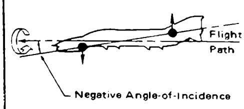
Third, as the roll rate increases, these periodic variations of sideslip and angle of attack occur at the same frequency as the natural airplane directional and longitudinal motions (most easily seen with stab aug or CAS off, these are the "Dutch roll" and the "short period" modes of motion), reinforcing them until a divergence or catastrophic failure occurs. This reinforcing effect may be best understood as a resonance between inertial and aerodynamic forces, leading to ever-increasing yaw and pitch excursions from the flight path.

SOME SOLUTIONS

Now that the reasons for roll coupling are clear, how can it be avoided? Generally, changes of mass distribution are impractical, but rate dampers in the pitch and yaw axes can reduce coupling into the Dutch roll and short period modes of motion by damping the motions themselves, thereby raising the critical roll rate for divergence. Other preventive measures also are normally required, such as limiting roll travel to less than 360 degrees. This restriction limits the time duration that the destabilizing forces can reinforce the yawing and pitching modes of motion and thereby keeps sideslip and angle of attack within acceptable limits.

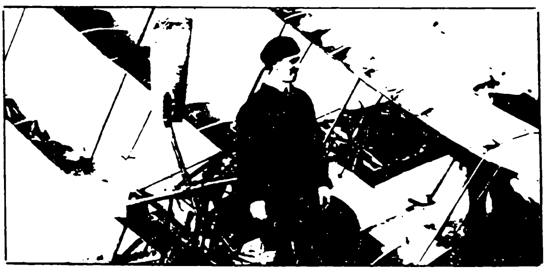
Other preventive measures involve placing restrictions against full deflection rolls at less than one, or less than zero g's, in order to limit angle-of-incidence problems. Lateral stick stop devices have also been used to lower maximum roll rates in some fighters. Similarly, CAS and fly-by-wire control systems, employ lowered aileron gains and deflections, or use electronic roll rate limiters in order to keep roll rates less than critical for 360-degree rolls. Such limiters are normally dependent upon flight conditions to avoid poor

FIGURE 7
ANGLE-OF-INCIDENCE EFFECT



transient lateral response in low speed flight conditions.

In summary, today's high performance fighter airplanes are typified by high fuselage densities and little rolling inertia in order to attain the high speeds and good rolling performance required. Accordingly, they suffer from a cross-coupling resonant condition when gyroscopic and inertial forces associated with high roll rates overpower normal aerodynamic stabilizing forces, leading to divergence and departure from controlled flight. Unlike loss-of-control departures at high angles of attack, these cross-coupling departures occur primarily at high speeds and low angles of attack where roll rates are highest. Unfortunately, if this type of departure does occur, the results are usually catastropic due to the extremely high airloads. Even though rolling limitations may sometimes seem unnecessary, they do have a very firm grounding based on some very real problems. Suitable respect for these limitations can go far towards making high performance flight safer and more enjoyable. So the next time you hear someone grumble about "unnecessary rolling restrictions," point out these dangers and explain why the restrictions exist. After all, you're an expert now! ■



"THE MORE THINGS CHANGE . . ."

Whether this accident in 1909 to famous early aircraft designer A. V. Roe's first triplane may have resulted from a whifferdill, roll-coupling, or other divergence is lost to aviation antiquity. However, Mr. Walker's article here reminds today's jet drivers that they are just as much at the mercy of the immutable laws of physics and aerodynamics as were the very first flyers, but with results often not as easy for the pilot to take as this one evidently was.

(Photo used courtesy of FLIGHT INTERNATIONAL MAGAZINE)

(PUBLISHED 1978)

F-15 ACCELERATION LIMITATIONS..

The rugged F-15 airframe has been certified for a service life span of from 25 to 30 years — it's possible that your son could two decades from now strap into the same Eagle you're flying today! Whether that actually happens however, depends on the care and handling you give that airframe in day-to-day operations at the present time. The following two articles describe some situations that can cause an Eagle to "age prematurely;" and what you can do to make sure your F-15 is ready when your son is . . .

The Overloaded Eagle

By PAT HENRY/*Chief Experimental Test Pilot*

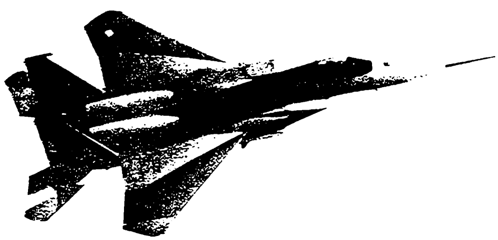
Back in 1972, when we first started flying the F-15, it became immediately apparent that we had a super-agile bird, very well suited to the name "Eagle." It was forecast at that time that a major problem in service use would be how to tame the young lions — i.e., produce aggressive; max performance fighter pilots without literally clipping their wings through excessive G limitations. Alternatives such as placards or administrative controls are not too palatable, but neither are the end results from over-G's: shortened aircraft service life, or worse yet, a build up of structural fatigue that brings on the *coup de grace* — structural failure.

In the Category I test program, aircraft loads were measured during the structural demonstration flights. As a fallout of these test points, stick force/G and available load factors were determined, essentially throughout the envelope. As should be expected for an aircraft with relatively high turning capability at low speeds, there is sufficient "G" available at optimum maneuvering airspeeds to literally "pull the wings off." That's simply a basic fact of life for a fighter airplane with a high lift wing and tail power to match.

For general interest, the maneuver used in flight test to set up these unsymmetrical acceleration test points, hopefully with a high degree of control over the variables involved, is a Rolling Pull Out (RPO). The RPO is basically a bank-to-bank roll at a given load factor, lateral stick, airspeed, and altitude. Getting these parameters to come together simultaneously, and stay together during the maneuvers, is the tricky part. In an RPO, the test pilot establishes a wind-up turn at the desired conditions, then comes straight across with the stick either one-half or full, for the lateral contribution. if you do it just right (and the flight records will tell the story in minute detail), you will have introduced full lateral stick in less than 0.2 seconds, while maintaining exactly the same longitudinal stick force/position. For some flight conditions, this is a relatively simple task — others are a lot dicier, particularly at high airspeed and low altitude, where the aircraft is extremely responsive in pitch. Roll-to-pitch coupling can also be present in varying degrees to help you overshoot the target G, so a graduated build-up to the desired end point is always pursued. It should be obvious by now that we treat these points with respect, and the reasons are twofold — difficulty of hitting the test conditions without overshooting and the ramifications of overshooting.

Whether we overstress any given aircraft during test flying or it occurs out in service use, the end results are the same, namely increased costs. Short term costs include unscheduled maintenance in the form of inspections and repairs, plus the inevitable reduced aircraft availability. The major long term effects will be decreased service life for the entire airframe, and the accompanying cost of premature replacement.

In care you're skeptical about the frequency or magnitude of G overstress, take a close look at the two charts that J.T. discusses in Figure 4 of the next article; I think you'll be shocked! (At least I hope you will be!) I can't urge you strongly enough to log all excessive G excursions; after scrutinizing the charts I'm sure you'll agree with me why. Since most aircraft don't have Signal Data



F-15 71-283 was used for Category I Load Tests at Edwards AFB. This was Eagle #4, and the first with raked wing tips. Painted black lines on wing and vertical fin were photographed from camera mounted in aft cockpit. From this film, wing and fin deflection could be measured.

THE COLD HARD FACTS

Recorders installed, the burden is on us, the pilots, to track these events. The burden is also on us to observe the limits and minimize the overshoots. There are ways to do this for us through the control system, but most pilots find that approach distasteful. Program management also finds it expensive, so the ball is still in our court.

If projected life cycle costs so justify, a mini-computer program may eventually be developed to compute instantaneous load factor limit, and to provide visual and aural warning as that limit is approached. That option isn't cheap either, so it's still the responsibility of each and every one of us to know and observe the design limits of the aircraft. So much for my sermon; Mr. Johnston will now lay on some technical support for my exhortations. ■

F, G, and n_z

By J. T. JOHNSTON/*Section Chief - Loads*

After recently reviewing the load factor (n_z) levels from F-15 service usage and after discussions with several pilots, we feel it's time for a review of the Dash-1 Acceleration Limitations and in particular the difference in the symmetrical and unsymmetrical flight limits. For starters, Figure 1 is a reproduction of the Symmetrical and Unsymmetrical Acceleration Limitation Charts from the Flight Manual. In order to simplify the pilot's task of remembering the limits, the charts are uncluttered with different configurations and speeds. What the pilot can do and cannot do is pretty obvious.

Recapitulation of the number of exceedances of acceleration limitations, from data available to us, and the level of load factors presently being experienced with the F-15, gives us cause for deep concern. We understand the ease with which the aircraft can be maneuvered to extreme load factors, but if we expect to keep these "Eagles" flying for the next several decades, pilots will have to back off from these high G's today. So much for that.

Before we look at some of our service data, let's define symmetrical and unsymmetrical maneuvers -

- Symmetrical: Those flight conditions which load the airframe up due to angle of attack (AOA) only. That is, no lateral stick or roll is involved. A typical aircraft wing loading for symmetrical maneuvers is shown in Figure 2.
- Unsymmetrical: Those flight conditions where roll or yaw angles are generated, such as rolling and

". . . if we expect to keep these "Eagles" flying for the next several decades, pilots will have to back off from these high G's today . . ."

FIGURE 1 - ACCELERATION LIMITATIONS

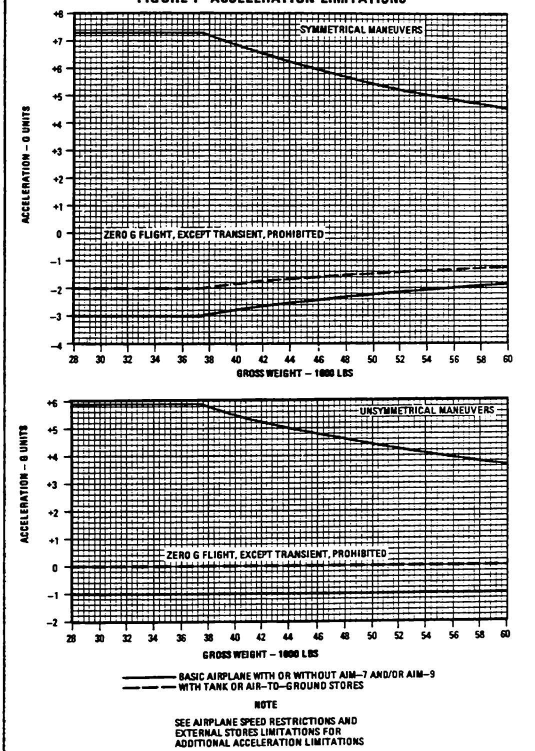
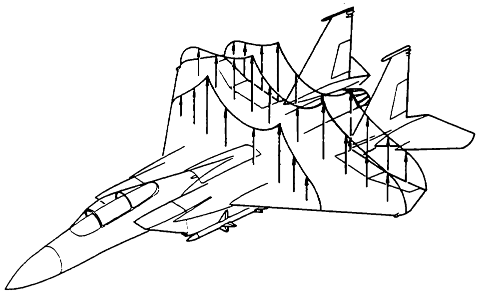


FIGURE 2 - TYPICAL SYMMETRICAL WING LOADING



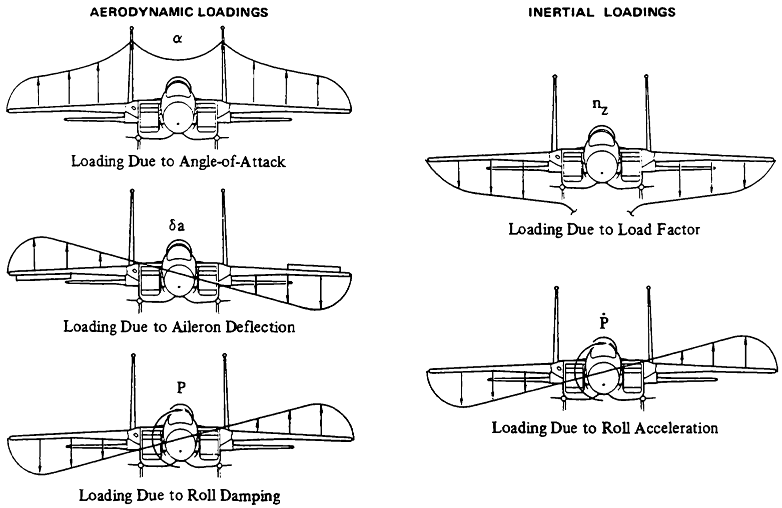
sideslip maneuvers. When roll is introduced into the maneuvers, the wing loading is a combination of loadings from angle of attack, aileron deflection, roll damping, roll acceleration, and inertial loads from load factor. All of these various loadings are shown in Figure 3.

In defining the structural requirements, all possible combinations of these loadings are investigated for unsymmetrical maneuvers executed between a minus 1 and plus 5.86 G's (80% symmetrical). The purpose of the reduced load factors for unsymmetrical maneuvers is to account for "delta" (change in) loading resulting from the unsymmetric parameters described in Figure 3. This technique has proven to produce the optimum airframe structural weight.

As long as the pilot observes the published acceleration limits for symmetrical and unsymmetrical maneuvers, aircraft loading will remain within design limits. However, numerous cases of excessive G, and in particular excessive unsymmetrical G limits, have been observed. Figure 4 shows the results of data reduced from one F-15 during 105 flight hours. The facts show that the aircraft has exceeded both symmetrical and unsymmetrical limits at least once every other flight (ironically this aircraft never had an "over-G" reported). The Symmetrical Limit Chart shows 59 occurrences of the aircraft being over-G'd; the Unsymmetrical Limit Chart shows 55 occurrences.

Let me explain how this information is obtained. One in five F-15s is equipped with a Signal Data Recorder Set (SDRS). This black box continuously records twenty-one measurands whenever the aircraft is on internal power. The purpose of this recorder is to measure the parameters for calculating loads to be used in the service life analysis. (For those who may be interested, the SDRS and its part in the F-15 fatigue tracking program was discussed in DIGEST Issue 5/1977.)

FIGURE 3 - AERODYNAMIC AND INERTIAL WING LOADINGS



The instrument used by the pilot to determine if the acceleration limits have been exceeded is the cockpit accelerometer (G Meter) shown in Figure 5. For simplicity, only two positive G units are referenced, a 7.3 at 37,400 pounds and a 5.1 at 53,300 pounds. The allowable load factors for all other weights are found on the Acceleration Limitation Chart (Figure 1). An important point for pilots to remember is that for high acceleration rates the cockpit accelerometer will indicate a lower G force than the aircraft is actually undergoing (because of the distance from the cockpit accelerometer to the aircraft's actual center of gravity). With this in mind, it is of extreme importance that all over-G incidents be reported during debriefing.

For positive load factor exceedance there are two types of inspections, a simple minimum effort type for exceedance between 7.3 and 8 G's and a more detailed inspection for exceedance above 8 G's. These acceleration limits are for symmetrical maneuvers; for unsymmetrical maneuvers, the acceleration limits are 80% of the symmetrical.

For ease in determining what type inspection will be accomplished when the aircraft has been over-G'd, it is helpful to maintenance personnel to know the approximate weight of the aircraft at the time the over-G event occurred. This information can save them hours of unnecessary work.

In conclusion, if we keep those load factors within limits, particularly at supersonic speeds, and observe the 80% symmetrical load factor limit when executing roll maneuvers, Eagles will be around for a long time to come. ■

FIGURE 4 - SIGNAL DATA RECORDER SET (SDRS) SUMMARY FOR ONE F-15 AIRCRAFT

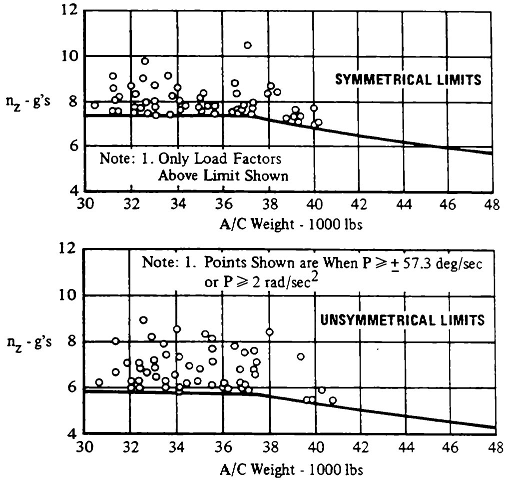
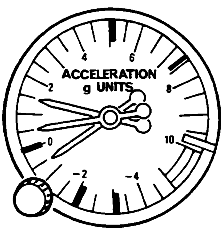


FIGURE 5 - COCKPIT ACCELEROMETER

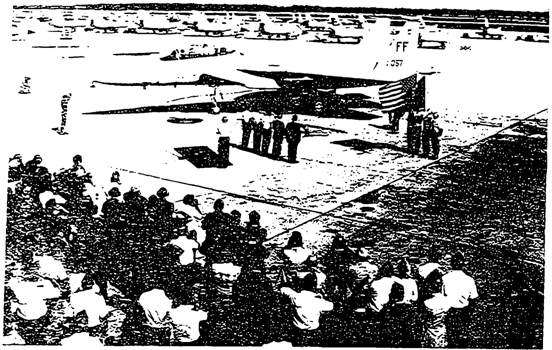


+7.3G	■	MAXIMUM POSITIVE LOAD FACTOR AT 37,400 POUNDS OR BELOW.
+5.1G	■	MAXIMUM POSITIVE LOAD FACTOR AT 53,300 POUNDS.
ZERO G	■	ZERO G EXCEPT TRANSIENT PROHIBITED.
-2.0 G	■	MAXIMUM NEGATIVE LOAD WITH TANKS OR AIR-TO-GROUND STORES.
-3.0 G	■	MAXIMUM NEGATIVE LOAD FACTOR AT 37,400 POUNDS OR BELOW WITHOUT TANKS OR AIR-TO-GROUND STORES.

REFER TO ACCELERATION LIMITATIONS FOR ADDITIONAL RESTRICTIONS.

NOTE

THE ACCELEROMETER INDICATES UP TO 1G LOW WHEN PULL-IN RATES ARE HIGH (1 G /SEC OR MORE). THE G INDICATIONS ON THE HUD ARE ACCURATE.



Millionth-hour aircrew - General Jerome O'Malley and Lieutenant Colonel Paul Hester - shut their F-15 down in preparation for "red carpet" reception on flight ramp at Tyndall AFB.

EAGLE OVER MILLION MARK

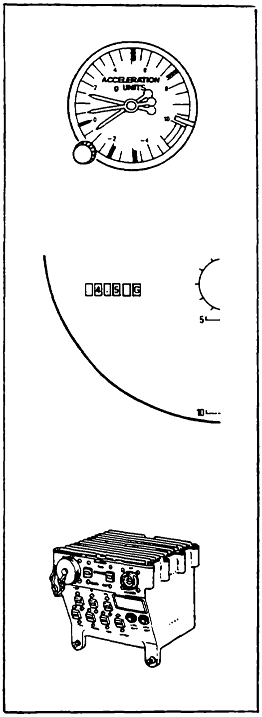
26 October 1984 was a red-letter day for McDonnell Aircraft Company and the United States Air Force. It was on that bright, sunshiny day at Tyndall Air Force Base, Florida, that the F-15 Eagle air superiority fighter officially became a "millionaire" - reached a million hours in the air!

When General Jerome F. O'Malley, new commander of the Air Force Tactical Air Command, and Lieutenant Colonel Paul Hester, 94th Tactical Fighter Squadron operations officer, landed F-15D S/N 80-057 at Tyndall, they were fifteen minutes into the second million hours, after a one hour fifteen minute flight from Langley Air Force Base, Virginia. With more than 800 aircraft operating from 14 locations around the world, Eagle flight hour # one million could actually have occurred at any number of places, but what more appropriate location and symbolic situation than Tyndall AFB and the concluding ceremonies for "William Tell 1984"?

F-15's have been operational at Tyndall for approximately a year, and are being flown there by the 325th Tactical Training Wing. F-15's from around the world were temporarily at Tyndall to participate for just the second time in a William Tell air-to-air weapons competition. Attention of the entire fighter aircraft community was therefore focused on the aerial event underway at this northern Florida base; and on this particular day all eyes at Tyndall were on the big red "apple" painted on the ramp as General O'Malley expertly brought his Eagle to a stop at that exact point, for a formal color guard reception.

It took just one month short of ten years of operational service for the F-15 to reach its millionth flight hour; November 1974 saw the airplane dedicated to squadron service with the 555th TFTS at Luke AFB, Arizona. In its first decade of utilization, the Eagle has become the "safest" fighter in aviation history - only four aircraft have been lost per 100,000 flight hours. In accepting a plaque from MCAIR in recognition of its accomplishments, General O'Malley said, "The F-15 a super performer. It has exceeded every goal we set for the airplane and is the safest fighter aircraft we've ever flown. It is a testament to the high state of Air Force readiness today, and a centerpiece of american technology and know-how." ■

General O'Malley accepts "million hour" plaque from William S. Ross, MCAIR vice president and general manager, F-15 program.



In the article "P, G, and n_z" (Issue 1/78 of the DIGEST), I noted that the F-15 cockpit accelerometer (G-Meter) indicates a different G than the actual Gs at the aircraft center of gravity (CG) during accelerated maneuvers. Since publication of that article, inquiries from the field have been received, asking why this is so, and if the Gs displayed on the Head Up Display (HUD) are correct.

An explanation of the location and characteristics of the F-15 accelerometers may help in answering these questions. As illustrated above, there are three devices in the F-15 which provide "load factor" information — the G-Meter (gage/pointer) on the cockpit left main panel; a digital readout in window #8 of the HUD display; and an exceedance counter set inside Door 6R. All three operate from accelerometers located at var-

(PUBLISHED 1978)

ACCENT ON ACCELEROMETERS

By J. T. JOHNSTON/*Section Chief Loads*

ious distances from the true aircraft center of gravity; and it is this "distance" which governs accuracy and reliability of the information provided. If you could pick up these three Eagle instruments at the local department store, you might find them offered for sale as "Good," "Better," and "Best" in terms of the value of the acceleration G load data offered to pilots and maintenance personnel. But before I get into details, let me remind you strongly that, regardless of the *relative* accuracy of the three devices, they all provide information that must not be ignored for all three are charting the "lifestyle" of your F-15 as it progresses toward a long (or shorter) life.

The cockpit accelerometer used in the F-15 (and the F-4) is a mechanical device, consisting of a pointer attached to a spring and mass to indicate the level of local acceleration. The readings indicated by the instrument are fine as long as there are no angular accelerations (rotational motion of the aircraft about its center of gravity resulting from rapid pitching maneuvers).

The accuracy of the cockpit accelerometer for high performance aircraft has been questioned as far back as the early days of the Phantom. This led to the caution note in the F-4 Flight Manual, explaining that the accelerometers may read ½G low, possibly lower when pull-in rates are high. In the F-15 Flight Manual, the discrepancy is 1G.

The F-15 cockpit G-Meter is located at F.S. 277.4, some 24 feet ahead of the aircraft CG, which makes it extremely sensitive to any aircraft angular accelerations (Figure 1). To illustrate the difference between G-Meter reading and the CG load factor, the graph (Figure 2) shows a portion of a test run which had high G rates (angular acceleration). This difference is principally due to the term $\Delta n_z = \dot{q}r/g$ where "$\dot{q}$" is the pitching acceleration and "r" is the distance from the aircraft CG to the instrument mass CG. There are other factors which will also contribute to the erroneous readings given by production cockpit accelerometers, such as static friction in the pointer and drag of the max G hand.

G data displayed on the HUD is obtained from acceleration information provided via the Central Computer. The CC receives a signal from the Lead Computing Gyro set, which has its accelerometer located in the avionics bay in the forward fuselage at F.S. 316.3. This display is somewhat more accurate than that presented by the cockpit G-Meter because it does not have mechanical friction or pointer drag to contend with, and because its accelerometer is more than three feet closer to the aircraft CG. But again, there is an error factor because of the actual distance between accelerometer and CG.

Due to its proximity to the aircraft CG, the most accurate acceleration readings on the F-15 are registered by the exceedance counter accelerometer. This instrument is located in the right hand main landing gear wheel well, and is just inches from the aircraft CG. The information it registers is transmitted to, and recorded on, the exceedance counter located behind door 6R of all F-15A/B aircraft, and on a Signal Data Recording Set (SDRS) tape cassette installed in every fifth aircraft. This is the source of the most detailed acceleration data on the Eagle.

Editor's Note: As the DIGEST was going to press, the finishing touches were being put to a video tape titled "THE OVERLOADED EAGLE." An expansion of the articles on F-15 acceleration limitations by Pat Henry and J. T. Johnston that appeared in our first 1978 issue, this is an excellent film of particular interest to those of you out there concerned with prolonging the structural life of the Eagles. The Air Force tape number is CVT-F-15A-11K00-1, and it should be at all F-15 using organizations around the latter part of December. ■

FIGURE 1 - INCREMENTAL LOAD FACTOR DUE TO PITCHING ACCELERATION

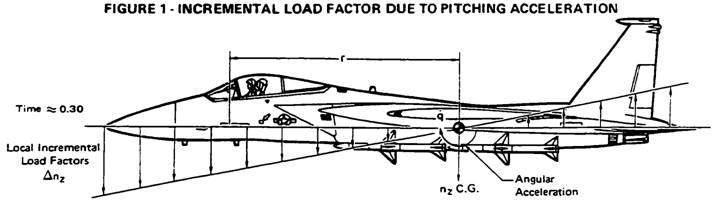
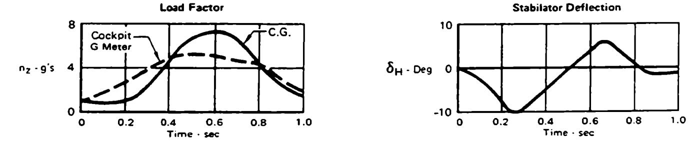


FIGURE 2 - TIME HISTORY



"OWS" THE F-15

(PUBLISHED 1981)

PROTOTYPE TESTING

By CAPTAIN RICHARD BANHOLZER /*422nd Fighter Weapons Squadron, Nellis AFB, Nevada*

During the past year, the Operational, Testing, & Evaluation (OT&E) section of the 422nd Fighter Weapons Squadron, Nellis AFB, Nevada, has been testing a prototype Aural-Warning device known as the Overload Warning System (OWS). The system was designed and built by McDonnell Aircraft Company for production incorporation into the F-15 and will be retrofitted in all F-15s in service. Evaluating McDonnell's latest black box was a pretty interesting program for us because it was a problem area that has existed since the Eagle hit the field, and I'd like to tell you a little about it.

OVERLOAD PROBLEM

The extreme maneuverability and high energy states attainable by aircraft such as the F-15 require the pilot to have extreme "G" awareness throughout an engagement. Currently, the operational employment of the F-15 is hampered by the absence of both a mechanization capable of providing the pilot with accurate heads-up real-time load factor monitoring, and a system for warning him of an impending over-G situation. The pilot does have a Heads Up Display (HUD) readout and the standard cockpit accelerometer, but both offer only relative accuracies, do not show G at the aircraft center of gravity, and are not always in his field of view.

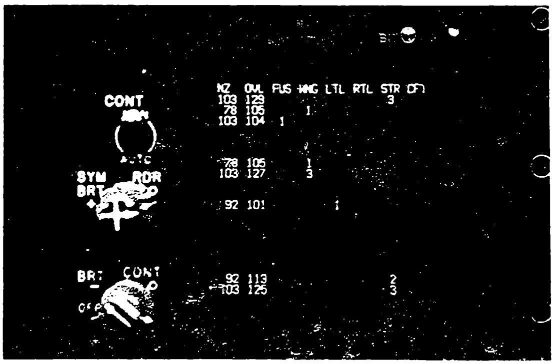
The acceleration limits shown in the DASH ONE do not thoroughly represent the F-15 capabilities. Symmetrical acceleration limitations at gross weights below 37,400 pounds are constrained by Air Force specifications limiting the aircraft to 7.33 Gs. This safety factor is needed because it is difficult for the pilot to ascertain where in the G envelope he is operating due to changing roll rate, Mach Number, altitude, gross weight, etc. The OWS will, by a series of tone and voice warnings, inform the pilot when he is on the threshold of overloading the aircraft structurally.

Additionally, when the aircraft exceeds T. O. acceleration limits, maintenance workload is adversely impacted by manhours required for over-G inspection. Many times these inspections are conducted on the pilot's "best guess" on what his flight conditions were at the time of the over-G. The OWS will provide maintenance with an aid in determining inspections needed in the event of an overload. An onboard computer will indicate which component of the structure has been overloaded. These parameters can also be called up by the pilot and displayed on the Air Navigation Multiple Indicator (ANMI). A sample display can be seen in the photo below. This information will be used to determine what to inspect and to what degree, thus greatly reducing the manhours to inspect the aircraft after an overload condition has occurred.

OT&E

MCAIR developed several prototype OWS warning mode systems, and our OT&E was conducted to determine what type of tone and/or voice combination was preferable and at what percentages of maximum allowable load limit the warnings should occur. The first system tested provided the pilot with one of three cockpit-selectable options, all of which initiated a warning at 85 percent of the maximum allowable load. One option (all voice) was eliminated because the pilot was unable to determine where he was in the 85 to 100 percent range. In the other two options, there was a distinct tone/voice change at the 95 percent point, but the pilot was unable to determine how much he

(Continued on Page 50)



Laboratory test simulation of OWS digital display system on ANMI. Although data can be called up by pilot, primary use of cockpit instrument is to inform maintenance personnel what to inspect and to what degree components have been overloaded in flight. Data comes from Central Computer. First column shows load factors (Nz) associated with percentages of overload (OVL) in second column. (Decimal points are omitted in first column — 103 indicates 10.3 load factor.) Remaining six columns are the components monitored by OWS — fuselage, wing, left tail, right tail, external stores, and conformal tank. Single digit numerical values are approximations of degree of component overloads — numeral 1 indicates overload between 100 and 109%; 2 is between 110 and 119%, etc. Incidentally, we sincerely hope the Eagle pilot never sees overload numbers as high as our laboratory simulation indicates; the overload warning tone/voice system would have alerted him long before numbers like these would show up!

OVERLOAD WARNING SYSTEM

ENGINEERING APPROACH (PUBLISHED 1981)

By J.F. KUNZELMAN/*Technical Specialist, and* J.T. JOHNSTON/*Branch Chief - Loads*

It's unlikely that the "G" meter in fighter aircraft will ever be replaced, but there are better things on the horizon. In the not too distant future, the F-15 Eagle will have an "Overload Warning System" (OWS) aboard. This new system will audibly warn the pilot that he is approaching the aircraft structural limits; tell him if a structural limit has been exceeded; and inform the ground crew afterward what component was overloaded and how severely. No more guess work!

Notice, we used the word "overload" in lieu of "over-G." This is important since for any given G level, the loads imposed on the aircraft structure can vary drastically depending upon such factors as: Mach Number, aircraft configuration, and degree of lateral stick input. Ironically, while data compiled from the Signal Data Recorder (SDR) tells us that the F-15 airframe is being overloaded in certain regions of the flight envelope, reserve structural capability exists throughout a large percentage of the envelope which is not being used effectively. SDR information also verifies that a significant number of over-G conditions are not being reported, most likely because they cannot be recognized. A typical case involves training with a centerline tank aboard (heavy aircraft), initial engagement at transonic speed around 15,000 feet, yank and bank for an AIM-9 shot. DASH ONE limits under these conditions for unsymmetrical maneuvers are around 5.0 G (Figure 1).Training close to home is partly responsible, but we've yet to see a 5.0 G writeup regardless of the cause.

For those skeptics who dislike new gadgets in the aircraft and think the OWS is a quick fix to stop abuse of the aircraft, it should be noted that as far back as 1969 studies were conducted for the US Air Force to show the F-15 load factor capability throughout the flight envelope based on various criteria. Most of you are familiar with the varying F-4 aircraft load factor handbook limits which range from 6.0 to 8.5. The F-15 was designed to 7.33 G at the critical speed and altitude within the flight envelope. As a result, reserve capability in terms of higher G limits is available elsewhere. Based on results of a thorough test program, it is now time to unleash the Eagle and let it fly to these known capabilities. The OWS is designed to provide this capability to the pilot while at the same time providing protection for the airframe.

Conceived as a means of addressing both the operational and maintainability aspects of the over-G problem, OWS is designed to provide a system that monitors all parameters (Mach, altitude, roll rate, lateral stick displacement, normal load factor, external stores loading, and total fuel quantity) necessary to continuously determine aircraft structural overload conditions. It will then provide timely alerting to the pilot of an impending overload situation through the use of an Aural Tone Generation System, as discussed in these two articles.

The core of the OWS is in the Central Computer and the new Programmable Signal Data Processor (PSDP). This new PSDP will gather additional information from such equipment as the Roll/Yaw and Pitch Computer, Fuel Quantity Indicator, and Armament Control Set, and transmit this data to the Central Computer. The Central Computer, with its existing information (speed, altitude, etc.) and reprogrammed aerodynamic and inertial data for computing forces acting on the aircraft, will continuously and in real time (20 times a second) generate the aircraft structural loading state. If a loading condition exceeds the 85 percent level, a message is sent to the PSDP to turn on the warning tone. An upper G limit of 9.0 has been included to protect the large mass items (engines, AMAD, etc.) from being overloaded.

McDonnell funded the OWS program and started hardware development of the OWS back in 1978. MCAIR test pilots evaluated the system and pressed for further development. When data from F-15 operational units made it apparent that a warning system was needed, a prototype was available which could be tested. The USAF funded a test program for evaluation under simulated combat conditions at Nellis AFB. This allowed the system to be tested in the environment necessary to optimize the warning system.

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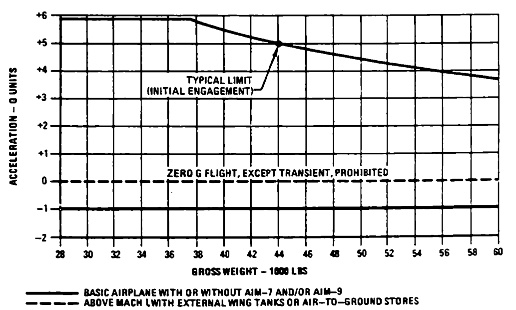
FIGURE 1 – ACCELERATION LIMITS (UNSYMMETRICAL MANEUVERS)



"OWS" THE F-15

PROTOTYPE TESTING (Cont'd)

was above 95 percent. This led to a three-step system — a 4 HZ interrupted tone at 85%, 10 HZ interrupted tone at 95%, and a voice "over-G" at 100%.

The next phase was to optimize the percentages at which the various warnings came on. We did not want the warnings on so early as to be a nuisance (as we found with the all-voice system), but we needed them on early enough to give sufficient warning of an impending overload. Through the use of Signal Data Recorder (SDR) information from past AIMVAL/ACEVAL trials, and from Air Combat Maneuvering Instrumentation (ACMI) sorties with the prototype OWS, it was found that pilots average about a seven percent overshoot from the first warning during a moderate G onset rate. More specifically, if the pilot is in a smooth pull with increasing G and the initial warning comes on at 6.5 Gs, he will normally react and arrest the Gs onset around 7Gs. (Remember that this is an average figure, and this system will not prevent a two-handed snatch over-G). With these figures as a basis, the final percentages were established as the initial tone at 85% design limit load; the second tone at 92% design limit load, and the third "over-G over-G" voice warning at 100% design limit load.

During the Nellis program, the OWS was tested in many operational scenarios. We started with single ship calibration and verification flights, progressed to 1 V 1 pilot familiarization flights and finally conducted a four-month flight evaluation involving 2 V 2, 2 V many, communication saturation, and Red Flag sorties. To determine if there was any audio interference, the system was evaluated against other tones in the aircraft such as AIM-9L, Radar Warning Receiver (RWR), Ground Controlled Interception (GCI), and the departure warning tones.

RESULTS

It was found that the warning tones or voice do not cut out any cockpit audio. They do however, add another input to pilot workload which we became accustomed to after one or two flights with the system. *The important factor was that pilots began to rely on OWS as an aid to flying their aircraft.* Typical pilot debrief comments were:

- "It allowed me to be more aggressive with the aircraft."
- "It gave me the extra G I needed to get my pipper on him."
- "I could concentrate on tracking him and not worry about looking at the HUD G meter."
- "I was surprised at how easily an over-G can occur in the transonic Mach regime."
- "I could perform a break turn at maximum G."
- "I never realized how many F-15s I really over G'd and never knew it."

It was shown that the roll rate required to start decreasing the symmetrical limit was very hard to achieve. It sometimes required lifting one's leg to get the stick against the lateral stop (a good trick at 6 or 7 Gs in an air-to-air engagement), but we did find it easy to achieve maximum-G shortly after takeoff in the critical transonic regime (a normal start engagement condition). We found the 9-G aircraft limit was rarely needed during an engagement, and when it was, it was a super energy/pilot depleting maneuver. (For you former F-4 jocks, you can compare it to the slats on the F-4E: they're there if you need them, but only when you can afford the resulting energy depletion.)

Some pilots found that they tended to use more G than was necessary to perform an optimum maneuver. A good example of this occurred in the butterfly dart pattern and was documented on ACMI. Without the OWS, most pilots initiated a 6 to 7 G turn when cleared to fire. This G loading was then reduced as the turn progressed, resulting in a relatively high energy state at the 180 degree of turn point. With OWS, initially, most pilots pulled to the 95 percent tone warning and held it throughout the turn, resulting in a very quick 180 degree turn, but a very low energy state at the end. This necessitated an acceleration maneuver and subsequent straight line flight path across the circle to reach firing parameters. *It should be remembered that the OWS warns that you are approaching "maximum G" and not necessarily "optimum G" for the maneuver being performed.*

While the OWS should be used only as an aid, and not change the way the F-15 is employed, it will be a welcome addition to the F-15 Weapons System for many reasons. It will be added to the aircraft with no additional cockpit switches and no additional aircraft weight. It will not only enhance the tactical utility of the F-15 by expanding the operational flight envelope, but will also provide Maintenance with more accurate information on aircraft loading and reduce inspection requirements. Pilots will not require additional sorties for familiarization once the OWS is installed in their aircraft. The bottom line is that it will tell you positively when the aircraft is approaching a structural overload condition and should be treated accordingly.

In closing I'd like to pass on a significant bit of advice and information for those of you who may consider "punching off" an over 7.33G reading because it's a 9G aircraft — DONT! Without the OWS installed, your F-15 Eagle is restricted, rightfully, to current DASH ONE limitations. ■

Captain Richard "Dick" Banholzer is a test pilot assigned to the 422nd Fighter Weapons School, Nellis AFB, Nevada. He has over 2400 flight hours in the F-4E and F-15 aircraft, and is currently assigned to the F-15 Operational, Testing, & Evaluation section of the FWS. He graduated from the Fighter Weapons School in 1977, and the USAF Test Pilots School, Edwards AFB, California in 1978.

OVERLOAD WARNING SYSTEM

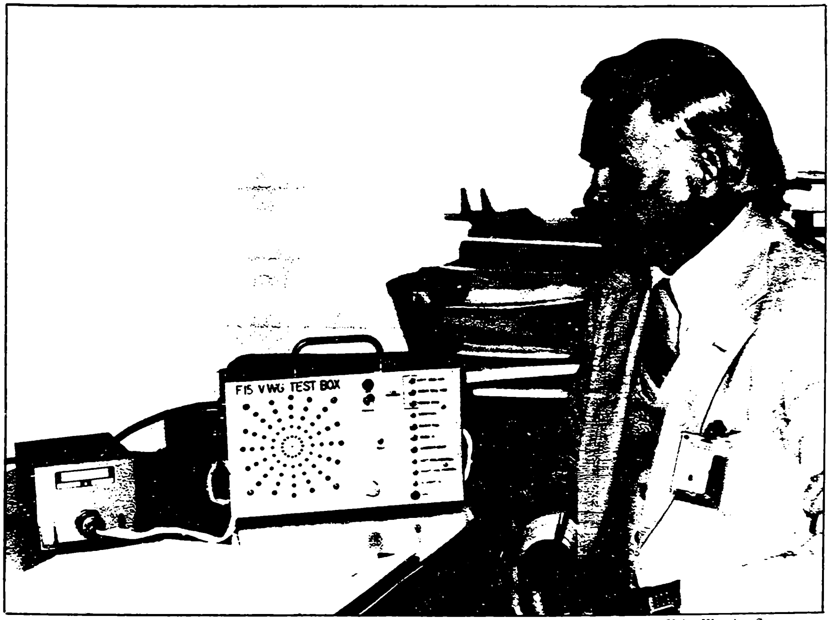
ENGINEERING APPROACH *(Cont'd)*

The OWS is patterned with both the pilot and ground crew in mind. For the pilot, there will be no new switches or gadgets. The warning tones have been set to limits which will give ample warning but not become a nuisance. There are no mechanical limits in the system, so if the need arises you can ham-fist it till your socks fall down.

For the ground crew, overload inspection will be much more simplified. There will be no need to go to a T.O. to see if the aircraft was really overloaded. There will be no need to inspect the entire aircraft if only a portion of the structure, such as the tail surface or wing pylon, was overloaded. There will be no need to do a teardown of the aircraft if the aircraft was overloaded by a few percent. A readout on the Air Navigation Multiple Indicator (ANMI) will tell you directly what, where, and how much must be inspected. We'll still show the Gs too, but the nitty-gritty will be displayed in percent of allowable load, and T.O. inspections will be related accordingly. One hundred percent, that's the limit regardless of the Gs.

For those of you who think this new system will bring on a rash of overload inspections, relax - in over 100 hours of flying at Nellis to the expanded envelope with the system installed, not a single overload was encountered.

The OWS will be installed on all production USAF F-15C models serial number 80-0033 and up, and F-15D models serial number 80-0056 and up. Retrofit will be on all F-15A/B models, F-15C models serial numbers 78-0468 through 80-0032 and F-15D models 80-0561 through 80-0056. As yet no TCTO numbers have been assigned. As the finishing touches are put on the system and we progress toward production deliveries late in 1981, more details about this airframe-saving system will be passed along to you. ■



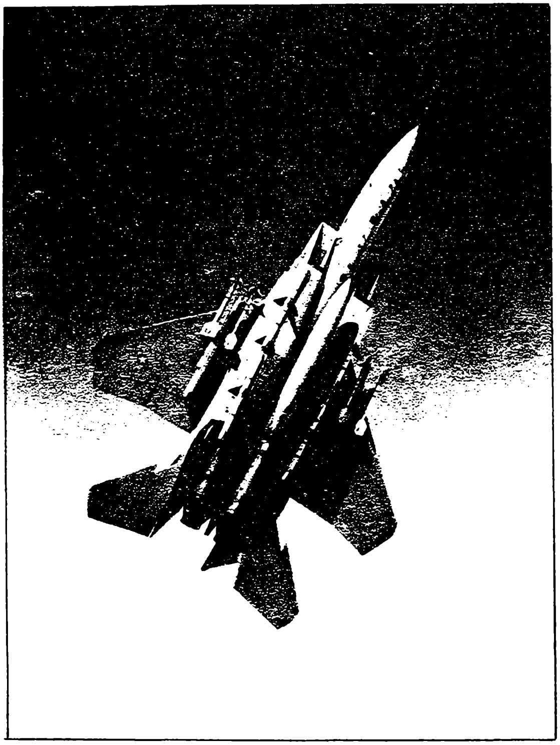
Co-author of this article, J.T. Johnston, with two units used in OWS test program. The smaller box on the left is the Voice Warning Generator as actually installed in the F-15 aircraft. It produces digitally synthesized voice warning signals associated with engine control, engine AMAD fire, fuel, landing gear, and over-G conditions. Larger unit is a voice warning test box containing switches, volume control, and speaker for evaluating quality of signals. Both devices were designed, developed, and fabricated by McDonnell Douglas Electronics Company.

PILOT'S PRIMER ON OVERLOAD WARNING SYSTEM...

(PUBLISHED 1982)

INSIDE OWS

By PAT HENRY/*Chief Experimental Test Pilot*



MCAIR is in the business of manufacturing high-strength, high-performance fighter airplanes. We've been doing this for over 40 years now, but despite their designed-in structural integrity, every one of our models — from the Phantom I in the early 'forties to the Hornet in the early 'eighties — could be overloaded (and possibly racked up) if the pilot did not maintain constant awareness and control of the flight parameters leading to these excessive stresses. True enough, but several generations of fighter pilots would attest that this control is much "easier said than done." Take the F-15 for example.

Available G in the F-15 is greater than the structural limitations of the airframe throughout most of the flight envelope. With only passive systems on board (accelerometer and HUD display) the pilot can easily over-G his Eagle without realizing it. To avoid going beyond the acceleration limitations, he must observe many different details, i.e. weight, roll rates, unsymmetrical limits, etc. This can make the job of tracking allowable load limits complicated. All this will be eliminated by the "OWS."

The F-15 onboard Overload Warning System monitors all parameters relative to structural loading of the aircraft. Inputs to the system are computerized and then translated into an aural warning, alerting the pilot when he is on the threshold of an over-G excursion. The system will be a valuable aid that provides cues to help pilots exploit the Eagle's capabilities better. The aircraft will have the capacity to be flown more aggressively — up to 9G's, giving him better advantage in combat conditions. Additionally, the OWS will be welcomed by the maintenance complex because it will remove guesswork from over-G inspections by informing the ground crew what has been overloaded and to what degree.

The DIGEST has presented several discussions on various aspects of this general subject, beginning back in Issue 1/78 when Pat Henry, Chief Experimental Test Pilot, and J. T. Johnston, Loads Section Chief, presented "The Cold Hard Facts of F-15 Acceleration Limitations." Air Force flight testing of the prototype OWS was described in Issue 1/81 by Captain Dick Banholzer of the 422nd Fighter Weapons Squadron at Nellis AFB, Nevada. In that same issue, Mr. Johnston (along with MCAIR Technical Specialist, J. F. Kunzelman) discussed the engineering approach behind the new warning system. In 1978, Messrs Henry and Johnston also collaborated in production of a video tape titled "The Overloaded Eagle," which was made available Air Force-wide. Now that OWS is appearing in production Eagles and scheduled for all-model F-15 retrofit, we asked Pat Henry to update our readers with some operational information on this highly effective way to assure a long and structurally happy life for the Air Force's finest fighter . . .

With the F-15, the "magic number" has always been 7.33. Over the years, we have devoted much effort to educating pilots concerning the significance of that number, and thereby reduce the number of over-g events. Dire things could happen to the airplane (and, administratively, to the jock) when you lived beyond your g means. Well, with the Overload Warning System (OWS), the magic number is now 9 . . . maybe.

We really haven't changed our minds, or the airplane. The answer lies in the ability to be *selective* and only go to the elevated load factors where the airplane can take it. Looking at Figure 1, it becomes immediately apparent that the Eagle enjoys a huge chunk of envelope where 9 g's are structurally permissible if below the basic flight design weight of 37,400 pounds. With this envelope programmed in the central computer and with OWS incorporated, we can now safely allow the airplane to operate at higher g levels while reducing the amount of maintenance inspections and aircraft down time - an absolutely undeniable cost/benefit situation!

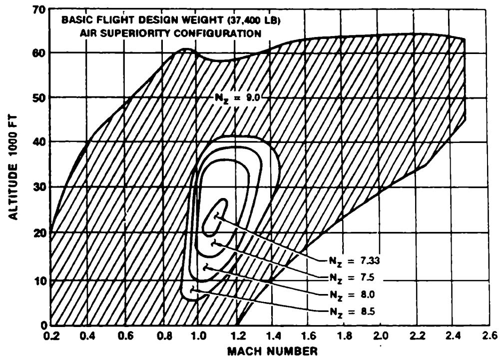
The OWS system is now being delivered in all new F-15C/D's, starting with F-15C #182 and D #28 (USAF Serial Nos. 80-0033 and 80-0056). Retrofit is in accordance with TCTO 1F-15-654/5 and will eventually extend back to include all operational A/B/C and D models. We here at St. Louis, and some fighter types in the field, are just getting acquainted with the production system, so this article will attempt to describe operation of the system as well as some problems already noted and in work. Please bear in mind that this is a dynamic system, and later changes may eventually make some of this material obsolete.

Switches, Knobs, and Stuff

The first physical evidence you've drawn an OWS equipped bird for your mission will be seen during preflight. In the nose wheel well, just aft of the Avionics Status Panel, is a dedicated OWS reset switch (more on this in a minute). Once in the cockpit, all will look the same until you fire up the Head-Up Display and Vertical Situation Display.

The new HUD display (Figure 2) is a dead giveaway; Current g's now share window 8 with maximum allowable g's. (Note the absence of decimal points and minus signs; we had to give them up to get it all in, but that's no problem in flight - if you can't tell the difference between 1.2 g's and 12 g's, you're already in trouble!) A word about the readouts. Current g's will often sit near 1 g, but not right on it. A

FIGURE 1 - F-15 MANEUVERING ENVELOPES



display of 09, 11, or 12 is not uncommon; beyond that warrants corrective action. As for allowable g's, we don't envision that pilots will fly around with one eye glued on window 8 to see what's permissible during ACM. Over a period of time and varied flight conditions, however, the readout will change in accordance with Figure 1; and pilots will inevitably develop a mental picture of their structural envelope. Allowable g's displayed are symmetrical, of course, not rolling g's.

The remainder of the OWS displays take place on the VSD. The pilot can cue these displays via the Navigation Control Indicator keyboard by selecting CCC with the Data Select knob, M2 in the Destination Data window, and entering R10 or R11 via the keyboard - very similar to setting up for HUD film titling. Let's say you do this during preflight while the Inertial Navigation Set is still aligning (although it's done the same way airborne or whenever), what you'll see will probably look like Figure 3. In some rare instances, it may look something like Figure 4 or 5, with leftover data from maintenance action or a previous flight - we don't want to make this too easy!

Figure 6 shows the location of all possible data which can show up on the VSD. Line 1 will be a summary of the worst conditions seen during all the high g events of the flight (or previous flight, perhaps) - the highest load factor, overload percentage, and severity code for all reporting stations. Line 2 will be the most recent high g event recorded (85% design limit load or greater). The remaining lines are in groups of three and concern themselves only with a particular reporting area such as forward fuselage or wing. Speaking of severity codes, Figure 8 is a tabulation showing what the codes mean relative to abusing your bird. The

FIGURE 2 - F-15 HUD DISPLAYS

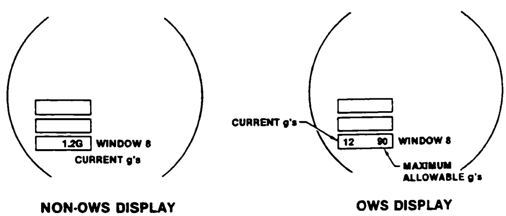


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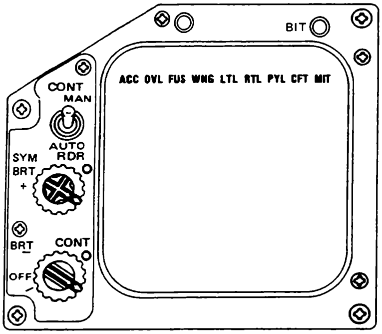
FIGURE 3 - VSD OWS DISPLAY COLUMN HEADINGS



ACC = ACCELERATION - LOAD FACTOR
OVL = PERCENT OF OVERLOAD
FUS = FORWARD FUSELAGE
WNG = WING
LTL = LEFT TAIL
RTL = RIGHT TAIL
PYL = PYLON
CFT = CONFORMAL FUEL TANKS
MIT = MASS ITEMS

FIGURE 4 - INITIAL CC LOADING DISPLAY

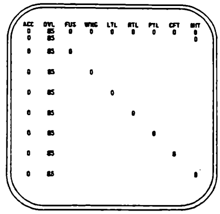
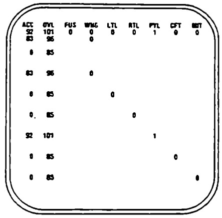


FIGURE 5 AFTER LOADING EVENTS



codes are used by maintenance to determine specific corrective action.

There's another new and interesting thing to look at on the VSD. Any time you select VSD BIT and hold down the BIT-Initiate button, you'll bring up the display shown in Figure 7. Now while doing this, move the control stick with your right hand and note that the stick force readout changes in proportion to your lateral stick inputs, left and right. It it doesn't, something is amiss; and the OWS may not be seeing lateral stick inputs. The OWS automatically reduces allowable load factor in *proportion* to lateral stick, as the method of protecting the aircraft from asymmetric or "rolling" g's. You can readily see the advantages of this mechanization - the full reduction is not imposed until full lateral stick is reached. Due to different roll rates obtainable at subsonic versus supersonic speeds, the OWS is programmed for a max reduction in allowable load factor of 20% subsonic and 12% when supersonic. Next, still looking at the OWS test pattern, compare fuel quantity display with your fuel quantity indicator. Normally, you'll seldom see greater than 100 pound difference. Based on built-in tolerances in the fuel quantity system and OWS computations, our experience to date shows no need for troubleshooting unless the difference exceeds 500 pounds.

FIGURE 6 - OWS RECALL DISPLAY

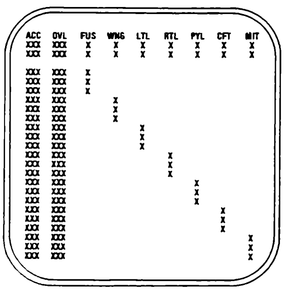


FIGURE 7 - VSD BIT DISPLAY

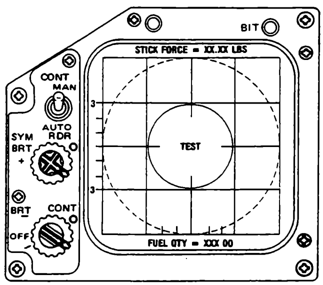
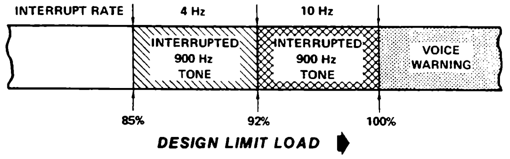


FIGURE 9 - OWS AUDIO OUTPUT



Continuing the Preflight . . .

Having thus amused and amazed ourselves while the INS was cooking, let's assume the align light is now flashing rapidly. As you move the INS mode knob from ALIGN to NAV, the previously recorded OWS info (e.g., Figure 5) will automatically clear in preparation for this mission's data. If you happen to have the OWS display on the VSD at this time, you'll see the transition to Figure 3.

There is one exception - by design. Note that line 1 of the Figure 5 OWS data shows a severity code of 1 (matching a 101% overload). Any entry associated with an overload - severity code of 1 or greater - will prevent clearing of OWS data from computer memory by cockpit switchology alone. To clear such data requires coordination by two people, one in the cockpit and the other in the nose wheel well. The one in the cockpit must first bring up the OWS display, via the NCI keyboard, using R11 (R10 won't cut it). The ground technician can then clear all the data by activating the reset switch in the nose wheel well for a one second or greater period.

Airborne Operation

Once airborne, the system will merrily compute allowable g's, taking into consideration over a dozen parameters, such as aircraft gross weight, flight conditions, etc., and keep you informed via window 8. Subsonically, you'll begin to see allowable g's decrease slightly above 0.9 Mach, per Figure 1. While maneuvering, the OWS will try and keep you honest via its modulated 900 Hz tone - the same one used for departure warning when you're flying sideways.

The aural warning schedule is shown in Figure 9. If maneuvering with 9.0 g's allowable, the 4 Hz interrupted tone should commence at approximately 7.6 g's (85% of 9 g's), and the step increase to 10 Hz interrupt rate at approximately 8.3 g's. (The crossover points - tone turn-on and increase to 10 Hz - evolved through ACM scenario testing by USAF pilots with a prototype OWS.) The voice warning is triggered if over 100% design limit load is reached (severity code of 1 or greater) and will say "Over g, over g."

FIGURE 8 - SEVERITY CODES

Code	% Design Limit Load
0	85 TO 100
1	100 TO 110
2	110 TO 120
3	120 TO 130
4	130 TO 140
5	140 AND GREATER

During, or right after, some of these high g excursions, compare the HUD g's and cockpit accelerometer g's with the audio. I think you'll find the HUD and audio tracking well - they speak the truth - and the main instrument panel accelerometer significantly low, particularly for rapid pitch rate maneuvers (g spikes). You may want to record some of these events with the Video Tape Recording System selected to HUD for closer examination after the flight. It's my personal conviction that the maneuver tone volume (not cockpit controllable) is too low - if I'm going to operate right up against the 100% design limit load line, I don't want that knowledge obscured by UHF inputs. That's something that can be adjusted if sufficient demand is generated by service pilots.

Negative G's

So far, only positive g's have been addressed, but the OWs will be watching also if you're an unloader. The structural problem for the aircraft is much less complicated under negative g's, so for the most part, the OWS will start triggering at 85% and 92% of −3 g's. This isn't a very large band to work with (−2.55 to −2.76 g's), so things will happen in a hurry.

We really don't recommend negative g maneuvering, even with the help of OWS. Some other neat things may happen in this regime - even properly serviced AMAD's often suffer generator drop-outs beyond -2 g's. If both generators drop off, so will the CAS; and you're liable to buy a bigger negative g step than you bargained for.

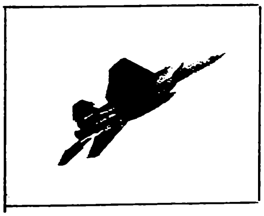
Program Improvements

Like all new systems, a certain amount of debugging has to take place as a function of time and in-service experience. For example, one very distracting anomaly has been the false triggering of massive indicated overloads while cruising peacefully at 1 g or so. We tracked this problem down to some extremely brief, high magnitude transients from the Air Data Computer which would drive the quick-response OWS through the roof while being ignored by the flight displays. A detect sensor and filter has been designed and flight tested which will desensitize the OWS without sacrificing any operational capability.* The sensor will identify illogical inputs by their abnormally high rate of change. The filtering will be achieved by terminating OWS operation during these short-duration spikes. For long-duration inputs, which can defeat the short-duration filtering, the OWS will be shut down for the remainder of the flight after approximately 30 seconds of continuous input. This will handle those failures which would trigger a continuous OWS tone or "over-g" message.

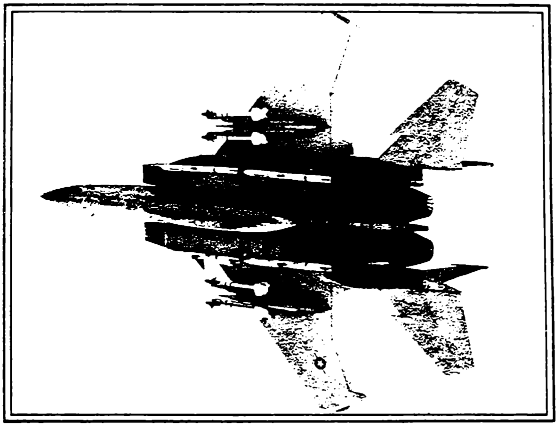
As touched on earlier, another area of possible improvement is the very subjective matter of tone volume levels. My personal feeling is that when depending on these tones to operate at or near 100% design limit load, their message has to get through, so they almost can't be too loud. We also have reports that other tone volumes (e.g., gear warning, high AOA) are too faint to suit many ears, so the entire spectrum is being investigated on a late model F-15C here in St. Louis.

As with all the other systems in the aircraft, we are very interested in feedback from service use to help us establish our priorities and apply our resources; so please let us hear from you (Commercial: 314-232-2142; Autovon: 263-6444).

*This modification has been submitted for USAF approval by ECP-01581C. ■



The DIGEST has presented several discussions on the F-15 "Overload Warning System" (OWS) in the past two years. Two articles in Issue 1/1981 presented the Air Force prototype test program and an engineering analysis of the system. After the production system was designed and aircraft delivered with OWS installed, a third article was written in Issue 2/1982 to familiarize pilots with this new capability for utilizing more of the Eagle's maneuvering envelope. First deliveries of OWS-equipped aircraft were to Langley AFB, Virginia, and after more than a year's experience, pilots there are enthusiastic about the system. MCAIR engineering provided additional informal information to personnel at Langley during this period; and now that more aircraft equipped with OWS are being delivered and a retrofit with TCTO 1F-15-655 is being accomplished, this informal data has been summarized to provide you with more on OWS.



(PUBLISHED 1983)

THE F-15 OVERLOAD WARNING SYSTEM

By JAMES W. HAKANSON/*Lead Engineer, Technology*

The F-15 Overload Warning System (OWS) was developed by MCAIR to increase aircraft maneuvering capability without any structural redesign or modification. Prior to the OWS, the "acceleration limitations" in conjunction with some of the "prohibited maneuvers" in the Flight Manual were used to protect the airplane structurally. These sections in the manual contained limits that were a tradeoff between capability and complexity. However, since the F-15 has additional capability and the CC (central computer) has no problems with the complexities, we gave the CC the additional task of performing real-time inflight structural loads analysis.

The central computer continuously calculates (approximately 70,000 operations per second) the forces on ten aircraft components (wing, tail, etc) and determines the allowable g. The computer summarizes this data through comparison and reduces the output to two numbers – one number (*least* allowable g's) that is displayed on the HUD (headup display) and another number (*largest* percent of allowable limit load) that controls tones/voice. The allowable g display and its tones/voice scheme were determined as a result of prototype testing by the Air Force at Nellis AFB, Nevada. (If you're interested in details on how this was done, you may want to look at the previous articles in the Digest.)

OWS Limits

Pilots have expressed a need for estimating OWS limits for mission planning. To accommodate this need, two symmetrical limit charts (Figure 1 and 2) were developed which are now part of the combat performance appendix section of the F-15 Flight Manual. One chart is required for the airplane and one for the interface between the airplane and conformal fuel tanks (CFT). Remember that asymmetrical maneuver limits are always more restrictive than symmetrical limits. When in the "second tone" (aircraft between 92 and 100% allowable limit load), g's must be reduced as lateral stick force is applied. Additional details concerning limits for each of the OWS measured components are presented below.

- FUS (Forward Fuselage) – This component should never limit maneuvering. Its primary purpose is to record the severity code in case the airplane exceeds 9.0 g's.
- WNG (Wing) – The symmetric limits are defined by the DASH ONE charts and the symmetric and asymmetric limits are displayed on the HUD.
- LTL & RTL (Left and Right Horizontal Tail) – Critical in the lower right hand corner of the envelope for negative g maneuvering and rolls when CFT's are carried. In this configuration, a severity code of greater than one would be difficult to achieve.
- PYL (Pylon) – Critical only for asymmetric maneuvers with full tanks or BRU-26/A mounted air-to-ground stores beyond half lateral stick. The allowable load factor is displayed on the HUD. (The individual bomb g limits are not programmed in OWS; therefore, the pilot must observe the limitations identified in flight manual Figure 5-6.)
- CFT (Conformal Fuel Tanks) – The symmetrical limits are described in the DASH ONE chart and displayed on the HUD. These also apply for asymmetrical maneuvers with smooth

lateral stick motion. CFT's are critical at high speeds/low altitude or for abrupt asymmetric maneuvers.

• MIT (Mass Items) — These items limit any maneuver to +9 or −3 g's.

OWS System Check

The OWS utilizes information obtained from the avionics systems, processes the data in the central computer, and activates the tones/voice through the integrated communication control panel (ICCP). There is no OWS "black box" as such and no OWS built-in test (BIT) system is necessary because the avionics which make up the OWS have BIT systems. (A block diagram of these systems is presented in Figure 3.) The philosophy behind this design is that when these systems are working properly, the OWS is working properly. When performing maintenance on these systems, a normal system check is required but an OWS functional check is not. The pilot can verify that the OWS is working (in flight or on the ground) by observing that:

• Allowable g's are displayed on the HUD and current g's are of reasonable value.

• The armament control panel (ACP) displays the actual airplane configuration.

• The systems supplying information for the OWS are up. These systems are of prime importance for flight with or without the OWS, and if inoperative, it would be obvious (for example, CAS, airspeed/altitude, and fuel quantity).

When performing an initiated ICCP BIT, you will not hear "Over G" but you will hear "Over" and "G" somewhere in the voice sequence. The voice warning system employs a vocabulary which it recites during BIT. The various messages are assembled from words or portions of words from the vocabulary. The message "Over G" uses the "Over" from "Over-Temp" and the "G". If the BIT light goes out, the tones and voice are present.

A "Closed Loop" ground checkout of the OWS can be done if you think it is necessary. It would require hooking up a WOW box, A TTU-205, and aircraft power. Put one stabilator full leading edge up and the other full down. The WOW box must be set for weight-off wheels and the TTU-205 increased from about MACH .4 at sea level to a maximum of 1.2 at sea level. The tones will be heard and the horizontal tail codes will appear on the VSD. The voice warning will occur for 30 seconds (after TCTO 839, continuous before) and OWS will then turn off. The Mach numbers at which these events happen are a function of the AOA probe position. It is not feasible to activate any component other than the horizontal tail on the ground.

Lateral stick force and OWS fuel quantity are displayed on the vertical situation display (VSD) during VSD initiated BIT. This provides an easy way of verifying these inputs to OWS. The lateral stick force is displayed to a resolution of .01 pounds; therefore, it is normal for the lateral stick force to jitter, even when not touching the stick. The force should be about 18 pounds at either lateral stop. The OWS fuel quantity should be within about 800 pounds of the fuel quantity indicator.

HUD Allowable g's Display

Display of the allowable g's on the HUD is a tool to assist the pilot in using the system. It is a prediction of the g level where the aircraft structure will be loaded to 100% of the allowable limit load. This prediction is a fairly complex task and there is not always perfect agreement between allowable g's and the g level where the voice activates. The pilot would probably never notice this disagreement in flight; however, it could be seen when reviewing the HUD video tapes. *The tones/voice are based on actual flight conditions, not predictions and are the prime means of using the system and structurally protecting the airplane.* The tones are activated as a function of percent (85% 1st tone and 92% for the 2nd tone) of allowable limit load, not percent allowable g's displayed on the HUD. The allowable g's displayed on the HUD are generally reduced during asymmetric maneuvers. It is doubtful that a pilot could use this feature effec-

FIGURE 1
F-15A/B/C/D
OWS SYMMETRICAL ALLOWABLE LOAD FACTORS

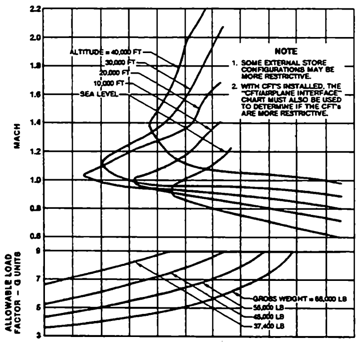


FIGURE 2
F-15C/D WITH CFT
OWS SYMMETRICAL ALLOWABLE LOAD FACTORS
CFT/AIRPLANE INTERFACE

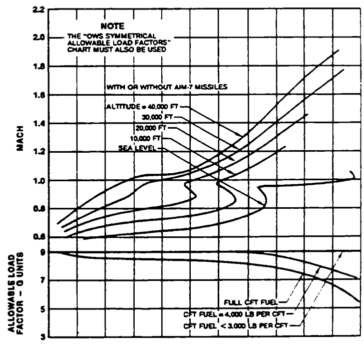


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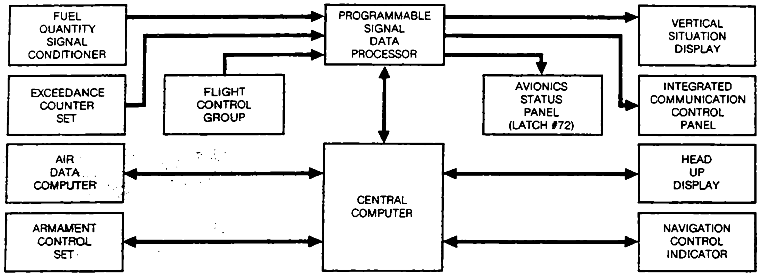
FIGURE 3
OVERLOAD WARNING SYSTEM BLOCK DIAGRAM



tively; however, it may be of use in post-flight analysis.

Severity Code 1's

The tones/voice scheme was devised in order to permit the pilot to use the airplane close to its structural limits and yet have sufficient time to check the maneuver without structurally overloading it. While realizing that occasionally the pilot will spill over into the Severity Code 1's unexpectedly, we have confidence, based on analysis and experience, that it is safe to continue a mission with Code 1's. (If you have recently been chewed out by the Ops Officer for bringing home a Severity Code 1, it may be a good idea to show him this article.) Here are a couple of examples of how a pilot can get a Code 1 unexpectedly —

- Holding a constant load factor with increasing Mach number in the .9 to 1.0 Mach region.
- Holding lateral stick and load factor while decelerating through Mach 1.

On the other hand, however, it is important not to fly in the voice warning region. This would lead to overshoots of Severity Code 2 and above. It is also important to point out that with the OWS, structural inspections take on a new significance. Structural failure and/or deformation can be predicted quite accurately since the actual load is being recorded. Therefore it's recommended that you fly the airplane using the beepers in order to keep "Betty" quiet. If you do spill over and she starts squawking and you get only a Severity Code 1, MCAIR feels it's OK to press on and continue the mission. The RTB criteria, however, is established by the USAF.

When flying with AIM-7 simulator plugs ("Simplugs") you can subtract 1% overload per "Simplug" from the wing column. For example, if you get 112% overload with four "Simplugs" aboard the actual overload is 112 − 4x1 = 108%, which in this case is a Severity Code of 1 and not 2. This only applies to the wing.

OWS and Conformal Fuel Tanks

All OWS airplanes are equipped with software to accommodate conformal fuel tanks with and without missiles. CFT's will rarely be the limiting component, especially if a few thousand pounds of fuel have been transferred. When the CFT's are full and Sparrow missiles are carried, the OWS may be sensitive to roll acceleration (lateral stick slams). The key ingredient here is smooth stick motion, trading g's for rolling motion when in the tones.

When flying an aircraft configured with CFT's, you may notice an increase in the number of horizontal tail warnings. An airplane with CFT's has an increase in nose-up pitch moment which cause the stabilators to work harder to balance the airplane. These warnings will occur at high speed and low altitude during rolls and push overs.

Upgrading the OWS

As experience was gained with the overload warning system, we learned several things of importance for both aircrews and maintenance personnel. As a result, the OWS has been upgraded. Discussed below are changes that enhance the system for the pilot and for maintenance operations. The changes will eliminate some OWS squawks and reduce maintenance manhours.

Shortly after OWS airplanes came off the assembly line, it was found that some air data computers were capable of generating large data spikes. For example, 50 milliseconds of Mach 3 or a −45 degrees of angle-of-attack (AOA). These spikes would result in "overloads" of 256% with Severity Codes of 5 while flying in 1 g level flight. To desensitize the OWS to high magnitude/short duration data spikes, TCTO 1F-15-839 (Software Changes to the Central Computer) was developed. Also, it was found that failures could occur which would turn on the "Over G Over G" voice warning and keep it on throughout the flight. With incorporation of the TCTO, the OWS will be automatically turned off after 30 seconds of continuous "Over G Over G." Remember any time the OWS HUD display is gone, you must observe the non-OWS g-limits. A third improvement is that if the aircraft is configured with external tanks but they are not entered in the ACP, or the tank-present signal is missing, OWS will be inoperative. Therefore, before squawking the OWS, check your ACP set-up.

The overload warning system has been used successfully for over a year now, and the more pilots learn about the system the more confidence they have in it and the more capability they get out of the airplane. OWS has been a big boost for the F-15 Eagle by greatly increasing the maneuvering capability of the aircraft and reducing maintenance. We will continue to keep you updated on the system through the Digest. In the meantime, if there is any additional information you desire regarding the system, please contact your local MCAIR Rep or give us a call. ■

PROPULSION ►

(PUBLISHED 1974)

Powering the Eagle

an Overview of the P&WA F100 Engine Development and Status

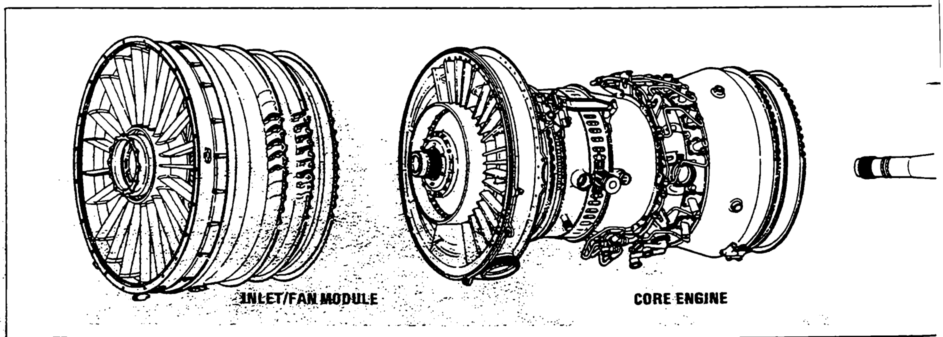
BY PAT HENRY/*Experimental Project Pilot*

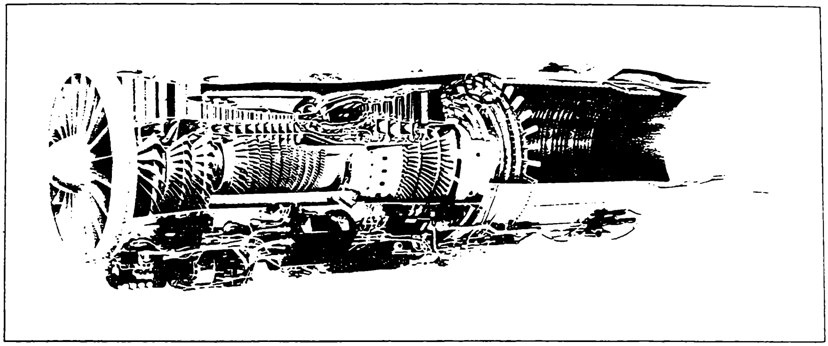
When the McDonnell F-15 airplane took to the air for its first flight on 27 July 1972, it was also the first flight for the Pratt & Whitney F100 engine. Both airplane and engine fared very well on that initial excursion, have experienced few growing pains since, and are just about ready to drop in on your flight lines. Therefore, I thought you might be interested in a little history as to how we got where we are today with the F-15 propulsion system.

The decision to embark on simultaneous development of a new airframe and a new engine was made back when the Eagle was just a collection of lines on the drawing board. This unique tack could only have been considered for a multi-engine aircraft with the inherent safety factor of a back-up engine. It also set the stage for a challenging development program, which I'll try to summarize for you here.

Because most of you are new to the Eagle, a general description of its propulsion system might be in order first. The F-15 propulsion system consists of the air inlet system, the basic engine (sometimes referred to as the "core" or "gas generator"), low pressure compressor (fan), afterburner, and the jet fuel starter (JFS). In handbook prose, the F100 engine is a low bypass, high compression ratio, dual spool, augmented turbofan.

Looking aft in order of appearance we see the low pressure compressor, which is a three stage "fan," followed by the high pressure compressor providing ten more stages of compression. Approximately 35%-45% of the air exiting the fan is bypassed around the high pressure compressor. Next is the combustion chamber, a single, annular, ram-induction type featuring continuous ignition. The drive turbine follows, consisting of four stages; the first two driving the high pressure compressor and the last two driving the fan. (The cockpit readout of Fan Turbine Inlet Temperature (FTIT) is taken from pickups between the second and third stages.) In the afterburner, the turbine



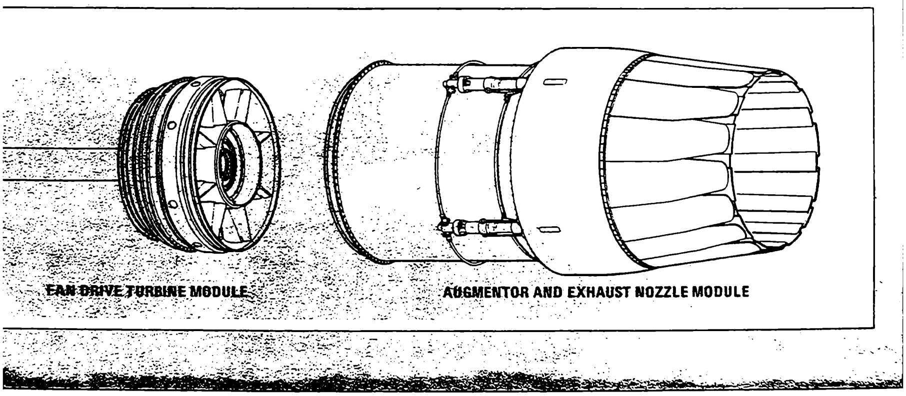


discharge gases and the relatively cool fan bypass air are mixed and burned in five A/B segments. Last, but far from least, is the exhaust nozzle-an item of far more interest and importance than in non-fan engines.

Nestled between the engines is the Jet Fuel Starter, a small gas turbine that provides the Eagle with the capability for completely unassisted starts - no A/C power or aircart required. The Air Inlet System, which is separate and independent on each side, consists of four interconnected variable ramps, a bypass door, and an Air Inlet Controller. Normal positioning of the ramps for optimum inlet operation is completely automatic.

(Back in the 4th Quarter 1973 DIGEST, Colonel Wendy Shawler gave you a status report on the F100 engine development program. I'll have to take you over some of the same ground in order to explain how early operating limitations the Colonel discussed then were worked out.) The flight test program actually began several months before first flight, with the arrival in St. Louis of prototype F100 engine #002. This engine was installed on a test stand that incorporated a complete aircraft accessory drive assembly – generator, hydraulic pumps, Jet Fuel Starter – the works. Thus, while Flight Test probed the engine for handling characteristics and soft spots, Engineering was also evaluating the many components closely associated with the engine. The stand had the capability of introducing hydraulic and electric loads as predicted by Engineering to add authenticity to the testing.

There are certain known characteristics peculiar to turbofans that provided a starting point for investigating this new beast. The most important operational difference between an A/B equipped fan engine and a straight turbojet is susceptibility to stalls during A/B operation. When lighting, modulating, or cancelling A/B the pressure in the tailpipe is trying to rise and fall dramatically. These pressure spikes have direct access to the back of the fan, ▶



right up the bypass duct. If the pressure spike is large enough, be it up or down, the fan will stall. This always results in a very attention-getting bang, and can even lead to a stall and stagnation of the high pressure compressor. (We'll discuss the wonderful world of stagnation in more detail later, perhaps in a companion article.)

So how do we handle these pressure spikes? With a very sophisticated and quick-reacting exhaust nozzle. The convergent nozzle area (Aj) actually achieves two important goals by regulating the pressure in the tailpipe - it not only provides stall margin for the fan during engine transients, it also fine tunes the speed of the fan by establishing just the right amount of back pressure. Would you believe, unlike some turbojet engines you've been flying, the nozzle indicator is now one of the prime gauges for your hawk-like eyes to monitor during engine handling? Believe it!

In exercising old #002, we soon learned that A/B lights on the test stand presented a two-fold problem. First, because the tailpipe was stuck in a hush-house and we weren't smart enough yet to tell by watching the nozzle indicator, we couldn't be sure of solid lights in min-A/B. Furthermore, we ran the risk of extremely hard auto-ignition lights at high A/B fuel flows if we missed the segment one light-off (A/B ignition is only on for approximately one second to avoid late lights). Our testing led to refinements in the A/B hardware and the unified fuel control, and to considerable operational experience, all of which enhanced safe A/B operation in the early flights.

We are continuing to this day to test and improve the A/B so that it can be used at any time and in any manner the pilot deems necessary. Toward this end we now have an A/B that can be used throughout the flight envelope. The lights are soft and reliable, and the fan has sufficient stall margin to cope with any pressure spikes the nozzle doesn't smooth out. If a light is missed, the engine no longer tries to bite you - an A/B reselection starts the A/B ignition timer for another cycle and you're off to the races.

Besides inducing stalls through hard A/B lights, we found that rough engine handling could sometimes get us in trouble. In particular, the engine was sensitive to throttle "Bode's" near idle (chop to idle followed by a throttle reversal). This had obvious operational implications. I don't mean to imply that fighter jocks are routinely heavy handed on the throttle, but if you're forced to approach the throttle quadrant with the same caution and forethought as does the airline Captain your adversary is going to have you for lunch!

Our test stand experience led to very early flight testing in this "off idle" area. We found plenty to complain about, but our P & W friends were quite responsive. The final outcome is that production engines now have an optimized RCVV (Rear Compressor Variable Vane) schedule for maximum stall margin. Working in conjunction with this is a combustion chamber minimum pressure limit (the famous 42 psi min Pb) which in effect provides an altitude bias to the idle RPM. The higher you go, the higher goes idle RPM - driven up by the minimum allowed burner pressure - thereby giving adequate near-idle stall margin at all altitudes. So, latter-day Lufbery's, feel free to use that throttle as the situation dictates!

Another apparent characteristic of fan engines is slow spool-up and, while this may be no big deal for airliners, it's flat unacceptable for fighters. Because of our background, all of us at McDonnell tend to use the J-79 as a baseline by which to judge the F100, and when it comes to engine acceleration (and quick thrust response) the old 79 is a hard act to follow. Producing rapid acceleration in an F100 presents a two-fold problem. It's hard enough getting that big fan moving, but in addition, no temperature overshoot is permissible as is used with the J-79. This is necessary to prevent heat distress since the turbine is already working at state-of-the-art temperature extremes. In flight, the slow response of the early engines bothered us across the board - it affected everything from raw acceleration to precision instrument landings. However, Pratt & Whitney maintained a steady program of improvements which introduced changes to both UFC (Unified Fuel Control) and EEC (Engine Electronic Control) for better engine acceleration and thrust response.

The UFC has the basic accel-decel fuel flow schedules, whereas the EEC has more of a supervisory role. This electronic control monitors the key engine parameters and makes inputs, usually through the UFC, to set and maintain the upper limits, such as maximum FTIT and fan speed. The fuel flow schedules in the UFC have been optimized for maximum spool-up rate; the EEC has been modified to accept the highest possible rate of FTIT increase while still guarding against a temperature overshoot. The end result is that you'll be flying the most responsive fanjet on the market. Be they small throttle inputs or slams to the firewall, when you ask for power . . . you've got it.

The list of improvements that have come from ground and flight testing the F100 is too long to cover in detail in just one article. Three of the most important from an operational point of view have been briefly discussed; take my word for it, there are numerous others and product improvement is a continuing thing.

One of our prime objectives in flight test was to uncover deficiencies of this new engine so that improvements could be developed before the airplane was introduced into service use. Another primary goal has been to define the operational envelope of the machine in the hope there will be no surprises when you start flying it. To that end, we have spent countless flight hours with fully instrumented engines determining such things as the airstart envelope, best airstart techniques, the undesirable characteristics of unsuccessful airstarts, the afterburner envelope (which can vary as a function of pilot technique), the various (but happily infrequent) failure modes of the engine, and its performance in all the corners of the envelope.

Many of these operational characteristics will be discussed in detail in later articles. Therefore, your questions and comments are strongly encouraged so we can cover the areas of prime interest first, and in sufficient detail. If your response to these gab sessions is just a fraction of what I predict it'll be for the airplane itself, the "Writer's Cramp" will be my pleasure. ■

F100 Status Report

By PETE PILCHER/*Senior Pilot*

One of the design goals that an airframe manufacturer must address during the development of aircraft is the integration of a powerplant into the weapon system - successful operational fighters are built around powerplants that are dependable, in the proper thrust class, and economical to operate. Another goal of the design team is to provide an engine that requires minimal attention from the operator - if an engine can be trusted to respond correctly to the operator's demands without supervision, we then have a "fighter pilot's engine."

Some engines have earned this reputation, while others have not. The J79-GE engine in the F-4 enjoys such a reputation, but had to earn it; in its early operational career, the J79 gave us problems. The F100-PW-100 in the F-15 is not yet, in my opinion, a fighter pilot's engine, but with improvements and experience, I expect it to become so.

During early testing of the F100, we identified several problem areas:

- Starting irregularities
- Off-Idle stalls
- A/B handling - stall/stagnation
- High idle thrust

We have completely put to rest the off-idle stall situation. Starting irregularities and high idle thrust can be tolerated by operater compensation. We have not fully solved the A/B handling problem to date, but we're working at it. One of the latest efforts in this area was conducted at Edwards AFB last spring and summer when one of the pre-production F-15s was instrumented and outfitted with known good and bad engines from Langley AFB. Our goal was to understand and explain why some engines were stall/stagnation prone even though they passed normal trim checks. This testing gave us insight into the mechanics of the stall and gave the engine maker insight into ways to both prevent stall and improve recovery after stall.

The F100 is frequently called a high-technology engine and with good reason. It is complex in mechanization and control; has a high thrust-to-weight ratio; and operates at high internal temperatures. The engine is certainly an advancement in the state of the engine art; no other operational fighter engine and no engine developments currently funded for fighter applications match the F100 thrust/weight capabilities.

New F-15s are coming off the line with engine serial numbers between 900 and 1000. These new engines are operating relatively trouble-free; they start and run properly with the exception of an occasional stall in the upper left hand corner of the envelope (high altitude/low speed).

With this background, let's review where we stand today and let's go through just what we see in the various phases of flight, from start-up to shutdown.

The current production engines are starting rather well. On rare occasions, if the engine does not continue to accelerate through the 35 to 55 percent N2 RPM range, the pilot must add fuel flow by placing the engine start fuel switch in SEA LEVEL (HIGH). Although rare, we still have stagnated starts in which the N2 RPM turns around while the FTIT continues to climb. Sometimes the pilot hears the stall as a pop. If the throttle is secured, the engine cooled by continued spinning by the JFS, and then restarted with the bleed selected to that engine, it almost always starts properly the second time around. The airplane driver who closely monitors engine gages will notice the closed loop idle come on the line (usually 5 to 45 seconds after the engine gets to idle) by the 2 or 3 percent increase in N2 RPM.

For those of you who enjoy chasing electrons, Figure 2 shows a simplified electrical schematic of the system. As you see, power is supplied from the Engine Control/Essential 28 Volt DC Bus, which means that so long as at least emergency generator output (normal mode) is available, power will be supplied to the engine start fuel bypass system. If the main generator is off the line, the derich solenoid is receiving power and approximately 100 pph of fuel will be bypassed back to the fuel pump, thus reducing the starting fuel flow. Now, at approximately 50% N_2, when the main generator comes on the line, the time delay relay energizes. It holds its breath for 30 seconds, then closes; in so doing, it removes power from the derich solenoid and the bypassing terminates. This can be noted by about a 1/2 percent increase in RPM. If you choose to check the operation of the fuel bypass system at this point, just hold the Start Fuel switch in the ALTITUDE position. This action energizes the override relay, which in turn allows the derich solenoid to be powered. A corresponding 1/2 percent RPM decrease is evidence that the relays and logic are functioning.

Okay, So What?

Having completely mastered the starting fuel bypass system, you might reasonably ask how can that knowledge help? If the system always worked as advertised, and if the engine always started successfully, you wouldn't need the knowledge. However, history has shown that occasional (?) troubleshooting may be required. Also, since it's a little more difficult to walk away from an aborted airstart, we may generate some interest in that arena.

The thoughts I would like to pass on about airstarts are based on observing hundreds of starts on prototype and early-model production engines. A separate contractor-conducted airstart program on the final F100(3) production engine was never run. Contractor engine development halted in October 1974, prior to delivery of the first F-15 to TAC, when engine test funding was terminated. However, the basic airstart considerations should not change dramatically from one engine series to the next.

For identification purposes, we categorize airstarts either as "spooldown" or "windmill." As the names imply, the first type includes any airstart attempted while the engine is still winding down. We're guessing that 99% of all operational (vs test) airstarts will be of this variety. The windmillers are those wherein N_2 has stabilized at some given RPM, or is increasing, when restart is initiated.

Talking primarily to spooldown airstarts, the absolute best point at which to "pressurize" (throttle to idle to begin restart) is debatable. There are certain basic factors, however, that set practical upper and lower limits. It is generally good to allow RPM to drop below 50% for a couple of reasons. At about this point, the engine start bleed strap (7th stage compressor bleed) will open, increasing compressor stall margin. The start fuel bypass system will also energize to the automatic derich mode as the generator drops off the line. If the flight condition is such that the engine wants to windmill near or above 50%, these factors become academic because there is so much airflow available. At the low RPM end, I prefer to pressurize no lower than 20% because, for low airspeed airstarts, the spooldown rate can be such that 12% or lower could be reached prior to getting light-off. Below 12%, the engine very soon runs out of electrical power to drive the ignition system, and fuel pressure to open the minimum pressure side of the P & D valve. Airstarts have been attained with as low as 9% N_2 indicated, but light-off cannot be relied upon in that region.

Another key factor, and a difficult one to pin down exactly, is FTIT at time of light-off. It has been very obvious that the cooler the engine, the better the chances. Therefore, if the operational situation permits this luxury, you'll be better off letting the engine cool down somewhat prior to initiating restart. A rule of thumb of 500° FTIT maximum has been offered in the past, but there is nothing magic about this temperature - let it cool down while spooling down, within practical limits.

If it's *really* your lucky day, and you find yourself needing airstarts on *both* engines, there are only two reasonable approaches to the problem. One of them, the nylon letdown, hopefully can be held in abeyance while the other option is tried. Say we start off by noting a sudden thrust loss. While immediately pushing over if possible, to retain airspeed, check for stagnations. If FTIT is low or dropping, you've probably suffered fuel starvation to the engines and your future is looking very bleak. If FTIT is holding or rising, you've got stagnations - the more normal mode of total engine failure. After pulling both throttles to idle, exercise care to shut down only one of the engines, thereby leaving the other in stagnation to provide hydraulic control power. You may only get one shot at it, so do your best to get in the heart of the airstart envelope. Our experience with stagnations in the past would indicate that the engine will suffer this severe environment without shedding pieces. However, the possibility exists that it may freeze up once the throttle is chopped.

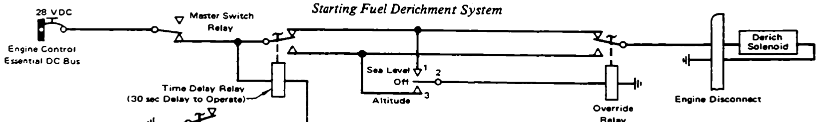
What If ....?

Let's assume you're hot-footing it down into the middle of the airstart envelope, trying to get a light-off, and nothing is happening in the engine room.

Again, our experience is that no light-off is probably the result of insufficient starting fuel flow; too rich mix-

FIGURE 2

Starting Fuel Derichment System



Control Switch		Fuel Flow-PPH	
Position	Function	RPM<50%	RPM>50%
Sea Level	Override	450	450
Off	Auto	350	450
Altitude	Check	350	350

tures are good lighters, but lead to hot starts. Therefore, after a reasonable wait, you can improve the light-off chances by holding the Engine Start Fuel switch to "SEA LEVEL," thereby manually overriding the automatic bypass system. What is a reasonable wait? Give it about 20 seconds, anyway. The engine needs 10 to 15 seconds, on the average, to energize the P & D valve, fill the fuel manifolds, and fog (atomize) the fuel. Try not to rush the count - we all know how time flies when you're having fun.

At the other extreme, if you're in the envelope, but getting hot starts, perhaps the basic fuel flow schedule is too rich or you're not getting automatic derichment. In this case, by manually holding the Start Fuel switch to "ALTITUDE" and going as fast as the situation permits, you're doing all that's possible to achieve a good start. Now aren't you glad they put the panel way back in the right hand corner?

In closing, let me just touch on windmill airstarts. Our limited experience with these has shown that they're almost always good, probably because the airspeed required to sustain RPM puts you well into the envelope. If anything, those starts may tend to be slightly lean - stand by to override the bypass system by using the "SEA LEVEL" position to get a light-off, or to help accelerate a cool, hung engine up to idle.

(PUBLISHED 1976)

"gotta find a home..."

One of the top-forty tunes of a decade ago focused on the home-hunting habits of the common boll weevil. The song shared the weevil's feelings about the farmer, and maybe about the world in general. Anyway, the song always came back to the weevil's lament, "I gotta find a home."

Well, weevils aren't the only critters that are searching for a home; as another song put it, "Birds do it, bees do it" too! And that's what this article is all about: the tendency of little creatures to seek comfort and security within the nooks and crannies of any potential homesite — even aircraft.

For instance, a European Phantom was reported to have a true airspeed and Mach indicator problem in which both indicators read low in flight, yet checked out fine on the ground. Having experienced this type of trouble a number of times before, the McDonnell Field Service Engineer suggested a visual inspection of the pitot tube. Lo and behold: an obstruction about six inches into the tube. Further investigation revealed it to be a bee's nest in which the material looked much like the filter from a cigarette.

In another instance, a "mud dauber" fabricated a home in the end of a pitot tube, a home which looked for all the world like another cigarette filter plugged into the end of the tube.

In yet a third case, a small bee (about a quarter-inch long) made its way well into the pitot plumbing. In this case, the obstruction was so far back that it could not be seen. Needless to say, troubleshooting of an obstruction like this can be(e) a real problem.

Instruments are not the only portion of the aircraft that can be bothered by bees. Another report tells of a pitch trim problem. The aircraft discrepancy sheet indicated that too much nose-up trim was required to maintain level flight. Ground tests did not expose the problem but another flight of the aircraft brought a repeat gripe. At this point, the pitch trim venturi was checked, and guess what? Another "little feller" had found a home, cutting off air to the bellows. Removal of the bug restored the trim system to normal operation.

Magnified insect remains removed from pitot static system. (APPROACH Magazine)

An East Coast Marine aircraft had come out of calendar check after an extensive time in storage. During the initial engine runup, it was noted that the left-hand engine bypass bellmouth remained open. Troubleshooting of the system revealed that both the pitot tube and static port for this bellmouth were completely plugged. The culprits, in this case, were some industrious South Carolina mud daubers who were quickly done out of house and home.

Wasps and bees are not the only creatures to create problems of this sort. In another instance of pitch trim trouble, the venturi was found to contain the carcass of a "Texas-size" beetle.

No aircraft is immune from invasion by insects. Though the likelihood of bug, bee, and beetle intrusion is greater in warmer, tropical climates, there is no place hosting Phantoms and Eagles that can give you an absolute guarantee of a bug-free environment.

So, what's the solution? Probably the best suggestion is that every pitot-static opening be properly covered when the aircraft is not in flight. The simple fact that a cover is installed is not sufficient; the cover must be installed properly and securely.

When one of these problems does arise, use good judgment during troubleshooting. Consider what functions of the aircraft are affected, and see if any associated systems are affected as well. In checking pitot static systems, it is often easy to isolate a problem to a specific location by installing test equipment at various points in the system. However, no test equipment can surpass simple, down-to-earth common sense trouble isolation.

So, getting back to one of the songs we mentioned at the beginning of the story — "Birds do it; Bees do it; Even educated fleas do it"; but it's a whole lot better that they not do it within aircraft pitot, static, or pitch trim plumbing. If that beetle, bug, or bee has "gotta get a home," let it be in that subdivision down the road, not in your Phantom or Eagle.

Warner-Robins AFB/15 Nov 1978 . . . "USAF F-15 73-108 flew a successful FCF late yesterday, and all specified parameters of inflight JFS operations were met."

JFS AIR OPERABLE

To Eagle Drivers, the cockpit decal above should be a very welcome sight! Because the incorporation of the airstart capability is an expedited program, F-15s with that capability are now in the field even though the flight test programs have just been completed. This article presents the preliminary flight test results as well as other advance information about the new JFS capabilities.

(PUBLISHED 1978)

By PAT HENRY/*Chief Experimental Test Pilot*

Commencing with F-15 No. 77-0076 (F-288), delivered to the USAF in August of this year, and after incorporation of TCTO 1F-15-572, the Eagles you fly will have the capability to use the Jet Fuel Starter (JFS) for inflight assisted starts. We are still gaining experience with this long awaited capability, but already certain pros and cons have come to light. Your ability to use this added feature to possibly save an airplane (and thus avoid a mountain of paperwork) will be greatly enhanced by a thorough understanding of the JFS and its interfaces; something well beyond just finding the switch and pulling the handle.

To set the appropriate stage and help hold your attention, assume for the moment that you're the proud, but slightly worried, owner of an F-15 glider — you've just experienced a double engine failure! Generators are dropping off, and the Caution Light Panel looks like a Christmas tree — so quick, what do you do?

Obviously, you're going to keep one engine in stagnation for hydraulic power and lower the nose for airspeed. Regardless of altitude, if the dual engine failure has occurred at a fairly high airspeed — and this becomes a judgment call — you'll certainly want to try one spooldown airstart as soon as the restart RPM and FTIT window is reached. However, from this point on, your actions could be quite different, and your probability of flying home greatly enhanced, with inflight JFS. Let's come back to this sticky wicket after discussing the inherent capabilities of the JFS.

From the flight test results to date, it appears that good jet fuel starters can run as high as 25,000 feet. There are many, many variables involved, most of which are insufficiently understood by the engineering community at this time and not measurable by the pilot anyway. Therefore, I'm predicting a wide variance of JFS capability as a function of (JFS) age, its fuel nozzle condition, manifold losses, centerline stores, etc. You can't do much about these except check the JFS performance during FCF's and rid yourself of *all* centerline equipment (including pylon!) during an actual emergency. You can, however, be aware of the envelope and aim for the conservative side. Figure 1b shows the envelope performance we recommend based on flight checks at St. Louis and Edwards AFB.

Figure 1a shows that at 17,000 feet and below, we are enjoying JFS operation out to 450 knots, a speed that should provide sufficient torque to

FIGURE 1 – JFS INFLIGHT ENVELOPE

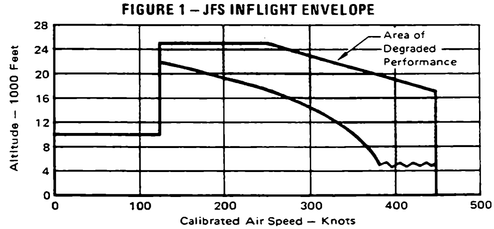


FIGURE 1a–DEMONSTRATED

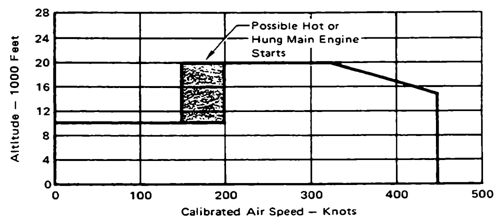


FIGURE 1b–RECOMMENDED

spool-up an engine from zero RPM with ram air alone. It did not seem practical to investigate JFS capabilities beyond this speed. On the slow side, if you can fly comfortably, the JFS can run. Even though age and wear can degrade JFS performance, thus lowering its airstart ceiling, we feel that 20,000 feet should be routinely attainable. A much more dramatic degradation is seen with any kind of centerline store, including a pylon alone; hence, the recommendation to clean off that station when you really need the JFS. An area of degraded performance is shown, rather than a hard line, because the variables of centerline store type and JFS condition make it impossible to accurately predict performance for any given combination.

light gives you positive feedback that you have discharged the accumulator to start the whole chain. Inflight, at least with one engine running, you don't have the audio cues you're familiar with to verify JFS spool-up, light-off, and engagement. Perhaps in the quiet environment of a double engine failure, you'll hear some of this; but we're not too anxious to explore this area. Similarly, the JFS Ready light is your best, if not only, cue that the JFS start has been successful.

Speaking of successful JFS starts, if no Ready light is observed within 10 seconds of handle pull, *it is of paramount importance that the JFS switch be promptly turned off*. Two of the anomalies noted during the test

FIGURE 2 – DESCENT RATE COMPARISON(1)

Altitude (1000 Ft)	Speed (Kts)	Engine Condition	Descent Rate (Ft/Min)	Elapsed Time: 20K - 10K (Min:Sec)
20	350	Both Windmill	8843	1:08 (2)
		1 Windmill 1 Idle	7723	
	220	Both Windmill	3312	3:01 (2)
		1 Windmill 1 Idle	2715	

(1) Estimated; Clean Configuration, 33,000 Lb. Gr. Wt.

(2) Assumes Constant (20K) Rate of Descent. Actual Rate will Decrease with Lower Altitude.

Just as your main engine thrust is reduced as a function of lower air density, the useable JFS torque diminishes with altitude. Therefore, we could possibly find ourselves in a region (very high in the JFS envelope) where any given JFS may run, but not have sufficient excess torque to successfully start an engine. If you attempt an engine engagement in this region, you will probably have a near stand-off, i.e., very little engine acceleration after light-off, until your glide takes you to lower altitudes (which shouldn't take too long with both engines out).

SYSTEM INTERFACES

It is normal to see a JFS Low Light on the Caution panel after the starter handle is pulled. Our experience indicates that most JFS bottles will recharge in one to two minutes; but depending on hydraulic pressure and accumulator pressure switches, it could take as long as four minutes. This is of interest, for different reasons, depending on whether you're in a training/FCF environment or saddled with the real thing. For one, the Low program were hung and slow JFS starts; either can generate unacceptably high temperatures in the JFS. Since your only insight to an abnormal start is via the Ready light, this further underscores the necessity for an operable light.

Since the JFS accumulators are also your emergency utility hydraulic power source, we believe it is only prudent to test the JFS inflight with both accumulators charged, i.e., no JFS Low light, and then only discharge one bottle during the test. Then, if the accumulator fails to recharge after the JFS start, you have not deprived yourself of *all* this hydraulic back-up. (Since the emergency brakes use the first accumulator, they will be lost in the event of no recharge, but emergency landing gear extension and nose wheel steering should still be available from the second accumulator.) Furthermore, only pulling the first bottle, insofar as practical, complies with the recommendation to attempt only single accumulator starts inflight — two accumulators discharged simultaneously could accelerate the JFS too rapidly, resulting in missed ignition.

To maximize safety while flight testing here in St. Louis, we install a jumper wire, which permits the JFS to run even with both engines operating. Inflight, therefore, we can start the JFS and confirm its operation without shutting down an engine. This is particularly useful when testing the JFS in the upper left hand corner of the envelope where an engine, if secured, would smartly wind down to zero RPM.

MEANWHILE, AFTER THE STALL . .

Now, I'd like to pick up the double engine failure script from the third paragraph; but let me preface the discussion with an explanatory note. What I'm presenting is my emergency philosophy — how I'd hope to handle the problem if so unlucky as to lose both engines. Of course as squadron pilots, you must follow established procedures/guidelines. While it is certainly not my intention to undermine such doctrine, I feel that open discussions are always appropriate, and could lead to improved procedures for everybody.

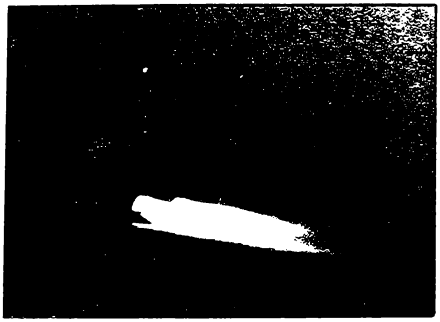
Dive or Glide?

Obviously, there are two generalized initial altitude conditions — either inside the JFS envelope or above it. With one engine in stagnation to support the hydraulic requirements of the flight control system and emergency generator, my choice would be to transition to a 220 knot glide, versus 300, 350, or whatever, as soon as the JFS envelope (Figure 1b) is reached and punch off ALL centerline garbage.

From high altitude, I still see the need for the 350 knot dive, or at least an attempt to reach this condition down to the JFS envelope. First, this will give you the best chance for an early spooldown airstart (on the non-hydraulic support engine). Second, it would give you an early look at whether this engine can even achieve a light-off, whether inside the airstart envelope or not. It would be very reassuring to know the engine is getting both fuel and ignition. Third, a low descent rate glide from very high altitude can mean an excessively long period in stagnation for the back-up engine. We know the stagnation is progressively deteriorating that engine, but there's no way to measure when we're about to punish it to death — literally.

You may ask, "Why the 220 knot glide once inside the JFS envelope?" This would be very close to the speed for max L/D, and therefore, an opti-

(PUBLISHED 1980)



F-15 Dual Engine Restart Procedures

By GLEN LARSON/ *Engineering Test Pilot*

Eagle engines ordinarily give off the reassuring glow pictured above. However, for that rare occasion when the light from both sides suddenly diminishes, here are some newly developed dual engine restart procedures.

Double engine failures or malfunctions of any kind are no fun; and fortunately, the problem rarely occurs in the F-15. Since it is an isolated occurrence and no one has a great deal of experience in handling dual engine problems, we felt that some research and simulation effort was needed. Numerous dual engine-out simulations were conducted in the Goodyear simulator at Luke AFB, where we found that pilot technique often departed from flight manual philosophy.

We examined the problem and suggested a procedure that is a reasonable resolution consistent with engineering design and pilot behavior. The procedure was presented and accepted at this year's F-15 Flight Manual Review Conference. In short, our goal was to maximize the probability of regaining engine operation regardless of the failure cause. Of course, all situations cannot be covered by a single procedure; and to quote the flight manual, "you must determine the most correct course of action using sound judgment, common sense, and a full understanding of the applicable system(s)."

The simulation effort revealed some interesting pilot techniques. Many pilots will tend to lower the nose excessively. If the problem was introduced at very low airspeed and extreme nose-up attitude, the pilot tended to enter a 70-80° dive, remain in the dive, and occasionally go supersonic while attempting to clear a dual stagnation. This technique drastically reduces the time available to clear the stagnation and often results in some pilot disorientation. A dive angle of approximately 10° will generally sustain 350 KCAS and sufficient windmill RPM on the engines to retain hydraulic power. For example, for a *clean* aircraft gliding at 350 KCAS at 10,000' MSL, with one engine stagnated and the other windmilling, the actual glide angle will be 12.8°. Remember, this is to sustain RPM on a previously windmilling engine. If you have let RPM go to zero, airspeeds of 450 KCAS may be required to get the engine windmilling again; but 350 KCAS will sustain RPM at 18-20% with *normal* flight control demands.

A major pilot concern was that some source of hydraulic power, usually from a stagnated engine, be retained at all costs. As you can see, an engine windmilling at 18-20% is adequate unless you constantly cycle the flight controls, thus imposing a *continuous* demand on the system. It is important to note that normal gliding flight does not tax the hydraulic power available from a windmilling engine. (By the way, when was the last time you practiced flying on the standby instruments? Remember, with RPM on both engines below approximately 45%, the main generators will drop off the line and the primary flight instruments will freeze at their last readings.)

Since we are addressing a specific procedure, it's best to examine each step with its supporting philosophy. Dual engine problems are usually associated with stagnations; therefore, the procedure is oriented to a high altitude, low airspeed problem. Assume you are at 35,000', 150 KCAS, and both engines start giving you problems:

Step 1 - Both throttles - CHOP TO IDLE (Military if in A/B).

This assumes that the first indication of a problem was a stall (it usually is) and is an effort to clear the stall. Unfortunately, it wasn't your day, and the engines entered a classic stagnation.

Step 2- Throttle (right engine) - OFF WHILE ESTABLISHING 350 KNOTS.

Lower the nose to establish a 350 KCAS glide while shutting an engine down. We recommend the right engine due to lower hydraulic demands on that engine, which results in a lower spooldown rate and a higher RPM for a given airspeed and altitude. If maximum FTIT is a consideration, then the left engine may be a better choice. If the problem is due to a flameout, the right engine is always the best choice.

Step 3 - Perform restart procedures.

It is important to emphasize the spooldown start procedure. It is not *necessary* to wait for a stabilized RPM before attempting a restart. A spooldown start is performed by moving the throttle out of cut-off at or above 25% RPM. Since time is critical, we recommend initiating the start attempt at 25% RPM even if airspeed is low or FTIT is high. This procedure gives the *best* chance for a restart. Placing the throttle in the mid-range position instead of Idle will deliver thrust 8-10 seconds sooner. This "tiger start" technique may go against your instincts, but it is the best way to get power back — fast! This technique allows the engine to accelerate quickly and minimizes the chances of a stall.

Placing the throttle to mid-range

allows the engine controls to bypass the Idle operating condition and move directly to the condition called for by the pilot. Since the engine doesn't have to establish a stabilized idle, time to regain thrust is reduced; and as an added benefit, stall margin is increased. If the throttle were placed at Military, exactly the same sequence occurs, *except* that the EEC comes into play. If the original problem was related to an undetected EEC problem, then the original stall or stagnation may reoccur; therefore, placing the throttle to mid-range is the optimum choice. If you move the throttle out of cut-off as the engine spools down through 25%, the RPM will continue to decrease to some value below 25%, stabilize, then begin to increase as the engine relights, and FTIT will increase shortly after RPM.

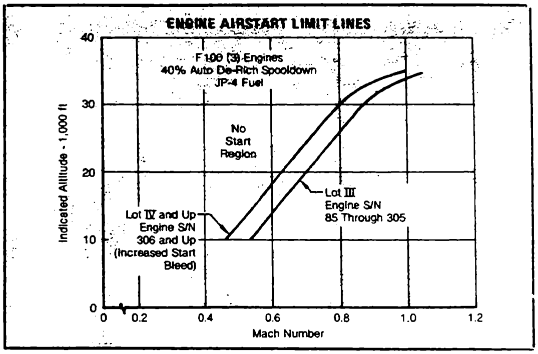
We elected a spooldown restart instead of using the JFS on the first engine because the engine start envelope is larger than the JFS operating envelope, and the 350 KCAS glide during the start will descend the aircraft into the JFS envelope. Some other relevant considerations during the first start are —

- The upper limit of an airstart for the subsonic case is 35,000 ft.
- Avoid steep dives since time available for restarts is drastically reduced, and ejection at high speed in a steep dive may be out of the envelope.
- 350 KCAS is more than sufficient speed for a "zoom and boom" maneuver, if necessary.
- Since the other engine is still in stagnation, a 350 KCAS glide will allow you to move on to clearing the stagnation on that engine as soon as practical, thus reducing the thermal stress on that engine.

As a point of interest, spooldown airstarts are routinely performed on all production acceptance flights with virtually a 100% success rate. The starts are performed on the start limit line shown in the chart, usually at 10,000 ft/.46 Mach and 30,000 ft/.8 Mach. The following points from the chart are representative of the lower airspeed limits at which starts can be obtained: 30,000' and .85 Mach (330 KCAS); 20,000' and .65 Mach (320 KCAS); 10,000' and .46 Mach (260 KCAS) for all engines, Lots III and IV. Lot IV engines can be started slightly slower. In any case, a 350 KCAS glide will be adequate below 35,000 ft.

Step 4 - At RPM increase on engine being started or if restart is unsuccessful, shut down the other engine.

This step requires a bit of thought.



At what point during the attempt on the first engine do we abandon it and move on to the other engine? The issue is somewhat academic since the first engine will do one of three things:

- Start and run fine.
- Not light off, in which case it probably wouldn't start anyway.
- Go back into stagnation, so there's no point in wasting time on it.

In any of the above cases, it's best to move on to the other engine when RPM increase is noted or if there is no start in a reasonable time. A "reasonable time" is best defined as a function of altitude available. Obviously, at 5000 ft AGL, a few seconds is long enough, while at 30,000 ft MSL, you may have the luxury of waiting a full minute or more. As a guideline, it takes 10-12 seconds from the time you move the throttle out of cut-off for the fuel manifold to fill and establish the proper fuel-air mixture in the combustors. Indication of a relight should be apparent within 12-14 seconds after moving the throttle out of cut-off. If time permits, using the High/Low position of the engine start switches may be of some help; but remember, the object is to get either one running as quickly as possible.

Step 5 - Perform restart procedures.

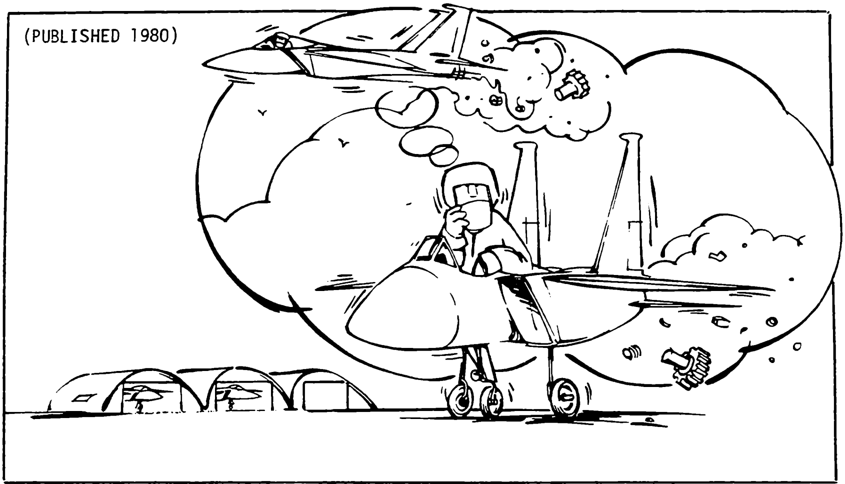
At this point, we have given up on or succeeded with our efforts on the first engine and this step depends on whether or not you have airborne JFS capability. If you do, fire it up and commence a JFS-assisted restart per the flight manual. It's not necessary to use the JFS, since a spooldown start will work as described before, but a JFS-assisted restart is another effort to maximize the probabilities of regaining an operating engine ASAP.

A word of caution: If you plan to use the JFS, be careful when shutting down a stagnated engine. Don't hold the fingerlift full up while moving the throttle full aft since that will activate the micro-switch, which causes the JFS logic to attempt to engage the JFS as soon as it is started. If engine RPM is 30-50% when the JFS attempts to engage, a "crash engagement" occurs which results in a sheared starter shaft. An engineering change has been proposed to solve the crash engagement problem; but until it is approved and incorporated, caution must be exercised.

In summary, using spooldown "tiger" starts gives you the best chance of restart. Use a spooldown "tiger" start on the first engine and don't waste time attempting multiple efforts. Move on to the other engine and use the JFS, if available, or a spooldown "tiger" start and maintain a 350 KCAS glide to maximize time available.

During the entire procedure, hydraulic power is *always* available. During first engine restart, hydraulics come from the other engine. During second engine restart, hydraulics come from a running/restagnated/windmilling first engine. In the event of a no-start on either engine, hydraulics come from the second engine while engaged to the JFS, if available, or windmilling engines.

Now that you've gained a better understanding of the systems involved and optimum procedures, you can analyze the situation and take the best course of action to resolve your problem. If you have the opportunity, I highly recommend a few minutes in the simulator exploring dual engine malfunctions, corrective actions, and standby instrument flying. ■



Don't CRASH Engage Your JFS

By GALEN STANLEY/ *Senior Systems Safety Engineer*

"...During first engine start, the JFS engaged normally, accelerated through 50%, disengaged, and returned to idle. The engine stagnated and the pilot noticed the FTIT climbing through 600 degrees as the RPM decayed through 45%. He immediately raised the fingerlift and pulled the throttle to off. As he did, the JFS accelerated and the CGB re-engaged the decelerating engine. The RPM and FTIT dropped to zero while the JFS continued to run at 100%. The JFS switch was placed in the 'off' position, and the aircraft ground aborted. Investigation revealed the CGB stub shaft had failed at its designed shear section..."

As you read the above excerpt from a recent report, how many of you asked yourselves if this could also happen during an attempted *inflight* JFS-assisted *restart*? Well unfortunately it can, so let's see why this potential problem exists.

It Works Like This

To fully understand how you can get into this fix, a brief description of the engine start circuitry on aircraft with Air-Operable JFS capability is needed. I'll only talk about the right engine circuit to avoid confusion, but the left engine circuit is the same as far as this situation is concerned.

When the right master switch is ON, power is provided to the right engine start switch (actuated by the fingerlift), whenever the RPM is below approximately 50%. Momentarily actuating the start switch (lifting and releasing the fingerlift) will energize the right AMAD select relay, designating the right AMAD/engine to be engaged by the JFS. By the way, the left and right select relays cannot both be energized at the same time; and once one of the relays is energized, it will remain energized until the RPM exceeds 50% or the master switch is cycled or turned completely off. To illustrate the point, if you were starting with external power, you could lift and release one of the fingerlifts before starting the JFS, and then during JFS start, the corresponding AMAD/engine would engage automatically once the JFS reached the proper operating speed and pressures.

There I Was...

The Flight Manual emphasizes the importance of attempting/considering normal inflight restarts before attempting a JFS-assisted airstart during a dual engine out situation. Suppose you follow the book's advice, have no luck and decide to shut down and attempt a JFS-assisted airstart. The net result is that you have had two opportunities to inadvertently engage the start circuit while shutting down the engine. If during either shutdown, the fingerlift was held full up while moving to the full cut-off position you will get some kind of inadvertent or out-of-sequence engagement. Let's look at the possibilities.

If the engine start circuitry has previously been activated, the JFS, when started, will immediately engage and attempt to accelerate the engine. If the engine RPM is below 30%, the engagement should be normal and not result in any problem.

If the start circuit is already energized and the engine RPM is above 30% when you start the JFS, you stand a good chance of shearing the CGB stub shaft. This is also true if you inadvertently actuate the start circuit while shutting down the engine with RPM above 30% and the JFS already running. In either case, if the shaft fails the JFS will accelerate to 100% and stay there. JFS restart capability for that engine has been lost.

The problem with this failure is that you will not know what has happened. What you will see is that neither engine is coming up to JFS motoring speed, no matter which fingerlift you raise; and inflight it will be almost impossible to tell that the JFS is at 100%. The only way out of this one is to de-energize the start circuit on the side with the failed shaft by cycling its master switch.

It's Up To YOU

Well, now that you know *why* the problem can exist, and *how* you can get yourself into it, let's see *what* can be done to prevent it. If you experience a dual engine stagnation, try a spooldown (throttle to idle at 25%) airstart as you attempt to establish a 350 knot dive into the JFS envelope. If the spooldown attempt is unsuccessful (for example, hot start) your best option is a JFS Assisted Restart. If you follow the book -

1. **Throttle (right engine) – OFF**
2. **Centerline stores and pylon – JETTISON**
3. **JFS switch – CHECK ON**
4. **JFS handle – PULL AND RELEASE**

the engine RPM should be at or near 30% before the JFS reaches the speed necessary to engage. Thus, the odds of damaging the CGB shaft are low.

The first step, throttle-off, is extremely important as it starts the RPM decreasing back below 30%, while the other steps set the JFS up to assist the restart attempt. If you can afford an additional few seconds, waiting until approximately 30% RPM before pulling the JFS handle will virtually eliminate the possibility of shearing the CGB shaft due to an inadvertent engagement during a JFS start.

If you are forced to start the JFS with engine RPM above 30% or if you shut down an engine between 30 and 50% with the JFS running, you could shear the CGB shaft and accelerate the JFS to 100%. This condition would be obvious on the ground but is extremely difficult to detect in flight. Therefore, if you attempt an inflight JFS engagement and do not get an RPM increase, quickly cycle both master switches and try again. If you still get no response, cycle the master switches again and try the opposite engine.

NOTE: Rapid cycling of the engine master switch will de-energize the start circuit without affecting fuel flow to a running or stagnated engine.

An Ounce Of Prevention

The best way to avoid the problem described above is to avoid activating the start circuit during engine shutdown. At the present time only your careful movement of the throttles into cutoff without hitting the start switches will prevent start circuit activation but we don't want to have to rely on "technique" in a dual engine out situation. MCAIR is investigating ways to eliminate the problem completely; but in the meantime your throttle technique remains very important. If you want a chance to test your skill (without damaging hardware), try this drill when you go out to fly. After starting the JFS, place both throttles at idle. When ready to start the right engine, place the throttle in cut-off using your normal technique and see if the JFS engages. Before the second engine start, lift the left fingerlift and release it as soon as you start to move the throttle aft. The odds are good that you will get an inadvertent engagement on the right engine but you will be successful in avoiding it on the left.

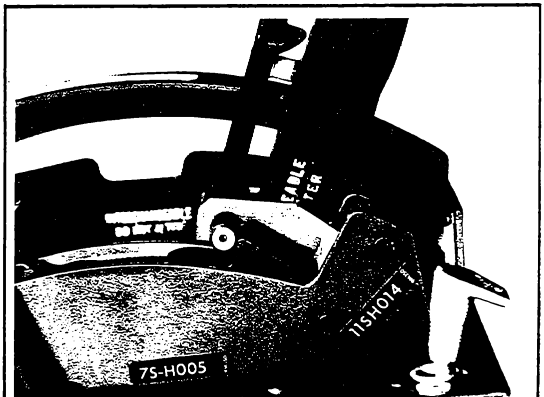
Editor's Note: An Interim Operational Supplement has been issued against the F-15 "Dash One" to add the following statement in Section III under Starting, Abnormal Engine Start, Engine Fails To Accelerate Normally, after Step 2; and under Inflight, JFS Assisted Restart, after Step 7:

CAUTION

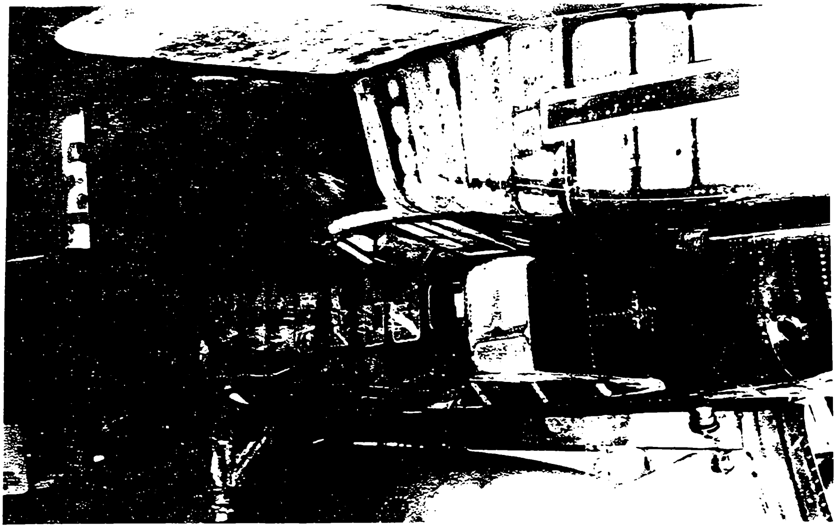
"Exercise caution when shutting down an engine with the JFS running. Release the fingerlifts prior to reaching the cutoff position to prevent immediate JFS re-engagement above 30% RPM."

TO 1F-15A-1S-73 applies to A and B models, while 1F-15C-1S-10 pertains to C and D models.

Incidentally, MCAIR test pilots Pat Henry and Glen Larson recently had the opportunity to experiment with the F-15 simulator at Luke AFB, which has been modified to include Air Operable JFS. They report that with this added capability, the simulator is also a good place to practice your shutdown technique as well as Dual Stagnation and JFS-Assisted Restart procedures. Sounds like a good idea to us. ■



Throttle quadrant engine start switches. During shutdown care must be taken to ensure fingerlift cam follower (left) does not engage start switches (right).



(PUBLISHED 1980)

F-15 ENGINE FIRE
No Foul Systems in the Eagle

By FRANK BIANCA/*Flight Safety Engineer*

Consider a currently hypothetical situation. You are a hundred miles into the enemy's homeland, egressing in full augmentor toward the forward line of troops. Your wingman suddenly calls out, "Lead, you're on fire!" Holy roastin' rickshaw! Now what do you do? There are no fire indications in the cockpit, and all the engine instruments are normal. Your friendly wingman now informs you that it is the right engine, and it is burning out to the wingroot. Also, pieces are falling off your Eagle. Time to gather your gear, punch out, and spend the rest of the war looking out from behind iron bars? Maybe, maybe not.

The F-15 was designed and built with many safety/survivability features, one of the most significant of which is the design of the aircraft to prevent or survive inflight fires in the engine and tail sections. The Eagle has no history of explosions due to engine fires.

Before we go one word further, I want to emphasize that this article does not suggest that just because the aircraft was designed to survive inflight fires a decision to eject should be delayed or not considered. You are the one "on the spot," and your judgment alone should decide whether to stay or go. However, the decision should be based on knowledge and, by knowing your safety systems and procedures, you should be able to make the correct decision to continue flying or to eject. Information concerning how the engine and tail areas were built to resist fire, what limitations affect the fire detection and extinguishing system, and why the emergency procedures were written as they are will give you a big edge in making the critical decision.

A major feature in the F-15 to prevent catastrophic fires is the location of the fuel cells. *No fuselage fuel is located aft of the forward edge of the engines.* If a turbine disintegrates from battle damage and throws pieces through the airframe, the fuel cells are not directly in line for the debris. Also, a bullet through a fuel cell should not cause a leak into the engine compartment. The three-inch fuel lines in the engine bays are titanium, covered with a self-sealing material that should keep small holes from causing large fires.

All items not interfacing with the engines have been eliminated from the engine compartment. Components such as the hydraulic pumps and the generators, which make excellent spark producers when they are hit or malfunction, are now separated from the engine compartment. The only hydraulic line in the engine area is there for the arresting hook. Even if that line is broken, only the small amount of hydraulic fluid left in the line will leak out, unless the hook switch is activated.

The other components in the engine compartment that could be possible sources of ignition are electrical leads to the fuel flow meters, fire extinguisher, and the arresting hook. Also, the Environmental Control System (ECS) crossover bleed ducts and primary heat exchanger cooling air duct are in the engine area. Electrical current to the fire extinguisher and hook is applied only when those items are activated. As for the cooling air ducts, even though they carry relatively low temperature air, they are made of titanium for fire protection.

Firewalls have been used to separate the engine compartment, airframe mounted accessory drive (AMAD), and jet fuel starter. These firewalls are made of titanium and designed to withstand a 2000°F fire for ten minutes. Other strengthening items have been built into the aircraft to resist structural failure and buckling during a fire.

Why, back in our hypothetical emergency situation, did you not see a fire light? A quick check of the maintenance manual shows that the most rearward part of the fire detection system goes only to the end of the compressor area. The Eagle has had some real barn burners going in the augmentor section where there are no fire detectors. So if your wingman yells "Fire!" and you have no indication in the cockpit, that might give you a clue as to what is "cooking."

The F-15 has a fire extinguisher to help save your day, but it cannot be expected to handle all fires simply because of its limited discharge port locations. When discharged into an engine bay, the extinguishing agent is released on the outside of the engine case between the engine and airframe. It lasts about 0.5 seconds and is sent overboard by the normal ventilation system which was designed to expel volatile fumes from around the engine case. Fire extinguishing agent swirling around the outside of the engine case may make you feel better because you have done something, but it will have no effect on a fire burning inside the augmentor section.

What really makes the F-15 capable of withstanding fires is the combination of the handbook emergency procedures with the design features just discussed. In the event of an engine fire, here is what the DASH ONE tells you to do (along with a little information on the reasons therefor) —

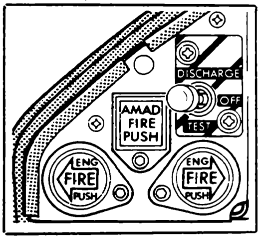
1. Throttle - IDLE

Pull it out of afterburner and back where you can see if you have the correct engine analyzed before shutting it down. If the fire was caused by augmentor operation, it might go out at this time. The engine could then be left in Idle for landing.

If the Fire Warning Light remains on,

2. Fire Warning Light - PUSH

Do not wait too long to do this item. Pushing the light shuts off the fuel outside of the firewalls, so that the only available fuel is that which is still in the engine supply line. Since the engines, when operating, are nothing but controlled fires anyway, pushing the fire light will stop the controlled fire and should stop the uncontrolled one, unless there are other problems, such as oil spraying from the engine. Even without a fire light, if the fire is confirmed, push the respective fire light. The fuel shut-off valve works when the button is pushed whether the light is on or not.



If the Fire Warning Light still remains on,

3. Throttle - OFF

4. Fire Extinguisher - DISCHARGE

As the DASH ONE indicates, shutting off the throttle after closing the main shutoff valve keeps from trapping fuel in the engine supply line. The fire extinguisher might help if the fire is outside of the engine case.

If fire persists,

5. EJECT

If anywhere up to step five, the Fire Warning Light goes off,

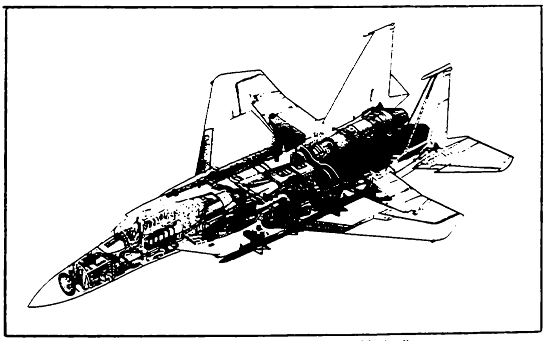
1. Fire Warning System - TEST

2. Monitor other fire indications closely

Remember, there is a firewall between the engines. If one engine has given a fire indication, followed by the other, you might have quite a fire behind you. A fire in the AMAD section should be handled promptly by correct use of the emergency procedures.

"Step Five" is where the judgment mentioned earlier comes in. One person's definition of "persists" might be different from the next. On one incident the F-15 experienced, the engine fire light went out in 30 seconds. On another, the fire went out in seven minutes. In a combat situation, had the second pilot given up on the airplane early, it might have meant the difference between walking home through enemy country or flying the next day. Every fire is different. You must decide if yours is going to ultimately destroy your aircraft.

Whether a fire is "persisting" or about to go out is hard to determine. With the engine fire light pushed, all fuel should be cut off to that particular engine compartment. If the fire appears to increase in intensity, then something else must be wrong, and your decision to eject must be made more quickly. In peacetime or in combat, go through the emergency procedures. Give them a chance to work before you depart your aircraft. Remember that many safety survivability design features have been built into the tail section to help the Eagle survive a fire. Understand what you have working for you, then make your decision. ■



F-15C/D "glass view" shows location of fuel cells.

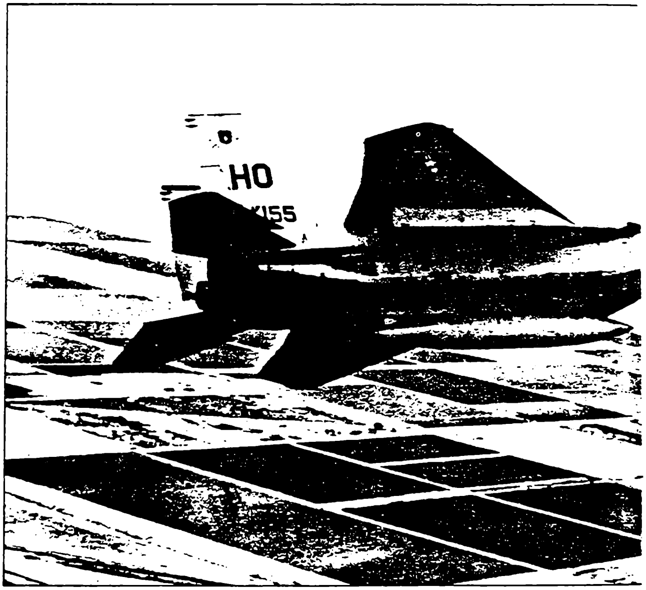
(PUBLISHED 1980)

AN EMERGENCY WAS DECLARED!

"NO. 2 ENGINE WAS SHUT DOWN IN FLIGHT DUE TO AN UNUSUALLY LOUD BANG ON THE RIGHT SIDE PRECEDED BY FLUCTUATING RPM AND FTIT. PITCH AND ROLL CAS DROPPED OFF AFTER THE ENGINE WAS SHUT DOWN AND WOULD NOT RESET. AN EMERGENCY WAS DECLARED AND DESCENT BEGUN FOR A STRAIGHT-IN APPROACH. AT 7 DME THE GEAR AND FLAPS WERE LOWERED AND THE AIRCRAFT IMMEDIATELY ROLLED TO THE RIGHT. THIS WAS CAUSED BY A SPLIT FLAP (RIGHT SIDE DID NOT COME DOWN). THE FLAPS WERE PUT UP AND THE PILOT TOOK THE AIRCRAFT BACK UP TO DO A CONTROLLABILITY CHECK. DURING THE CLIMB THE AIRCRAFT HAD UNCOMMANDED ROLLS TO THE RIGHT ABOUT EVERY MINUTE. THESE ROLLS WERE PRECEDED BY WHAT SOUNDED LIKE A HYDRAULIC SURGE. IN ADDITION, THE AIRCRAFT EXPERIENCED UNCOMMANDED ROLLS TO THE LEFT WHILE PERFORMING CONTROLLABILITY CHECK. THE FLYABLE AIRSPEED WAS 220 KNOTS, WITH ANYTHING UNDER THIS COMPOUNDING THE SERIOUSNESS OF THE UNCOMMANDED ROLLS. ON SHORT FINAL THE AIRCRAFT WAS ATTEMPTING TO ROLL RIGHT EVEN WITH LEFT AILERON/RUDDER. TOUCHDOWN WAS AT 200 KNOTS FOLLOWED BY MAX BRAKING AND DEPARTURE BARRIER ENGAGEMENT. AT 140 KNOTS ON ROLL OUT THE PILOT EXPERIENCED AN UNCOMMANDED PITCH UP."

The data above was the first part of a TWX (teletypewriter) message received at the MCAIR Home Office on 13 February 1980. The report was from Wayne Witt, a company Field Service Representative with the 49th Tactical Fighter Wing at Holloman AFB, New Mexico; and it set in motion an extremely interesting set of events that will be detailed for you in this article.

The second paragraph of Mr. Witt's message described the immediate action at Holloman following the incident with F-15 S/N 77-0155 -



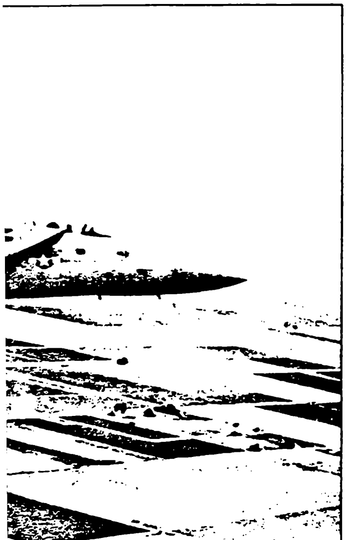
"THE AIRCRAFT WAS IMPOUNDED AND A COMPLETE CHECKOUT OF THE FLIGHT CONTROL SYSTEM WAS MADE. THE AILERONS WERE FOUND OUT OF RIG. THE LEFT SIDE WAS 1-1/2 INCHES LOW AND THE RIGHT SIDE WAS 1/2 INCH HIGH. ALSO THE RIGHT SIDE STABILATOR/RUDDER SWITCHING VALVE WAS FOUND TO BE INOPERATIVE. NOTHING WAS FOUND TO CAUSE THE SPLIT FLAP CONDITION. IT IS SUSPECTED THE AILERON SWITCHING VALVE WAS NOT SWITCHED DUE TO THE WINDMILLING NO. 2 ENGINE."

F-15 Single Engine Flying and Flight Controls

What followed in the ensuing two weeks provides an excellent example of "creative problem solving" on the part of several organizations and individuals, both military and contractor. Pilots, project engineers, safety engineers, and product support specialists representing both the USAF at Holloman and MCAIR here in St. Louis collaborated in an intensive investigation and resolution of some very disturbing "flight control transients."

USAF Major John E. Cunningham, 49th TFW/MAQ, starts out our discussion . . .

Since the F-15 has been at Holloman, we have experienced some problems with the engines. Some of these problems resulted in landing with one



good engine and the other one windmilling. So what, you say? The F-15 has so much power (even with one engine) that a single engine landing is "no sweat." Right? Well, maybe and usually.

What I'm getting at is the paragraph hidden in the LANDING section of the DASH ONE on "Single Engine Operation." The latter part of this paragraph discusses "repeated flight control transients" which are caused by a windmilling engine. The key sentence in the paragraph is the last one, which says to " Monitor hydraulic pressure as windmill RPM sufficient to cause transients may occur with the tachometer indicating 0%."

After a couple of flight tests and discussions with MCAIR engineers and test pilots, let me tell you what we discovered about these flight control transients.

On the 29th of February, I flew Aircraft No. 77-0155 in an attempt to simulate the flight control transients experienced on the flight reported in the TWX by Mr. Witt. The right engine was cut off and allowed to spool down to zero RPM. At 15,000 feet MSL, 230 KCAS, the PC-2 pressure was reading 1500 psi and the hydraulic light illuminated. At 190 KCAS, the RPM went to zero and PC-2 was 500 psi; the PITCH and ROLL CAS dropped off (as expected) but would not reset (a malfunction we thought was fixed). Up to now, no flight control transients had occurred.

As I accelerated through 220 KCAS, PC-2 pressure increased to 1200 - 1400 psi. I heard several "rumbling" noises and the aircraft started a slow, deliberate roll to the left. I countered the roll with right stick and rudder. After two or three seconds of this, the YAW CAS dropped off and the left-roll tendency quit. With four inches of right stick and some right rudder, guess what happened? When I neutralized controls, the rapid right roll stopped and the aircraft flew OK. YAW CAS reset but PITCH and ROLL CAS remained off (a malfunction we have since fixed). By maintaining PC-2 between 1200 and 1400 psi, the same sequence began again about 20 seconds later - left roll, countered by right stick and rudder, YAW CAS dropping off (a malfunction we have since fixed), and so on.

Deciding this was no good, I slowed the aircraft to 180 KCAS and allowed PC-2 to drop below 1000 psi (about 500 psi actual). At the lower speed, no transients occurred. I lowered gear and flaps, slowed to "on speed" (150 KCAS), and the plane flew fine. After raising gear and flaps, I accelerated to 240 KCAS, PC-2 went to 1200 - 1400 psi, and here we go again!

Pat Henry, MCAIR Chief Experimental Test Pilot, flew two similar profiles at St. Louis on 3 and 5 March and experienced the same flight control transients except at speeds around 340 KCAS instead of 240 KCAS. Conclusions from all of these test flights and consequent engineering analyses are that the flight control transients are normal and occur when the windmilling-engine PC pressure builds to 1200 - 1400 psi. The increasing PC pressure causes the hydraulic switchover valves to operate (causing a noticeable "thump" or "rumbling" sound), followed by flight control transients.

Windmill RPM won't tell you much about what is happening in this situation. At the low end, tachometer readings are not accurate and won't necessarily increase as PC pressure builds. What will tell you when to expect a flight control transient is the PC pressure on the windmilling engine. The DASH-1 refers to "monitor hydraulic pressure"; the windmilling-engine PC pressure is what it is referring to. By maintaining an airspeed which will keep the suffering PC pressure above 2,000 psi or below 800 psi, you will be able to prevent any flight control transients. After losing an engine and prior to landing, I recommend a controllability check. During the check, monitor PC pressure while you determine the flight characteristics for landing.

This recommendation and the above guidance are consistent with MCAIR findings. They have talked with SPO and recommend these changes to the DASH-1. MCAIR is also working on changes to the maintenance tech orders so the guys who fix 'em can do a better job. My hat is off to all the contractor people for their assistance in obtaining a good fix for aircraft 77-0155 and for assisting in better identifying the 'flight control transient' phenomenon!

One last point to ponder: In this situation, adding airspeed for the wife and kids could ruin your whole day.

Major Cunningham has summarized the entire investigative effort for you, but there were a lot of interesting steps along the way. In troubleshooting after the original incident (but before the functional check flight), the right stabilator/rudder switching valve failed the T.O. procedure and the ailerons were out of rig. The valve was replaced; the ailerons were ▶

rigged; and the aircraft was checked out for flight. Ten flights were subsequently made with only one reported problem - a hydraulic leak at the brake which resulted in a ground abort but was unrelated to the situation.

After much telephone message traffic between Missouri and New Mexico, on the 27th of February the Flight Control Incident Investigation Team (FCIIT) at MCAIR sent a TWX to Holloman. The message noted that the cause of the uncommanded rolls had not yet been determined but presented several possible causes. More troubleshooting and an FCF were requested, with the flight to duplicate the incident conditions as closely as safety would permit. After Major Cunningham's flight and excellent debriefing, the FCIIT requested similar flights by MCAIR. Chief Experimental Test Pilot Pat Henry reports as follows . . .

As mentioned in John's report, our findings with two late model production aircraft out of St. Louis were essentially consistent with his data from the Holloman bird. One of our aircraft consistently gave just one flight control transient as PC pressure passed the hydraulic switching valve range (≈800psi decreasing and ≈1400 psi increasing pressure), while the other one exhibited multiple switching and accordingly, multiple flight control transients. Each bird seems to have its own "personality" in this region. Bear in mind, however, that the multiple switching was accentuated by deliberately stabilizing at the speed that produced just the right output pressure from the PC pump of the windmilling engine. I found it very interesting that at these very low windmilling speeds, you can actually tell more about what the engine is doing by observing PC pressure than indicated RPM.

The important messages from all this testing, in my opinion, are that you can anticipate the flight control activity by monitoring PC pressure instead of engine speed, and that the hydraulic switchover valves will have finished doing their thing long before you reach on-speed for landing. Furthermore, I find it comforting to learn that we can expect hydraulic pressure, and therefore control surface power, at much lower airspeeds than were previously surmised.

We had never dwelled in the 800-1400 psi range during pre-production flight test (despite many opportunities to do so with the prototype engines) because we would either be acceling for an airstart or deceling for landing. We are most grateful for the opportunity to explore this interesting region further, thereby providing better understanding through articles such as this and through improved guidance in the DASH 1. We are also appreciative of the thorough flight check and documentation by Major Cunningham, which were instrumental in solving this riddle.

One area that didn't match between St. Louis and Holloman as well as I would have liked was the airspeed required for a given engine speed. It is important because our engine restart guidance is based on flight test data that indicates an airspeed of 350 knots or slightly greater is a good (and sometimes necessary) target to shoot for during engine restart (non-JFS assisted). At high altitude, this speed puts you into a more favorable airstart envelope, thereby avoiding hot starts (which are a big waste of time, particularly in the dual engine out scenario). At all altitudes for the right engine, and all altitudes above approximately 10K for the left engine, this speed should be sufficient to sustain RPM above 10-12%, the minimum for ignition and starting fuel. For a given test condition, say 10K/350 knots, the exact windmill RPM will vary somewhat from one airframe/engine combination to another due to component friction and hydraulic loads. However,

No, this is not the "nine good men and true" of the United States Supreme Court in mufti - our camera just happened to catch the MCAIR Flight Control Incident Investigation Team (FCIIT) at a particularly solemn (and therefore rare) moment. Picture obviously was taken early in the investigation of the Holloman incident, while the team was still puzzled by some knotty aspects of the situation. Smiles came later, after they solved the problem! From far left (clockwise), team members are - Jim Hedges, Unit Chief, Electronics; Perry Hoffman, Lead Engineer, Avionics Laboratory; Carl Scherz, Section Chief, Guidance & Control Mechanics; Dave Thompson, Chief F-15 Technology Integration Engineer; Dave Nothstine, Chief F-15 Program Design Engineer; Ford Miller, Product Service Hydraulics Specialist; Dale Cattoor, Product Service Senior Avionics Specialist; Mike Peterson, Systems Safety Engineer; and Pat Henry, Chief Experimental Test Pilot. According to Mr. Thompson, head of FCIIT, the ultimate goal of the team is to so reduce unexplained flight control incidents that it works itself out of a job! You can help by bringing the real puzzlers to its attention.

even those engines that are on the low windmill RPM side exhibit an extremely low spooldown rate as they approach their stabilized speed. Therefore, there is plenty of time for the engine to pressurize its fuel manifold and light off if the DASH 1 guidance of throttle to Idle at a minimum RPM of 25% is practiced. John's bird looked a little optimistic with respect to windmill RPM in that airspeed required to sustain a given engine speed was lower than average. My concern is that some false optimism would be generated as to what minimum airspeed to shoot for in the dual engine failure case. We still think 350 knots is a good target until transitioning to a JFS assisted restart.

Naturally, the TWX from Wayne Witt at Holloman prompted an early meeting of the Flight Control Incident Investigation Team — this was exactly the type of non-routine situation that "FCIIT" had been set up to handle! There were enough out-of-the-ordinary aspects to this incident to qualify the problem flight of F-15 77-0155 as "Number One" on its list of investigations. And the flight tests at HAFB and MCAIR added a significant layer of data to that obtained from the original incident. The group met several times during the two week-period to sift through all the information. Systems Safety Engineer Mike Peterson summarizes the team's analysis . . .

We looked at all the available data in an attempt to understand several things —

• Why would Pitch/Roll CAS not reset?

• Why were the transients recurring, and at lower than expected speed for switching valve transients (indicating that the engine was windmilling at a lower than normal airspeed)?

• Why would Yaw CAS drop off just before the hard right roll?

• Why were the transients much larger than expected from switching valve transients?

Previous experience with the F-15 flight control system identified the fact that when an engine windmilled down, some control transients would occasionally occur as the switching valves went through the "test" cycle (a check for downstream leakage). This normally occurred as the airspeed was slowing through 300 knots. And certain other events of the incident could also be explained —

• Pitch and roll CAS dropping off was normal because the right stabilator CAS servo lost PC-2A hydraulic pressure when the stabilator/rudder switching valve was in the test cycle. (But once the valve switched to back-up Utility pressure, these should reset.)

• The split flaps occurred because lowering of the flaps decreased PC-2 hydraulic pressure, which then caused the right aileron/flap switching valve to go to the test mode. The right hand flap did not extend immediately because of the lack of hydraulic pressure.

• The out-of-rig condition of the ailerons should not have contributed significantly to the transients.

After our first time through the entire incident "scenario," everything appeared to be explained by the T.O. indication of a failed right stabilator/aileron switching valve. However, there were enough doubts in our minds that we contacted the Holloman safety office and our base reps for more information. They told us that the incident airplane had been released from impoundment and had flown several times with no flight control malfunctions! End of problem, right? Wrong. There were too many "doubting Thompsons" on the FCIIT to just let an intriguing incident like this solve itself so simply. So we got a detailed debriefing from the incident pilot, Captain Bob Debusk.

His information explained some of the things that were puzzling us, and clarified the exact nature of the uncommanded rolls and pitch up. The pitch up, which had occurred on landing rollout, was not an aircraft malfunction — this Eagle just didn't want to stay on the ground at the high landing speed. The hydraulic surge was more like a rumbling noise, and discussions between Holloman and MCAIR pilots established that this was the switching valves operating. Captain Debusk also more specifically defined the transients as always being the rumbling noise followed by a slow left roll, then Yaw CAS drop off, followed by the hard right roll. Now there were enough unanswered questions to make everybody on the team happy, and we went back into session!

We requested that Holloman fly an FCF on the incident airplane, even though it had flown several problem-free missions since the incident. We asked for a duplication of the flight conditions and that as much data as possible be recorded about aircraft characteristics as the right engine was windmilling down. This was the flight discussed earlier by Major Cunningham; and his records of airspeed, RPM, PC pressure for numerous points, plus his statements on aircraft characteristics were extremely helpful in analyzing the incident. In this flight, the flight control transients and the CAS drop offs were duplicated, and we now knew that merely replacing the switching valve had not solved the problem.

Additional checks of the hydraulic system were generated and we also requested that other systems that could cause rolls, such as the flaps and AFCS, be looked at again. It was at this point that we also asked for some flights by MCAIR for comparison with the Holloman data. These were the flights just discussed by Pat Henry, and his experiences under the stipulated test conditions were sufficiently similar to Major Cunningham's that our decision to look deeper into the hydraulic system was reinforced.

No malfunctions were found with the flap circuitry during troubleshooting, so we concentrated our efforts on ▶

FCIIT

MCAIR has established an interdisciplinary group to investigate unusual flight control incidents. Not intended to serve as a sounding board for routine situations, the "FCIIT" (Flight Control Incident Investigation Team) is usually called to action only for major unexplained anomalies such as the one described in this article from Holloman AFB. The team also meets monthly to analyze the significance of trends which may be developing in accumulated minor flight control incidents reported through normal channels.

The FCIIT is currently composed of representatives from several company departments, including Engineering Technology, Design Engineering, Systems Engineering, Systems Safety, Flight Test, and Product Services. Other technical department specialists may also serve in specific situations. The group also works closely with military base personnel through on-site MCAIR service engineers.

The team's primary purpose is to examine and reveal cause/effect relationships between flight control and other aircraft systems which would not be anticipated or indicated when incidents are analyzed from a single discipline point of view. Reports are prepared on each incident investigated, with summaries distributed to affected agencies and operational activities via MCAIR Field Service Representatives. The reports include incident details, troubles/problems found, and corrective action taken or recommended.

the switching valve. Some novel troubleshooting techniques finally revealed that a pressure switch in the PC-2 reservoir had failed! This switch failure prevented Utility pressure from going to the switching valve. With no Utility pressure to the switching valve, the valve would stay in "test" or else go back to PC pressure when the pressure was high enough, thus resembling a failed switching valve. The pressure switch was replaced and the aircraft passed all flight control checks. Another flight was made on 77-0155 with the right engine shut down, and the Yaw CAS did not drop off. Pitch and roll CAS could be reset and the transients were very small, as would normally be expected. We had finally located the "root" of the problem, and everything else began to fall in place.

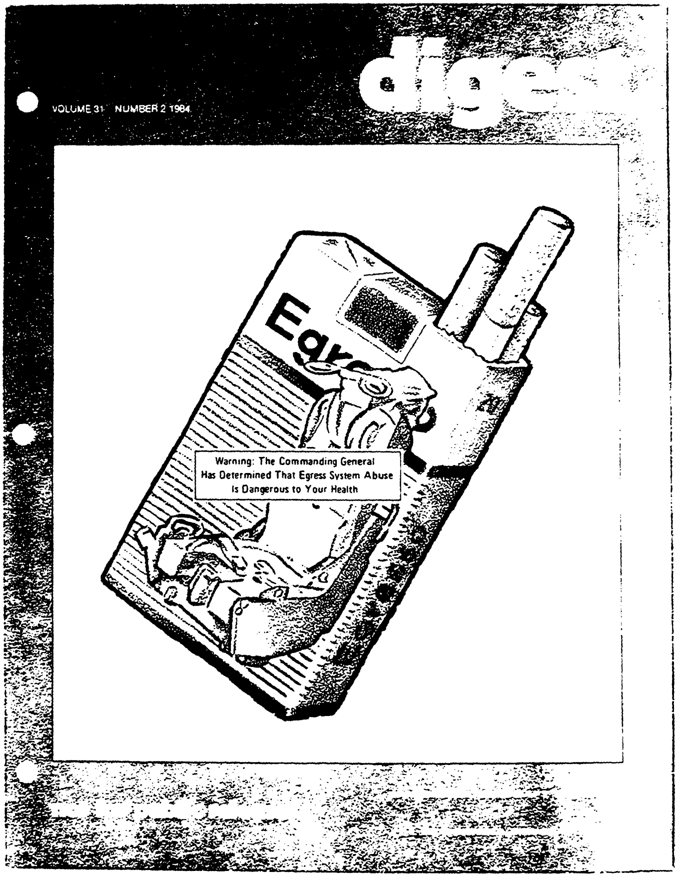
In recapping the situation, here is what the FCIIT analysis produced —

- The pressure switch in the PC-2 reservoir had failed, thus denying Utility System pressure to the switching valve;
- Pitch/Roll CAS failure to reset was caused by lack of backup Utility pressure to the CAS half of the stabilator actuator;
- Transients at lower than expected airspeeds resulted from unexpected variations in engine windmill characteristics;
- The larger than expected transients and Yaw CAS drop off were the result of having one rudder unpowered (loss of PC-2 pressure and failure of Utility backup pressure).

The final result of the investigation was a fuller understanding of a very disturbing flight control incident, which in turn led to some recommended changes in a couple F-15 technical manuals. The "Single Engine Operation" information in the Flight Manual should stress that split flaps might occur and noises may be heard. Hydraulic pressure is the best indicator and the pilot should avoid airspeeds where PC pressure fluctuates between 800-2000. A revision to the Troubleshooting Manual (T.O. 1F-15-2-5) was recommended to improve troubleshooting of the PC system pressure switch. A phase inspection of the pressure switch was also recommended, and design changes will be considered if necessary. The FCIIT has closed its books on this one.

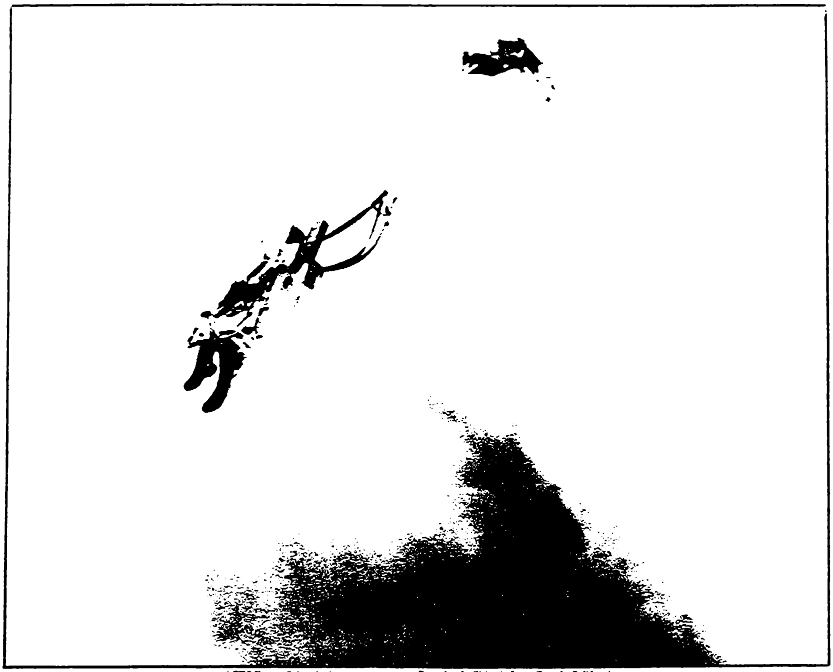
Without the Flight Control Incident Investigation Team, the excellent cooperation of personnel of the 49th Tactical Fighter Wing at Holloman AFB, and support from MCAIR Engineering and Flight Test, Eagle 0155 might still be doing uncommanded (and unexplained) rolls around the skies of New Mexico! We hope this discussion has given you an insight into the highly effective partnership developed between MCAIR and the customer for the resolution of problems. ■

EGRESS ►



VOLUME 31 NUMBER 2 1984
digest
Egress
Warning: The Commanding General
Has Determined That Egress System Abuse
Is Dangerous to Your Health

(PUBLISHED 1977)



ACES II seat firing during test program at Douglas facilities in Long Beach, California.

ACES II

ADVANCED CONCEPTS EJECTION SEAT

Come next July, your F-15A/B cockpit is going to have a new look. That's when Eagles in Block 18 (USAF 77-0061 for "A" models and 77-0154 for the "B") will incorporate the "ACES II" ejection seat. The DIGEST will come at you with technical details on operational and maintenance aspects in an upcoming issue, but for now, we'll just offer a quick scan of what looks to us to be a very neat seat!

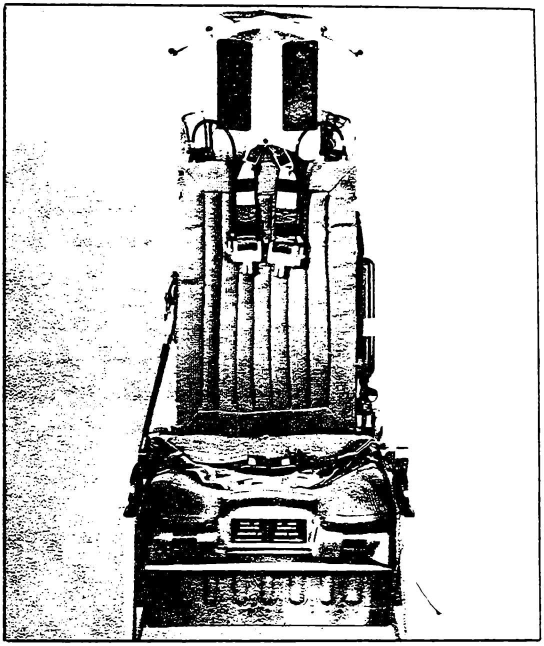
Manufactured by the Douglas Aircraft Company component of McDonnell Douglas Corporation, this "advanced concepts ejection seat" is scheduled for installation in the MCAIR F-15, General Dynamics F-16, and Fairchild A-10. Thus it looks like you tactical pilots, regardless of which high performance aircraft you fly, will sit in "ACES." You should sit pretty comfortably, and very safely.

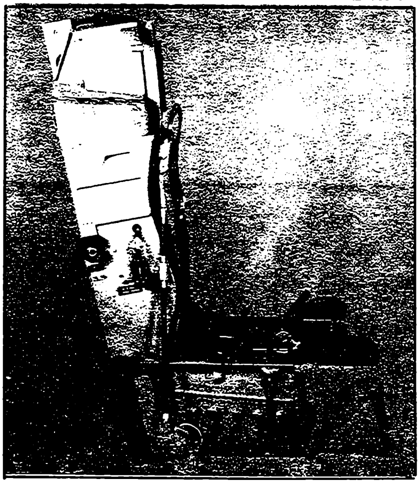
Production delivery of the first of 441 seats procured by the Air Force occurred on 28 October at Long Beach. Subsystems of the ACES-II seat were tested and qualified as a part of the USAF/Douglas ACES-I research and development program; and qualification tests of the ACES-II system were completed in June of 1973. Douglas is well equipped to handle the ACES program, having produced more than 7000 ESCAPAC ejection seats for their own A-4 Skyhawk and for the A-7, B-57, S-3A, and current versions of the A-10, F-15, and YF-16.

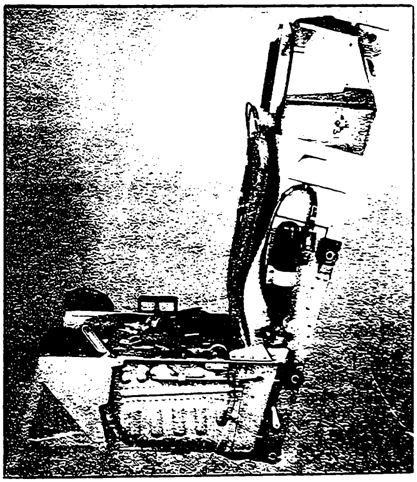
This new seat will give Air Force flight crews a much better chance to survive emergency ejections and land safely in a range from zero altitude/zero airspeed to altitudes up to 50,000 feet and airspeeds of 600 knots. Compared to earlier designs, the system is especially effective at low speeds, low altitudes, and adverse aircraft attitudes. For example, a pilot can eject safely from altitudes under 200 feet while flying inverted at 150 knots. Earlier designs required nearly twice that altitude for safe recovery under similar circumstances.

Key to the advanced concepts design is a solid-state electronic system which sequences and controls performance of the recovery subsystems built into the seat. The electronic system outperforms ballistic and mechanical systems used on previous seats. It works with seat-mounted altitude and airspeed sensors to activate one of three recovery modes for each ejection situation.

Other improvements over earlier ejection seat designs include a vernier rocket to provide pitch stabilization and a specifically designed drogue parachute to provide yaw stabilization and deceleration of the seat until the main chute deploys. It uses a mortar to deploy the main chute and reefing system to open the parachute in stages, which lessens strain on the crewman. In event of a mishap that doesn't require ejection while the aircraft is on the ground, the crewman need only use a single-point release to free himself from all restraints. ■

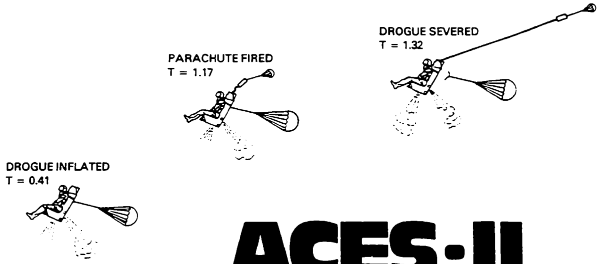


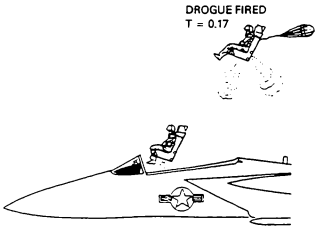




ECP-703

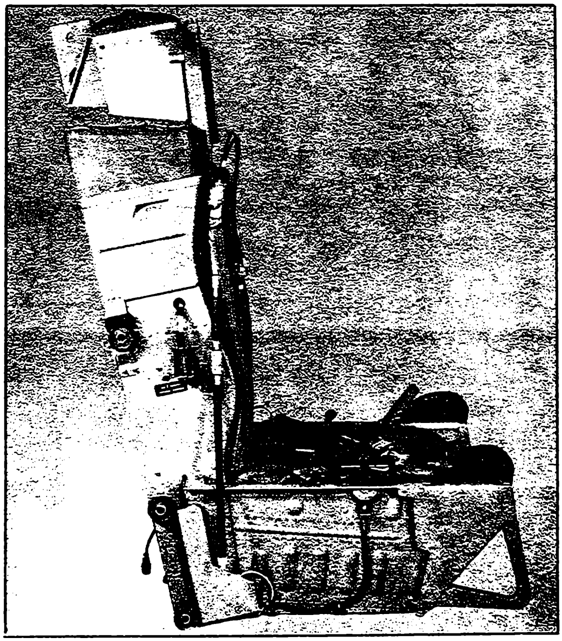
(PUBLISHED 1978)





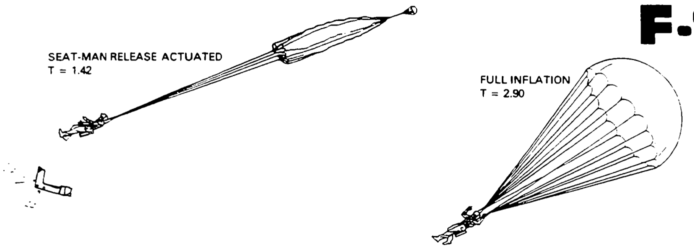
ACES-II

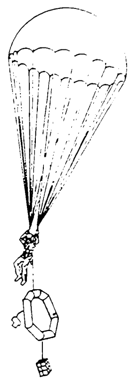
The F-15A/B will soon incorporate the "ACES-II" ejection seat system in accordance with ECP-703. ACES-II was designed and manufactured by the Douglas Aircraft Company (DAC) component of McDonnell Douglas Corporation, and is furnished GFE to the F-15 program. DAC has published considerable technical documentation providing general information concerning ACES-II intallation in several current USAF aircraft. In this article, MCAIR system safety specialist Jack Sheehan passes this information on to you, with specific applications to the Eagle.



ACES-II is designed to provide improved aircrew survivability with increased visibility and comfort while enhancing total egress system reliability and maintainability. It has been qualified from zero through 600 KEAS during extensive development testing. Unlike systems used in previous McDonnell airplanes - which incorporate face curtain or lower ejection handle initiation, ACES-II utilizes side-mounted initiation handles in accordance with MIL-S-9479. We can already hear some of you out there saying "Why in the world did they do that?!" since you are all familiar with the D-ring ejection control method. Actually, life science data over the past 15 years indicates an operational advantage to the side ejection control; and dynamic centrifuge testing substantiated the ease of operation at high-G forces over the face curtain/center pull D-ring. And for most USAF crews, flight training began in the T-33/-37/-38 which use side ejection handles, so you actually learned this mode of initiation first, regardless of which mode you graduated to in line fighters. The rationale for side controls is simply to increase accessibility, allowing for optimum body position and limb restraint.

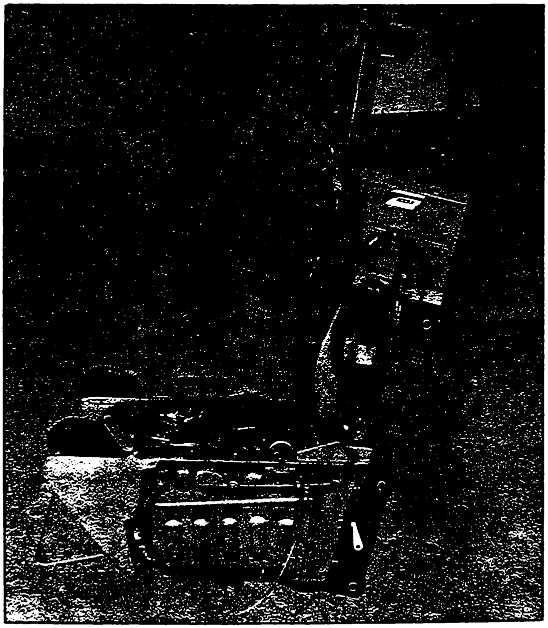
F-15A/B





EJECTION SEAT

It's known by two acronyms - "ACES," which stands for Advanced Concepts Ejection Seat; and "HTES", which is short for High Technology Escape System. "Advanced concepts" and "high technology" are the key words in talking about this new aircrew protection system - in the form of such innovations as solid state electronics, multiple operating modes, instantaneous seat stabilization and high speed deceleration, and a balanced high speed/low speed drogue and recovery parachute system. Let's take a look at these and other features of this Block 18 and up replacement for the ESCAPAC IC-7 crew escape system.



ACES-II is a lightweight, advanced-performance escape system that incorporates high technology state-of-the-art subsystems. Electronic event sequencing by solid-state, redundant timing circuits is used in conjunction with proven electro-pyrotechnic components to achieve high reliability and maintenance-free service.

The system is composed of several subsystems - seat structure, guide rails, seat adjustment actuator, firing controls, propulsion, pitch control, trajectory divergence (F-15B only), drogue parachute, recovery parachute, recovery sequencer, harness release mechanism, survival kit, restraint, and emergency oxygen. These systems combine to provide stabilized ejection and rapid, minimum-distance recovery performance

Stabilization under low-speed ejection conditions is achieved by a gyro-controlled vernier rocket ("STAPAC") that provides pitch control for thrust line-cg offsets in excess of ± 2 inches. At high speeds, stabilization and deceleration are achieved by a drogue parachute.

Multiple recovery modes permit the functions and timing of the recovery system to be selected for each of three ▶

ıodes to optimize performance ıroughout the escape envelope. The rogue and recoverv parachute sys-ɘms are independent so that the rogue need not be deployed in a ›w-speed, low-altitude ejection here immediate deplovment of the ·coverv parachute is essential. At gh speeds, the operation of the ·ogue and recoverv parachutes is ,erlapped to increase system decele-tion.

An electronic sequencer controls e sequencing, timing, and initiation the recovery system functions. ode selection is performed by the coverv sequencer in conjunction ith an environmental sensing system hich senses airspeed and altitude ›nditions. The complete system, in-uding the recovery sequencer power pplies and the dual pitots for air-eed sensing, is mounted on the seat.

The drogue and recoverv parachute stems are configured so that, at the aximum speed conditions, the de-·leration forces applied to the crew-an are within recommended limits. ıis approach is essential in obtaining ficient use of the drogue and re-›very parachutes under lower speed ›nditions. The drogue is sized to the aximum speed case, and the drogue attachments are located so that the forces are applied in the "eye-balls out" direction in which they can be most readily withstood by the crew-man. A reefed recovery parachute is used to permit deployment of the parachute at relatively high speeds. The early deceleration due to the reefed parachute contributes to the achievement of minimum-dis-tance recovery.

The emergency oxygen is delivered automatically upon ejection. The sur-vival kit can be selected to auto-matically deploy four seconds after seat/man separation.

Ejection Seat Subsystems

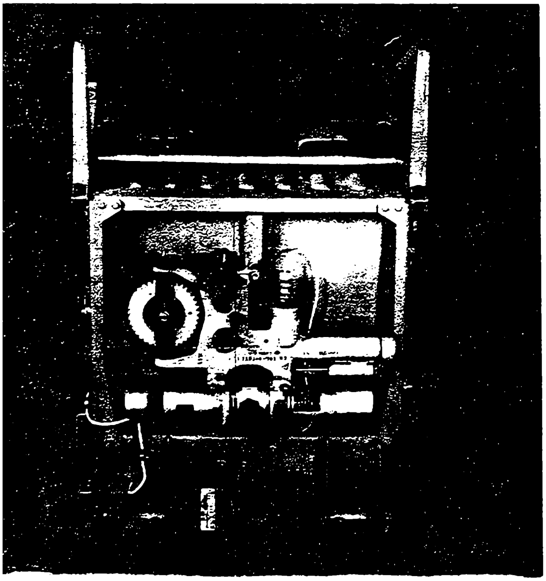
Seat structure and ejection guide rail subsystems are of conventional aircraft-type construction and fabri-cated primarily of aluminum alloy. Unique features of the other subsys-tems are described below.

SEAT ADJUSTMENT ACTUATOR

The seat adjustment actuator pro-vides five inches of vertical adjust-ment. The actuator is bolted to attach-ment brackets in the aircraft and to the base of the rocket catapult. Height adjustment is made by raising or lowering the entire seat along the guide rails. The actuation switch is mounted above the left console.

PROPULSION

Propulsion is provided by a Type CKU-5/A rocket catapult. The catapult is integral with the rocket motor, and the normal velocity at separation is 43 feet per second. Peak catapult accele-ration is approximately 15 g. Nominal rocket impulse is 1150 pound-seconds.



"STAPAC" Assembly

PITCH CONTROL

Pitch control is achieved by the "STAPAC" gyro-mounted vernier rocket system. This seat stabilization unit is mounted beneath the seat bucket and action is initiated by the recovery sequencer as the seat leaves the guide rails. The gyro is spun up ballistically and linked to the vernier rocket to correct offsets in excess of ±2 inches.

TRAJECTORY DIVERGENCY

Trajectory divergency is provided on F-15B ejection seats only, and insures that there is no interference between the two crewmen and their escape system components. Trajec-tory divergence is achieved by a small rocket which is initiated by the recovery sequencer as the seat leaves the rails. The rocket is positioned to cause the seat to roll, and the redirec-tion of the main rocket thrust results in lateral motion and trajectory dis-placement. The rockets for the F-15B front and rear seats are installed on opposite sides of the seat buckets to produce opposing roll actions.

DROGUE PARACHUTE

The drogue parachute provides stabilization and deceleration under high speed conditions and stabiliza-tion during high altitude descent. The drogue is not deployed under low speed (Mode 1) conditions. The drogue extractor and parachute are stowed at the rear of the ejection seat in separate compartments.

The drogue gun is initiated by the recovery sequencer as the seat nears the top of the guide rails and propels a one-pound drogue slug. The drogue-slug deploys the extraction chute and is then detached from the chute by a static line. The extraction chute de-ploys the drogue chute via a high-drag bridle and tow line. The drogue parachute bridle has a two-point attachment to the seat.

When deployment of the recovery

parachute commences, the drogue parachute is detached from the seat by shaped charge cutters which sever the bridle at the two attach points. The cutters are initiated by the recovery sequencer 0.15 second after the parachute mortar has been fired.

RECOVERY PARACHUTE

The recovery parachute is a mortar-deployed 28 foot C-9 canopy. It is deployed in the reefed condition to alleviate high deployment forces imposed on the crewman. Two reefing cutters with 1.15 second delay are installed on the parachute skirt. The parachute canopy is packed around the mortar and the mortar is initiated by the recovery sequencer. The mortar propels the parachute off the seat, deploying the suspension lines first; then the canopy deploys skirt-first as the packing container strips off.

If automatic initiation of the recovery parachute mortar does not occur, parachute deployment is initiated automatically when the seat/man release system is actuated by the recovery sequencer.

RECOVERY SEQUENCING

The recovery sequencing subsystem selects the recovery mode appropriate to the escape environment and executes the recovery sequence.

The environmental sensor contains two altitude-compensated dynamic pressure transducers and a static pressure transducer. Pressure inputs are obtained from two pitots mounted on the parachute container and from a static port which is open to ambient pressure behind the seat. The dynamic pressure transducers are set to switch from a low-speed to a high-speed position at 250 knots at sea level. The altitude compensation causes the switchover speed to decrease as altitude increases. The static pressure transducer is set to switch from a high-altitude to a low-altitude position at 15,000 feet at zero airspeed.

The recovery sequencer contains electronic logic circuits and electronic time delay circuits. The logic circuits interrogate and interpret the speed and altitude transducers in the environmental sensor and select the recovery mode. The time delay circuits provide the correct event-time sequence for each recovery mode. Electrical power is provided by thermal batteries which are initiated by gas pressure from the catapult stage of the rocket catapult. Until initiation by the rocket gas pressure, the thermal batteries are completely dormant and there is no electrical input to or output from the recovery sequencer. The timing sequence is started by a "sequence start" switch which is actuated by a striker plate as the seat moves up the guide rails. Actuation of the "sequence start" switch by removing or installing the seat in the guide rails has no effect because the thermal battery has not been activated.

The recovery sequencer is fully redundant in that two identical, independent electronic systems are provided to execute the recovery sequence. Each redundant system fires one of two bridgewires in each of the squibs used to initiate the ballistic components. If the recovery sequencer fails to initiate the recovery sequence, the crewman can pull the RESTRAINT EMERGENCY RELEASE handle (see Harness Release Subsystem), which releases the seat lap belt and shoulder harness and deploys the pilot chute.

SURVIVAL KIT

The survival kit consists of a fabric case which houses the liferaft, rucksack, and an auxiliary container. The survival kit stows in the seat bucket beneath the rigid seat pan which is supported by the seat structure. The liferaft and rucksack are attached to the survival kit case by a dropline. The auxiliary container, which is for stowage of items to be retained with the crewman, is secured inside the survival kit. Webbing straps secure the kit case to the two survival kit attach fittings.

A control on the seat pan allows the crewman to preselect automatic deployment of the liferaft and rucksack, and a control on the kit permits deployment to be initiated manually following seat/man separation. In the automatic mode, the kit case is opened by a ballistic cutter four seconds after seat/man separation.

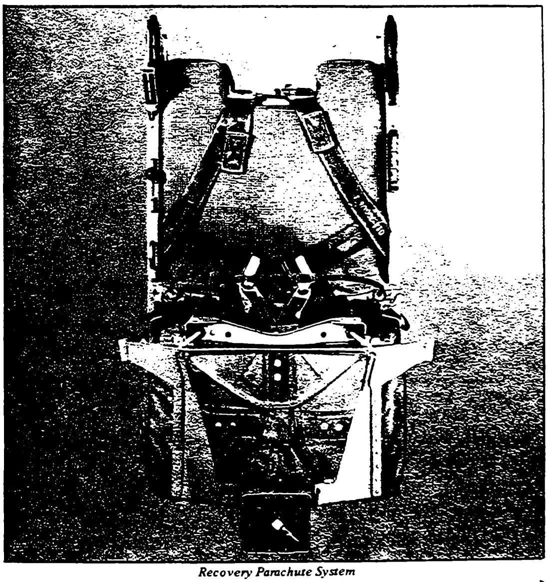
Provisions are made for installation of the AN/URT-33C Radio Beacon. Access to the function switch on the URT-33 is provided in the seat and, when the switch is on, the beacon will automatically go into operation during seat/man separation.

RESTRAINT

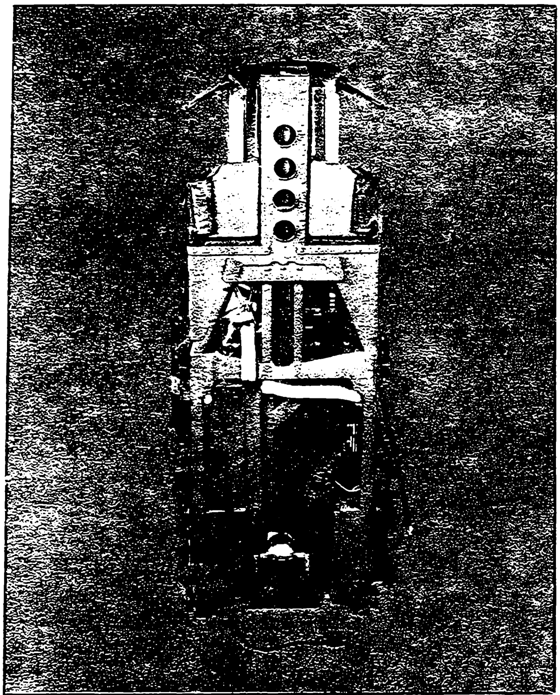
Restraint of the crewman during ejection or crash landing conditions is provided by a lap belt and inertia reel. Dual inertia reel straps pass around rollers on the parachute risers and are secured to the seat by a locking pin. The inertia "lock and unlock" control is located on the left side of the seat bucket.

HARNESS RELEASE SUBSYSTEM

ACES-II provides different methods for crewman harness release - one as an automatic part of the ejection



Recovery Parachute System



Rocket Catapult

process and one as a manually-initiated ballistic operation during emergency ground egress. The manual system also serves as backup in the event of failure of the automatic seat separation system during ejection.

• Automatic - Following ejection, the harness release system releases the crewman from the seat automatically. The mechanism is powered by a thruster which is initiated by the recovery sequencer 0.25 second after the parachute mortar is fired. The thruster rotates a bellcrank which, by means of rods and cables, mechanically withdraws locking pins to release the lap belt, inertia reel straps, seat pan, and parachute mortar from the seat and the pilot chute from its compartment. The crewman is released by withdrawal of the lap belt and inertia reel strap attach pins. Release of the seat pan latch allows the pan to rotate as the survival kit is withdrawn from the seat bucket during seat/man separation. Once the bellcrank has been rotated open, it remains locked in that position.

• Emergency Ground Egress - The seat also incorporates a "Rapid Escape Divestment System" that permits the crewman to leave the cockpit during a ground emergency, by actuation of a single handle. This system is composed of ballistic-actuated quick-release mechanisms on the parachute risers and survival kit retention straps, energy transfer lines and manifolds, and an initiation device. The initiation device includes an interdiction function which is connected to a guide rail-sensing arm. Initiation of this system, which is possible only when the seat is in the aircraft ejection guide rails, is accomplished by actuation of the RESTRAINT EMERGENCY RELEASE handle on the right side of the seat bucket. This actuates the parachute and survival kit quick-releases and at the same time releases the lap belt from the seat. As the crewman stands to leave the seat, the hoses for oxygen and other services are disconnected by brute-force quick-disconnects.

• Seat Separation Backup - After ejection, the RESTRAINT EMERGENCY RELEASE handle serves as a manual backup in the event of failure of the automatic seat separation system. In this case, initiation of the Rapid Escape Divestment System is prevented by the initiator interdict device, and actuation of the RESTRAINT EMERGENCY RELEASE handle releases the crewman from the seat and the pilot chute from the recovery parachute container.

For maintenance purposes, the harness release subsystem can also be actuated by a control located behind the cushion on the upper right-hand seat side, where it is inaccessible to the crewman during flight. When the Equipment Release cable is actuated, the harness release mechanism locks in the "open" position. The mechanism can be reset by pressing the Equipment Release reset located on the right-hand side of the seat.

EMERGENCY OXYGEN

A 22 cubic-inch emergency oxygen supply is provided. The cylinder assembly is on the left side of the seatback where it is visible for inspection and can be readily removed. The hose is routed to the CRU-60/P connector on the crewman's torso harness. The system is actuated automatically in an ejection by a lanyard connected to the cockpit floor. A manual control ("green ring") is also provided on the left side of the seat bucket.

Change 3 of T.O. 1F-15A-1 is dated 15 January 1978, and adds information on the ACES-II ejection seat to Sections 1 and 3 of the DASH ONE. The discussion presented here explains the theory of ejection seat operation, and you should carefully read Change 3 for the details of seat description and ejection procedures for the three modes of operation provided by this system. Your Egress and Life Support personnel should also be consulted for answers to your questions.

Actuation of the side-mounted ejection controls (either or both) fires the seat-mounted initiator, which supplies pressure to power retract the inertia reel and (via a quick-disconnect) to initiate the canopy emergency escape sequencing system. The canopy emergency escape sequencing system initiates canopy removal and provides high-pressure gas back to initiate the rocket catapult. In an F-15B sequenced automatic ejection, the rear crew member is ejected 0.40 second before ejection of the front crewmember.

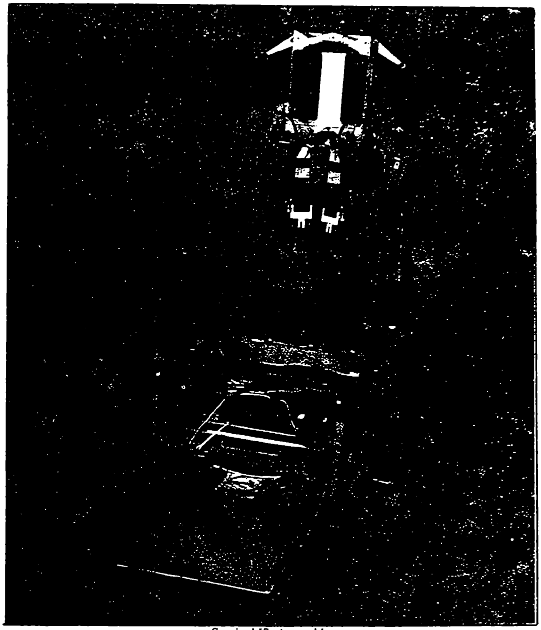
When the catapult cartridge ignites, high-pressure gas ported from the catapult chamber to the recovery sequencer initiates the thermal battery power supplies. As the seat moves up the guide rails, the pitots on both sides of the parachute assembly are exposed to the airstream. Pitot and static pressure inputs to the environmental sensing unit act on speed and altitude transducers. The recovery sequencer interrogates the transducers and selects the appropriate recovery mode. Movement of the seat up the guide rails activates the emergency oxygen supply and disconnects the egress system gas disconnects.

As the seat approaches the top of the guide rails, the recovery sequence start switch is closed by a striker on the guide rails, initiating the pitch stabilization system STAPAC (and in the F-15B the trajectory divergence rocket) after short time delays. The initiation of other subsystems depends upon which of three recovery modes is in operation. The portion of the flight envelope appropriate to each operating mode is shown below.

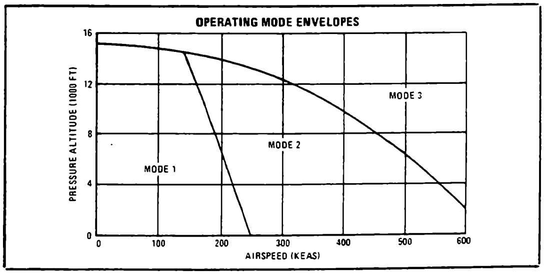
Mode 1 operation is a low speed ejection in which the parachute is deployed almost immediately after the seat departs the aircraft. The recovery parachute mortar is initiated 0.37 second after rocket catapult ignition. As the mortar propels the parachute assembly away from the seat, the 1.15 second delays in the reefing line cutters are activated and the pilot chute deploys. The harness release thruster is actuated 0.25 second later and the deploying parachute separates the crewman from the seat. The parachute inflates to the reefed configuration and, when the reefing line cutters actuate, the parachute fully inflates. If automatic survival kit deployment was preselected, the kit opens approximately four-seconds after seat/man separation and allows the life raft and rucksack to deploy.

Mode 2 operation is a high speed ejection in which a drogue chute is first deployed to slow the seat, followed by deployment of the parachute. Drogue deployment is initiated just prior to the seat leaving the guide rails. Projection of the drogue-gun slug deploys the extraction parachute which, in turn, deploys the drogue parachute. The recovery parachute mortar is fired 1.00 second after drogue initiation; and 0.15 second after the parachute mortar is fired, the drogue bridle is severed. The remainder of the sequence is the same as Mode 1.

Mode 3 operation is a high altitude ejection in which the sequence of events is the same as Mode 2 except that man/seat separation and deployment of the recovery parachute are delayed until Mode 2 altitude and speed conditions are attained.



Survival Kit Assembly



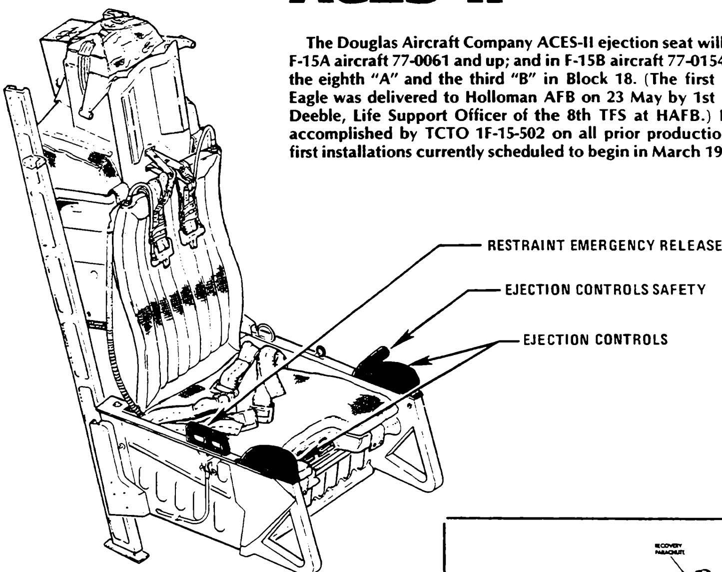
NOTE: This has been an "introductory" presentation intended only for general ACES-II familiarization. On the seats you will be receiving, some details, including hardware modifications and equipment routing, may not be as described or pictured herein. Please check manuals and change documents for completely up-to-date information.

KNOW YOUR F-15A/B

(PUBLISHED 1978)

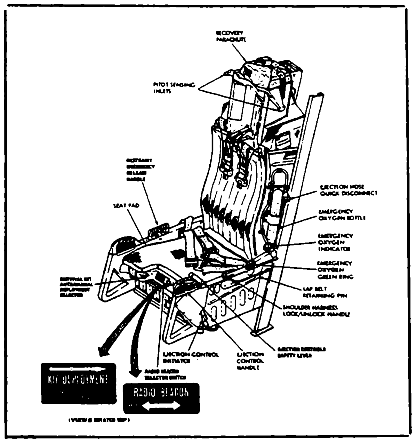
ACES-II

The Douglas Aircraft Company ACES-II ejection seat will be installed in F-15A aircraft 77-0061 and up; and in F-15B aircraft 77-0154 and up. This is the eighth "A" and the third "B" in Block 18. (The first ACES-equipped Eagle was delivered to Holloman AFB on 23 May by 1st Lieutenant Ken Deeble, Life Support Officer of the 8th TFS at HAFB.) Retrofit will be accomplished by TCTO 1F-15-502 on all prior production aircraft, with first installations currently scheduled to begin in March 1979.



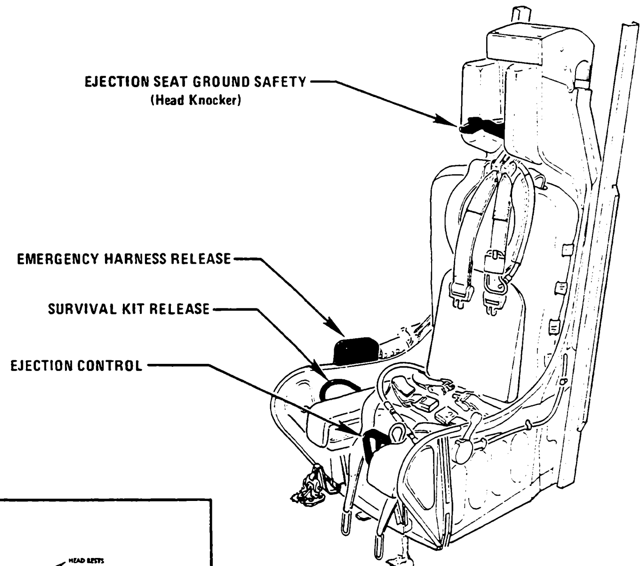
ACES-II uses only one "safety streamer" - the 3943401-1 Ground Maintenance Safety Pin Assembly.This is a two-pin assembly - insert one pin in hole under RESTRAINT EMERGENCY RELEASE handle and other in hole under left ejection control handle. An Ejection Controls Safety lever immediately behind the left ejection control handle is rotated up and forward to lock both ejection control handles. These are the only system safeties - don't forget or ignore either of them.

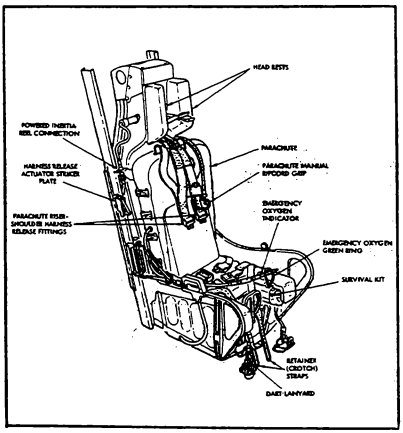
ACES-II is designed for better reliability, maintainability, . . . and personnel safety. Where you may have been used to seeing numerous red safety streamers in and around the cockpit, ACES-II is safed for you with only the two devices noted above. They and the Canopy Ground Safety Locks provide your cockpit life insurance policy. However, merely because it is quicker and easier to safe the ACES cockpit for ground operations, don't become careless or complacent; your life insurance policy is only as good as you are!



EJECTION SEATS

ESCAPAC





The Douglas Escapac IC-7 ejection seat will be phased out of the F-15A/B aircraft by TCTO 1F-15-502 retrofit. However, there are some 300 airplanes to be changed, and this seat will be in the inventory for a good while yet. Therefore, while our primary purpose in this special article has been to familiarize you with ACES-II, this would be a good time to remind you of safety aspects of the IC-7.

Escapac has even fewer safety devices to worry about than ACES. The "Head Knocker" (Ejection Seat Ground Safety Handle) is required to be down and locked for all ground operations; and the 68D110015-1001 Ejection Seat Safety Pin must be installed in the M-99 Initiator for all cockpit area maintenance operations. The Canopy Ground Safety Locks are also required. ■

(PUBLISHED 1975)

First Flight of the Eagle

(without a canopy)

By DENNY BEHM/*McDonnell Experimental Test Pilot, Edwards AFB, Calif.*

In a test program, not everything you find out about an airplane's characteristics is intentional. Thus it was that on the 10th of February, 1975, at approximately 1100 hours in a Cat II flight over North Edwards (AFB), it became possible for McDonnell test pilot Denny Behm to evaluate "topless" performance of the F-15. While it wasn't part of the planned test profile, it's interesting that this first F-15 canopy loss occurred during the airplane's final ECS qualification flight! As Denny indicates below, he certainly got a quick and sudden look at a heretofore unexplored portion of the Eagle's "environment" . . .

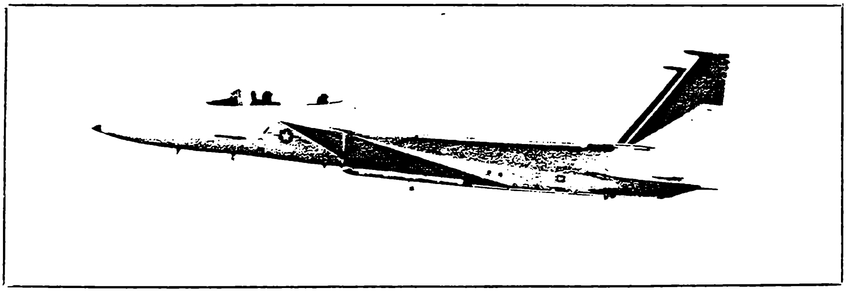
"During a recent flight in TF-1, I had the unique opportunity to experience flight without a canopy. The flight was scheduled to go out to 2.0 M at 50,000 feet, but shortly after takeoff the canopy departed unannounced! I was climbing through 20,000 feet at 375 knots when the canopy unlock light came on. My first thought was to get the power back and descend. Unfortunately, this didn't help since the unlock light was followed in short order by the canopy handle moving aft out of the locked position. Just as I began to reach for the handle to lock it, the canopy moved aft slightly and disappeared. Canopy departure was clean with no damage to the aircraft.

"The first couple of seconds following explosive decompression were confusing to say the least. The cockpit immediately filled with dust and kneeboard cards, but these cleared in just a few moments. By now the airspeed had dropped to 300-350 and the cockpit environment became rather pleasant. Cockpit temperature stayed comfortable and the noise level dropped to the point where normal communications were possible with the ground for the rest of the flight. The slip stream was well outside my normal sitting position. I was able to remain in my normal position with my visor up with no noticeable problems. It would have been interesting to find where the slip stream was actually located, but I didn't have the nerve to ease my hand far enough out to tell.

"The aircraft handling qualities were unaffected in the clean and power-approach configuration. There was no noticeable buffet at any speed from 350 knots to landing. The only unusual thing about landing the aircraft was the very uncomfortable feeling that I might fall out of the cockpit if I were to lean to one side or the other! If you're an Eagle driver, you know the feeling of being exposed in the normal cockpit. Without the canopy, it's almost unbelievable.

"This flight was a solo operation so we didn't get any definite opinions about back seat windblast. However, it appears from the way the personnel leads (which were tied down) were whipped around that the back seat would be somewhat less comfortable than the front.

"It was interesting to me that a situation which was one moment very frightening became a quite pleasant experience when all the dust settled. As a matter of fact, I thought as I was coming down final that had I been able to toss a long white scarf over my shoulder, the air superiority loop would really have been closed!"



After Mr. Behm's uneventful landing, TF-15 71-0290 was impounded until the next day, when Ed Jeude, McDonnell Canopy Design Specialist, arrived from St. Louis for a close look at the canopy system. With company Field Service Engineer Dick Doty and USAF technicians MSgt Morris Garrett and TSgt Douglas Nelson, Mr. Jeude made up a detailed analysis sheet and examined the system.

They discovered that the canopy locking mechanism would not go fully overcenter because a newly replaced push-pull control cable was bottomed out. The Overcenter Warning switch should have signalled this condition, but it was also out of rig - a double failure. Additional steps have been added to the checkout procedure to prevent reoccurrence. ■

MEMO TO MANAGEMENT

(PUBLISHED 1976)

MEMO

290-2905
17 May 1976

SUBJECT: CANOPY LOSS ON F-15 AIRCRAFT 74-115 (F76)

To: J. E. Krings *

CC: I. L. Burrows, H. H. Cole, H. F. Creel, J. F. Dobronski, W. S. Ross, E. R. Shields

From: W. H. Brinks

1. The second functional check flight on F-15 74-115 proceeded normally for the first twenty minutes. An A/B takeoff and intermediate power climb was made to the vicinity of Columbia, Missouri. Approximately five minutes were spent in cruise flight between 37 and 38 thousand feet at .9 Mach. At 37,000 feet, .9 Mach, both afterburners were lit to complete a high speed functional check to Mach 2.0. (The high Mach acceleration was aborted on Flight No. 1 because of a "canopy unlocked" light at Mach 1.6, 40,000 feet.)

2. Mach numbers were transmitted every tenth to Flight Test engineers to record acceleration performance. The acceleration was commenced with the seat height full down and visor down per my personal checklist. "Mach 1.5" was the last verbal transmission prior to canopy departure at approximately 1.6 Mach, 37,000 feet. No "canopy unlocked" light was observed prior to canopy departure.

3. The first cue that something had gone awry was an extremely loud, continuous roar. This was accompanied by some airflow buffet, which moved my head moderately. I was aware that moderate suction forces were attempting to move my body upward. Absolutely no problems associated with explosive decompression or temperature changes were observed at any time.

4. About two seconds after the roar commenced, the "canopy unlocked" light confirmed my suspicion that the canopy was no longer on the aircraft. The throttles were gently reduced to Military, and the speed brake was extended. Gradual throttle reduction below Military was monitored to ensure the EEC "lock-up" feature had worked correctly.⁼ (This probably kept the airplane faster for a longer period than desired, but who needs more than one emergency at a time?)

5. At 1.5 Mach, head motion stopped. Attempts to transmit the nature of the problem to ground personnel and other aircraft were unsuccessful, due to high ambient noise obscuring the transmission sidetone. The IFF switch was moved to "emergency" at this time to communicate the nature of the emergency. It was after the head motion stopped that the "head knocker"

⁼EEC - Engine Electronic Control

*Jack Krings/Chief Test Pilot; Irv Burrows/Deputy Program Manager, F-18 (former Chief Test Pilot); Hermon Cole, Director, F-15 Design Engineering; Henry Creel/Director, F-15 Engineering; Joe Dobronski/Director, Flight Test; Bill Ross/Vice President, Flight & Laboratory Development; Russ Shields/Director, Program Engineering.

(ejection seat safety) was discovered in the down position. I remembered bumping my head against the head rest prior to takeoff, so I assumed that the stagnation pressure about the top of the seat had moved the handle to the down position.

6. At about 400 knots, I heard the first ground transmission from Center requesting the reason for the emergency squawk. After explaining the situation, emergency vehicles were requested from MAC Radio and an airborne F-15 was requested to join for a visual check. Coherent two-way radio transmissions were only possible after reducing speed to the vicinity of 350 KCAS.

7. The aircraft was configured for a landing at 6,000 feet, 200 KCAS, over an unpopulated area. Controllability checks were uneventful, and a final visual check by the second F-15 confirmed aircraft integrity. My final impression prior to landing was, "What fantastic visibility!"

W. H. Brinks
Project Pilot-Experimental

You have just read verbatim the official company internal report of a recent incident involving the F-15 Eagle. We couldn't think of a better way to present test pilot Bill Brinks account of a recent Eagle canopy loss incident than to reproduce his original report to McDonnell Flight Test and F-15 Engineering management.

This represents the second inadvertent inflight canopy departure (one F-15, one TF-15) in the more than four years of Eagle flight operations, and caused no harm to either aircraft performance or pilot safety. However, incidents of this nature bear their own messages and implications for other flight crewmen, and we therefore felt justified in letting you "inside the operation," so to speak.

Both of the inflight canopy losses have occurred during single crewman operations, so we still have no data relative to back seat conditions. As far as the front seat is concerned, both Bill Brinks and Denny Behm (DIGEST Issue #1/1975) report similar experiences — no effects from the explosive decompression; coherent communications possible at about 350 knots and under; normal controllability of the aircraft; and no uncomfortable cockpit temperatures. Both pilots were struck by the visibility and the heightened feeling of "exposure" during landing.

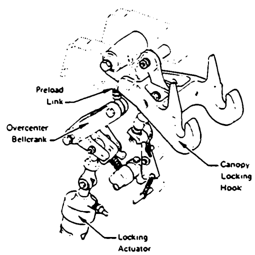
Following Bill's incident, the aircraft was impounded and an investigation team formed from representatives of Flight Safety, System Safety, Design, Strength, Liaison, Tech Pubs, Manufacturing, and Inspection departments. An initial visual inspection of the canopy locking mechanism revealed no obvious discrepancies or failures, so the team established several "possible cause" factors for detailed examination —

- Inadvertent canopy control movement.
- Control system failure commanded the actuator to unlock.
- Canopy jettison system fired.
- Structural failure of canopy locking hooks, latches, or locking mechanism.
- Manual release/jettison system on canopy unlocked.
- Canopy system not properly rigged.

Most of these "possible" causes were eliminated after investigation of the system and supporting structure. The canopy control handle was in the "locked" position. The canopy remover (pyrotechnic jettison thruster) was not fired. Aircraft locking mechanism was in the locked "overcenter" position. There were no aircraft structural failures and no foreign objects in the cockpit (such as broken glass or broken hooks). A functional check of the locking mechanism was satisfactory. Although structural failure of components on the canopy itself was considered improbable, the possibility could not be completely ruled out because the forward portion of the canopy has not been recovered.

The most probable cause as established by the investigating group was in the area of canopy system rigging. The canopy appears to have separated from F-15 74-115 because improper adjustment of the preload link allowed excessive aft position of the canopy and then sufficient aft movement of the canopy from cockpit pressure to disengage the canopy latches.

The canopy locking mechanism provides a positive means of retaining the canopy when correctly rigged; however, the system is complex and care should be taken to see that it is rigged properly. Some changes in procedures and dimensions have been developed which should improve the rigging process — you should see these changes in the near future.



Probable area of rigging deficiency

(PUBLISHED 1980)

ACES II Seat Care

By DON PERSON/ *APG Specialist*

That age old habit of stowing the survival kit straps and seat lap belts over the seat side structure, and not properly securing the CRU-60/P oxygen/ headset connector when not in use is now causing serious problems, and here's why.

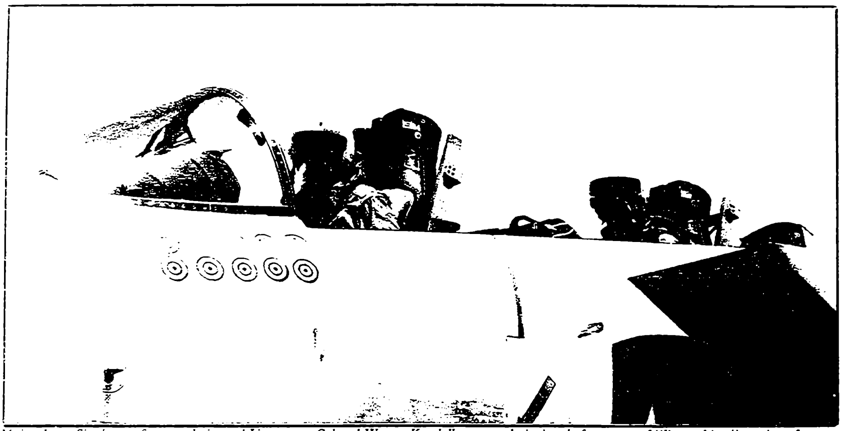
The clearance between the "ACES II" ejection seat side structure and the right console of the F-15 is approximately one inch. During up and down travel of the seat, the survival kit straps, the lap belt buckles, and the CRU-60/P connector can become wedged between the seat side structure in the area of the seat restraint release handle and the console. Fortunately, up to this time F-15 seat damage has been limited to bent and buckled upper seat flanges and broken support clips for the seat restraint handle. Other aircraft equipped with the "ACES II" seat have not been so lucky. Their damage has consisted of pieces broken out of the seat upper flange. Perhaps this damage may seem minimal to you but, and this a big but, *damage of this kind cannot be repaired at the local level. Depot level assistance is mandatory.*

With a little care and concern this unnecessary damage can be prevented. Next time you leave the cockpit of your Eagle make sure you stow the CRU-60/P connector in its storage plug on the right console, connect the lap belt buckles, and leave the survival kit straps on top of the survival kit cushion, as shown in the photograph below. This action takes only a few moments of your time and it can save an expensive piece of *your* life support equipment. ■

(PUBLISHED 1977)

FLYING THE "BALD" EAGLE

By MAJOR JERRY SINGLETON/*Deputy Director F-15 Joint Test Force, Edwards AFB, California*



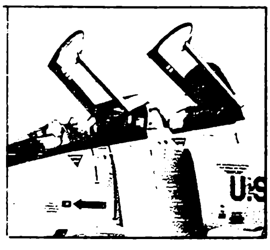
Major Jerry Singleton, front cockpit, and Lieutenant Colonel Wayne Kendall, rear cockpit, just before start of "live subject" portion of recent Canopy-Off test program on TF-15 Eagle at Edwards AFB, California. Major Singleton, author of article below, is Deputy Director of AFFTC Joint Test Force at EAFB; Lt Col Kendall is with Aerospace Medical Research Lab at Wright-Patterson.

Have you wondered what it would be like to lose a canopy and have to return to base (RTB) in that configuration? Hopefully, after you have read this article you will have a better idea of what it's all about. Fortunately, only a few USAF pilots have had this experience (258 canopies lost between 1965 and 1975). It normally is an unhappy event for the pilot, both in the air and on the ground (where he gets the third degree from the Squadron Commander, CO, and Chief of Maintenance). Because nobody wants to *unintentionally* fly a fighter without a canopy, INTENTIONALLY performing the maneuver may sound like an insane idea. But flying the Eagle "bald" was in fact a carefully planned test at the Air Force Flight Test Center (AFFTC), Edwards AFB, California, in the spring of this year.

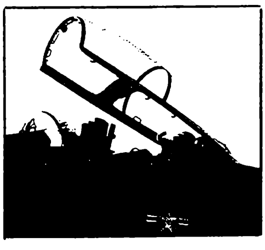
For those unfamiliar with the TF-15, it should be pointed out that there is NO protection for the backseater in the form of either a separate canopy or a windscreen, as is obvious from the photo above. Therefore, these tests were performed at the request of the Tactical Air Command to determine answers to three serious questions:

- What happens to the aft cockpit occupant in the event of an inadvertent canopy loss?
- What kind of protection should be provided for the backseater?
- What emergency procedures are required in the event of a canopy loss?

Everyone involved in this program



F-4 and F-15 canopy systems photos here show obvious differences between aircraft design concepts. All Phantoms are two-place aircraft, with separate canopies and "dog-house" between front and rear cockpits. Both items provide protection for backseater so that he is not directly affected by a forward canopy loss. Only every ninth Eagle produced is two-place, and a single-piece canopy is used to cover both seats.



was of the opinion that answers to these questions could result in lifesaving information for the rear cockpit occupant of the two-seater Eagle. To get these answers, a comprehensive, four-phase test program was developed — (1) high speed taxi test with an instrumented anthropometric dummy in aft cockpit; (2) flight test with dummy; (3) high speed taxi test with human subject in aft cockpit: and (4) flight test with human subject.

TEST PROGRAM PLANNING

Prior to the first test, several planning meetings were held. The first step involved the Configuration Control Board, which had to approve the Class II modification to the aircraft (TF-1, 71-0290). TF-1 was the first two-seat TF-15 produced and has been a mainstay in the F-15 Development, Test, and Engineering (DT&E) test program. The modification included canopy and canopy actuator removal; changes to the ejection seat system to make it operable with the canopy off; installation of a mirror system in place of the Vertical Situation Display in the rear cockpit to allow use of a video camera to constantly monitor the back seater; and installation of a fixed mirror in the front cockpit to permit observation of the GIB (Guy-in-back).

The second step was the review of the proposed test by a Technical Review Board consisting of AFFTC and McDonnell Douglas engineering experts (aerodynamics, stability and control, and human factors). The basic test plan, as conceived by the F-15 Joint Test Force (JTF), was approved with one important addition. It was decided that a thorough evaluation should include a supersonic point to evaluate shock wave effects. Therefore an acceleration to 1.2 Mach at 20,000 feet was added to evaluate this phenomenon.

"Dummy" positioned in rear cockpit of TF-15 prior to first ground tests of canopy-off program. Helmet contained 15 sensors, and eye-mounted transducer was also used to measure impact pressures.

DUMMY AND EQUIPMENT

For the first two tests, a 95th percentile anthropometric dummy (to represent the worst case*) was obtained from the 6511th Test Squadron (a detachment of AFFTC) at NAS El Centro, California. In addition, Captain Ron Hill, the project engineer from the Human Factors Branch at Edwards AFB, obtained an instrumented helmet used by the Flight Dynamics Laboratory of Wright-Patterson AFB in the F-16 canopy-off testing at NASA Ames, NAS Moffet, California. This helmet had 15 high-response transducers installed in the front, top, back, and sides to measure impact pressures from the airflow.

The dummy also had a microphone installed in each ear to measure the sound response in decibels (db) in an attempt to determine how difficult it would be to have successful air-to-air and air-to-ground communications without a canopy. Finally, as project manager/pilot, I carried an audiometer in my flight suit leg pocket to measure noise levels in the front cockpit. Photos of the dummy, instrumented helmet, and "ear" microphone arrangement are shown here.

***Editor's Note: "Worst case" in this test means that the dummy was selected to represent the largest stature of any potential backseater. An article on "human factors," as they affect aerospace design is being prepared for an upcoming DIGEST.**

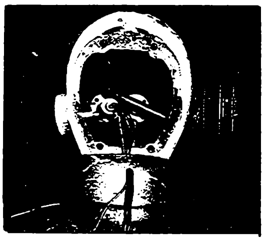
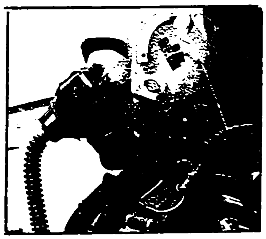


Photo at left shows rear of dummy's head. with microphones installed in each ear to measure noise levels in open cockpit environment. Problems occurred with dummy's equipment early in first test flight, as shown in photo at right. Oxygen mask was forced upward over eye transducer and chin strap moved out of proper position. Several sensors on helmet and visor had been lost during earlier acceleration in this flight.



DUMMY TESTS

The dummy ground tests were completed on 26 May 1977. Two ground runs to 150 KIAS were completed for an initial evaluation of the dummy instrumentation. All the data gave a green light for the actual flight test, which was accomplished the following day.

The flight was "photo chased" by an F-4 flown by the F-15 JTF director, Lt Col Ken Dyson. It was an uneventful flight until the first acceleration was approximately one-half complete. An acceleration to 500 knots at 5000 feet MSL was started, but at 280 KIAS the dummy lost its visor with two-thirds of the 15 installed transducers. While this was a setback (although not unexpected as the rear seat was positioned "full-up" to obtain the worst case results), the test run was continued since the dummy also had a transducer in the left eye.

At 485 KIAS, further problems occurred as the oxygen mask on the dummy suddenly came up over the eyes, and the chin strap was pushed up under the nose. Unfortunately, this caused the test to be terminated as all effective impact instrumentation was now lost. As a result, the "supersonic" effects were not evaluated.

It should be noted that throughout this acceleration the front cockpit was relatively unaffected. Voice communications were never in question and head/body buffet was negligible until above 350 KIAS. At this point, the seat was lowered full down from my normal sitting position (seat up 3/4); and this reduced the slowly increasing head buffet to a very comfortable level.

DUMMY FLIGHT TEST ANALYSIS

Dummy flight test results indicated two potentially important items: First, there appeared to be a strong possibility that the GIB would be subjected to unbearable sound levels. The db meter pegged at 110 db at 50 knots on the ground tests and at 130 db at 85 knots on the flight tests. However, we were unsure as to how well the ear cups in the dummy's helmet were sealed with the head, and this caused us to question the data (results from the human flight test demonstrated no auditory problems). Second, the helmet instrumentation told us that the dummy never felt 0.80 psi impact pressure.

AGARD (Advisory Group for Aeronautical Research & Development) data told us that 0.80 psi, which equates to 180 KIAS freestream velocity, would subject a human to eyelid flutter if his visor were up. To stay on the safe side, it had been decided by the Safety Review Board that the velocity that resulted in 0.80 psi would be the maximum to which we would subject a human (in case he lost his visor). While a GIB would survive this velocity in an inadvertent canopy loss, it was still a point where he could be severely injured if caught unprepared.

> **"FOR REAL"**
>
> **On 16 September 1977, USAF officers Captain Loren Shriver and Lieutenant Colonel Dick Cooper became the first Eagle two-man flight crew to experience unintentional loss of a canopy in flight. MCAIR experimental test pilots Denny Behm and Bill Brinks had previously undergone the same experience, Bill in an F model and Denny in TF-1 without a backseater, but Lt Col Cooper is the first man to have occupied the aft cockpit in a "for real" canopy loss.**
>
> **Captain Shriver and Lt Col Cooper were returning from a radar tape evaluation mission over Edwards AFB, California. They were at 20,000 feet, 0.9 Mach, Two to Three G's inverted, when the canopy departed the aircraft. Details of the incident are not available at the moment, but neither crewman was injured or incapacitated in any way. Lt Col Cooper reportedly experienced no problems with retention of his helmet; in sharp contrast to the difficulties undergone by Lt Col Kendall in the planned canopy-off test program.**
>
> **If possible, the next issue of the DIGEST will offer first-hand reports from both crewmen involved in this incident, as well as a maintenance analysis of the loss by Dan Drapp, MCAIR Senior Design Engineer, canopy systems.**

HUMAN TESTS

The "real" dummy, Dr. (Lieutenant Colonel) Wayne Kendall of Wright-Patterson's 6570th Aerospace Medical Research Lab, had volunteered (no kidding) earlier to be our guinea pig. (Dr. Kendall also had been the pilot/flight surgeon who had participated in the F-16 canopy-off tests at NASA Ames.) During the TF-15 Eagle tests he used all his own personal equipment to make the test as realistic as possible.

The only instrumentation available was the video tape camera to show the effects of the windstream on him. A deadman's switch also was installed as a part of the Class II Mod to the aircraft. Dr. Kendall had to hold a button down on the switch to keep a warning light from illuminating in the forward cockpit. If he let go, the light would come on and the aircraft would be slowed down immediately. This was actually a backup in case cockpit communication was impossible due to high noise levels.

The human tests were started on 1 June with two taxi tests to 150 KIAS on the 15,000 foot Edwards runway. Communications and wind blast effects were acceptable. One run was done with the aft seat full down, and one with it full up. There was a significant difference in wind effect, confirming the obvious (?) fact that full down was a better environment for the GIB.

The flight test was completed on the next day. Takeoff was made with the aft seat full down and remained there most of the flight. Airspeed was held to 175 KIAS initially and the aircraft climbed to 5000 feet MSL. Cockpit communications were possible with Dr. Kendall either sitting up or taking "protective measures" (bending forward as far as possible with shoulder harness unlocked). Accelerations were made in 20 knot increments for safety considerations.

It was quickly discovered that 200 KIAS was the maximum speed at which intelligible conversation could be successfully completed between cockpits, even when taking protective measures. It was further determined that front seaters can understand back seaters up to 250 KIAS, but 200 KIAS is the maximum for conversations going both ways, Remember, this was with the aft seat full down, an abnormal seat position during operational missions. The deadman switch installation now paid off as it allowed us to safely continue the acceleration without intercom.

The acceleration was terminated at 415 KIAS due to loss of air-ground communications in the front cockpit. There was so much noise from the mask flutter of the GIB that I could no longer understand transmissions from either the ground or the airborne chase. Although the deadman switch was never activated, Dr. Kendall admitted he was near his limit with the seat full down and bending forward under the glare shield as best he could.

WHAT HAPPENED BACK THERE

The flight ended after 50 minutes. Dr. Kendall was obviously exhausted from his ordeal; however, his only injury was a small abrasion on the right cheek that was caused by his oxygen mask strap flapping against the skin. An audiogram was performed immediately after the test and no temporary hearing loss was found. The noise level in the rear cockpit was not painful, which confirmed our earlier doubts above the noise data we recorded during the dummy tests.

We learned several important facts during the test run:

• Dr. Kendall had to use one hand to hold onto his helmet (elbow pointed straight ahead, not sideways) above 280 KIAS. He felt sure he would have lost the helmet above that airspeed. Buffet and vibration made it impossible for him to see the instrument panel clearly. Flying the aircraft from the rear cockpit would not appear to be possible above approximately 200 KIAS (hopefully one would never have to try it).

• Eye "tearing" with the visor down was severe above 350 KIAS, and it became difficult to breathe above 400 KIAS. Clearly this is not a comfortable environment! Significant effort was required to maintain the forward protected position. It was Dr. Kendall's opinion that, "with the rear seater caught sitting up, he would not be able to lean forward until the aircraft was slowed to below 300 KIAS."

• Seat position was also evaluated up to 250 KIAS. The rear seat was raised to approximately one-half up at 200 KIAS. The environment worsened considerably. Dr. Kendall was not willing to go above 250 KIAS with the seat in that position. Obviously, full up would have lowered that airspeed even more. Most pilots fly the rear seat almost full up for best visibility.

• Speed brake and angle of attack effects were also evaluated. The small preproduction speed brake was extended every 20 knots to determine if it had any adverse effects on the GIB. None were noted. It was thought that perhaps increased alpha would help protect the rear cockpit. Angle of attack was increased to 21 units at 200, 250, and 300 KIAS but no beneficial or harmful effects were noted.

CONCLUSIONS

I hope no one ever has to make use of this information; but I know our friend "Murphy" is lurking nearby. As long as we fly, the opportunity for canopies to be lost exists. So here are a few primary pointers we derived from this interesting JTF program —

• If a canopy loss is the only problem, the front seater is in a benign environment and can easily RTB safely. Communications are normal and recovery is a piece of cake. If you slow to 250 KIAS or less, you will hardly know the canopy is gone.

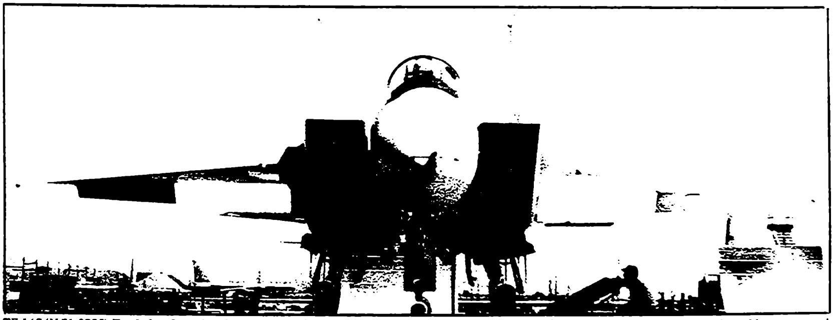
• Don't either of you stick your arm or even a pinky near the slipstream. That's a quick way to ruin a happy ending to a real emergency. However, you can raise your arms to give HEFOE signals or tighten your oxygen mask, lower your visor, etc. (Within the normal confines of where the canopy originally was, front seater arm or body movements can be made with zero problems; back seater should restrict movements to those suggested in the next paragraph.)

• Some kind of protection should be provided for the back seater. A canopy lost during ACM at 350 KIAS or greater could have severe effects on the GIB. Most probably, the helmet would be lost and vision would be impossible. In the past, the GIB has always ejected under these circumstances due to severe disorientation. Hopefully, these test results will be publicized well enough so that this will not have to happen in the future. If you are in this situation, lower the seat and lean forward as far as possible. This will provide the best possible environment. If the helmet is not lost, hold onto it with one hand (elbow forward, not sideways) and pull forward with the other hand (grab the lower instrument panel above the rudder pedal wells).

• Communication between cockpits will not be possible until below 200 KIAS, so just sit tight. The front seater will be able to talk to the ground below 400 KIAS. He should slow the aircraft using speed brake and moderate G's (remember the GIB will be trying to lean forward) as quickly as possible.

• Once you get below 200 KIAS, no sweat — you can even talk if the GIB has retained his helmet. If above 25,000 feet, you would also obviously want to descend as (1) it's cold, and (2) oxygen mask flutter will make a good face seal impossible.

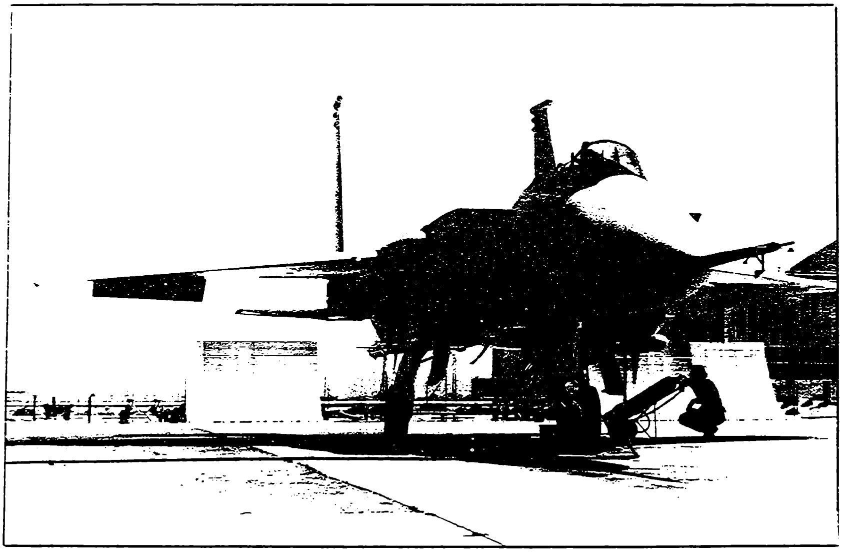
• Expect lots of junk in the eyes, even with visors down. Even though our cockpits had been vacuumed with both seats removed, dirt and debris were real problems for both crewmen. It was so bad on my first flight that I was constantly blinking and tearing. ■



TF-1 (S/N 71-0290) Eagle has been involved in several flights without canopy, two on purpose and two unintentional. First accidental loss occurred early in 1975; second took place three months after conclusion of test program described herein. (Readers should note that Major Singleton's article is intended to present only a summary of this canopy-off program; the complete technical report has been prepared by Project Engineer, Captain Ron Hill, in limited distribution document AFFTC-TR-77-21.)

". . . 71-0290 took off that day with an unlocked canopy and a switch sufficiently misrigged to indicate a safe system. . . "

BALD EAGLE FLIES



The last issue of the PRODUCT SUPPORT DIGEST contained an account of the F-15B Canopy-Off test performed on aircraft No. 71-0290 at Edwards Air Force Base in June 1977. On 16 September 1977, USAF officers Captain Loren Shriver and Lieutenant Colonel Dick Cooper performed an unintentional repeat of that test.

Lt Col Cooper, the back seater, was putting the airplane through its paces on the way back from a radar mission over Edwards AFB when the canopy left the airplane. He recovered from the maneuver and relinquished control to Capt Shriver. Both crewmen were able to control the aircraft and suffered no ill effects.

An investigation was begun immediately to determine the cause of the canopy loss. The canopy was recovered and showed no signs of latch or locking roller failure. The canopy remover had not fired. Attention was then directed to the canopy locking mechanism and control system to find a probable cause. The canopy locking mechanism was found unlocked; the front canopy control handle was found in the full forward (locked) position; and the rear canopy control handle was found 10° aft of the full forward position. The canopy lock position switch was open (canopy unlocked light on). Applying force to either handle would close the canopy lock position switch (canopy unlocked light off), but the locking mechanism could not be locked. 71-0290 took off that day with an unlocked canopy and a switch sufficiently misrigged to indicate a safe system.

Since both handles were reaching a full forward position without locking the canopy, a teardown of the canopy locking mechanism and control system was instigated to determine why the mechanism could not be locked. Several reasons were found -

- The teleflex control cable and forward gearbox were so badly worn that apparently a tooth was slipped.
- The plunger on the canopy lock position switch was bottomed out, preventing the mechanism from locking.

(PUBLISHED 1977)

By DAN DRAPP/*Senior Engineer - Design*

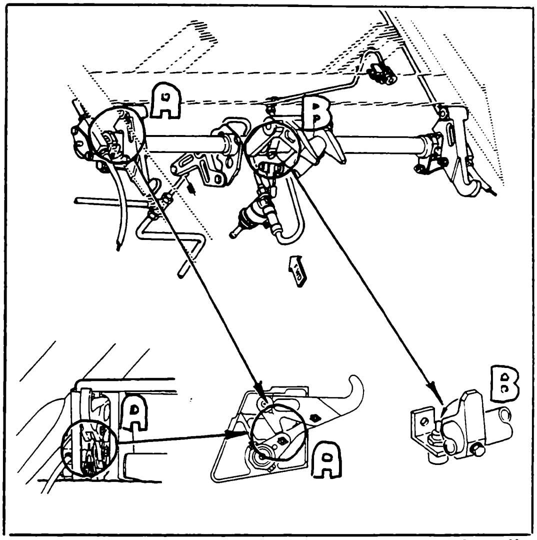
"BALD" AGAIN

• A structural interference was discovered between the canopy remover bellcrank and both the pressure regulator cam and detent feel cam on the canopy cross-shaft.

The slippage that occurred at the forward handle explained the difference found between the positions of the forward and aft handles. The cause of cable wear is application of excessive force to the canopy control handles. Attempting to lock the mechanism with the noted interferences or before the canopy is full forward will damage a cable.

Before drawing some conclusions on these facts as discovered, let's briefly review the "mechanics" of the F-15B canopy control system -

Canopy operation can be controlled by handles located in the front or rear cockpits or by an external handle. Canopy motion is accomplished hydraulically, and locking is accomplished mechanically through an overcenter link at the locking hooks. Movement of any control handle will also move the other control handles through the direct mechanical interconnection of the internal and external control cables and the canopy locking torque tube. Placing of a control handle into the hold or lock position will hydraulically lock the canopy actuator, independent of the position of the canopy itself. Premature selection of the lock position, before the canopy is completely forward, stops the canopy motion. Damage to the control cable will result due to the interference between the locking hooks and the locking rollers.

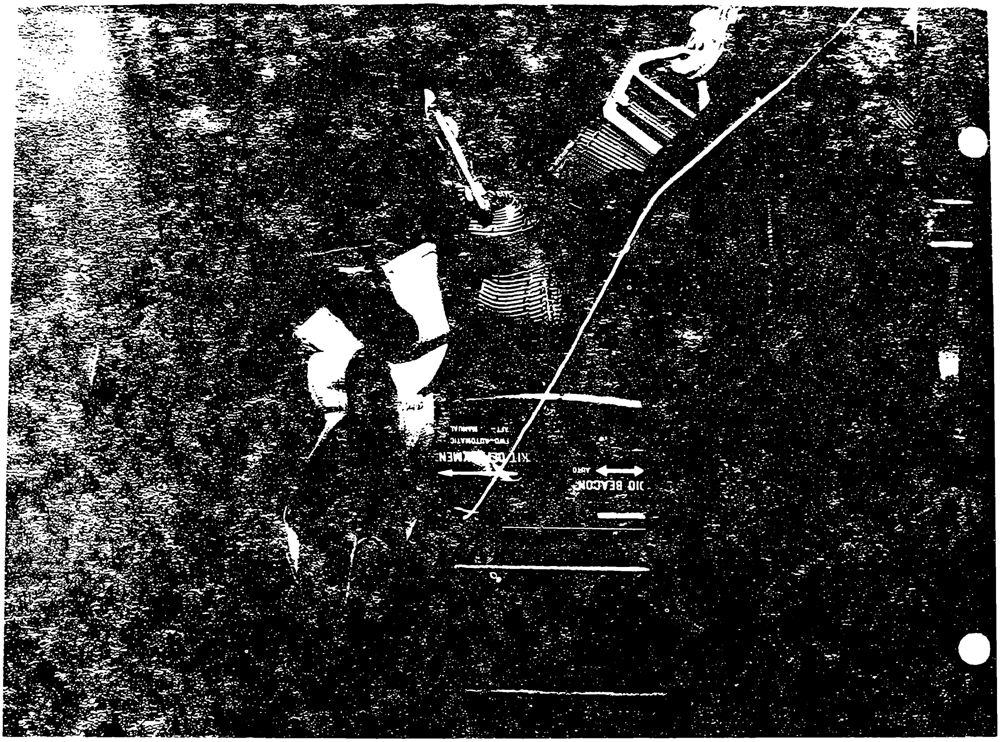


Top sketch circles two areas that were primary parts of Bald Eagle's canopy loss problem. (A) shows locking hook/overcenter link area where overcenter stop on either left or right side of canopy torque tube must be making contact to assure the mechanism is locked. The canopy lock position indicator switch (B) was misrigged (plunger bottomed out), preventing the mechanism from locking or from giving an indication that the canopy was unlocked.

The canopy departed from this airplane because the locking mechanism was not locked. Murphy convinced the pilot that it was locked, however. This incident, again, points to the importance of properly rigging indicator switches. The damage and interferences in the canopy control system should have shown up immediately in the form of a canopy unlocked light on the main instrument panel *had the Canopy Lock Position switch been rigged correctly*. This light is the pilot's only visible indication that his canopy is locked. Not only did the switch not give the pilot warning, but the switch itself caused an interference in the system.

Another important item to remember when closing the F-15B canopy is the 10-second delay required after apparent forward motion of the canopy has stopped, before putting the control handle from down to lock. If the canopy is not completely forward, the locking hooks will strike the canopy locking rollers and resist handle loads applied through the cable. Observing this delay will reduce the possibility of overloading the control cable. ■

One final question. Since this is the second inadvertent canopy loss on this particular airplane, do you think that the importance of indicator switch rigging can be overemphasized?



Pop Goes the Life Raft!

(PUBLISHED 1979)

By PAT HENRY/*Chief Experimental Test Pilot* and FRANK MANIE/*Senior Engineer, Design*

During the course of a recent accident investigation, the potential results of an uncommanded in-cockpit life raft inflation were brought into sharp focus. While highly improbable, there is evidence that it does happen, and that it could contribute significantly to a hazardous incident or even accident. Since it would be an immediately alarming event for the pilot, to say the least, this article hopefully will help prepare him mentally for the quick reaction necessary to cope with this unpleasant potentiality.

The scenario we investigated, and are reporting on below, is that of a life raft suddenly trying to inflate with the pilot still strapped comfortably (up to that point) in his seat in the airplane. This huge canvas balloon that's trying to instantly mature underneath an unfortunate crewman can break the survival kit latches as it attempts to escape its confines. When this happens, it is guaranteed to cause personal discomfort of rapidly increasing severity, and more than likely some incapacitation, as you will see below.

The Investigation

The Life Support, Crew Station Engineering, System Safety, and Flight Operations groups here at MCAIR collaborated to investigate this situation, using an ACES II seat and a laboratory test setup. The object of the investigation was two-fold, thus requiring two consecutive tests. The first test was designed to establish the force, in terms of lapbelt load, that would be induced by life raft inflation; the second was to quantify the human tolerance level and reaction to those loads.

Test One — The Dummy

For the first tests we used a 5th percentile Alderson dummy seated in an ACES II ejection seat. All attaching hardware was standard USAF equipment except for the lapbelt, which was instrumented with strain gages to measure tension. The survival kit was a production "fly-away" ACES II kit, fully packed, including the LRU-16/P life raft and the FLU-2A/P carbon dioxide cylinder. Both a pressure transducer and a direct reading gage were installed in the inflation assembly to record life raft pressures. Initiation was via a static line attached to the CO_2 bottle.

Upon actuation of the CO_2 bottle, the life raft partially inflated within the survival kit container. Pressure within the life raft reached 45 psi almost instantaneously. The raft cammed the seat pan latch open and rotated the seat pan up approximately 2.0 inches along the rear edge, raising the dummy in the seat accordingly. Lapbelt tension reached 450 pounds within 3.0 seconds and rose to 475 pounds within 50.0 seconds!

While these figures may not seem significant at first, consider that the 45 psi measured in the raft increases with altitude. Thus the lapbelt tension of 475 pounds, equivalent to a downward load of 950 pounds across the mid-section, would also increase with altitude, producing a tension of 540 pounds, or a total load of 1080 pounds at a cockpit altitude of 14,000 feet!

Test Two — Live Subjects

Obviously, only a dummy would "sit still" for something like the highest numbers recorded in the first tests. So our second series of tests were conducted on real live "volunteers," who would add a subjective aspect to the investigation by being able to say "OUCH, this is enough!" at the proper time. We proceeded to these second tests somewhat cautiously, but anxious to see exactly what human reaction to loads of these magnitudes would be.

This test setup was a little more elaborate than our first one because of the safety precautions required. The seat, attaching hardware, and survival kit were from the first test, but in place of the CO_2 bottle, we used a controlled nitrogen source with a subject-held dump valve switch. This was necessary to command the inflation value while maintaining the applicable rate of onset, while allowing the subject to terminate the test in the event the forces became too great.

The test subjects were subjected to gradually increasing lapbelt loads, each time deflating the raft and recharging the nitrogen source to the next higher value. The tests were continued until lapbelt tensions of slightly over 400 pounds (800 pounds total load) were reached. At this point we seemed to be approaching the subject's willing tolerance level, with no doubt that the significance of the problem had been clearly demonstrated.

The Conclusions

After the tests were completed, we analyzed the data and drew the following conclusions:

- While the test did not establish the ultimate lapbelt tension physically tolerable by an aircrewman, it was generally concurred that lapbelt tensions of about 400 pounds are extremely uncomfortable and would require immediate corrective action.
- Within the limits of the tests performed, no involuntary reactions were noted; however, it can be reasonably assumed that the total surprise of the inadvertent raft inflation would draw a good portion of the aircrew's attention immediately, and would center his attention on resolving that specific situation.

Ron Warren, MCAIR Life Support Technician, positioned in test seat. He holds safety dump valve switch in right hand while the nitrogen source can be seen on his left. Gages indicate both tank (source) pressure as well as raft pressure. The photo on preceding page shows area between survival kit pan and cover where knife could be inserted to puncture inadvertently inflated life raft.

- The maximum time that the lapbelt load is tolerable appears to be unpredictable from these tests and possibly variable from subject to subject. It is fair to conclude that a crewman would make every possible effort to relieve the tension as soon as possible rather than live with the condition until landing the aircraft.

The technical and quantitative results of a life raft inflation as measured in our simulations are obvious in the numbers recorded. It is somewhat more difficult to clearly convey the qualitative and emotional results of such a traumatic event.

All subjects, pilots and engineers alike, were unanimous in describing the rapidly mounting pain and alarm as the seat rose and attempted to squeeze one in half with his own lapbelt. Even though sitting there in the relative comfort and security of our Life Support Equipment Lab (at zero mach and floor level) we all experienced an apprehension which built almost instantly — probably because there was no discernible end point to the rapidly increasing pain. This concern and pain immediately commanded almost all of one's attention. It is easy to imagine how much more alarming it would be inflight due to the hostile environment and obvious potential consequences. There's no doubt in our minds that the pilot is going to stop doing almost everything, except flying into the trees, to address this new problem. It is a real attention getter.

None of us subjects were the least bit interested in subjecting ourselves to loads representative of what the dummy felt during the actual inflation test. When one realizes that those loads would be significantly higher again at elevated altitudes, the picture becomes even more sobering.

The Recommendations

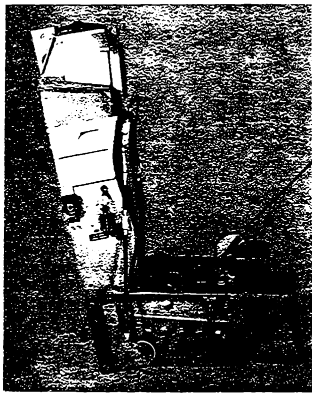
Based upon the results of our investigation, we offer three recommendations in the event you should suddenly find yourself vying for cockpit space with that huge balloon:

- Don't Release the Lapbelt — This may sound obvious, but during the surprise and alarm of an airborne inflation, you might be tempted to go with your first impulse, namely, to relieve the lapbelt tension by releasing the belt. If you do, your rapidly growing raft is going to try and drive you through the main instrument panel.
- Puncture the Raft if Possible — The photo at the left shows the most viable access to the raft. Due to the attach point design of the survival kit cover, it should raise up as much as an inch across the front, giving a wide target area for knife point deflation.
- Descend as Soon as Possible — If you're unable to relieve the pressure in the raft, descending to as low an altitude as possible will minimize the pressure differential between the raft and ambient air, thereby keeping expansion force to a minimum.

Let us conclude by reminding you that we aren't trying to sound like alarmists — we recognize the probability of any one of you having to cope with an uncommanded life raft inflation is extremely small. Nonetheless, the potential is there and its impact is very serious. In the absence of more realistic simulation, we hope that this report will help you to be prepared for the unexpected. ■

(PUBLISHED 1980)

ACES-II SEAT SAFETY LEVER



BEFORE TAXIING

16. INS mode knob - INS (when aligned)
17. Altimeters - SET
18. Ejection seat - ARMED
    a. [illegible]
    b. Ejection controls safety lever - ARMED (ACES II)
19. JFS LOW light - OUT
20. BIT lights - CHECK
    Investigate any unusual BIT lights before taxiing.

Suppose — just suppose — that for some strange and sudden reason, you have decided to vacate your airborne Eagle via the ACES-II ejection system. You did remember, way back in Step 18 of your BEFORE TAXI checklist, to flip the Ejection Controls Safety Lever back to ARMED, didn't you? While it only takes a moment — now or then — to move the lever, spare moments are in short supply when you're at 2000 feet and out of luck. Do your things at the right times and avoid what could be one of life's most embarrassing moments, where you find yourself all dressed up and no way to go!

FLIGHT CONTROLS ►

(PUBLISHED 1975)

A Broad-Brush Look at...

THE F-15 HYDRO-MECHANICAL CONTROL SYSTEM

By B. P. "PERRY" HOFFMAN/*Electrical Engineer, Flight Control Section, Avionics Engineering Laboratories*

At the beginning of any aircraft design program, the customer specifies his requirements and desires. In the case of the F-15, handling qualities were rigidly spelled out by the USAF: "*The aircraft must meet or exceed Level II requirements throughout its operational envelope without the aid of electronic augmentation.*" Military Specification MIL-F-8785B(ASG) defines all the details of flying qualities sought in an aircraft. For the sake of this article, the following brief definitions should suffice:

- Level I - Flying qualities clearly adequate for the mission Flight Phase.
- Level II - Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.
- Level III - Flying qualities such that an aircraft can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both.

In short, this means that the basic hydro-mechanical control system must be such that a pilot can complete an air-superiority mission without a bunch of electronic boxes doing it for him. To better explain Level II handling, an F-4 Phantom (in contrast with the F-15) is incapable of meeting Level II requirements throughout its maneuvering envelope *with* SAS (or Stab Aug) operating.

Within this article we'll explain how the controls of the F-15 Eagle satisfy this requirement. In later issues of the DIGEST we'll go a little deeper into the system to shed some light on the role of electronics in increasing control capabilities to Level I handling qualities.

CONTROL STICK BOOST/PITCH COMPENSATOR

Since the Eagle's flight controls are designed with a fighter pilot's needs in mind, the end result is a blend of specification requirements and pilot desires. Any reference to the similarity between a conventional century series fighter control system would be difficult. It's obvious that both contain control sticks and control surfaces; however, in the F-15 it's what's in between that makes the difference.

The part that's "in between" is what we call the "CSBPC," or Control Stick Boost/Pitch Compensator. This device is the "brains" of the F-15 mechanical control system and contains two major assemblies known as the "Pitch/Roll Channel Assembly" (PRCA), and the Aileron Rudder Interconnect (ARI).

Since any aircraft responds differently to a given control surface input, depending upon the flight condition and extent of maneuvering, considerable sophistication must be employed within the mechanical control system to assure uniform response to pilot commands. The PRCA and ARI units help the basic hydro-mechanical controls provide the maneuvering capabilities and handling qualities required to satisfy the Level II specifications.

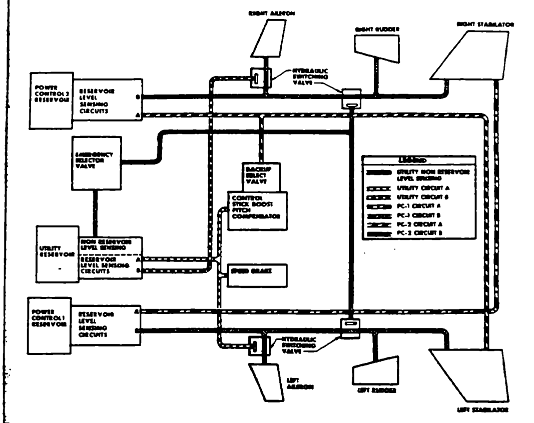
Since the applications of the PRCA and ARI in the Eagle are quite involved, we won't discuss them in detail at this time. Instead, we'd like to consider the total hydro-mechanical control system now with coverage of individual axis and electronic portions in forthcoming issues of the DIGEST.

HYDRAULICS

The F-15 control system is powered by three separate hydraulic systems: Power Control One (PC-1) driven by the left engine, Power Control Two (PC-2) driven by the right engine, and a Utility system which contains two pumps, one on each engine. Each system is provided with a switchover valve which senses system return pressure. If pressure falls below a pre-selected value, required pressure is regained through a switch to another system.

Referring to the hydraulic system block diagram (Figure 1), you can see which hydraulic system powers which control system actuator. The PC-1 system powers the left side of the aircraft plus both stabilator actuators. The PC-2 system powers the right side of the aircraft plus redundant power to both stabilator actuators. The Utility hydraulic system is a backup system and

FIGURE 1 - FLIGHT CONTROL SYSTEM HYDRAULIC DIAGRAM



can provide power to the entire control system. The PRCA and ARI receive their hydraulic power from the Utility system with PC-2 as a backup. What this all adds up to is a system that can be safely flown and landed after a total loss of any *two* of the three hydraulic systems.

LONGITUDINAL CONTROL SYSTEM

At first glance the Longitudinal Control System (Figure 2) seems to be a conventional system, but as you look at component locations some interesting and important differences become evident.

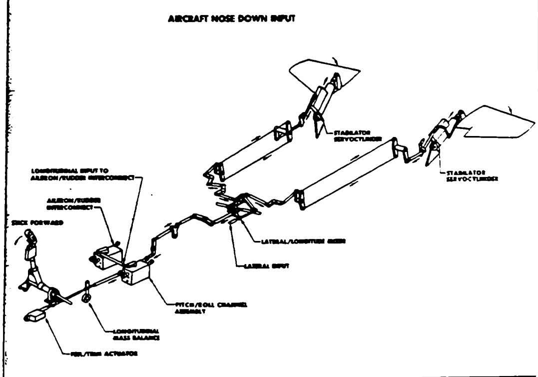
The feel trim actuator, located in the aft fuselage of most aircraft, is located below the control stick in the F-15. This reduces the amount of linkage, thus reducing control stick deadband, and lessens overall applied stick force.

Added safety is also obtained should there be a linkage separation downstream of the PRCA. If a separation *does* occur, a "fly-by-wire" capability is provided by the electronics and the pilot will still have positive feel at the stick. With a manual system such as installed in the F-15, a pilot may not even realize he has a linkage separation since the aircraft will fly and feel the same with or without the problem.

The Pitch/Roll Channel Assembly (PRCA) provides variable mechanical advantage of the pitch control system as a function of airspeed system data. It also aids in controlling stabilator deflection to eliminate the difference between commanded and actual load factors. This feature compensates for trim changes due to such things as speedbrake or flap extensions, external store separations, and aircraft speed changes. The combination of feel trim, variable mechanical advantage, and series trimming gives the pilot, as near as possible, a constant stick force per G and keeps the stick pretty well in the same place in the cockpit throughout the flight. The linkage friction within the PRCA is carefully controlled to reduce control stick breakouts. The feel trim actuator location and shortened linkages to the PRCA and its low linkage friction provide the pilot with smooth, light control stick breakout forces. The PRCA output is hydraulically boosted, eliminating any feeling by the pilot of excessive frictions downstream of the PRCA. In addition, the hydraulic boost provides a shear force for chips and other foreign objects.

Outside of the PRCA the pitch linkage is fed to a "mixer" linkage where it is combined with roll inputs. These give the stabilator inputs reflective of either pitch or roll. The F-15 stabilators are used collectively for pitch and differentially for roll. ▸

FIGURE 2 - FLIGHT CONTROL SYSTEM – LONGITUDINAL CONTROLS



LATERAL CONTROL SYSTEM

As you review the design of the Lateral Control System (Figure 3) you'll find some similarities to what we've just covered in the longitudinal system. The feel trim actuator is located below the control stick, and it is there for the same reasons mentioned for the pitch trim actuator.

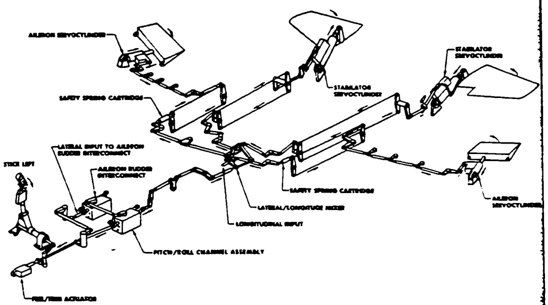
Variable mechanical advantage of the Lateral Control System is provided by the Roll Channel of the PRCA as a function of airspeed data. The stick-to-aileron ratio is also reduced as a function of *longitudinal* stick position. As angle of attack is increased, deflections are decreased for a given stick deflection. This eliminates the need for a pilot to remember to roll only with rudder during high angle air combat maneuvers. As the stick-to-aileron ratio is decreasing, the ARI is supplying information to increase rudder deflection.

Roll output of the PRCA is hydraulically boosted for the same reasons as is pitch output. The mixer linkage, referred to earlier, receives a lateral input which is transmitted to the ailerons and as differential signals to the stabilators.

A safety spring is provided, allowing continued roll control operation should one side become totally jammed. The aircraft can be safely flown and landed with one aileron and differential stabilator control. Aileron surface power is supplied by conventional hydraulic actuators.

FIGURE 3 - FLIGHT CONTROL SYSTEM – LATERAL CONTROLS

AIRCRAFT LEFT ROLL INPUT



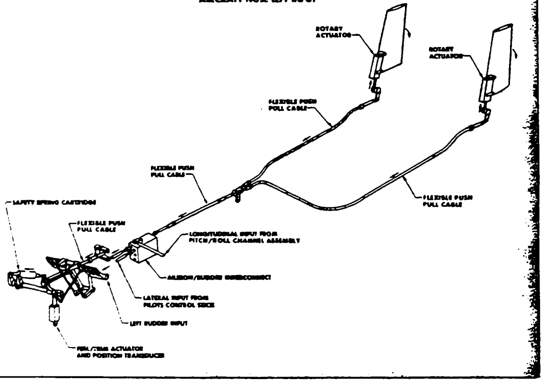
DIRECTIONAL CONTROL SYSTEM

The Directional Control System (Figure 4) is equipped with a feel trim actuator which is located forward and between the rudder pedals. A safety spring cartridge permits continued aircraft control and nosewheel steering in the event the rudder linkage jams. Should a linkage jam occur, mechanical control is no longer possible; however, pedal forces can be sent to the CAS electrically which will give "fly-by-wire" control of the rudders.

Mechanical pedal inputs are supplied to the ARI box, scheduling rudder control as a function of lateral and longitudinal inputs. The output of the Aileron Rudder Interconnect repositions a flexible ribbon which moves two rotary actuators, deflecting the rudder control surfaces.

FIGURE 4 - FLIGHT CONTROL SYSTEM – DIRECTIONAL CONTROLS

AIRCRAFT NOSE LEFT INPUT



EMERGENCIES

Should hydraulic power to the PRCA be lost, or if the pilot elects to select emergency modes of either roll or pitch through cockpit switching, the PRCA positions itself to a preset ratio, locks up, and allows adequate control for safe flight and landing. In this configuration, the functions of the PCRA and ARI packages could literally be replaced by simple bellcranks.

In addition, there is dual trim mechanization which prevents runaway trim. Takeoff trim for pitch, roll, and yaw can be achieved through a single switch setting.

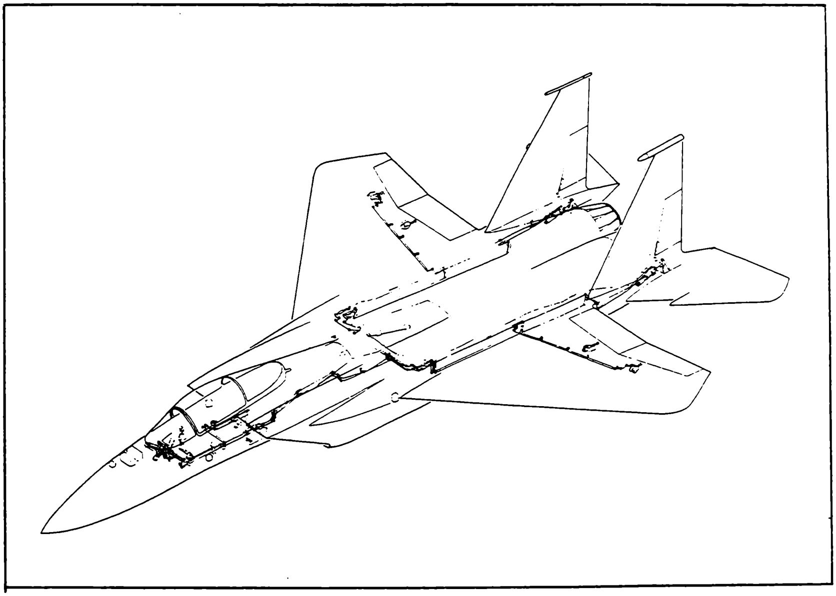
All control surfaces, including the stabilators, are balanced. Should control surface power be lost, or a mechanical disconnect occur, the surface will go to a trail position, permitting continued trim flight.

IN SUMMARY

We'll wrap up this introductory look at the Eagle hydro-mechanical control system by saying that the F-15 doesn't do anything by magic; you still have to pull on the pole to make the stabilator move. However, in the Eagle the distance the stabilator moves for a given input depends upon the PRCA. The same applies to the ailerons and rudders. If everything is operating normally you won't know just why, but you'll find that "it just feels good."

In future issues of the DIGEST, we'll get into the basic control system in more detail. In addition, we'll take a look at the Control Augmentation System (CAS), Stability Augmentation System (SAS), and Automatic Flight Control System (AFCS), and how they enhance and parallel the basic system. We believe that the Eagle has a good flight control system, and we hope these articles will help you understand why we feel this way.

Putting it all TOGETHER...



Now that you've had a chance to begin reviewing the engineering aspects of the F-15 flight control system, we'd like to remind you that there is an equally interesting article, aimed specifically at the aircrews, within our recently published Eagle Owner's Manual. In this article, Pete Garrison, Chief Experimental Test Pilot and Eagle Driver No. 2, passes on some first-hand cockpit views of what this system can do for you. To get a copy of this pilot-oriented booklet (which also discusses numerous other aspects of the F-15), ask your local McDonnell Field Service Engineer for publication P.S. 872, or write the Editor of the DIGEST direct. (Note: P.S. 872 was published as pages 24 - 47 of EAGLE TALK, Vol. I.)

(PUBLISHED 1975)

F-15 Flight Control System Part II

DIRECTIONAL CONTROL

By B.P. "PERRY" HOFFMAN/ *Senior Engineer, Flight Control Section, Avionics Engineering Laboratories*

Last issue we took a "Broad-brush" look at the overall F-15 hydro-mechanical control system; now let's get into the specifics of the directional control system. In future issues we'll review the longitudinal and lateral control systems as well as the impact of various electronic functions. Be on the lookout for each bi-monthly issue of the DIGEST so that you'll be able to get the full flight control story.

Directional control of the Eagle comes from two vertical fins and two synchronized rudder control surfaces (Figure 1). Conventional rudder pedals position the rudder control surfaces. All rudder pedal inputs go through the Aileron Rudder Interconnect (ARI) box, a part of the Control Stick Boost/Pitch Compensator. The ARI combines rudder pedal signals with functions of roll and pitch, providing turn coordination over a wide range of pitch and roll maneuvers.

Input authority to the rudder control surfaces in production F-15 aircraft is 15 degrees maximum. Lateral control stick inputs are scheduled within the ARI box for a maximum surface deflection between zero and 30 degrees depending upon longitudinal stick position.

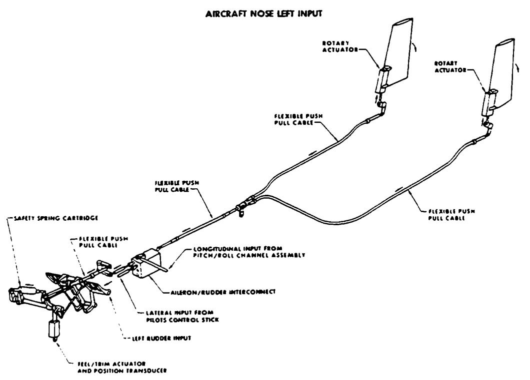
The ARI output is fed via flexible push-pull shafts to each of the rudder control surface actuators. The F-15 rudder power actuator is a part of the rudder hinge, allowing a smooth, streamlined surface with no linkage to wear or jam.

DIRECTIONAL TRIM

The F-15 feel trim actuator, forward and between the rudder pedals, receives its basic position signals from the rudder pedals through a common bellcrank and torque tube. The feel trim actuator establishes the neutral or zero force position of the rudder pedals by electrically extending or retracting the overall length of the actuator. Aircrew operation of the feel actuator is accomplished through actuation of the Yaw Trim switch, located just aft of the right-hand throttle. All trim circuitry is dual so that no single failure can result in runaway trim.

The Yaw Trim switch signal goes to the CAS roll/yaw computer where the switch commands are amplified by transistor relay drivers. This output is supplied to the yaw feel trim actuator, picking up relays within the actuator, powering its motor, and driving the actuator to a new position. Simultaneous with actuator travel, electrical signals are generated by a pair of Linear Voltage Differential Transform-

FIGURE 1 – FLIGHT CONTROL SYSTEM – DIRECTIONAL CONTROLS



ers (LVDT). These signals are fed back to the CAS roll/yaw computer and are used for three distinct functions. The signals:

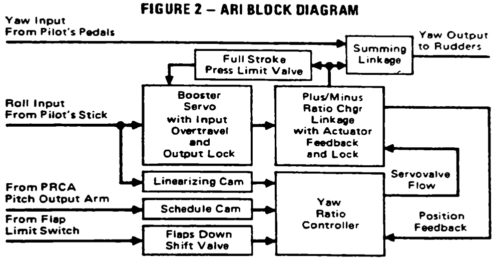
- Establish actuator neutral when the takeoff-trim button is held depressed.
- Limit actuator travel through use of voltage level detectors within the CAS roll/yaw computer to prevent driving the trim actuator into its mechanical stops.
- Advise the yaw CAS of a change in trim command so that the CAS doesn't defeat the pilot-inserted trim.

In addition to the trim LVDT's, another pair of LVDT's within the feel trim actuator measure deflection of the "feel" springs, and supply pedal force commands to the yaw CAS. Rudder deflections commanded by the CAS can add ±15 degrees with respect to the position held by the mechanical system, up to a combined maximum of ±30 degrees of rudder. The mechanical input pedal force per degree of rudder deflection amounts to 9.75 pounds on the pedal for each degree of rudder deflection. Twice as much rudder per pound force can be commanded when yaw CAS is engaged.

SAFETY SPRING CARTRIDGE – A safety spring cartridge is provided so that, in the event of jammed linkage, pedal forces can still be applied allowing CAS control of rudder operation. This provides an excellent "fly-by-wire" rudder system allowing safe return to the Eagle's nest. This also permits continued use of the nose-wheel steering system with a jammed rudder link. The same applies in the event of a linkage separation: CAS can again supply the pilot pedal commands to the rudder.

RUDDER PEDAL LIMITER – A rudder pedal limiter has recently been added to the rudder pedal torque tube. At Mach 1.5 or greater, a discrete signal from the left-hand air inlet controller actuates the pedal limiter actuator, physically restricting movement of the torque tube and pedals thus limiting rudder surface deflection to no more than 5 degrees. This prevents excessive rudder-induced rolls in a flight regime where roll/yaw coupling

FIGURE 2 – ARI BLOCK DIAGRAM



is a potential hazard. If the right-hand air inlet controller has not attained Mach 1.5 and the discrete signal has not been developed, or the limiter actuator has not extended to close the Maximum Extend Limit switch, a warning light illuminates to advise the pilot to use caution when operating the rudders.

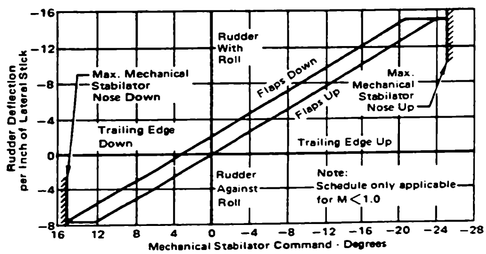
AILERON-RUDDER INTERCONNECT

The Aileron-Rudder Interconnect (ARI) box is the heart of the F-15 directional control system and is shown in simplified block diagram form in Figure 2.

Starting at the upper left corner of the diagram, yaw input from the pilot's pedals is fed directly into a summing linkage and out to the rudders. Since this function is a straight-through linkage arrangement with no hydraulic boost at the output, rudder linkage friction downstream of the ARI can adversely affect the ability of the rudder control surfaces to return to neutral when pedal forces are relaxed. This means that the maintenance man needs to eliminate all possible sources of friction within flex cables and bellcranks during any maintenance action. Roll input from the pilot's stick and pitch output from the Pitch/Roll Channel Assembly (PRCA) harmonize the rudder output through the Yaw Ratio Controller, the Plus/Minus Ratio Changer linkage, and the summing linkage.

The Flaps Down shift valve modifies the schedule allowing more rudder, sooner, with flaps down than is available through the flaps up schedule. Looking at the graph (Figure 3), note

FIGURE 3 – MECHANICAL LATERAL CONTROL - RUDDER INTERCONNECT
(Production Aircraft)



that with flaps *up* and 10 degrees nose-up stabilator, you can expect 6 degrees of rudder per inch of lateral stick (two inches of left stick equals 12 degrees of left rudder). Likewise, with flaps *down* and 8 degrees of nose-down stabilator, you'll get 3 degrees of rudder per inch of lateral stick (two inches of left stick equals 6 degrees of *right* rudder).

The Booster Servo at the roll input prevents rudder pedal commands from being fed back into the lateral control system. Coupled to the Booster Servo is the Full Stroke Pressure Limit valve which keeps the Booster from physically overloading the ARI structure as the ram reaches full stroke and simultaneous pedal inputs are applied.

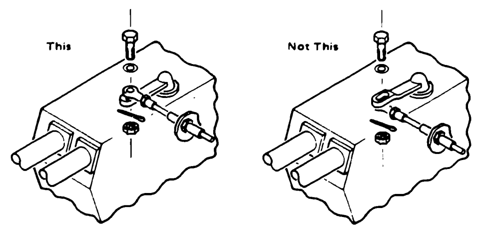
Since no ARI functions are required with the aircraft supersonic, a hydraulic shutoff valve located in the PRCA Pitch Ratio Controller turns off the supply pressure to the ARI unit when the aircraft reaches Mach 1. Rudder pedal commands are still available, as are the 15 degree CAS commands.

The Rapid Shutoff valve is actuated by the anti-skid wheel spin-up signal. Since we don't want the rudder to be controlled by lateral stick during cross-wind landings, lateral stick inputs to the rudder are turned off at ground-roll speeds of 50 knots or greater. The maintenance technician can duplicate this during preflights. While holding lateral and longitudinal stick inputs, note the rudder deflections, place the Anti-Skid switch to OFF, and the rudder should rapidly return to the trim position. Reselecting Anti-Skid should return the rudder to its deflected position within 25 to 35 seconds. You can get the same results by turning the Roll or Pitch Ratio switches to EMERGENCY. ARI will shut down, neutralizing the rudder, and the rudder will return to its deflected position when the Ratio switch is returned to the AUTO position.

A recent addition to the ARI is the Rapid Warm-up valve contained in the −17 ARI box installed in all production F-15 aircraft. The need for a reduction in the time required to attain ARI operation became apparent during the first winter operation of the system in St. Louis. On several occasions aborts and near-aborts were blamed on poor or missing ARI response to stick inputs. A number of corrective schemes were devised, including cycling of the stick several times and cycling of the ramps to get Utility hydraulic oil moving and heated up, but none of these could be relied upon to be effective. The manufacturer of the ARI box, Moog, Inc., then devised a thermo valve to bypass the hydraulic supply until the oil temperature reached about 140°F and installed it within the ARI unit. Once past 140°F, the valve opens fully.

An additional thermo bypass valve has been installed in the aircraft Utility hydraulic supply just prior to entering the PRCA to help speed the warming of oil during aircraft engine operation (this thermo bypass valve does not apply during external power operation). Though pilots or maintenance personnel will see little change in PRCA or ARI operation, there will be some increase in ARI turn-on time, as well as a definite increase in noise levels. A rather loud, low-pitched noise may be heard due to the input reducing pressure valve slowly building up to rated pressure. Though this may take up to 30 to 35 seconds, don't worry about it; it's normal, and the noise will go away.

FIGURE 4 – FLEX SHAFT RIGGING ON ARI



MECHANIZING THE ARI OUTPUT

At the output of the ARI, mechanical push-rod linkages are replaced with a flat steel ribbon riding on steel balls within a flexible housing. You may be familiar with similar cables used in some throttle systems. This unique system of linkage connects the single output of the ARI individually to each of the rudder actuator control valves.

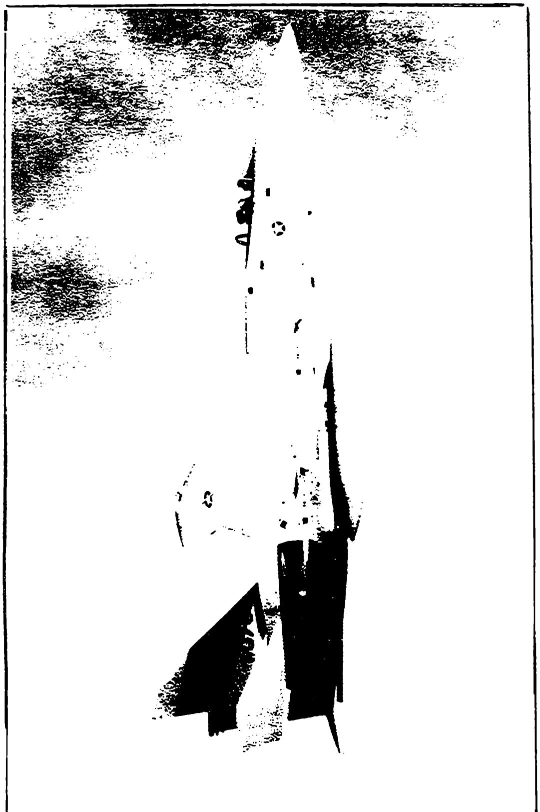
The control valves, operating from Utility hydraulic power, provide a rotary motion rather than the conventional extension or retraction of a hydraulic ram. Because of this design, the actuator requires no additional bellcranks or other linkages to change motion direction. Each actuator forms an integral part of the rudder hinge; the actuator is a self-contained unit and positions the rudder control surface in response to pilot pedal commands through the ARI, and electrical commands from the automatic flight control system (CAS).

If input linkage separation should occur, a pair of centering springs will return the input control valve to detent but yaw CAS commands of up to 15 degrees may be initiated through pilot pedal inputs allowing safe flight home. Electrical commands from the yaw CAS are received by an electro-hydraulic servo valve which in turn ports hydraulic pressure to the CAS piston. The CAS piston repositions the actuator control valve sleeve with resultant main actuator motion. Electrical feedback signals are generated by the CAS actuator Linear Voltage Differential Transformer, establishing a position authority on the surface. If a fault occurs anywhere in the CAS system that results in one rudder moving 3 degrees more than the other, yaw (and roll) CAS will automatically disengage.

MAINTENANCE CONSIDERATIONS

Conventional maintenance procedures apply to the directional control system, these are covered in the technical order. However, flexible shafts and cables require special care.

- Be especially careful when working with the flex shaft transmitting pitch information between the PRCA and ARI. Kinks or bends in this shaft cannot be tolerated and are cause for cable replacement.
- Be sure that the rod-end (bearing) is connected on the top of the ARI input arm; if it is connected to the bottom of the arm, interference with the ARI is likely and cable kinks will occur (see Figure 4).
- Be careful when handling flexible cabling; don't bend it too much or twist it any more than necessary. When connecting cables to bellcranks, strive for the best alignment possible by juggling clamps where necessary. On later delivered airplanes (beginning at about airplane 51), special adjustable support brackets will be available to assist in careful cable alignment. With cautious handling and installation, linkage friction and rudder surface hang-up will be minimized. ■



High thrust-to-weight ratio provided by F100 engines permits some spectacular combat maneuvering, such as this vertical climb by F-15A 77-070.

(PUBLISHED 1975)

F-15
Flight Control System
Part III
LATERAL CONTROL

By B.P. "PERRY" HOFFMAN/ *Senior Engineer, Flight Control Section, Avionics Engineering Laboratories*

Our last article dealt with the F-15 directional control system; now let's dig a bit deeper, progressing to the lateral control system. Lateral (or roll) control in the Eagle is obtained from simultaneous deflection of conventional ailerons located on the outboard section of each wing and differential stabilators. The amount of aileron/differential stabilator deflection per inch of lateral stick movement is controlled by the Pitch Roll Channel Assembly (PRCA), with scheduling based on both the output of the PRCA pitch boost servo (longitudinal stick position) and airspeed. The net effect is a proper blend of control deflections required for maneuvering throughout the aircraft envelope, and yet the pilot is given approximately the same feel no matter what the flight condition might be. Let's see how some of these requirements are mechanized.

LATERAL TRIM

Referring to Figure 1, follow the lateral linkage from the control stick to the lateral feel trim actuator. Note that the actuator is mounted in parallel with the overall control linkage. This is just a simple way to say that the system linkages are not shortened or lengthened; the trim actuator merely moves the total system.

The feel trim actuator performs two equally important tasks: it establishes the zero force position of the control stick and provides the pilot with an artificial feeling of maneuvering stick force. The zero force or "hands-off-stick" position may be varied as the pilot requires by activation of the stick grip button. The trim motor may also be repositioned through operation of the takeoff trim button which drives the actuator to a preset neutral position, streamlining the control surfaces.

Simultaneous with actuator travel, electrical signals are generated by a pair of linear voltage differential transformers (LVDT). These signals are used by the Control Augmentation System (CAS) computers, where they are compared in a preset voltage level detector which turns the actuator off when the proper level is reached. Another level detector stops the trim actuator at neutral if the takeoff trim button is held depressed.

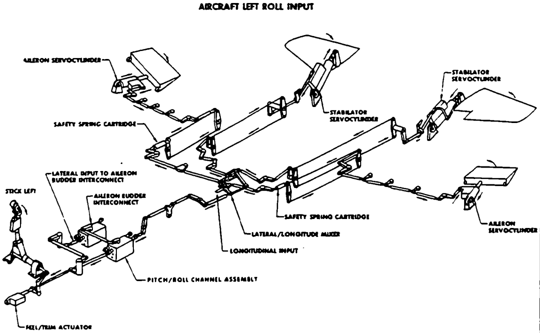
Outputs from these level detectors are supplied to the takeoff trim indicator light logic in the CAS computers. When this logic sees the same voltage level from all three channels, (roll, pitch, and yaw), the takeoff trim light illuminates, indicating to the pilot that his surface controls are properly positioned for takeoff. These LVDT signals serve yet another function in advising the CAS roll channel of changes in trim commands so that the CAS doesn't defeat pilot-inserted trim.

Lateral artificial feel force is provided to the pilot by dual spring gradients within the actuator. For the first inch of lateral stick travel, the force is 5 pounds (plus a 1.0 pound breakout); the gradient then drops to 3.67 pounds per inch of additional stick deflection. The dual spring gradient helps reduce lateral stick sensitivity around neutral. The LVDT signals and CAS circuits are dual redundant, providing a fail-safe operation in which the system shuts down to prevent runaway trim.

ROLL LINKAGE

As it leaves the lateral feel trim actuator, the linkage takes two paths. The first path travels to the ARI on the

FIGURE 1 FLIGHT CONTROL SYSTEM – LATERAL CONTROLS



right-hand side of the airframe. As we discussed in Part II of this series, this input supplies the lateral intelligence to the ARI. The second path continues down the left side of the aircraft to the PRCA which is the "brains" of the F-15 mechanical control system. Figure 2 is a block diagram of the PRCA and shows the data flow within the roll channel of the PRCA.

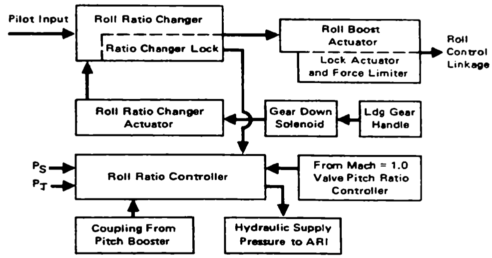
Roll Ratio Changer — The roll ratio changer, within the PRCA, contains the dual mechanical linkage required to vary the stick-to-aileron/differential stabilators gearing at a ratio of 4:1. Figure 3 explains how a parallelogram ratio changer does its work.

The dotted lever 1 pivot D is fixed to the PRCA frame while its pivot E varies with the position of the roll ratio changer actuator. Lever 2 has its pivot C fixed to the PRCA frame while pivot A attaches levers 2 and 3 together.

Diagram I shows the ratio changer actuator at maximum ratio as indicated by distances D to A and E to C being identical. Pilot stick inputs to point A displaces the output B by the same amount; that is, a 1:1 ratio. In diagram II, note that the ratio changer has been fully extended, placing pivots E and C over one another. Stick inputs to point A can rotate the linkages about E and C with only a small amount of output displacement for a ratio of 4:1. Diagram III shows an intermediate ratio. In this case, pivot E of the ratio changer actuator can be called upon to vary the ratios as dictated by the air data information fed to it, or by longitudinal control system position.

Presuming you've digested at least a part of that, let's press on. Within the ratio changer section, you'll find a ratio lock. This drives the ratio changer mechanism to the failed ratio in event of a loss of hydraulic supply pressure. In addition, the pilot may select the emergency mode to isolate a suspected malfunction. He does so by placing the Roll Ratio switch to EMERG(ency) which removes hydraulic pressure to the roll ratio channel of the PRCA. The roll ratio repositions itself to about one-half ratio in emergency, or 10° of aileron plus 3° differential stabilator which is more than adequate for normal flight and safe return to base. During emergency operation of the roll channel of the PRCA, the Master Caution light and Roll Ratio telepanel light will illuminate warning the pilot of a problem. Lateral control stick inputs into the PRCA during the transition time are quite heavy

FIGURE 2 PRCA – ROLL CHANNEL

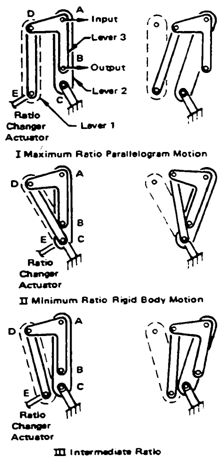


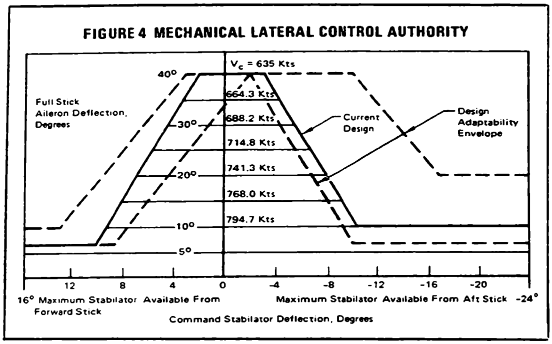
since the actuator is in the process of locking. However, after the short time required to lock, lateral control stick forces settle down to about twice that of a normal operating system. When the Roll Ratio switch is again placed in AUTO(matic), and the hydraulic supply pressure is available, normal system operation is restored.

Roll Ratio Controller/Roll Ratio Changer Actuator — The ratio controller and ratio changer actuator may be considered at the same time since the actuator simply provides the muscle for the ratio controller. The ratio controller receives pitot (Pt) and static (Ps) air inputs from the left hand probe. A cam-operated servo-mechanism controls hydraulic pressure to the roll ratio actuator, repositioning the ratio changer linkage to a new value. Figure 4 illustrates that both air data and longitudinal position affect the ratio controller. Longitudinal stick inputs to the roll ratio controller are the result of mechanical coupling to the roll ratio controller shaft from the PRCA pitch channel boost actuator. The combination of the air data and longitudinal inputs reposition the ratio changer and vary the control stick-to-aileron differential stabilator gearing.

The only situation in which the ratio controller cannot command the ratio changer actuator to move is when the landing gear handle is positioned to extend the gear. During the early days of F-15 flying, it became apparent that during crosswind landings more than the available roll power was needed to keep the upwind wing from rising. As previously stated (and shown in Figure 4), when the aircraft is slowed to land and the stick is either trimmed or held aft, roll power is "washed out" (the amount of aileron available with full stick is reduced). The problem was resolved by adding a solenoid valve to the ratio changer actuator which drives the ratio changer actuator to maximum ratio when the landing gear handle is placed in the down position. All production PRCA's have this feature so it is not possible to check aileron/differential stabilator washout on the ground without putting the gear handle up (with hydraulic pressure

FIGURE 3 SIMPLIFIED RATIO CHANGER DIAGRAM





applied, this just "ain't" a good idea).

A suitable ground check may be made by pulling the PRAD CONT (Pitch and Roll Adjust Device Control) circuit breaker. This circuit breaker removes dc power from the gear down solenoid and aileron washout may then be checked. With the stick at takeoff trim, apply full left roll deflection and note the position of the ailerons. While holding full left stick, slowly pull the stick aft, noting that the ailerons will begin to return to streamline stopping at about a 3 to 5 degree deflection as the longitudinal stick reaches the ¾ aft travel point. Returning the stick to neutral in pitch causes the aileron deflection to increase again to maximum. The same conditions may be seen for a right stick and for either forward or aft pitch inputs. Resetting the PRAD CONT circuit breaker removes the aileron washout function.

Another signal to the roll ratio controller is a hydraulic input from the pitch ratio controller "Mach = 1.0" sensor. The roll ratio controller contains a hydraulic shutoff valve which controls the hydraulic supply pressure to the ARI. This is the switching intelligence for turning off the mechanical ARI above Mach 1.0.

Roll Booster — The last major component in the PRCA roll channel is the roll boost actuator. The booster control valve is coupled directly to the output of the ratio changer and the valve directs hydraulic pressure to a conventional power cylinder to drive all the downstream linkages external to the PRCA.

The boost actuator has two purposes. First, it prevents any of the downstream linkage friction from being felt at the control stick. Second, the actuator output force is sufficient to provide chip and foreign object shearing forces. In event of a hydraulic failure, or if the pilot selects emergency operation, the boost actuator control valve input arm locks at neutral. Both sides of the boost piston are ported to return pressure, and the actuator functions as a fixed link. Pilot inputs must then physically move the actuator piston as well as all the downstream linkage. This is why the stick forces become a bit higher during emergency operation.

MECHANIZING THE PRCA ROLL OUTPUTS

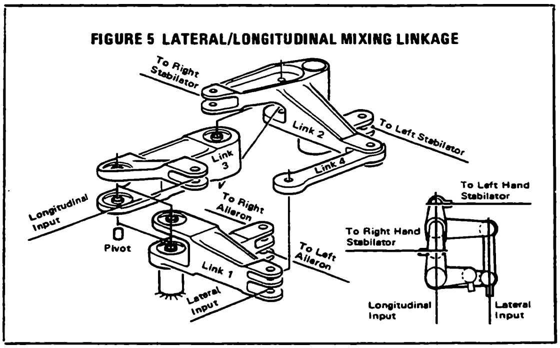
The output shaft of the roll booster actuator carries the modified lateral commands of the control stick through a conventional system of push rods and bellcranks to the next major component, the lateral/longitudinal mixing linkage.

Mixing Linkage — The mixing linkage receives both lateral and longitudinal control stick inputs, decides which control surface is supposed to move, and pulls or pushes the appropriate control rod to deflect the surface. Figure 5 shows an expanded layout of the mixer which fits together in the shape of a parallelogram. Referring to the expanded view, a lateral input deflects link 1 pushing one aileron rod while pulling the other. At the same time link 2, connected to link 1 by link 4, rotates, deflecting the stabilators differentially. A longitudinal stick input to link 3 rotates it, pulling link 2 which pulls or pushes both stabilator rods, giving collective stabilator.

A long, detailed explanation should not be necessary if you keep in mind that during *lateral* inputs all links move as a unit, rotating about the pivot. During *longitudinal* inputs, link 1 remains fixed and link 2 moves back and forth rotating about links 3 and 4.

Two Linkage Paths — From the mixing linkage there are again two linkage paths.

- The aileron path utilizes push rod linkage to the lateral safety spring cartridge. The safety spring cartridge is connected in series with the lateral control linkage and allows the other aileron (plus differential stabilators) to continue functioning even though the linkage in one wing is hopelessly jammed. The safety spring cartridge is attached to a system of bellcranks and cables, carrying the lateral command to the wing root area. Push rods and idler bellcranks then carry the command to the aileron power cylinder



control valve which ports hydraulic pressure to a single system actuator, deflecting the control surface.

The aileron power cylinder is a bit different than those used on previous McDonnell-built aircraft. The actuator body is fixed to the airframe and the linear operating ram is attached to the control surface. A mechanical feedback arm is connected between the ram and control valve to stop actuator travel when the input command is satisfied. In the event of total hydraulic loss to either actuator (Power Control is primary and the switching valve does not switch in Utility backup), the actuator contains internal valving which enables it to revert to an aileron damper.

- The differential stabilator path is quite similar to the aileron path, again utilizing a bellcrank/steel cable arrangement to carry the command to the aft torque tubes in each tail boom. The aft torque tube motion displaces each stabilator actuator control valve the prescribed amount and direction to cause differential deflections of the stabilator control surface.

The stabilator actuators, though more complex, are similar in design to the aileron power cylinders and are dual systems, containing CAS actuators. The stabilator power cylinders will be discussed in considerably more detail in the next article in this series, which will focus on the longitudinal control system.

(PUBLISHED 1976)

Mackay Trophy to Streak Eagle Pilots

Three happy Air Force pilots, Lt Col Roger J. Smith, Maj David W. Peterson, and Maj Willard R. Macfarlane (left to right in the photo) are shown after having been awarded the 1974 Mackay Trophy. As members of the F-15 Joint Test Force at Edwards AFB, California, the pilots broke all existing time-to-climb records earlier this year during the F-15 "Streak Eagle" program.

The Mackay Trophy is awarded annually to Air Force personnel who have made the most meritorious flight of the year. The trophy was presented to the Streak Eagle pilots by Gen David C. Jones, Air Force Chief of Staff, on November 24, 1975.

(PUBLISHED 1975)

F-15 Flight Control System Part IV

LONGITUDINAL CONTROL

By B. P. "PERRY" HOFFMAN/*Senior Engineer, Flight Control Section, Avionics Engineering Laboratories*

In the preceding article of this series, we discussed how the Pitch Roll Channel Assembly (PRCA) roll channel adjusts the ratios or gearing between the pilot's control stick and the ailerons. The pitch portion of the PRCA is in many ways identical, but is somewhat more complex. This complexity comes through use of a PRCA device called the Pitch Trim Controller (PTC), which automatically adjusts the longitudinal trim to maintain a constant pilot-selected load factor. We'll cover the PTC in some detail later, but meanwhile let's take a general look at the Longitudinal Control System (Figure 1).

PITCH FEEL TRIM ACTUATOR

Beginning at the extreme left of the diagram, the first major component affecting control stick operation is the pitch feel trim actuator. It is designed to be in parallel with the total longitudinal linkage.

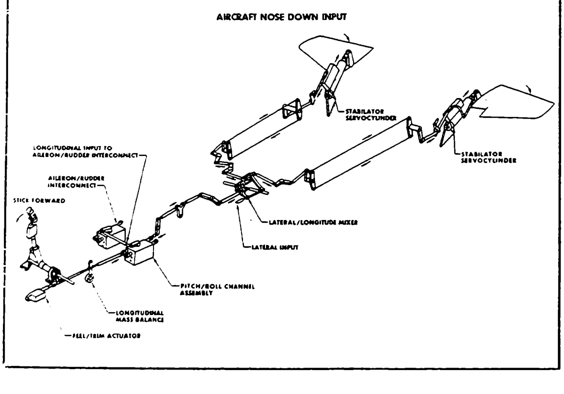
The zero-force or hands-off stick position is varied when the pilot presses the pitch trim switch on his stick. The pitch trim actuator moves the trim position of the control stick and linkage to satisfy pilot requirements. The only force the pilot normally feels when he moves the stick is generated by a dual-spring cartridge which is part of this trim actuator. These dual springs give the stick a higher force per inch displacement near the trim position and a reduced force per inch for larger stick inputs. This reduces the force a pilot has to hold for sustained high g maneuvers.

Electrical commands from the control stick trim switch or the takeoff trim button extend or retract the trim actuator. Linear Voltage Differential Transformers (LVDT) are mounted on the actuator and generate electrical signals which advise the CAS computer of the new trim position so that the CAS will not try to defeat the pilot-desired trim change. These same signals are also used by preset level detectors which shut the trim actuator off before the actuator reaches its mechanical limits.

When the takeoff trim button is depressed, the trim actuator drives to a pre-determined position dictated by preset detectors within the CAS computer. The electrical sensors and components associated with the trim system are dual so that a loss of any one element causes the trim to shut down, preventing a runaway trim condition for a single failure.

Moving aft from the trim actuator, there is a lead weight on an idler arm. This is not a "bobweight" for generating stick force per g, as is the case in the F-4, but is there simply to balance the control system so that sudden acceleration or deceleration of the aircraft does not produce stick motion.

FIGURE 1 FLIGHT CONTROL SYSTEM - LONGITUDINAL CONTROLS



PRCA

To begin with, let's break the PRCA pitch channel into its components (Figure 2) and see how each affects the longitudinal controls. The input rod offers two linkage paths with the main path tying directly into the ratio changer linkage, and a second input being supplied to the Load Factor Sensor portion of the Pitch Trim Controller.

Pitch Ratio Changer — The pitch ratio changer utilizes dual redundant parallelogram linkage identical in operation to the roll ratio changer described in our last DIGEST article on lateral controls. The only operational difference is that the pitch ratio changer gearing utilizes a 6:1 ratio, where the roll ratio changer utilizes a 4:1 ratio.

Pitot (Pt) and static (Ps) information is supplied from the left-hand probe to the ratio controller bellows assembly, repositioning a cam-operated valve supplying hydraulic pressure to the ratio changer actuator. The ratio changer actuator then drives the changer linkage, varying the stick-to-stabilator gearing as required.

If hydraulics are lost to the PRCA, or if the pilot selects the EMERG position of the pitch ratio changer by actuation of the Pitch Ratio switch, the pitch ratio changer will drive to its failed position. In the failed mode, the gearing ratio is one-half of its maximum value and all other functions (PTC and Boost) are inoperative.

Two additional functions are associated with the ratio changer (one is no longer used, but a note of explanation is in order in case you see it on a schematic diagram). Early in the program there was a gear down valve which drove the pitch ratio changer to maximum (for landing control) when the nose gear proximity switch actuated. The disadvantages seemed to outweigh the advantages, so the valve has been deactivated on current Eagles and the valve will not be in future PRCA's. For the maintenance man, this is a great help since he can simply place the Pitch Ratio switch to emergency and check the fail mode without having to simulate a gear up condition.

The second function is a safety feature controlled by the pitch ratio changer actuator. If a hardover failure of the pitch trim controller occurs at low pitch ratios, the pitch CAS is inoperative, longitudinal control could be lost. To guard against this possibility, a valve within the ratio changer actuator is opened when the actuator approaches the minimum ratio position. This valve supplies hydraulic pressure to an interlock piston within the pitch trim compensator, keeping it at a position where adequate control is always available.

The output of the pitch ratio changer is fed to the pitch boost actuator servo valve. The boost actuator output is mechanically linked back to the servo valve input, closing the loop. The boost actuator output also feeds the downstream linkage, which deflects the stabilators and drives the linkage scheduling the roll ratio controller and the ARI. As you will recall from our previous articles, the pitch output is used in roll to schedule the roll ratio as a function of angle of attack. Yaw is affected by this also, since yaw gain increases as roll gain decreases. *No more rudder rolls, guys; all you need for high angle of attack maneuvering is a lateral input from the stick.*

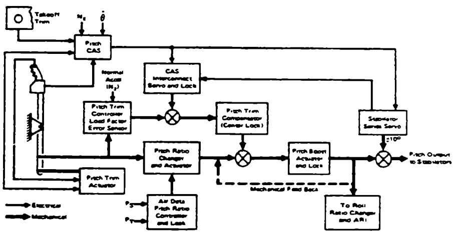
The pitch trim compensator is mechanically coupled to the pitch boost actuator control valve. The motion of this device adds (or subtracts) boost actuator displacement (pitch output to stabilators) to what the ratio changer output position is commanding. The action of the pitch trim compensator is controlled by the pitch trim controller.

Pitch Trim Controller — The pitch trim controller adds or subtracts stabilator deflection to compensate for such aircraft variables as increases or decreases in airspeed, speedbrake/flap extensions, or changes in loading. These changes are introduced without movement of the control stick. These combined PTC and pitch ratio changer outputs result in a nearly constant stick force and stick position per g within the Eagle's operating envelope (about four pounds of stick force results in one-half inch of stick displacement, producing a load factor of one g). The name given this feature is "series trim." The concept is not new; it's been tried on several previously-built aircraft. The uniqueness of this feature in the F-15 lies in the fact that it works, and does so without electrical inputs.

Stick inputs are fed into a high-class, accelerometer-controlled servo loop known as the load factor error sensor (LOFES), a part of the PTC. This stick input establishes the neutral or zero point it works around. For example, let us assume that a pilot, or the trim actuator, is holding the stick in a position commanding a load factor of one g. Any subsequent deviation from that setting will be sensed by the PTC accelerometer which will valve hydraulic pressure to the pitch trim compensator piston, repositioning the piston and commanding the required amount of collective stabilator to keep the aircraft at one g. This series trimming capability is true for disturbances created by flap, speedbrake, and landing gear extensions. Acceleration and deceleration are also compensated for, producing an essentially neutral speed stable airframe. Since the trim change we've described is "series," no stick movement is noted.

A rapid warm-up valve has recently been added to the PTC. Hydromechanical devices such as the PTC utilize extremely close tolerances in their construction and don't want to work well when the surrounding metal and hydraulic oil are cold. Because of this, a viscosity sensing bridge coupled to the PTC pressure inlet bypasses the input hydraulic oil through a small orifice which speeds the warm-up process and allows normal control operation sooner. All production PRCA's contain the warm-up fix, as do most flight test units.

FIGURE 2 PITCH CONTROL SYSTEM



LINKAGE

Control rods carry the PRCA output to the mixer linkage. The mixer linkage combines the pitch command with the roll system signals to drive the stabilator (a diagram of the mixer linkage was presented in the article about roll control in the last issue). Then a system of dual cables and bellcranks extends along the major length of the fuselage to the aft torque tube. The aft torque tube carries the pitch commands directly to both stabilator control valves, causing actuator displacement and resultant stabilator deflection. ▸

These cables may be a prime contributor to flight squawks such as "too much stick motion with no aircraft response," or "soggy longitudinal controls." Maintenance personnel should pay special attention to cable tensions. If readjustments are necessary, be sure that they are made to both cables to keep the end bellcranks parallel (the rigging pin can be installed in both ends). Refer to Technical Order 1F-15A-2-5 for the final word.

STABILATOR POWER CYLINDER

The final components of the pitch channel are the two stabilator power cylinders and if they don't work, it's a long walk home. A lot of thought went into the Eagle actuators; they obviously had to have considerable muscle to move the large control surfaces under the large air loads imposed. This part is relatively easy since you just make the piston bigger (more surface area times three thousand pounds of hydraulics equals horsepower).

We have, however, deviated some from traditional actuator design. The normal method has been to allow the total actuator barrel to move with commands until the barrel position matches the commanded valve position, stopping actuator motion. In the F-15, the actuator is firmly attached to the airframe and the ram is attached to the stabilator. A mechanical feedback linkage is then employed to shut off the input valve when the actuator has reached a new position. This method is deemed best from a structural standpoint and decreases the total mass which must move.

The actuator contains two identical pistons powered by separate hydraulic systems. Either of these systems can adequately control the stabilators. In the event of two hydraulic system failures, one-half the actuator remains functional with get-home-safe capability because the Utility hydraulic system automatically switches into one-half of the actuator if the PC system normally supplying that side is lost.

Finally, the pivot bearing location for the stabilator surface was selected to allow it to trail under total actuator failure. In other words, the F-15 actuator will not go hardover as was the case in older aircraft. Because of this, one power cylinder could fail but you could still fly home and land.

FLY-BY-WIRE CAPABILITY

Going back to the beginning of the pitch system, the pitch trim actuator is physically located as close as possible to the control stick. This permits the pilot to "fly by wire" with normal stick feel, using the CAS stick force sensor should the mechanical linkage between the stick and stabilator actuators break or disconnect.

Snicker if you must, but an early Eagle was flown back to the nest and landed safely with a total mechanical disconnect of the longitudinal controls. Control stick feel to the pilot was normal despite a control rod having been completely severed by an ECS turbine blade.

To allow this fly-by-wire capability, a centering, or detent, spring was added to the input valve area. If a linkage disconnect occurs ahead of the power cylinder, the valve centers itself. At this point, the CAS commands the actuator travel within its authority of plus or minus ten degrees of stabilator travel, enough to get home safely. Jammed linkage poses a different problem since the CAS must work around the jammed valve position, but even this is not impossible.

Our next article will pick up where we've left off here; we'll look into the overall CAS functions in the F-15. ●

(PUBLISHED 1976)

F-15 Flight Control System Part V

YAW and ROLL CONTROL AUGMENTATION

By PERRY HOFFMAN/ *Senior Engineer, Flight Control Section, Avionics Engineering Laboratories*

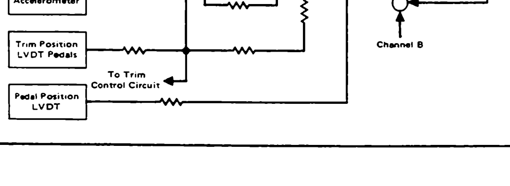
Having taken a look at the mechanical aspects of the F-15 Flight Control System, let's turn our attention to the *electronic* portion, the Control Augmentation System. Possibly the most frequently asked questions are: "Just what is the Control Augmentation System?" "What does it do for me?" "How does it do it?"

The Control Augmentation System (CAS) consists of two distinct functions. The first is our old friend, the Stability Augmentation System, otherwise known as Stab Aug or SAS. For those old-timers who can remember far enough back, this used to be called a Damper. The Stab-Aug, or Damper, portion of the F-15 CAS is designed to help stabilize the airframe, compensating for unwanted motion which might occur as a result of wind gusts or disturbances.

The second CAS function is its Control Stick Steering mode. This measures, compares, shapes, and smooths out pilot stick and pedal inputs allowing precise and comfortable control throughout the maneuvering envelope.

Why is CAS desirable? Well, we know that the F-15 airframe is basically stable, and that the manual flight controls are designed to give Level II handling without augmentation. (Military Specification MIL-F-8785B defines Level II handling as "flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.") Despite the basic stability of the Eagle, various flight conditions and varieties of store loadings could result in some pretty touchy handling situations were it not for the CAS. In addition, the CAS provides safe control of the aircraft should the basic mechanical system suffer failure or battle damage such as foreign object jams or shot-away linkage.

FIGURE 1. YAW CAS (SINGLE CHANNEL)



The bulk of this article will be concentrated upon the yaw and roll CAS (you'll see, shortly, that these two functions can't be separated). The pitch CAS will be the subject of a future article.

YAW CAS

Sideslip Control and Damper - Precise sideslip control is provided during maneuvering by the yaw CAS. Referring to the yaw channel block diagram (Figure 1), you'll see a Rudder Pedal Position Linear Voltage Differential Transformer (LVDT). As the pilot applies force to the pedals, the mechanical system begins to deflect the rudders. At the same time, the pedal position LVDT generates an electrical signal. As the aircraft responds to the pedal input, the yaw rate gyro and the lateral accelerometer will sense this motion. Their blended signals are compared to the pedal position LVDT signal within the Roll/Yaw Computer and the resultant signal output either adds or subtracts rudder control surface deflection as needed to obtain the proper response. Likewise, this combination of signals directs the servo amplifiers to deflect the rudder power cylinder and rudder control surfaces the correct amount and direction to dampen any unwanted yaw disturbance.

Elimination of Steady-State Sideslip - A circuit was added to reduce uncommanded and persistent sideslip (due primarily to rudder linkage friction or hysteresis) during supersonic flight and following a maneuver. The combined lateral acceleration and yaw rate sensor feedback voltages are compared with the rudder trim position and the combined output is used by the Proportional plus Integral (P + I) circuit to apply rudder surface deflection in a direction to eliminate the sideslip. With a maximum authority of ±3.75 degrees rudder, pilots can expect the ball to be pretty well neutral during flight.

There is, however, one disadvantage to the P + I circuit. If rudder surface hangup exceeds the P + I authority, yaw CAS cannot automatically trim the aircraft to zero sideslip. Thus, some manual pedal trim may be required to make up the difference, reducing sideslip to minimum. Manual trim should be applied slowly, or in small amounts with waiting periods, since the new pedal LVDT position affects the integration of the P + I circuit. If this procedure is not followed, it may appear to the pilot that he is chasing the trim.

Pilots can expect to see yaw trim changes varying in magnitude with different aircraft anytime the landing gear is extended or yaw CAS is disengaged. Both of these actions drive the P + I integrator to zero, introducing a yaw transient. So we may have solved the supersonic sideslip problem but created a problem at yaw CAS shutdown.

Maintenance personnel will have to locate and reduce system friction to a minimum. A good source of system friction can be found in the flexible cables. Here are a few points to keep in mind:

- Any kinks or rough spots are cause for cable replacement.
- At any attach point such as bellcranks or ARI output rods, attempt to line up the cable end exactly with its attach point. In other words, reduce any apparent side load; the nature of the ribbon cable is to increase friction loads as side force is exerted on the ends.
- Make changes in direction with as large a radius as possible with only minor twists.

To summarize, problems will go away when mechanical rudder linkage is kept friction-free.

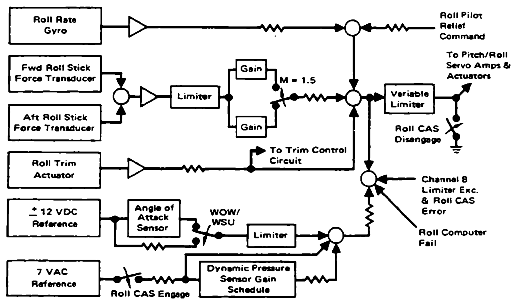
Failure Detection and Shutdown - All three CAS channels utilize a dual-channel "Fail-Off" system. There are several things that would cause the failure detection circuitry to shut down CAS: if spin is evident (yaw rate exceeds 41.5 degrees/second); a malfunction unbalances the yaw computation circuits; an imbalance between the rudder actuators; and failure of the rudder actuator shut-off valves. Let's amplify a bit on these situations -

- If CAS is causing or aggravating the spin mode, we want CAS off. Therefore, any yaw rate in excess of 41.5 degrees/second will cause yaw CAS to shut down. Roll shuts down as a result of yaw shutting down and pitch follows if the high yaw rate continues for a period of longer than 120 milliseconds.
- Yaw CAS will shut down for malfunctions which unbalance the yaw computation circuits. This may result from differences in one of the dual yaw rate gyros or lateral accelerometer output voltages which exceed a preset level. The system will also be shut down by an electronic failure within the yaw computation circuits.
- A shutdown of yaw CAS will occur if a problem in the system causes one rudder hydraulic actuator to mistrack the other by approximately four degrees for a period of one second or longer.
- The final cause for shutdowns applies to production computers Part Number 275E514G3 which will be installed effective F-15 ship 61 and TF-15 ship 14. Circuitry has been added to these computers which senses open wiring to the rudder surface actuator shutoff valve or a failure of the actuator shutoff valve itself. In the older G2 computers, an open in shutoff valve wiring renders the yaw CAS inoperative; however, the roll CAS will not be shut down and the roll and yaw CAS telepanel lights remain out. Because of the inoperative shutoff valve, electrical signals from the CAS will not affect the rudders; the only rudder movement will come from mechanical inputs. Since the rudders will not mistrack in this mode, there will be no shutdown.

During preflights (maintenance and aircrew), ground personnel should double check to insure that yaw CAS inserts an additional 50 percent (or 15 degrees) of rudder surface deflection, making a total surface movement of 30 degrees (these figures are approximate).

Aileron Rudder Interconnect - In order to improve turn coordination, roll rate signals are applied to the yaw channel. Since greater rudder deflection is required for turn coordination

FIGURE 2. ROLL CAS (SINGLE CHANNEL)



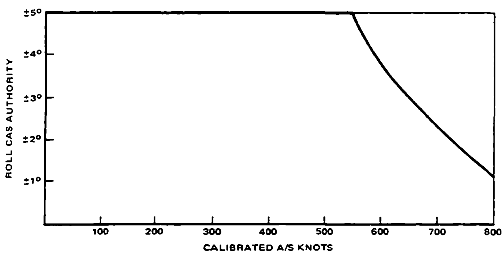
at high angles of attack, this roll rate signal is scheduled with angle of attack, increasing rudder deflection as angle of attack is increased. To minimize roll and yaw coupling tendencies, ARI is defeated at Mach numbers above 1.5 and for negative angles of attack by driving the ARI signal to zero. The ARI signal is also driven to zero with wheel spin up. This is one of the aids for better control during crosswind landings. With ARI operational during a nose-high rollout, and with lateral stick held to lower the upwind wing, ARI would add rudder in a direction to drive the nose into the wind.

Control Surface Actuators - The muscle for the Eagle's rudders is not the conventional linear ram-type actuator normally seen on aircraft flight control systems. To minimize space requirements, a rotary actuator was designed which is an integral part of the rudder hinge line. Not only does it receive manual inputs, but also electrical inputs from the servo amplifier. These signals control an internal piston which deflects the rudder control surface upon CAS command. The rudder actuator is also load-limited so that as inflight loads increase the actuator deflection is decreased accordingly, reducing unwanted tail loads. In the event of hydraulic pressure failure to either actuator, bypass valves prevent oil from escaping into the return line. In this condition, the rotary actuator becomes a self-contained surface damper.

Problem Areas - Yaw CAS (as well as CAS in general) has an excellent record of reliability. Once "infant mortality" rids the computers of problem components, the next malfunction is pretty far down the road. During production flights at St. Louis, we have replaced two rate gyro packages - one had a loose connection in an individual gyro, the other was out of tolerance gradient-wise. No accelerometer sensors have been replaced to date.

Some difficulty may be experienced when yaw CAS is initially turned on - the first pedal application may result in a yaw shutdown. The reason for this involves the locking rings which hold the rudder actuator CAS rams at zero while CAS is disengaged. During yaw CAS turn-on, these rings must move to unlock the CAS ram. Sometimes the unlock ring of one rudder actuator lags the ring in the other actuator. As rudder pedal force electrical signals attempt to move the rudders, one rudder CAS ram encounters the not-fully unlocked ring. This results in a slight lag in one of the rudders. If this lag exceeds four degrees, the failure detection circuits will be triggered. The corrective action, in this case, is to reset yaw CAS and you should be back in business with normal operation.

FIGURE 3. ROLL CAS AUTHORITY vs AIRSPEED



ROLL CAS

Roll CAS (Figure 2) provides stability augmentation (or roll damping) as well as supplying the maneuvering capabilities to satisfy Level I lateral control requirements throughout the F-15 envelope (Mil Spec MIL-F-8785B defines Level I as "Flying qualities clearly adequate for the mission Flight Phase.) Short period roll oscillations and aerodynamic disturbances are sensed by dual roll rate gyros whose outputs are shaped and amplified and sent to the *stabilator* actuators, not to the ailerons as one would expect (there are no electrical inputs of any kind to the Eagle's ailerons). The stabilator control surfaces operate differentially to stop the unwanted roll disturbance and restore stable flight.

Electrical commands from lateral force inputs to the pilot's control stick force transducer (located both forward and aft in the TF-15A) are first applied to a deadband circuit in order to desensitize the roll commands around the neutral point. The roll command is then switched as a function of Mach number. In this process, the larger gradient assures that the time-to-bank requirements are available at the lower speeds. The lower gradient reduces the roll/yaw coupling tendencies at Mach numbers in excess of 1.5.

Dual roll rate gyros measure aircraft response to a lateral control stick input. The roll channel of the roll/yaw computer then adds or subtracts differential stabilator deflections to assure the proper response. Roll CAS authority is a maximum of ±5 degrees differential stabilator relative to the position selected by the mechanical control system.

The roll CAS error is limited by functions of airspeed and angle of attack. The airspeed limit is employed so that excessive structural loads are not generated on the differential tail. The scheduling signal is derived by the Dynamic Pressure Sensor Unit of the Automatic Flight Control Set. This unit receives pitot and static pressures which drive dual potentiometers. The output is then shaped to provide the proper gain schedule as shown in Figure 3. The gain potentiometer excitation voltages are switched through a lag network upon roll CAS engagement to reduce transients.

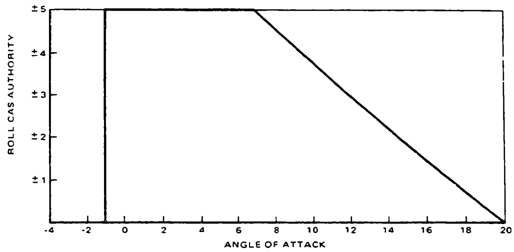
Additional roll CAS limiting is required to reduce the roll/yaw coupling for negative angles of attack, and at large positive angles, to minimize the adverse yaw. This provides an equivalent to the "aileron washout" function of the mechanical control system which was discussed in an earlier article. The angle-of-attack limiting is subtracted from the airspeed schedule as indicated in Figure 4.

No roll CAS is desired at angles of attack above 20 degrees. This prevents the adding of pro-spin controls through uncommanded pilot-induced CAS inputs, and roll damper inputs, at the higher angles of attack. The angle-of-attack schedule is switched to a fixed reference at wheel spinup to assure that full roll CAS authority is available for adequate crosswind control during landing rollout.

Failure Detection - As in the yaw CAS, there are a number of ways in which the system can detect and react to system abnormalities -

- Like the yaw channel, the Roll CAS sensors must track each other by preset limits. When these limits have been exceeded by any sensor, or if a failure occurs in the roll CAS computation electronics, roll CAS shuts down.
- Roll CAS also shuts down, and remains down, if yaw CAS fails or is turned off by the pilot. This assures that no adverse roll/yaw coupling will occur.
- Since roll and pitch CAS share the stabilator servo actuators, roll CAS will shut down if the pitch CAS fails. However, if there has been no failure of the roll computation, and if the stabilator servo actuators are still operative, the roll CAS can be reset and will operate normally. This is a pretty good troubleshooting aid. If a failure of pitch and roll CAS occurs, but roll can be reset, it is unlikely that the stabilator actuators, or the wiring to the actuators, are at fault. In this case, check the pitch CAS computation circuitry and the sensors.

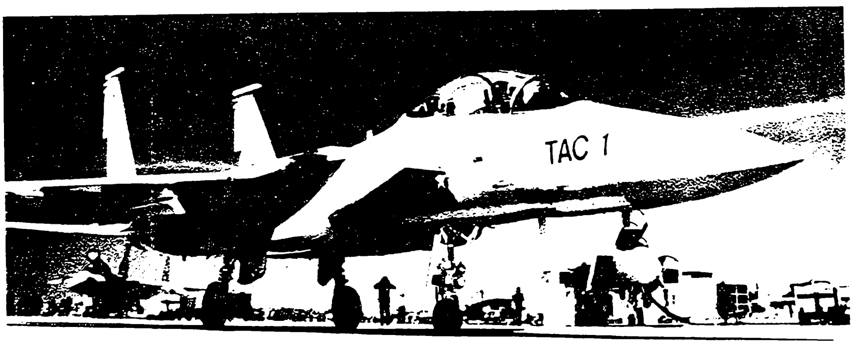
FIGURE 4. ROLL CAS AUTHORITY vs α



- The primary means of detecting failures within the roll and pitch CAS servo loops consists of monitoring the level of error of a differential pressure sensor (DPS) hydraulic ram (one in each stabilator actuator). As long as the servo signals are equal, the DPS error is zero and the system will operate normally. Failure of a servo valve, or electrical failure of an actuator LVDT, will drive one or the other DPS ram hardover. As a result, roll and pitch CAS will shut down. A lag network is employed to filter the DPS error signal being monitored, minimizing nuisance shutdown. With the DPS failure detection scheme just described, a fast-operating, high-authority CAS can be employed with an acceptable level of failure transients for hardover servo valves, or in the event an actuator LVDT output is lost.

This presentation of the F-15 Control Augmentation System will be continued in our next issue of the DIGEST as we take a look at the pitch CAS. ■

Introducing the... F-15B EAGLE



Introducing? What do we mean, *introducing* an airplane that's been flying since 1973? What we really mean is that, as of 1 December, all two-seater Eagles will carry the new model designation of F-15B. Headquarters USAF directed the change to make it completely clear that the two-place version is a "combat" aircraft in its own right and equal to the F-15A. Gone is the heretofore designation of "TF," which, at least by implication, limited the airplane to training capabilities.

And what better serial number for us to use to announce the changeover, than USAF 73-108, otherwise known as "TAC 1" - the first operational Eagle delivered to the Air Force, back in November of 1974? TAC 1 was the first of 38 two-seat fighters to have left the assembly line in St. Louis so far; some 37 more are scheduled. Thus you should be seeing F-15B's for a long time to come! ■

(PUBLISHED 1976)

F-15 Flight Control System Part VI

PITCH CONTROL AUGMENTATION

By B. P. "PERRY" HOFFMAN/*Senior Engineer, Flight Control Section, Avionics Engineering Laboratories*

To conclude this series of articles about the F-15 Flight Control System, let's look at the Pitch portion of the Control Augmentation System (CAS) and the Attitude/Altitude Hold modes. Please stay with me to the end; while I've attempted to remove most of the mysteries surrounding this complex electronic system, it hasn't been easy and I hope we all don't get too confused.

Pitch CAS/Single Channel - The operation of the pitch channel electronics is similar to yaw and roll in that Pitch CAS performs two functions. First, conventional stability augmentation improves ride comfort by reducing or eliminating undesirable aircraft motions from disturbances such as wind gusts. The second operation provides the pilot with precise control of aircraft performance by measuring the aircraft response to a given command, and adding or subtracting stabilator deflection as required to match the command to the "ideal."

Stability Augmentation - As shown in Figure 1, the prime sensor used for damping the unwanted pitch oscillations is the pitch rate gyro. When the aircraft receives a change in its flight path, the resultant rate of change is sensed by the rate gyro. A corrective signal is generated by the pitch rate sensor and is fed to a buffer demodulator, then a rate canceller (which eliminates steady signals), and to a variable limiter (which eliminates switching transients during landing gear operations). The rate signal is then summed, shaped, and sent to a structural filter which reduces frequencies that would cause coupling to the airframe, producing unwanted stabilator oscillation.

The variable limiter performs two functions. When the Pitch CAS switch is reset, the limiter slowly increases CAS authority from zero to 10 degrees. Secondly, it forces the roll and pitch channels to share the 10 degree authority over the stabilator actuators by limiting the amount of CAS series servo deflections either channel can command when the mechanical pitch deflection is greater than 18 degrees nose up.

The modified corrective rate signal is then applied to a servo amplifier which electrically commands the servo valve to extend or retract the 10 degree CAS series servo (internal to the stabilator power cylinder), repositioning the main power cylinder control valve, and porting hydraulic pressure to the power cylinder main ram piston. When the main ram piston deflects, it repositions the stabilator control surface in a direction to stop the unwanted airframe disturbance. An electrical follow-up signal is generated by the 10 degree CAS series servo Linear Voltage Differential Transformer (LVDT) which opposes the corrective rate signal input. When sufficient series servo deflection is obtained to match the rate signal input, series servo deflection stops. The aircraft rate of change in its flight path will grow smaller, and the follow-up LVDT signal starts to return the series servo to neutral. When the aircraft flight path is again stabilized, the rate signal is zero and the CAS series servo is at neutral.

A condition where no aircraft rates are being generated and corrective action is being taken by pitch stab aug is pretty hard to come by. The pitch stab aug is constantly working to maintain a stable airframe. (For the sake of simplicity, we only considered a single channel rate disturbance.)

Pitch Control Augmentation - Looking again at Figure 1, find the forward and aft pitch force sensors (F-15A or TF-15A). Longitudinal stick force commands from one (or both) control stick force transducers are summed together and fed through a deadband, dual-gradient circuit (prevents oversensitivity near null and reduces stick forces during sustained high "g" maneuvers). The resultant stick force command then goes through a 0.2 second pre-filter for smoother system responses to sharp pilot inputs (this improves tracking characteristics). The shaped command signal is then sent through the same structural filter as the stab aug rate signal, and on to the variable limiter. The variable limiter, used for stab aug and pilot commands, has the same reason for being in the circuit and operates the same as was explained in the stab aug section.

The pilot command is then sent equally to the left and right servo amplifiers which electrically command the servo valve to extend or retract the 10 degree CAS series servo, repositioning the main power cylinder control valve, and porting hydraulic pressure to the power cylinder main ram piston. Deflection of the piston repositions the stabilator control surface in the direction desired. Two LVDT follow-up signals are generated by the movement of the stabilator power cylinder. The main ram LVDT signal provides the intelligence for the variable limiter, telling the limiter just where the stabilator control surface is located. The other follow-up signal is again the CAS series servo position and stops series servo displacement when the input signal level is matched.

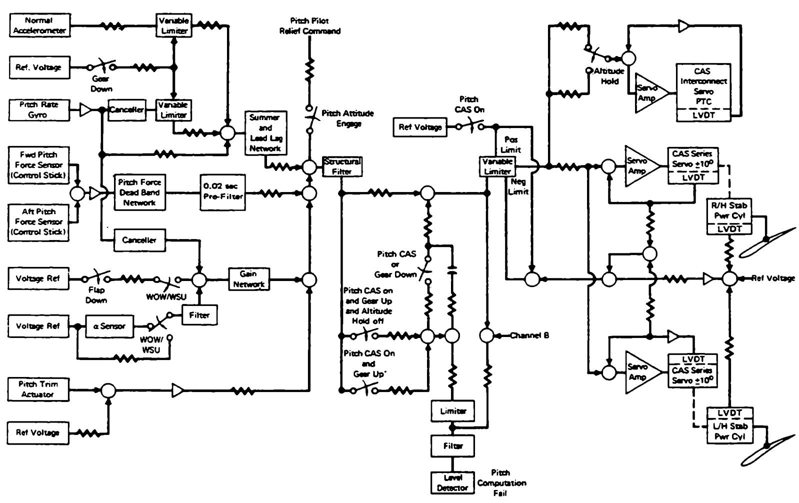
Aircraft response to a pilot's stick force command is measured by the pitch rate gyro and normal accelerometer sensor outputs. These signals meet at the summing/lead lag network. If the measured response after summation does not agree with the control stick force command, the difference is fed to the stabilator actuators, adding or subtracting control surface deflection until the difference is zero. This is called "blending of command and surface deflection" to achieve the ideal aircraft response.

Up to now, we've considered an aircraft with landing gear and flaps up. When the gear handle is positioned down, normal acceleration signals and the pitch rate canceller circuit are eliminated. The removal of normal acceleration is necessary to get rid of transients due to aircraft impact with the runway. Removal of the pitch rate canceller circuit at the same time insures stable longitudinal control during approach and landing. The reduction of these two signal levels is achieved through variable limiters which have a one-second time constant for fade in or out.

Angle-of-attack signals are also used by the pitch CAS to inhibit stalls and match the pitch CAS to the mechanical control system stabilator command characteristics during high angle of attack maneuvers. The stall inhibiting circuit subtracts a portion of the pilot command signal proportional to angle of attack above a threshold determined by flap position. The threshold is higher with flaps down due to added lift which increases the stall boundary. Pitch rate signals are added to the angle-of-attack signals to provide stall inhibit anticipation during rapid maneuvers. The angle-of-attack signal is switched to a pre-set value by the weight-on-wheels switch, or by wheel spin-up signals from the anti-skid sensors, removing stall inhibition during ground operation.

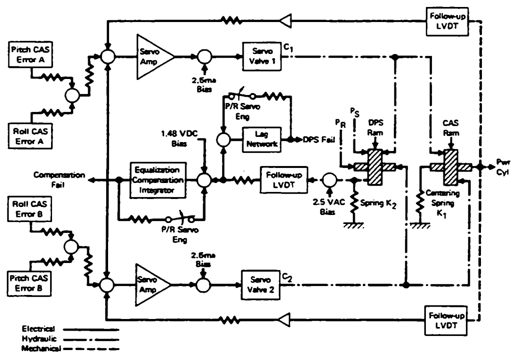
Pitch trim signals are also fed into the pitch CAS to tell the system what trim value the pilot requires. If this trim signal were not present, any manual retrim selected by the pilot

FIGURE 1 – PITCH CAS BLOCK DIAGRAM



would be defeated by the CAS returning the aircraft to the original trim position. Pitch CAS commands are also used to deflect the CAS interconnect servo, located in the Pitch Trim Compensator module of the PRCA. These commands serve two functions. First, they insure that the mechanical and the CAS systems are tracking each other, minimizing any disagreement that may exist if the pitch CAS disengages and the mechanical system takes over. The second feature allows the CAS interconnect servo to carry an offset from null, allowing the CAS series servo (within the stabilator power cylinder) to maintain its full ±10 degree authority.

FIGURE 2—PITCH/ROLL SERVOLOOP



Pitch CAS Engage/Disengage Logic - When the following conditions are satisfied, pitch CAS engagement is possible by placing the engage switch to ON or if on, pulling it back to RESET and then ON.

- Pitch CAS equalization error below failure-detect threshold (no pitch computation error).
- Differential pressure sensors and compensation output below failure - detect threshold.
- Aircraft yaw rate below disengage threshold (41.5 degrees/sec) and aircraft is not spinning.
- CAS interconnect servo has not failed.

The pitch CAS switch reset pulse will cause the CAS series servo shutoff valves and the CAS interconnect shutoff valves to be energized. In addition, the differential pressure sensor failure - detect circuit and equalization integrator will be activated.

Activation of the series servo and interconnect servo shutoff valves latches the necessary logic to keep the shutoff valves engaged. Simultaneously, another latch circuit activates the pitch CAS engage limiter, fading from 0 to 100% authority (±10 degree series servo), extinguishing the telepanel fail light.

Disengagement of pitch/roll CAS occurs if any of the pre-engage conditions indicate a failure. Roll CAS may be reset if the failure has not occurred within the stabilator power cylinder or power cylinder wiring. As long as the CAS series servos and differential pressure sensors have not failed, placing the Roll CAS switch to RESET will re-engage roll CAS.

Some nuisance disengagements of CAS may randomly occur. As long as they can be successfully reset and stay set, there is no cause for alarm. Repeated shutdowns, that reoccur (after reset) when completing similar maneuvers, should be "griped" so maintenance can seek out the cause. To help them, pilots should include in writeups all known information such as airspeed, altitude, g load, and/or any additional flight conditions which may affect the CAS.

If shutdowns are caused by a CAS interconnect servo failure, pilots may select PITCH RATIO EMERGENCY and then reset pitch CAS. Placing pitch ratio in EMERGENCY inhibits the interconnect failure detect logic. Pilots then have the option of flying at about one-half mechanical pitch ratio without hydraulic boost from the PRCA (this means you'll have higher stick forces). Also, mechanical ARI will be inoperative. With pitch ratio EMERGENCY selected, and an operational pitch CAS, sufficient longitudinal control is available for most maneuvers including adversities during the landing phase. Whatever method you select, leave pitch CAS shut down with full mechanical ratio (plus operational series trim from the pitch trim controller), or select emergency pitch ratio and reset pitch CAS. Slow down to a reasonable "q" before experimenting.

Failure Detection - The primary means of failure detection for Pitch CAS is the monitoring of voltage levels generated by a differential pressure sensor located in each stabilator power cylinder. Normally the voltage level will be zero (no failures) which satisfies an equalization integrator circuit of the CAS and shutdown will not occur. A lag network is employed to filter the differential pressure sensor output signal in order to minimize nuisance failure shutdowns. But suppose a failure does occur?

Due to the high authority of the pitch CAS (±10 degrees), failure transient control is provided and its operation can be understood by referring to Figure 2. If a failure occurs, one servo valve will be driven hardover by full supply hydraulic pressure and the CAS ram will begin to displace in the direction of the failure (note that each servo valve is biased so that null failures within servo amplifiers or servo valves will result in hardover servo control pressure). As the CAS ram begins to displace, an error is generated at the input of the remaining servo amplifier. This causes its servo valve to establish a counteracting force on the ram. Since both servo valves are connected to the same hydraulic source, the resultant forces seen by the CAS ram are opposite and equal, causing the ram to stall. This condition is called "force fight." The CAS ram will then begin to center at a 0 to 1 degree per second rate determined by spring K1. Small mistrack and deadband errors below the equalization compensation networks are dealt with by the "force fight" technique and result in small engage transients. These are too small to cause aircraft displacement and can be ignored.

Looking again at Figure 2, note the

differential pressure sensor (DPS) ram and its associated connections. With no failures, pressure C1 and C2 on the right hand side of the DPS ram are equal and balance the combined pressures of Ps + Pr and spring K2 on the left side of the DPS ram. In this condition, equilibrium exists (no motion of ram) and the equalization compensation integrator output is zero (CAS remains set, no failures).

If pitch CAS component failure occurs, such as described in the "force fight" explanation, the equilibrium of pressures C1=C2 on the right side of the DPS ram are upset and the DPS ram begins to drive slowly hardover. The DPS, LVDT signal is fed to the equalization compensation integrator which starts to slew in a direction to reduce the LVDT signal to zero. If the integrator signal has not reached zero in three seconds, a shutdown pulse is generated by the CAS and both stabilator ±10 degree series servo shutoff valves are deenergized. Controlled orifices and spring-controlled locks center the CAS and DPS rams.

There is one normal condition where the DPS senses an error which is not a real failure. If during ground checks of Pitch CAS operation, the control stick is held hard in any one corner with sufficient force to command full CAS authority (pitch and roll), you may get a shutdown. As this procedure is a function of technique, it can't be totally relied upon as a valid check of DPS operation. Some pilots have even experienced a similar condition during landing rollout.

As you can see, any type of failure within the pitch CAS electronics, sensors, and hydromechanical components can create an imbalance in either of the dual channels. With the DPS scheme of failure detection, you'll get a shutdown of Pitch/Roll CAS.

No matter which scheme of failure detection is chosen, some surface deflection must occur before an action to stop it can be taken. The "force fight" method with DPS shutdown detection was chosen for the Eagle to minimize the transients felt by the aircraft.

Pitch Trim Compensator Failure - Pitch trim compensator failure techniques are similar to those described for the stabilator power cylinder/DPS. Failure detection is accomplished by comparing the sum of servo valve control pressures, with the sum of return and regulated hydraulic supply pressures. A spring-loaded differential pressure spool is employed to compare the pressure, and if the hydraulic pressures are sufficient to overcome the detent spring loads (component failure of PTC interconnect servo), the ram deflects and opens redundant failure-detection switches. Opening of any of the PTC switches creates a failure pulse which shuts down the pitch CAS. Pilot action with a PTC failure was discussed earlier.

Equalization of PTC failure detection is unnecessary since dual channel pitch CAS commands are averaged at the servo amplifier. This assures the same command to each channel and the PTC servo gains are considerably lowered. As a result, channel mistrack displacements are higher, but since the servo drives the PTC at a slow rate, the resultant transients are acceptable.

Roll/Pitch Pilot Relief Modes (Attitude Hold) - Attitude Hold modes may be engaged if all the following pre-engage conditions are satisfied:

- Roll attitude interlocks are present (yaw/roll CAS engaged and roll outer loop signal is below pre-engage threshold) with no roll stick force applied.
- Pitch outer loop signal is below pre-engage threshold (equivalent to a steady state command of 0.25 g.)
- INS attitude valid (central computer and ADC operational).
- Aircraft normal acceleration greater than O g and less than + 4 g.
- Autopilot disengage switch (paddle) closed.
- Pitch CAS engaged.

With these conditions satisfied, the solenoid-held Attitude Hold switch will remain engaged. Disengagement will occur when any one of the pre-engage conditions is not met. The Attitude Hold mode will maintain aircraft attitude with ±45 degrees of pitch attitude and ±60 degrees of roll attitude. If the aircraft is maneuvered outside of these limits, the Attitude Hold switch will remain engaged, but the holding functions are eliminated. Maneuvering back within the attitude hold limits will again re-engage attitude hold.

Maneuvering within the attitude hold limits can be accomplished without disengagement. Force applied to the control stick in excess of one pound actuates the control stick steering mode, repositioning the controlling surfaces in the same manner as described in Roll and Pitch CAS pilot command inputs. While the aircraft is maneuvering, the Pitch and Roll Attitude Synchronizer is unlocked and allows synchronization to the new attitude commanded by the pilot. When sticks force is reduced below one pound, Attitude Hold modes re-engage.

Both the Roll and Pitch Attitude Hold modes have solid state synchronizers. These follow the attitude information from the INS platform attitude gyro, keeping the attitude signals below the pre-engage threshold limits.

There is an additional input to the roll synchronizer, roll rate, when the Attitude Hold mode is engaged and the pilot is maneuvering the roll control stick steering. Under this condition, the roll rate signal is sent to the roll synchronizer, causing it to lead the changing roll attitude. If this slight lead were not used, pilots would experience "roll rebound" or, if the stick was released at 30 degrees of right wing down for instance, the aircraft would roll back to 25 degrees because the roll synchronizer did not keep up with the aircraft roll rate. The rate signal is switched out prior to engagement of the Attitude Hold switch. As the roll rate signal appears as an error signal to the roll synchronizer, the pre-engage lever detector limit may be exceeded, preventing engagement of the Attitude Hold modes.

(Altitude Hold) - Altitude Hold mode may be engaged if the following pre-engage conditions are satisfied:

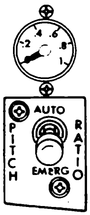
- Altitude Hold engaged.
- INS vertical velocity signal valid.
- ADC altitude error signal valid.
- Magnitude of aircraft vertical velocity less than 2000 feet per minute.

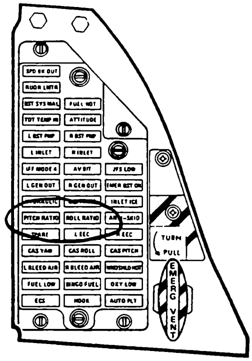
When these conditions are met, the solenoid-held Altitude Hold switch will remain engaged. Disengagement of altitude hold will occur if any of the pre-engage conditions are not met.

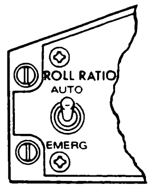
Altitude error signals from the Air Data Computer, a vertical velocity signal from the INS, and cancelled pitch attitude are blended to generate an altitude hold error command. The resultant signal is sent equally to the stabilator actuators, deflecting them in a direction to return the aircraft to the engaged altitude. Pitch attitude error signals are switched out during altitude hold operation, but are used to operate the pitch synchronizer so that it will be aligned should the altitude hold be disengaged and *attitude* hold is again engaged. Finally, all signals are faded in and out, thus minimizing transients. ■

F-15

(PUBLISHED 1979)







Pitch & Roll Ratio Lights

By PERRY HOFFMAN/*Senior Engineer, Flight Control Section, Avionics Engineering Laboratories*

"Give me a complete explanation of PRCA ratio warning light illumination . . . in words of two syllables!"

When MCAIR flight controls engineer Perry Hoffman was out on a recent accident investigation, he was given that unique request by a thoroughly confused USAF Eagle Driver. Mr. Hoffman has been writing with great clarity on flight controls topics for the DIGEST for many years, but even he found the F-15 pitch and roll ratio warning lights to be more than "two-syllable" topics. PRCA is a complex component with an involved interrelationship with the mechanical control system, including those warning lights associated with system problems. Therefore, while Perry graciously consented to preparation of yet another DIGEST article, he has been forced into more than a few polysyllables in the following discussion. Even so, we feel if you can find a quiet spot and give Mr. Hoffman your undivided attention for a little while, you'll find this to be an "illuminating" presentation! (Incidentally, this and many other topics are also discussed by the same author as a part of P.S. 951, a reprint collection of previous DIGEST articles on the general subject of F-15 flight controls. While there aren't too many two-syllable words in them either, we've been told by both mechs and pilots that there has never been anything written better. See your rep for a copy, or ask us for one.)

ROLL RATIO LIGHT

There have been several attempts to explain in basic terms what the roll ratio light tells you, the pilot, as well as what to do if it's on. In my opinion, there are two ways to attack the explanation, and I'm going to use both. First, the bottom line — if you have a roll ratio light illumination at Mach 1.5, don't do any excess lateral maneuvering; just slow down below Mach 1.0 and continue your mission subsonic. If the light remains on or illuminates when you select gear down for landing, there may not be sufficient lateral control to hold the upwind wing down into the crosswind, so plant the nose shortly after the mains and allow longitudinal stick to go to neutral; then apply lateral stick into the wind and there will be more aileron/differential stabilator available to hold the wing down. For those who want to know why we say this, read on while I try to clarify the design rationale.

Back in the beginning, the F-15 roll ratio warning light was designed to do just one task — tell the pilot when hydraulic power was lost in the roll chan- of the PRCA. (There were a couple of variables in the form of airspeed switches, but these switches rapidly went away.) So for some time the design was such that when the roll ratio light illuminated, three possibilities existed — (1) Hydraulic failure, the utility pressure drops, and for some reason PC2 does not switch in and pressurize the PRCA; (2) The pilot selects roll ratio emergency; or (3) The light circuit itself malfunctions. Simple?

Well, simple things often become complicated very rapidly. Early flight tests indicated a tendency toward roll coupling at high supersonic mach numbers and we found these coupling tendencies could be eliminated by reducing the total aileron deflection to 2/3 authority supersonic. So, the roll ratio controller output was limited and the ARI deactivated above Mach 1.0. This was accomplished via a hydraulic switching signal generated within the pitch channel of the PRCA, and by the same hydromechanical scheduling device that computed pitch ratio. But now how do we warn the pilot the 2/3 ratio shift has or hasn't taken place? Using another available supersonic signal which is generated by the engine air inlet controller, it is possible to warn the pilot of a problem. When the Mach 1.0 signal reduces the roll ratio authority to 2/3, a switch activates to arm the roll ratio warning circuit. As the aircraft accelerates through Mach 1.5, the left air inlet controller discrete signal energizes a multicontact relay, which recognizes the armed 2/3 authority circuit and completes the ground path, thus keeping the warning light out. Thus a roll ratio warning light illuminated at Mach 1.5 now indicates a failure of the roll ratio controller to shift, not just a loss of hydraulic pressure. Maybe not so *simple* as our beginning bird, but it works!

Incidentally, you might note that this arrangement provides no warning for the region between Mach 1.0 to 1.5. No problem, because roll ratio failures are relatively benign at those speeds. The rudder pedal limiter ($\pm 5°$) is also activated by the Mach 1.5 relay. The rudder pedal limiter is another device designed to protect the pilot from lateral coupling at high supersonic speeds. A "RUDR LMTR" light indicates improper scheduling, i.e., below Mach 1.5 the limiter remains engaged or above Mach 1.5 it fails to engage.

By now I'm sure even the most conscientious reader might be getting confused, so let's attempt to clarify in the form of "action" and "indication" by the system and by pilot reaction.

Action: Mach $\geqq$ 1.0

Aileron-Rudder Interconnect (ARI) is switched off.

Roll ratio or lateral stick to aileron/differential stabilator is reduced to 2/3.

Indication: Mach 1.5

If roll ratio has switched, warning light remains out. If roll ratio has *NOT* switched, warning light illuminates.

Rudder pedal limiter activates maximum rudder pedal movement equivalent to $\pm 5°$ of rudder.

Pilot Reaction: Maneuvering

All warning lights out, no problem!

Roll ratio light illuminated, do not attempt large lateral maneuvers, and AVOID any *lateral maneuvers* at *LESS* than 1 G. You could get into an undesirable situation by the *roll ratio* being *maximum* and the *ARI not* being *shut down*.

Reduce speed toward subsonic. If light goes out at Mach 1.5, there is a strong probability of a normal control system below Mach 1.0. If the warning light remains on at Mach 1.5, the probabilities are that you have an electrical malfunction of some sort (left air inlet controller mach switch, rudder pedal limiter relay, or associated wiring).

The main thing to remember if the light remains on below Mach 1.0 is that you won't induce any hazardous flight conditions with normal maneuvers. Perform a controllability check in the lateral directional axis, since the biggest effect of a roll ratio malfunction in the subsonic regime is partial loss of turn coordination at high AOA. In other words, the aircraft will no longer be a "feet-on-the-floor" handling bird.

Pilot Reaction: Landing

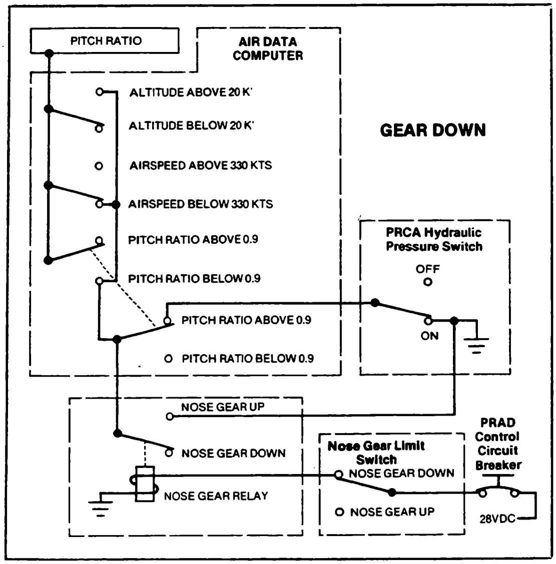
When the landing gear is extended, a new set of conditions present themselves. Normally the roll ratio drives to maximum (crosswind consideration). If the roll ratio warning light should illuminate at gear extension the probabilities are that the roll ratio is not at maximum; no biggie, unless you have a strong crosswind. You just won't have sufficient roll power to hold the wing down when the control stick is aft of neutral while holding the nose high during landing rollout.

PITCH RATIO LIGHT

Pitch ratio is varied in the F-15 for one primary purpose — to prevent oversensitive pitch control in those regions where a little stabilator gives a lot of "G." Everywhere else in the flight envelope, it tries to give you the maximum usable stabilator control. Normal operation of the pitch CAS can pretty well mask even large errors in the desired ptich ratio, so the variable ratio is primarily to help you handle the Eagle when CAS is off. The pitch ratio indicator keeps the pilot informed of what the ratio is doing and you are probably used to associating certain readings with flight conditions, but how do we warn you when the pitch ratio is not performing as advertised?

A continual monitor of pitch ratio would require some sort of computer but then if a problem came up, you wouldn't know if it was caused by the monitor or the pitch ratio. Instead of building a monitoring computer, a relatively simple logic system was devised as a check to make sure the pitch ratio is not grossly mispositioned in the critical areas of flight — low altitude, high speed, and landing. This logic can best be understood by leading yourself through the two diagrams which show the circuits for gear down and gear up conditions. The logic goes like this: If you can't work your way from the "PITCH RATIO" box through the maze and get to "ground" (⏚) you have a condition that will illuminate the "PITCH RATIO" light.

This "short cut" logic does lead to one small problem wherein the warning light can lie to you if you are low and fast with landing gear down. With the landing gear down, the primary purpose of the warning circuit is to advise if the pitch ratio is less than 0.9, indicating reduced control authority for landing. Below 20K and at speeds around 220 to 300 kts, the pitch ratio begins to schedule down below 0.9; thus if the landing gear is down at



these conditions, the light comes on. Pitch ratio will read between 0.9 and 0.75. In this rather limited speed range with gear down, the pitch ratio caution light can illuminate, but the system is working as designed. So, if you have a habit of flying fast (260 to 300 kts) with the gear down, you may get an occasional objection from the pitch ratio light.

With the gear up, the primary function of the warning circuit is to advise that the ratio has not scheduled below 0.9 in the region below 20K at speeds above 330 kts. With the gear up and the altitude above 20K or airspeeds below 330 kts, there is no position of the pitch ratio that will cause a warning light.

What should the pilot do when the pitch ratio warning light illuminates? A better way to put it might be — what should he *not* do? First, he should not panic; second, he should not ignore it!

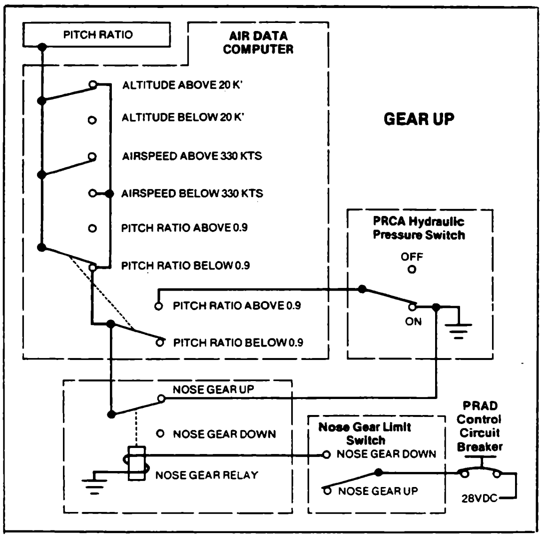
Some pilots tend to ignore warning light illuminations because we have had many just plain switch failures that have given false warnings with the system operating normally. Don't do this! Treat each unexplained light illumination as a real problem and exercise caution until you convince yourself everything is normal.

On the other hand, there is no need to panic. Assume straight and level flight and monitor the light; note the flight conditions and pitch ratio indicator as exactly as possible (for post-flight writeup); perform controllability check in pitch axis; and spot check a couple of flight conditions and insure pitch ratio indicator is reading properly. Examples of spot checks you can make:

- Stabilize in trimmed flight at 10K, 350 kts CAS — pitch ratio indicator should read 0.4 ± 0.15.
- At 20K, 375 kts CAS, pitch ratio indicator should read 0.2 ± 0.15.
- At 30K, 300 kts CAS, pitch ratio indicator should read 0.7 ± 0.15.

When at least two of the indicator readings are within the specified tolerances, all's well; continue monitoring the indicator and warning light circuit and press on. Approach and landing speeds should yield a pitch ratio indicator reading of 0.9 to 1.0 and a warning light out. If the ratio is below 0.4 and warning light illuminates, select pitch ratio emergency and perform a cautious landing. If pitch CAS is operational even with pitch ratio in emergency, no big problems will exist and you may not notice any degradation in handling qualities.

Write it up when you debrief; intermittents are difficult to troubleshoot but there is a problem that should be fixed. Give the debriefer all the conditions observed when the light came on; many times even the slightest detail points a finger at the bad guy.



CONTROL STICK FORCES

While I've got your attention on the subject of F-15 flight controls, let's briefly discuss a couple points concerning stick forces which came up during a recent flight safety meeting here at MCAIR. Seems that these points also relate to the pitch/roll ratio switches.

When making the stick force transducer check in the DASH ONE Pre-Taxi Check (Step 13) some pilots have experienced a marked increase in longitudinal stick force as forward stick force is applied. This is a normal indication if the pitch ratio has not attained the fail position of 0.4.

The reason for this increase-in-force phenomenon begins when the pitch ratio switch is placed in EMERGENCY. Hydraulic pressure is removed from the PRCA pitch channel and the pitch ratio linkage now begins to move toward 0.4, being forced by very large springs. In order to drive the pitch boost actuator and ratio changer linkage to their fail positions and not cause any reverse stick drive, a small reservoir of oil (approximately two ounces) is provided. When the pilot moves the stick longitudinally before the ratio has driven to 0.4, this small supply of oil is immediately depleted. Now he is physically moving dual pitch ratio linkage, a dragging pitch boost actuator, and a pitch trim controller all attempting to drive to fail or lock position, plus the aircraft longitudinal control leakage aft of the PRCA.

As you might suspect, forces get high just within the PRCA itself. They have been measured at 12 to 14 pounds without aircraft linkage attached. So when 0.4 is attained, pitch forces will be about three times their normal value (pitch ratio in AUTO). If you move the pole early, expect high stick forces which make you believe the pole is in concrete. The best approach is to wait until the ratio is in fail (0.4) and then push the stick forward. (The test isn't valid anyway, unless the pitch ratio is totally failed.)

Another point to consider is that if you select EMERGENCY in pitch or roll some systems will give a slight kick or jump in the stick. This may also be seen if utility hydraulics fail and the PC2 system does not rapidly pick up the PRCA. This is normal and to be expected, and the kick may increase in magnitude if stick force is applied at time of PRCA failure. ■

(PUBLISHED 1974)

The First Hundred

The F-15 Eagle lifted off the ground for its maiden flight on 27 July 1972. In the two-plus years since that day, the Eagle has recorded over three thousand flights and the number of people to ride in the Eagle has passed the century mark.

We believe in the capability and potential of the F-15, and have since its earliest drawing-board stages. But the confidence of engineers, product support specialists, manufacturing people, and the "hundred-and-one" other specialties that go into making a successful aircraft, is not half as important as the confidence required by the man in the cockpit - the man for whom the weapon system will become a vehicle of victory or defeat.

Reactions from some of the first hundred people to fly in the Eagle (their names are listed below) surpass any comment we, on the ground, might come up with. We'd like to share here a capsule collection of these reactions:

"The Eagle, though not a small airplane, has 'the feel of a fighter' - it is fun to fly."

Irv Burrows (Eagle Driver 1)

"A fighter with explosive performance that handles like a dream."

Pete Garrison (Eagle Driver 2)

"The handling qualities of the F-15 are excellent . . Cockpit visibility is really great."

Col Wendell Shawler, USAF
(Eagle Driver 3)

"At maximum performance takeoff and climb, the F-15 is the closest thing to a booster rocket Apollo or Gemini space launch."

Astronaut (BGen) Tom Stafford
(Eagle Driver 55)

Know ye all mankind by these presents that

is now, and forevermore, exalted among the Eagles for having this day flown the aircraft destined to dominate the skies throughout this decade and the next, — the United States Air Force F-15 Eagle.

In so doing, he joined that honorable company of individuals who are forever enshrined among the

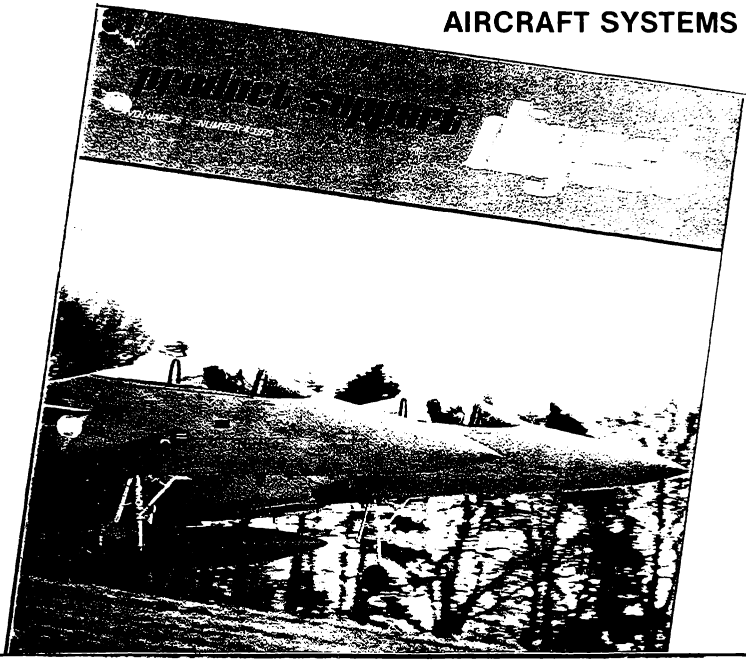
to fly the F-15 Eagle.

In witness whereof I have affixed the Eagle Seal this day of

Chief Eagle

MCDONNELL AIRCRAFT COMPANY

I. L. Burrows, Jr.	Col J. W. Wood	Capt M. E. Durbin	C. W. Martin
C. P. Garrison	MGen D. Brett	D. Malvern	Col W. L. Skliar
Col W. H. Shawler	BGen W. D. Druen, Jr.	Dr. J. L. McLucas	MGen B. Peled
P. Henry	Gen G. S. Brown	Dr. W. B. LaBerge	Col R. R. Moore
Maj C. W. Powell	LCol W. Schob	BGen T. P. Stafford	LGen D. S. Sweat
Capt M. E. Sexton	BGen H. M. Lane	LGen W. J. Evans	Col W. H. Heermans II
D. D. Behm	Maj D. W. Peterson	VAdm W. D. Hauser	LCol W. E. Roth
J. E. Krings	Maj W. R. Macfarlane	Maj W. S. Vrablic	MGen J. A. Knight
Maj R. Smith	Capt B. E. F. Foster	LCol J. V. Corbisiero	Col L. D. McClain
Col W. J. Knight	Capt J. S. Rodero	Gen J.J. Catton	LGen F. M. Rogers
LCol F. W. Bloomcamp	Capt J. H. Doolittle III	W. S. Ross	BGen R.M. White
C. A. Plummer	W. L. Lowe	C. R. Lucas	BGen F. A. Humphries, Jr.
S. H. McIntire	C. E. Rosenmayer	Col J. D. Mirth	Maj N. K. Dyson
C. D. Pilcher	LGen D. F. James, Jr.	BGen R. F. Titus	BGen R. A. Rushworth
Maj J. K. Spiers	Capt D. D. Carson	Sen H. W. Cannon	BGen M. F. Doyle
R. G. Martin	Cdr G. W. White	LCol J. G. Rider	J. W. Plummer
E. D. Francis	W. K. Wight	LCol R. C. Herrick	MGen R. E. Hails
J. Dobronski	G. S. Graff	BGen R. C. Mathis	MGen G. F. Blood
W. H. Brinks	Capt B. W. Hudson	C. E. Meyers	Dr. M. L. Currie
LCol A. J. Bergman	Gen R. J. Dixon	A. I. Mendolia	Col L. M. N. Wenzel
Sen (BGen) B. Goldwater	Capt D. R. Mangum	P. F. Oestricher	Maj G. J. Higgs
Maj C. P. Winters	Maj J. H. Thomas	E. K. Enevoldson	MGen J. T. Burns
Gen W. W. Momyer	Gen M. Khatami	Capt G. K. Muellner	Col B. Morrell
MGen B. N. Bellis	BGen A. Azarbazin	BGen C. E. Yeager	Gen J. W. Vogt, Jr.
Col D. R. Spruill	Gen S. C. Phillips	Col H. L. Orthman	TSgt R. L. Sweebe



Seeing is Believing... or Is It?

The photo on the front cover shows two F-15s taking off. Or are they landing?

The photo, shot by SSGT Jose Lopez, Jr. of Det 3, 1361st AVS at Rhein Main Airbase and originally published in the Bitburg SKYBLAZER, has created quite a stir.

The SKYBLAZER captioned it as a takeoff, but when sending the picture to us, explained that it was actually a landing. We wondered how one could really tell (unless you were one of the three people actually there at the time), since all the "normal" indicators — gear, flaps, speedbrake, etc., are either not in view or non-committal. So we queried a local "panel of experts" (six company Eagle Drivers who happened to be on the ground when we walked into the Flight Test office).

Exactly one half of the panel swears it is a takeoff shot (lead has rotated before the wingie, spacing is that of a takeoff, etc.) The other half guarantees it to be a landing shot (wingman is looking forward and he has touched down first, etc.)

In an effort to break the tie, we asked one more pilot who offered this quick response, "Actually it's a very clever shot, taken by an extremely deft photographer using a telephoto lens, of a simultaneous takeoff and landing on parallel runways!" If we buy that, can anybody tell us which Eagle is coming and which is going?!

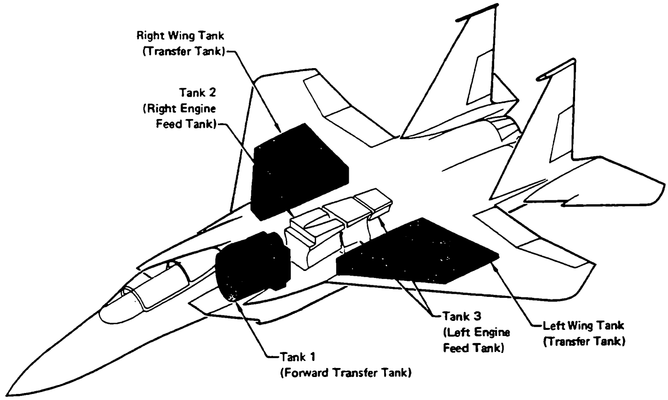
Truthfully, we're not at all sure what they're doing, but whatever it is, they're doing it in a mighty big hurry — we've seldom seen a photograph which so vividly catches the "let's get at it" attitude bred into the Eagle. Our compliments to the photographer for a super shot; our congratulations to the Bitburg crewmen for some obviously great coordinated action; and our notification to the MCAIR "Eagle Experts" that we are not likely to convene that panel again soon! [And now see page 173!] ■

Water (and Fuel) flow down hill... (PUBLISHED 1976)

F-15 FUEL GRAVITY TRANSFER SYSTEM

By GLENN HARPER/*Lead Engineer, Fuel Systems Design*

FIGURE 1 - F-15 FUEL TANK ARRANGEMENT



When early man first attempted to carry water from the stream to his fireside, he must have quickly learned the most basic law of fluid system design — "Water flows downhill." And shortly thereafter, he probably also observed the first corollary to that law — "The hill is never in the right place!"

The fuel systems of high performance fighter aircraft must obey that same basic law during any mode of operation that uses the force of gravity to transfer fuel from one tank to another. The fuel system of the Eagle includes a series of fuel transfer pumps to transfer fuel from the transfer to the feed tanks, but in the event of pump failure, a back-up system (the "Gravity Transfer System") lets gravity take over and perform the fuel transfer function. What complicates things for the airplane is that the "hill" is constantly changing because of aircraft attitude, acceleration, etc. — minor matters for our caveman but important to a pilot. However, this characteristic can be used to the pilot's advantage since he can manipulate the "slope" and "location" of the "hill" to improve the performance of the gravity transfer system.

This article will describe the operation of the F-15 "fuel gravity transfer system;" the effect on the system of the height and slope of the "hill" that the pilot provides; and what you — the jock — should do if a transfer pump failure occurs. Just a few simple rules related to aircraft pitch angle and throttle setting will suffice to bring you and your Eagle safely home.

GRAVITY FEED vs GRAVITY TRANSFER

The terms "gravity feed" and "gravity transfer" are frequently used interchangeably in F-15 fuel system discussions; however, they properly refer to completely different functions and the distinction between the two should be clearly understood. The term "gravity feed" (sometimes called "suction feed") describes the function which provides fuel from the feed tanks to the engines when no boost pumps are operating. This gravity feed system is used during the first engine start until the main generator comes on the line. This article addresses only the "gravity transfer" function which refers to a back-up method of transferring fuel from the "transfer" tanks to the "feed" tanks in the event of a failure of one or more transfer pumps.

BASIC FUEL TRANSFER SYSTEM

To understand the Eagle's fuel gravity transfer system, a brief look at its basic fuel system is needed. Fuel is carried in five internal tanks, as shown in Figure 1. Feed tanks 2 and 3, located near the center of the aircraft, supply fuel directly to the engines. Fuel is transferred to these feed tanks ▶

FIGURE 2 - F-15 BASIC FUEL SYSTEM

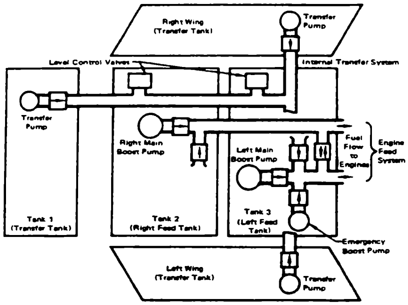
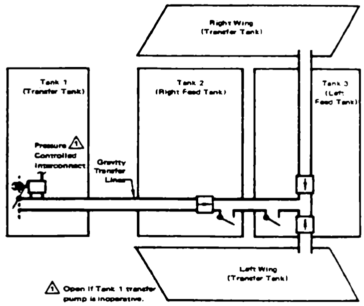


FIGURE 3 - F-15 FUEL GRAVITY TRANSFER SYSTEM



from three transfer tanks, one located forward of the ammunition container and one in each wing. A continuously operating transfer pump in each of these tanks maintains "full" feed tanks (except for brief periods of high fuel consumption). A schematic of this system is shown in Figure 2.

FUEL GRAVITY TRANSFER SYSTEM

In the event of a transfer pump failure, a "gravity" system allows fuel to flow to the feed tanks by way of independent gravity lines. A schematic of this system is shown in Figure 3.

In order for fuel to flow by gravity, the fuel level (not necessarily fuel quantity) in the *transfer* tanks must be higher than that in the *feed* tanks (check valves prevent flow out of the feed tanks). Figure 4 shows the relative locations of fuel in the transfer and feed tanks during gravity transfer.

Fuel is not transferred by gravity from a tank with a failed transfer pump until all other transfer tanks are empty because the transfer pumps which are still operating keep the feed tanks full and thus prevent establishment of the fuel head difference ("h") necessary to provide gravity transfer. The feed tanks cannot be refilled by gravity transfer since the fuel levels of both feed and transfer tanks will decrease simultaneously (except for a temporary increase in feed tank fuel level following throttle retardation from high power settings).

The pilot should note that FUEL LOW warning will occur while there is still a significant fuel quantity remaining in the affected transfer tank, even though the gravity transfer system is operating normally. This is due to the position of the fuel low level sensors in the feed tanks (Figure 4).

The primary factors which affect the operation of the system are "engine fuel demand" and "aircraft pitch attitude." Figure 5 shows how fuel availability is affected by engine fuel demand and aircraft pitch attitude for a pump failure in Tank 1. Figure 6 shows similar information for a wing pump failure. These two figures combine to show that for any transfer pump failure, maximum fuel is available when the engine fuel flow is limited to 3500 PPH/ENG or less, and aircraft pitch attitude is maintained between 3° and 7° nose-up. Transient throttle movements and attitude variations outside these bands are acceptable provided the aircraft is returned within these boundaries prior to feed tank depletion.

Figure 7 shows that the 3500 PPH/ENG limit can be maintained during cruise, loiter, or landing. In addition,

FIGURE 4 - RELATIVE LOCATION OF FUEL IN THE F-15 TANKS DURING GRAVITY TRANSFER

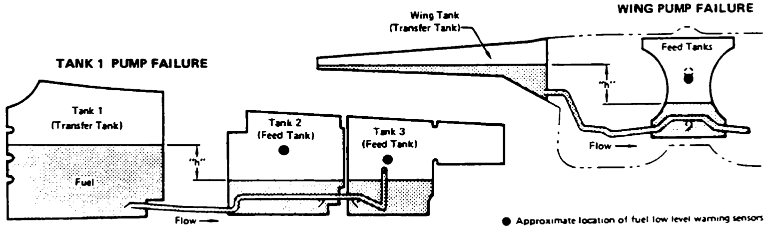


FIGURE 5 - EFFECT OF ENGINE FUEL FLOW AND AIRCRAFT PITCH ATTITUDE

(Tank 1 Transfer Pump Failure)

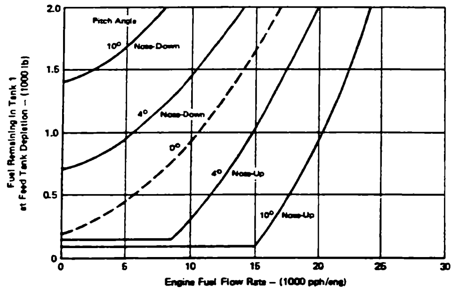


FIGURE 6 - EFFECT OF ENGINE FUEL FLOW AND AIRCRAFT PITCH ATTITUDE

(Wing Transfer Pump Failure)

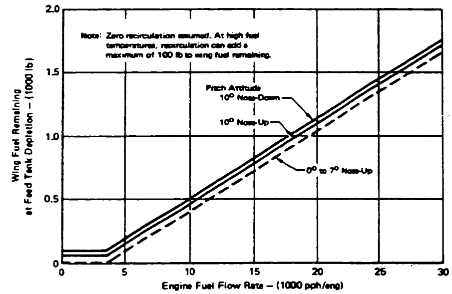
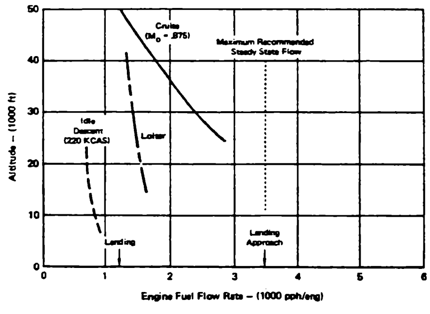


FIGURE 7 - TYPICAL ENGINE FUEL FLOW REQUIREMENTS

(Air superiority configuration)



cruise, loiter, and landing attitudes are normally within the 3° and 7° nose-up limit.

PUMP FAILURE DETECTION AND PILOT ACTION

Now that we have reviewed the F-15 basic fuel system and the gravity transfer system, we must answer the questions of how to detect transfer pump failures and what pilot actions are required once a failure is suspected.

The pilot can detect a transfer pump failure by monitoring the fuel quantity gaging system and the fuel low level warning light on the caution light panel. Any of the following indications should be reason to suspect a failed transfer pump:

- Premature low level warning.
- Fuel quantities remaining constant in one or more transfer tanks while others decrease.
- Feed tank fuel level decreasing with fuel remaining in any transfer tank.

If a failed transfer pump is suspected, the following action should be taken:

- Minimize fuel consumption and plan to be "on the ground" with normal fuel reserves.
- Do not exceed steady state power settings corresponding to 3500 PPH/ENG and attempt to maintain a 3° to 7° nose-up attitude by observing the pitch angle ladder on the head-up display. (Transients are acceptable outside these power setting and pitch attitude boundaries provided the aircraft is returned within the boundaries prior to feed tank depletion.)
- If failure occurs during takeoff or climb-out, continue to climb to safe altitude but do not exceed intermediate power.

TO CONCLUDE

The F-15 gravity transfer system allows an aircraft with one or more transfer pump failures to return safely to base. A measure of its performance is the amount of *unavailable* fuel left in the transfer tank at feed tank depletion. If the limitations are observed, the pilot can expect to land safely with no more than 150 pounds of unavailable fuel in any transfer tank having a failed pump. Sustained operation outside these limits results in more transfer fuel being made unavailable — that's sort of like early man trying to make water flow "uphill" from the stream to his fireside. ■

Contractor Case Study... (PUBLISHED 1982)

F-15 Engine Fuel Contamination

By ROGER SPETH/*Unit Chief, F-15 Fuel Systems Design*

A number of engine problems related to fuel contamination have occurred in the F-15 here at MCAIR and in the field during the past two years. The primary symptoms have been an inability to achieve augmentor operation and/or high throttle torque (spongy throttle) at engine cutoff. The problems have occurred most frequently prior to the second flight of new production aircraft (following a speed run on the first flight), or after extensive maintenance in the fuel system such as occurs at modification centers. Investigation of the engines has found the augmentor wash filter of the Unified Fuel Control (UFC) clogged with a mat of fibers and small particles. An extensive engineering and design analysis of the situation has been done that you may find interesting as an example of the approach a contractor takes to resolution of problems of this type.

FUEL FILTRATION SYSTEM

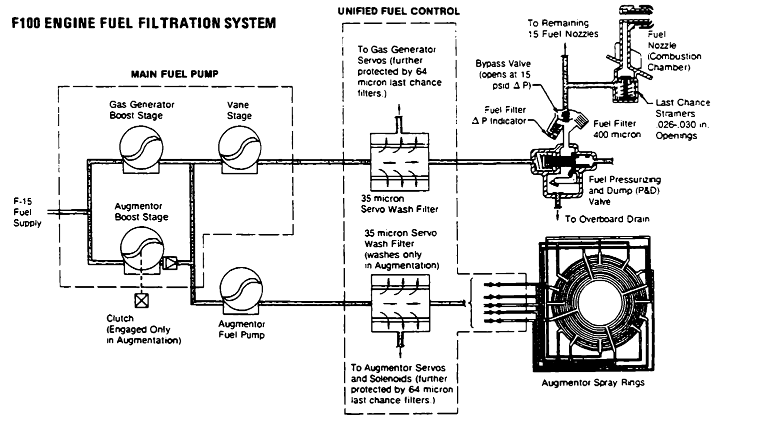
The path of fuel to the engines and the engine fuel filtration system is shown schematically below. Fuel to the engines is not filtered by the aircraft fuel system, for design reasons to be explained later. All fuel entering the engine passes through the main fuel pump gas generator or augmentor boost stage when in augmentation. Fuel flow to the gas generator is then further boosted by the vane stage before entering the UFC.

Within the UFC, fuel flow to the gas generator nozzles passes axially, unrestricted within a non-cleanable 35-micron cylindrical screen wall or servo wash filter, and is later filtered by a 400-micron cleanable woven-wire filter and integral "last chance" strainers at each nozzle. Some of the fuel passing within the servo wash filter flows off radially to the gas generator servos which control various engine functions. Both the axial flow, which tends to wash the cylinder walls of any collected debris, and the radial flow through the walls is continuous during engine operation.

When in augmentation, fuel pressure is increased by the augmentor boost stage of the main fuel pump, prior to entering the augmentor fuel pump. The augmentor fuel flow then passes axially within a filter of the same type as the one in the gas generator circuit and on to the augmentor spray rings. As with the gas generator fuel supply, radial fuel flow passes through the cylindrical filter wall to operate the augmentor servos, and flow to the augmentor servos is continuous whether the augmentor is operating or not. However, since the augmentor is only used intermittently during engine operation, there are periods of no flow to the augmentor spray rings during which the cleaning or washing action of the axial flow across the filter wall is eliminated. Thus at Mil Power and below, the augmentor filter acts as a "barrier type" filter which can be blocked by contamination. Contamination present in this circuit, however slight and whether introduced by the aircraft fuel system, the engine fuel system, or the fuel itself, can collect on the walls of this filter. If enough collects, flow to the augmentor servos is impeded and when augmentation is later selected, there is insufficient hydraulic muscle pressure to operate the augmentor servos to sequence the augmentor into operation.

Filter blockage in the UFC probably results from a combination of fuel contamination and a recent change to the servo wash filter design which better protects the servos from contamination, but has increased the filter sensitivity to contamination loading. The pore size of the wash filters in the UFC was changed from 64 to 35 microns without increasing loading capacity. In addition, the last chance filters at the servos within the UFC were reduced from 125 to 64 microns.

Data collected since the advent of the wash filter problem shows that problems can occur with as little as 27 miligrams of solid contaminant while other systems have operated satisfactorily until as much as 160 mg were collected. It should also be pointed out that problems do not routinely occur on every aircraft



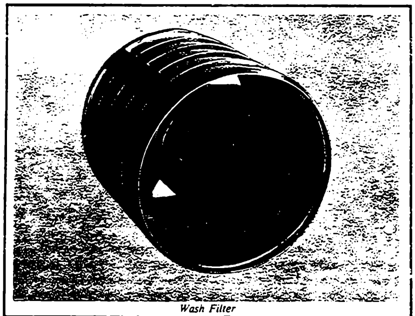
or every engine, with only about one engine in ten demonstrating an anomaly at MCAIR. However, our concern is that similar malfunctions may occur during field operations, particularly following aircraft fuel system maintenance or inadvertent servicing of an aircraft with fuel having high particulate or fiber contamination.

CONTRACTOR INVESTIGATION

Investigation at MCAIR has centered around the potential for contamination in the manufacturing process and aircraft operation. Various contamination sources have been considered including the fuel bladders, integral tanks, fire suppression foam, and the fuel source itself. Our fuel pits were inspected to assure conformance with MIL-STD-1518A requirements. Numerous meetings and discussions were held with the foam manufacturer and the various foam cutting subcontractors. These discussions covered all phases of foam handling from initial manufacture through receiving at MCAIR, including factory environment, cutting room environments, methods of marking for cutting, methods of cutting, packaging, and protection during shipment and storage. Meetings and discussions were also held with the fuel bladder manufacturer regarding cleaning and packaging of bladders. Finally, proper procedures were reviewed with our own shop personnel, including clothing worn and cleaning methods used on both the production and flight lines.

Many tests were run in our company laboratories to identify and quantify contaminants found in foam and fuel bladders. Wool and cellulose fibers have turned up frequently, which led us to look at such remote possibilities as introduction of wool fibers into the fuel by the felt tip markers used by one cutter to mark cutting patterns on the foam (a source which was never confirmed by the way). It was generally concluded from this work that while extreme methods could be used to extract minute quantities of contaminants from these parts, their cleanliness does meet accepted industry standards.

Two new production aircraft were selected at random from the MCAIR assembly line and subjected to extensive fuel sampling procedures at various stages in final production check out and flight test. A total of 39 engine fuel feed line samples were collected at various fuel flow rates from first defuel of the airframe to following the second



Wash Filter

aircraft flight. The highest contamination level measured was 1.4 mg/gal; most were below 0.4 mg/gal and all samples were acceptable for aircraft usage fuel. No trend was evident from the first sample to the last sample.

The investigation is continuing and emphasis is presently being placed on how to reduce the impact of the smaller pore wash filter on operations at MCAIR and in the field. The following procedures are currently being used during manufacture of new aircraft:

- Cleaning and packaging of fuel bladders with particular attention to lint and fiber contamination at the vendor.
- Proper bagging and storage of fuel tank foam. Foam is not unbagged until immediately prior to installation on production line.
- Use of lint free wipes, solvent, and vacuuming on assembly line.
- Increased attention to fuel supply filter cleaning and maintenance.
- Aircraft fuel system flushing during fuel gaging system calibration.

During flight test of new aircraft, engines are cycled into augmentation, if possible, at the conclusion of the first flight and at a fuel quantity of less than 3,000 pounds, in order to cause washing of the augmentor wash filter before landing. If an anomaly is noted in augmentor operation, or if a spongy throttle is noted on shutdown, the augmentor wash filter is changed before the second flight. If it was not possible to place the engines into augmentation on return, then a special augmentation run is made before the second flight. If an anomaly is noted in augmentor operation, or if a spongy throttle is noted on shutdown, the augmentor wash filter is changed before the second flight. (Unfortunately, cycling engines into augmentation is not a solution for the field for two reasons: inability to change the wash filter at Organizational or Intermediate levels and because of aircraft operating needs and scenarios.)

AIRFRAME MOUNTED FILTER

The question frequently arises, Why not simply put a filter on the airframe? The F-15 airframe/F100 engine fuel filtration philosophy is considerably different from past accepted practice. Prior to the late 1950's, aircraft fuel systems were designed with a large, low-pressure drop, airframe-supplied and mounted fuel filter. The filter usually consisted of a 200 mesh element with a 74 micron filtration capability. Typical examples of this design philosophy are the F-100, F-101, and F-105 aircraft, and more recently the F-16. A filter element bypass was either designed in or added on by retrofit to circumvent ice accretion problems associated with fine mesh filtration of cold, water contaminated fuel and subsequent loss of all engine power.

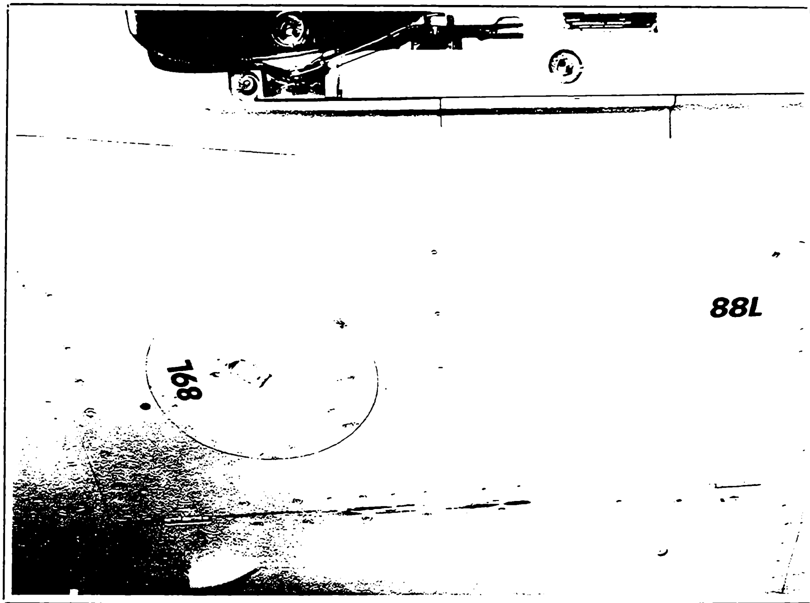
This type of filter would not resolve the current problem since the wash filter being blocked is 35 micron, much smaller than the upstream filtration. Also, bypass with icing would allow unfiltered fuel to be delivered to the engine. On more recent aircraft, the F-4, F-111, and F-14 for example, fuel filtration has been accomplished by 10 to 40 micron filters located on the engine between the engine supplied boost pump and the main engine high pressure pump. Such a fuel filtration concept results in a smaller filter due to the higher available operating pressure from the engine supplied boost pump, thereby enabling finer filtration matched more closely to the specific engine requirements. The change in primary fuel filtration from airframe to engine location has been made possible by greatly improved base fuel supply quality.

Fuel filtration on the F-15 airframe consists solely of a screen at each airframe boost pump inlet. These screens are 8 mesh, .023 inch diameter wire and result in a nominal filtration capability of 2600 microns. The F100 engine fuel filtration consists of 1500-micron non-serviceable basket type filters located in the various main fuel flow passages of the engine fuel control. Except for small wash type filters for the engine fuel control servo flows, and the 400-micron filter upstream of the gas generator nozzles, 1500-micron filters are the finest fuel filtration provided. These obviously will remove only gross contamination, and are insensitive to ice accretion and resultant filter element blockage.

The assumed design for an F-15 airframe-mounted fuel filter would place it in a portion of the fuel line in the engine compartment which necessitates stainless steel construction. This combined with the fact that the filter must accommodate all fuel flow (main engine plus augmentor) results in a unit which is both physically large and heavy. The added pressure drop increment caused by this fuel filter would require a redesigned airframe boost pump for a higher pressure rise. Additionally, this filter could remove only airframe-generated contaminants. The weight, cost, and system redesign penalties are not justified at this time.

CONCLUSIONS

While we are certainly concerned with contamination introduced into the aircraft system during manufacturing and maintenance, fuel specifications allow small quantities of contaminants. The F100 engine has been designed and qualified to operate satisfactorily under those conditions, and continues to do so except in the few instances noted in this article. Significant work has been done by MCAIR to insure delivery of "clean" aircraft, and test results show that the F-15 provides fuel to the engines that is considerably cleaner than requirements for fuel procurement, USAF fuel supply systems, and the engine itself. It is our opinion that, along with the steps being taken during manufacture here in St. Louis, problems in the field can be minimized through careful attention to the quality of fuel used and to avoiding introduction of contamination in the fuel system during maintenance. ■



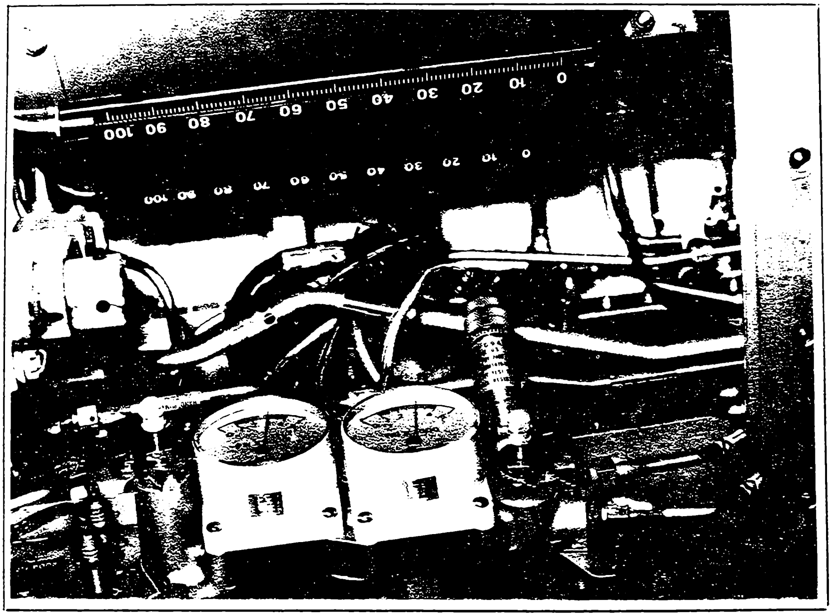
THE SECRETS BEHIND DOOR 88/89L

One of the most pleasant aspects of the fighter airplane business is the extent of cooperation between maker and user. (It may be that today's aircraft are so complex that everybody has to cooperate just to stay ahead of the game, but "why" is not so important as "how.") For example, we recently received a manuscript from an Air Force pilot, Captain Ken Fox, who is an F-15 instructor with the 58th TTS at Luke AFB, Arizona (sent to us by Charlie Osha, one of the McDonnell Reps there). The first paragraph of the captain's draft read as follows . . .

> This article is the result of research accomplished at Luke AFB. The Jet Fuel Starter (JFS) became a high interest item at Luke because of many JFS related ground aborts. The problems associated with the JFS ranged from low servicing/leaks, failure to start, failure to engage, to failure to disengage/shutdown. This article is not intended to make crew chiefs out of fighter jocks. It is, however, designed to make pilots aware of JFS operation, limitations, servicing requirements, etc. In that regard, we as pilots can help QC the JFS system by becoming more knowledgeable.

In its original form, Captain Fox's article applied to some specific problems at Luke. While we were certainly pleased to receive the data, we felt there was also an opportunity to expand the intent of his material to include some statements on the engineering aspects of the JFS. Therefore, we asked Mr. John Killoran, a Lead Design Engineer on the F-15 Project, to amplify some of the areas which Captain Fox's comments addressed, to make them more universally applicable to all F-15 users.

In our opinion, the final article presented here is an excellent example of maker/user cooperation with a purpose — cooperation between a USAF officer who recognized a problem; MCAIR field reps who provided liaison; and a MCAIR engineer who appreciated the opportunity to explain a complicated aircraft system. Captain Fox noted that his article was "not intended to make crew chiefs out of pilots," nor are Mr. Killoran's additions "intended to make engineers out of crew chiefs," but everybody should know more about the Jet Fuel Starting system after reading this material. ▶



Problems with JFS starting, and damage to secondary power system components during JFS starting, continue to occur in the F-15. Improper JFS hydraulic accumulator servicing, as well as starting and operating procedures, frequently contribute to these problems. This article is offered as a refresher on the workings of the hydraulic start system, and how it affects proper JFS starting.

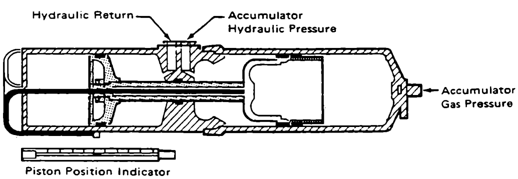
The JFS hydraulic start system major components are two hydraulic start accumulator bottles, two piston position indicators (attached to the accumulators), two compressed gas pressure gauges, all located behind Doors 88L/89L; and a hydraulic start motor mounted on the Central Gearbox (CGB). The accumulator bottles are of the cylinder and piston arrangement (see Figure 1) with nitrogen gas on one side of the piston and hydraulic fluid on the other side.

When an accumulator is discharged, the compressed gas drives the piston forward, forcing the hydraulic fluid through the start motor to spool-up the JFS to the point to where it can self-accelerate to idle. Proper servicing of the accumulators is necessary to ensure that sufficient hydraulic fluid, at the proper pressure for the prevailing ambient temperature, is delivered to the JFS.

The Piston Position Indicator (PPI) shows the percentage of fluid, by volume, in the bottle. Gas "precharge" is the initial amount (pressure) of gas available to be compressed when there is no fluid (zero PPi) in the bottle. Gas pressure is read on the two gauges inside Door 89L: the forward gauge measures pressure in the top accumulator, the aft gauge measures the pressure in the bottom one.

Proper servicing is achieved by first verifying that gas precharge is correct, and then adding hydraulic fluid to further compress the gas, thus providing adequate starting energy. Therefore, the PPI reading is just as important as the gas pressure gauge readings. As fluid is added, both the gas and the fluid are compressed, changing the volumes and pressures. These relationships are further affected by ambient temperature; a chart showing these temperature/pressure relationships is presented in T.O. 1F-15A-2-2-2 as Figure 2-7, Sheet 4. This chart further defines minimum

FIGURE 1 – JFS HYDRAULIC START ACCUMULATOR



servicing levels required to assure adequate JFS start energy; and is based on a constant quantity of nitrogen in the accumulator, so that once the correct gas precharge has been established, a no-leak system will not require reservicing with gas as the temperature changes. The zero PPI line on the servicing chart represents the resulting precharge pressure at various temperatures.

The T.O. chart and instructions must be used to service the system, but a handy preflight reference decal is available on the inside of Door 89L (or on the accumulators in some earlier aircraft). This decal, represented in Figure 2, can be used as a quick reference to determine if your JFS system is properly serviced. Look from the left side of the decal (ambient temperature) horizontally until you come to the block containing the number corresponding to the value read on the PPI scale. Then read vertically and you will find the minimum gas pressure required for JFS start. The photo on page 10 shows a properly serviced system at 55°F: PPI reading = 60, gas pressure = 3000 PSI.

If the gas pressure is low, this indicates a low precharge, and nitrogen servicing is required. Since the gas pressure gauge reading depends on both the gas precharge and the amount of fluid in the accumulators, the accumulator that requires gas servicing must be discharged (zero PPI) before setting the gas precharge. If the PPI reading is below the minimum value shown on the chart, fluid servicing is required. Remember, servicing the system with hydraulic fluid, either by hand pump or by a cart, only increases the fluid volume and has no effect on the gas precharge.

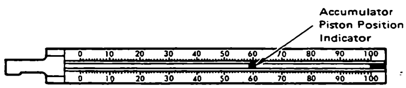
Proper accumulator servicing, both gas and hydraulic, is necessary for a number of reasons:

- Low precharge (high PPI reading) gives marginal start energy at low temperatures. Even if the JFS starts, it may not accelerate properly, causing extreme heat build-up which may result in damage to the gas generator.
- High precharge (low PPI reading) causes early start assist termination at high temperatures, again with the possibility of gas generator damage.
- The upper accumulator is also used for emergency braking, while the lower bottle also operates emergency steering and gear extension. Improper servicing may result in decreased capability for these vital systems.

In addition to proper servicing, there are several operating procedures which, when used regularly, will aid in problem prevention:

- Although simultaneous discharge of both start accumulators will not cause damage, it will create the need for manual hydraulic servicing if the JFS fails to start. Therefore it is not recommended except under certain low temperature conditions as detailed in T.O. 1F-15A-1, Section VII, "Cold Weather Operation", and then only on pre-TCTO 1F-15A-753 aircraft.
- Discharge of an accumulator while the JFS is turning may result in damage to the CGB hydraulic start motor and/or overrunning clutch. This can be prevented by waiting 20 seconds after the JFS fails to start before discharging the second bottle, thus allowing the JFS/CGB time to stop. This 20 seconds is a good time for the crew chief to check for leaks, verify accumulator pressures, and check for fuel at the JFS fuel drain.
- The CGB isolation decouplers have a centrifugally actuated flyweight switch which prevents extension if they are rotating. During the engine start sequence you must wait about 10 seconds after the first engine has reached idle before engaging the second engine. This gives the decouplers time to stop rotating, which must occur to permit the engagement of the second engine. This 10 seconds is the maximum time for decoupler braking, and has been shown to be longer than required for a CGB with a new clutch/brake assembly.

FIGURE 2 - PREFLIGHT REFERENCE DECAL



USE CHART FOR PREFLIGHT CHECK ONLY

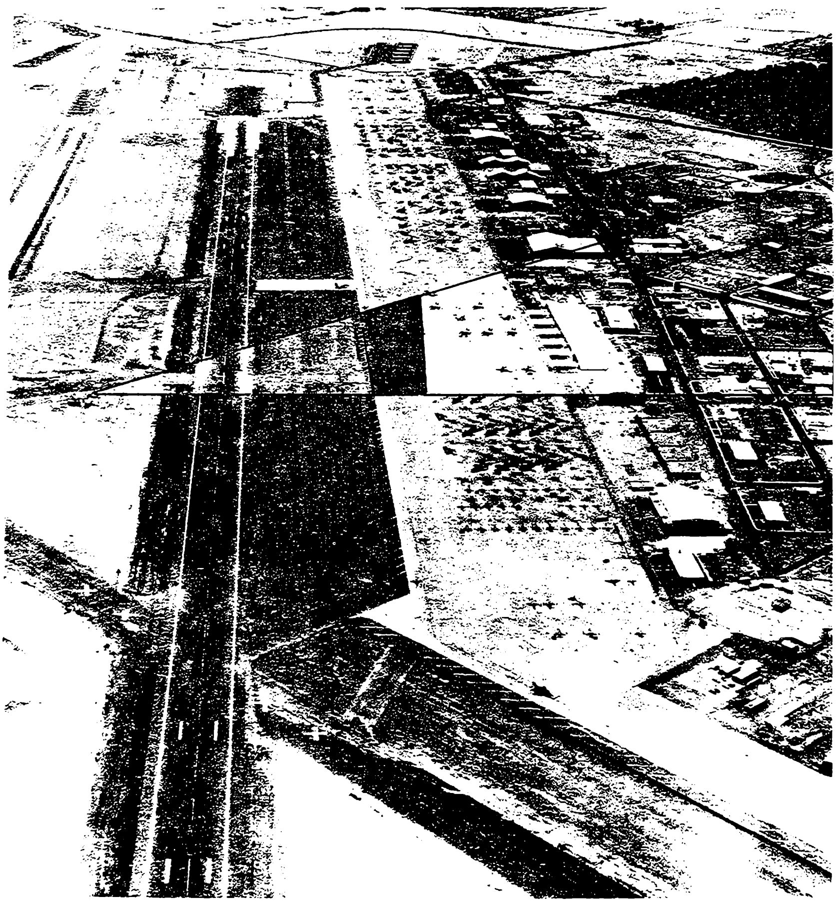
MINIMUM GAS PRESSURE (PSIG) vs MAXIMUM INDICATOR READING REQUIRED FOR JFS START

AMB TEMP °F	REFER TO T.O. 1F-15A-2-2-2 FOR SERVICING								
	2500	2600	2700	2800	2900	3000	3100	3200	3300
120					44	47	50	52	55
110	SERVICING REQ'D			43	46	49	52	54	56
100			42	45	48	51	54	56	58
90		40	44	47	50	53	56	58	60
80	39	43	46	49	52	55	58	60	62
70	41	45	48	52	55	57	60	62	64
60	44	48	51	54	57	59	62	64	66
50	46	50	53	56	59	61	64	66	68
40	49	52	56	59	61	64	66	68	70
30	52	55	58	61	64	66	68	70	72
20	54	58	60	63	66	68	70	72	74
10	57	60	63	66	68	70	72	74	76
0					70	72	75	76	78
-10					73	75	76	78	80
-20	SERVICING REQ'D				75	77	79	80	82
-30					77	79	81	82	84
-40					80	81	83	84	86

EDITOR'S NOTE: Captain Fox's article was originally prepared as a part of an intensive training program conducted at Luke AFB to reduce the JFS-related ground aborts being experienced there. According to MCAIR Rep Charlie Osha, this program (which also included classroom training, recertification training, OJT, and numerous spot inspections) has eliminated the greater portion of the improper servicing cases. One part of the training program may be of special interest to other F-15 users — a video tape on JFS problems and servicing. Prepared by the 58th TTS and directed toward aircrews only, it is informal and deals specifically with the problems encountered at Luke, but should be of real value to all F-15 users. The tape is available directly from Captain Fox, 58th TTS/F-15, Luke AFB, Arizona 85309 (Autovon 853-2329). ■

THE "NELLIS TRIANGLE"

(PUBLISHED 1978)



In the fighter airplane business, ninety-nine out of a hundred problems are really "no problem" — the causes are clear; the analyses obvious; and the solutions logical. But that number one-hundred! That's the way it was recently in a perplexing problem with the F-15 brake system. While 99 routine problems with the Eagle at Langley, Bitburg, Edwards, Holloman, and Luke were being just as routinely disposed of, Nellis AFB was wrestling with number 100, in a situation that for awhile was being likened to the famous and mysterious "Bermuda Triangle." While airplanes were not actually disappearing at Nellis, they certainly were doing some odd things. Here is the way the peculiar case was originally described in the first Trouble Report submitted to St. Louis by MCAIR Rep Phil Royer, back in February of 1977 . . .

The pilot made a writeup against F-15 75-0054 as follows: ". . . Brakes failed during taxi. Had normal braking after anti-skid turned off . . ."

"The pilot made a writeup against F-15 75-0054 as follows:" . . . Brakes failed during taxi. Had normal braking after anti-skid turned off . . ."

The entire brake system and anti-skid checked good in maintenance. The skid controller was changed (because we had one). Next flight the same writeup. This time the skid controller was once more changed and both Main Landing Gear (MLG) struts were serviced. Following this corrective action, a taxi check was requested. The pilot once more reported the same problem. Both MLG Weight-on-Wheel (WOW) switches and adapters were then changed. Again the aircraft was taxi-checked and again the brakes failed. However, this time the pilot reported that the brakes failed at only one area on the taxi-way.

A sergeant from the Electronic Shop climbed into Equipment Bay No. 5 and the aircraft taxi-checked again. At the same point on the taxi-way, the brakes would consistently fail. Several pilots and maintenance people confirmed that fact. It was determined that additional taxi checks were necesssary, and being one of the more vocal of the non-believers, I was invited to get headset, Y-cord, relays, extra skid controller, and test equipment and climb into Bay five.

The aircraft was taxied the entire length of the field (approximately two and one-half miles) twice. Only at this one location did the brakes fail. The aircraft was then taxied past "this" location 12-15 times and on each and every pass, the brakes failed. The area was later stepped off and found to be 120/150 yards in length. When the brakes would fail, the pilot could and did apply full braking with no effect on the aircraft. As the aircraft would exit the "ghost area," the brake pressure would slowly return, bringing the aircraft to a rather gentle stop.

During these failures, the relays from the WOW switches were removed one at a time and the input pin checked for voltage using a meter. No voltage was indicated and this had no effect on the loss of brakes. A spare controller was then slaved in and still the brakes failed. The wheel speed detectors were then disconnected one at a time and the brakes were still lost when taxiing in this area. Later both speed detectors were disconnected and the signal lead shorted (pin a to pin c). This also had no effect. During the only test conducted in my absence, a sergeant from the Electrical Shop rode in Bay 5. He noted that with both relays removed, the brakes would not fail in this area.

Another aircraft (75-055) was checked and found to have the identical problem. While testing this aircraft, the pilot turned anti-skid off, stopped in the area, applied full braking, and then turned anti-skid on. The aircraft started to roll almost immediately. The pilot allowed the aircraft to continue; and as he left the area the brakes slowly returned. The ACEVAL/AIMVAL pilots were requested to check their aircraft ('74 models) and they reported that 74-120 and 74-124 also had this problem. We checked with the 1st TFW, which was TDY here from Langley for RED FLAG, and they reported brake loss in the same area.

From all of this, very little has actually been learned. One of the few facts established is that there is an area roughly 120/150 yards in length where the anti-skid, if in the "ON" position, will dump all brake pressure. It has also been determined that with the anti-skid "OFF," there is never a loss of brakes. Also, this problem is on several and possibly all F-15A/B aircraft. The sergeant is positive that the loss of brakes is being signaled somehow through the WOW switch circuitry.

Base Civil Engineering checked the area and stated that there are no buried power cables at this location. The Base Facilities Manager and a representative from the Communications Squadron are presently trying to determine if any unusual RF energy is in this area. Meanwhile the aircraft continues to fly and the pilots continue to complain."

And now the story is picked up by MCAIR Engineering . . .

▶

"TRIANGLE"

BY ROBERT ASTON/ *Senior Engineer-Electronics*

After all local efforts failed to resolve the problem, MCAIR was invited to investigate this mysterious "brakes failed while taxiing" case. Our Electromagnetic Compatibility (EMC) Engineering group was assigned the task of resolving this unusual problem. Realizing that this type of problem must be researched at the site, a visit to Nellis AFB was arranged.

Upon arriving at Nellis, a review of the circumstances involving the so-called "Ghost Area" was conducted. A visit to the area was arranged for a first-hand view. One of our first observations was of a GCA antenna located between the parallel runways and approximately 350 meters from the taxiway. After plotting the location of the antenna in relation to the affected area on the taxiway (150 x 30 meters), our suspicions became aroused with respect to the effect of the GCA antenna energy output on the brake system components.

Since aircraft 75-0054 had previously experienced brake failure in the mysterious triangular area, it was assigned to us for test purposes. A special EMC monitoring device was attached to the aircraft brake system to observe any abnormalities while taxiing through "the" area. Several passes were made and the results indicated that the Weight-On-Wheels (WOW) switch was "changing state."

The anti-skid control was inputting this change in state as aircraft having just landed (touchdown protection) and, in the absence of a wheel speed greater than 45 knots, correctly dumped brake pressure for approximately 5 seconds. This delay protects against inadvertant brake application prior to landing and allows sufficient time for the wheels to spin up before braking. Thus our system was doing the right things, but obviously for the wrong reasons! Why was it happening while taxiing? The GCA (search radar) was turned off as our first step in the process of isolating the cause of the problem. With the GCA turned off, the brake system worked perfectly. (The same results were observed when the Anti-Skid System was turned off.) Conversely, the brake system failed only while these systems were operating simultaneously.

Using a field intensity meter, we determined the field strength of the GCA radar. It was recorded as 10 volts per meter at a GCA radar frequency of 2.8 GHz with a minimal variance either in or out of "the" area. (It should be noted that the F-15 WOW switch is qualified to 20 volts per meter at the 2.8 GHz frequency.) Armed with the data accumulated during the investigation and feeling that the solution was in sight, we returned to St. Louis.

Our intention was to duplicate the problem in the EMC Engineering laboratory, analyze the data, and recommend a positive fix. Unfortunately the problem could not be duplicated, either in the lab or on an aircraft; not even when the level of radiation was raised several orders of magnitude. Since the Nellis problem was a major concern to the 57th TTW Squadron Commander and of growing concern to MCAIR, a second visit to Nellis was immediately scheduled to try some different fixes.

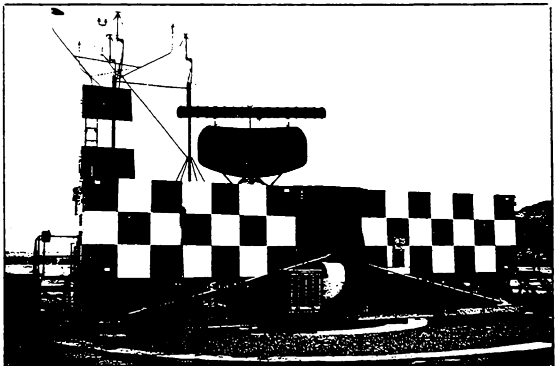
By installing an "old" type two-piece (without diode) WOW switch with the back plate removed, we were able to confirm that the problem was actually caused by oscillations in the WOW switch which were being impulsed by the one microsecond pulses from the GCA search radar. Several different fixes were tried during this visit before we got one that worked; the fix consisted of adding shielding to the four wires from the WOW switch to the splice area.

Since this was an experimental fix, it was removed from the test aircraft after successfully completing a series of taxi tests. The test aircraft wiring was returned to its original configuration and we returned to St. Louis to initiate the required paperwork to get this fix incorporated as soon as possible.

Everyone agreed that the "brakes failed while taxiing" problem was finally solved. But - unfortunately and for some unknown reason our "positive" fix encountered a flaw, a Murphy, or something. The interim fix was approved by the Air Force but when it was incorporated on the original Nellis test aircraft by Air Force personnel it "failed" the taxi test. Needless to say, back to Nellis we went!

By this time we were convinced that Nellis was the victim of some sort of sorcery. Why didn't the fix work for the Air Force Personnel, the same as it did for us? We were soon to find out why. While carefully reversing the procedure for installing the fix, we discovered that the splice area *was* shielded. We remembered that the fix we installed during our previous visit *did not* contain shielding in the splice area where the WOW switch wires are spliced together with those of an aircraft electrical cable assembly. (Apparently the added shield helped tune the noise, thereby increasing the coupling rather than isolating the noise.)

Several other fixes were tried and then we installed a different cable assembly (from the splice to the connector) which carries three groups of wires and their respective shields



GCA radar at Nellis AFB. Located between the parallel runways and approximately 350 meters from the taxiway, its frequency is 2.8GHz and power output is 500KW.

through from their beginnings to the connector. This cable assembly splices with the shielded wires from the WOW, Down Limit, and Wheel Skid switches. It was installed on the second shift and unknown to us, Murphy struck again; the WOW switch was also replaced because of a frayed shield. We tested this fix and it tested good, i.e. no more brake problems - so we thought.

Once again we intended to return the test aircraft to its original configuration. We then found out that the old cable assembly had been cut off the aircraft and it was too short to be reinstalled. Here we were with one serviceable cable assembly (unapproved modification incorporated), one unserviceable cable assembly, and an aircraft scheduled to fly the next day. How did we get into such a predicament?

In order to return the aircraft to its original condition and flyable, we had the experimental cable assembly modified to the original Air Force approved configuration. As we were doing this we also found that the old type two-piece (without diodes) WOW switch had been replaced by the new two-piece (with diodes) configuration. This made the results of the last test questionable; but we did feel that we had come up with a foolproof fix this time.

Who was it that said, "never count your chickens before they hatch"? Inasmuch as several months had elapsed since the initial pilot squawk, and after many encounters with frustration, the case of the "Nellis Triangle" was once again considered closed. Unfortunately, the old cliche, "time cures all ills" worked in reverse in this case. Our illusions of conquest over the imaginary EMI beast were soon to be shattered.

MCAIR had prepared an Engineering Change Proposal (ECP 704); the Air Force approved the change, and the healing hardware (one-piece WOW switch) was installed in Block 16 and subsequent production aircraft. Everything was going along real fine until one day in November 1977 when a message was received from Luke AFB citing a "brakes failed while taxiing" problem involving several Block 16 aircraft.

Our first observations at Luke confirmed an educated suspicion that the EMI environment there was different from the EMI environment at Nellis. In addition to the standard AN/APM-13 GCA radar set, Luke AFB also had an AT/APS-44 GCI radar operating approximately 800 feet away from the jinxed taxiway. The GCI unit measured at 1.3 GHz with a power output of 320 volts/meter in comparison with the standard GCA unit measured at 2.8 GHz with a power output of 10 volts per meter.

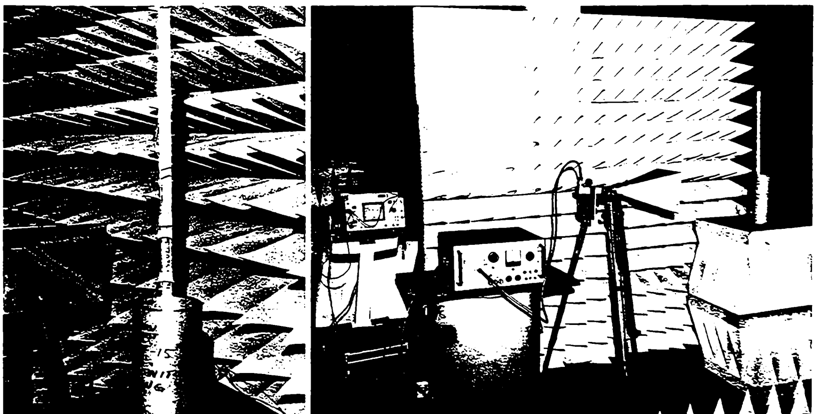
A new dimension was thus added to the basic EMI problem; and after unsuccessfully attempting several fixes, we decided to return to St. Louis for detailed tests. With the use of high power pulse generators in our EMI lab, we were able to duplicate the WOW switch malfunction encountered at Luke. The results of these tests produced two possible solutions to the problem:

Develop a new WOW switch with EMI feedthrough type filters on all input and output lines.

Install a time delay relay in the anti-skid system to obscure its sensing of the short changes-of-state of the WOW switch.

After considerable deliberation, it was decided to go with the new feedthrough filters. The new filters were successfully tested in our lab to levels as high as 700 volts per meter, but the real test came when the fix was installed in a test aircraft operating under the same EMI conditions. This test was performed at Holloman AFB. During the period between our investigation at Luke and the latest proposed fix by MCAIR, EMI brake failures had also occurred at Holloman AFB and Alconbury AS, England. Fortunately, these failures were basically the same as we had encountered at the other bases; therefore our previously proposed fix would not be affected.

Using the Air Force restrictions governing Electro-Explosive Devices (EED's), the minimum antenna-to-runway/taxiway separation was determined and the highest field produced was computed to be 500 volts per meter. We are now in the process of getting the new switches, with feedthrough type filters, qualified to levels of 700 volts per meter. Thus no matter where our aircraft operate or what frequency the radar uses, the WOW switches should not be affected. ■



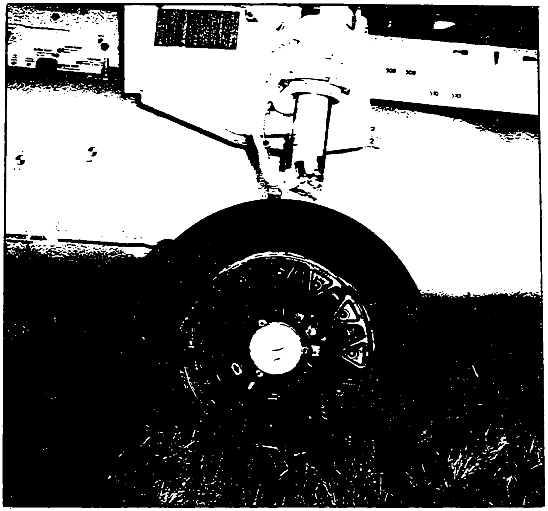
Lab set up used to simulate failure mode of WOW switches. A NARDA POWER PULSER excites standard gain horn, in the frequency of 3-5 GHz, which radiates on the simulated strut and WOW switch. Radar Absorbing Material (RAM-ECCO-SORB) is in the background.

(PUBLISHED 1983)

ANTI-SKID

The October 1982 issue of TAC ATTACK magazine discusses the latest of a long series of "don't trust the anti-skid" incidents involving F-4 and F-15 aircraft. It was a by-now familiar story; an Eagle pilot aborted a takeoff and . . . (1) applied the brakes, (2) thought the anti-skid system wasn't working, (3) turned the system off, (4) reapplied the brakes, (5) blew a main tire and destroyed a rim when the wheel locked up, (6) stopped eventually, shaken but unhurt, with a now out-of-service Eagle.

There was, as the magazine article notes, nothing wrong with the anti-skid system except the pilot's confidence level. Things just didn't "feel right" to him so he decided to put his faith in his feet, so to speak. However, we should be reaching an end to blown main tires with the introduction of the . . .



F-15 Pulser Braking System

By RAYMOND H. EHLE/*Senior Design Engineer*

In DIGEST issue 4/80, we gave you a broadbrush description of a proposed improvement to the F-15 wheel braking system. Called the "Pulser" system, it was in the development stage back then, but has since come a long way and is now going in both production and in-service aircraft, as described in this article. This improvement involves a new cockpit switch, brake pulser unit, electrical circuitry changes, some one-way restrictor valves, and a valuable new capability when it comes to "slowing down the Eagle."

Since the six-step "procedure" described above is almost guaranteed to follow the same path to a direct confrontation with the C.O., you who may be about to experience a real (not very often) or perceived (quite often) failure of the Mark III Anti-Skid System should be very interested in learning more about pulser.

IMPROVED BRAKING SYSTEM

The "pulser" system is intended as a primary back-up for the anti-skid system. Production incorporation began in Block 29 (F-15C S/N 80-0039; F-15D S/N 80-0058); retrofit is planned for all delivered A/B/C/Ds by TCTO 1F-15-763 (Improved Braking System), beginning in February 1983.

Service history has confirmed the extreme difficulty a pilot incurs in successfully managing the brakes at high speed without anti-skid protection. The probability is high for blowing one or both main tires under these conditions. However, the new pulser system will significantly reduce the likelihood of sustained F-15 wheel lock-up and consequent tire blow-out as a result of an anti-skid "off" situation. Automatic changeover from anti-skid to pulser in the event of a failure reduces pilot workload while providing maximum protection for the tires. All wheel brake/anti-skid features of the previous system are retained.

The heart of the pulser system is a 2Hz square wave generator (brake pulser unit) which cyclically dumps pilot metered pressure through the skid control valve. For each complete cycle, the brakes are applied for about 2/3 cycle and dumped for 1/3 cycle. The pilot feels this pulsating braking action in his harness as a series of sharp "jerks" which vary in intensity (force) according to pilot metered pressure and runway conditions. Pulser system test data from runway stops shows that the pilot acquires a much better "feel" for how much brake pressure to meter, and consistently meters below the skid level. But even if conditions demand a maximum-effort stop, the tires will be afforded maximum protection from blow-out. In addition to providing a "feel" for skidding tires, the distinctive sensation of the pulser assures the pilot that the back-up system is working.

Aircrews moving from an Eagle not equipped with the system to one that has it installed will not find any major changes in the cockpit. The only physical change has been removal of the old two-position anti-skid switch on the miscellaneous control panel on the left-hand console. In its place, you'll find a three-position switch marked "NORM," "PULSER," and "OFF" –

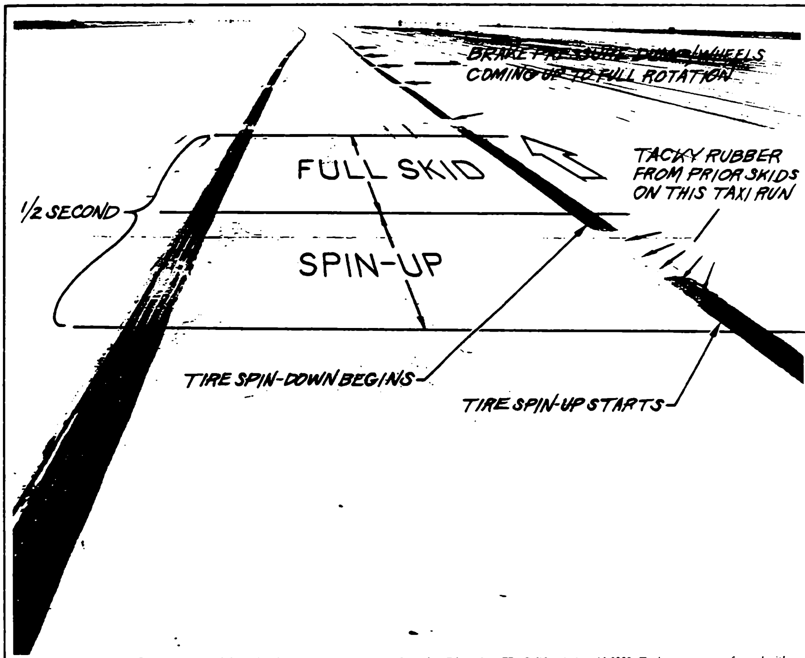
- NORM – With the switch in this position, normal anti-skid will be operating, plus the system will have the back-up pulser feature available in the event of a skid control system component failure. If a failure occurs and it is sensed by the skid control box, the pulser system logic will turn off the box and direct power from the brake pulser unit to the skid control valve.

• PULSER – Manually placing the anti-skid switch in this position turns off the normal anti-skid system and activates the brake pulser system.

• OFF – Setting the switch to this position turns off both the normal anti-skid and pulser systems. In this mode, only direct pilot-applied brake pressure is available.

In the event the anti-skid system detects a malfunction and automatically energizes the pulser system, the system can be returned to normal anti-skid by manually selecting PULSER, then NORM. If a malfunction still exists, the system will automatically return to pulser. If you encounter an anti-skid failure, the best technique is to select PULSER (if time permits) when the anti-skid light comes on and then switch to OFF after slowing to taxi speeds. The aircraft cannot be completely stopped with pulser brakes; it will continue to move forward very slowly if you attempt to park with the mode switch in PULSER.

One-way restrictors were added to both the normal and emergency brake lines to "slow down" the brake pressure application rate. What this means to the pilot is that he may feel a little difference when managing the brakes. With the one-way restrictors, there is less tendency for the brakes to grab the first time emergency brakes are applied.



"Tale of the Tires." Extensive tests of the pulser braking system were conducted at Edwards AFB, California in mid-1980. Taxi tests were performed with F-15C 78-648 at various gross weights and speeds up to 134 knots and 54,000 pounds. Photograph above shows typical "evidence" left on the runway of a full brake application test in which the pulser system brought the 35,000 pound aircraft to a safe, no-blown tire stop in 2000 feet from an initial 90-knot start. Engineering analysis of tire marks has been added to picture to indicate how each of those hundreds of black marks on your runway has its own story to tell.

Maintenance technicians will notice the addition of a few new components in the wheel brake system. Two new electrical boxes are located in the right outboard forward section of the nose wheel well. The forward box is the brake pulser relay panel, just aft is the brake pulser unit. Each box is tied into the wheel brake electrical system with one electrical connector. Four one-way restrictor valves are added to the wheel brake hydraulic system. One is located in each of the normal/emergency brake lines, downstream of the power brake valve.

During certain steps of the functional checkout and troubleshooting procedure, you will notice a loud "banging" noise. This is not abnormal or unusual, and is nothing more than hydraulic pressure being dumped when the pulser system is activated.

LANDING GEAR CONTROL AND INDICATION IMPROVEMENT

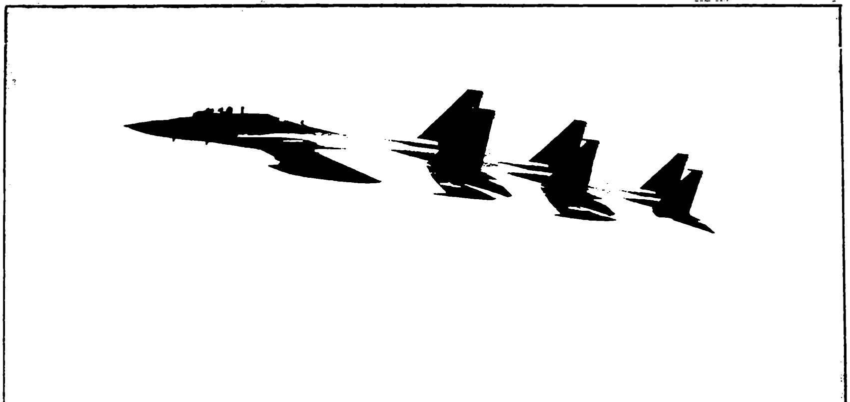
Prior to incorporation of the pulser system, the anti-skid system and the anti-skid warning light would be off any time the landing gear handle was up and the gear was down and also when ▶

the landing gear circuit breaker was pulled or popped. Along with the pulser system, a change was also made to make the anti-skid light come on for both of those conditions, i.e., warn the pilot that the anti-skid system was *not* on. Under these circumstances, the pilot would have to *select* pulser because there was no anti-skid protection and the system would not automatically switch to pulser.

There are two situations when the landing gear handle is up and the gear down. The first occurs only during a functional check flight (FCF), when the handle must be up in order to flight test the emergency gear system. The second occurs during normal gear retraction. At the moment the gear handle is placed in the up position, the gear is down and will remain full down until the gear doors open. During this time interval, the anti-skid light would be on, dutifully warning the pilot that the anti-skid system is off. We noted, however, that this condition could cause concern to pilots seeing an anti-skid warning light and a master caution light during the busy period of takeoff, even though there was no problem in the system.

To eliminate this misleading anti-skid warning light condition, a change was made on Block 31 production aircraft F-15C S/N 81-0039 and up and on F-15D S/N 81-0065 and up. Also, there will be a full A/B/C/D retrofit by TCTO 1F-15-791 (Landing Gear Control and Indication Improvements). With this change the anti-skid system will be on any time the gear is down, regardless of the position of the gear handle. The anti-skid light will only illuminate to warn of an anti-skid failure. A particular "failure" of which pilots need be mindful involves the landing gear circuit breaker. If the landing gear circuit breaker (pedestal panel) is pulled or popped, the anti-skid light will be on, warning of a non-operative anti-skid system. Under this condition, the automatic change-over circuitry is also inoperative and the pilot *must* select PULSER.

The "pulser" system has been a joint USAF/MCAIR development, and should prove to be a real time and money saver, not only because of fewer tire blow-outs but because of a reduction in the incidents that usually occur after a blow-out. . . such as the one below. ■



F-15 "Threagle." Holloman AFB, New Mexico appears to have come up with an extremely long range air superiority fighter, if you could believe there is such a thing as a one-cockpit, three-wing, six-tail Eagle! In truth, it was some exceptional formation flying by three pilots of 49th Tactical Fighter Wing, captured in this exceptional photograph by Douglas photographer Harry Gann back in 1978.

some practical pointers on the...

F-15 INERTIAL NAVIGATION SYSTEM

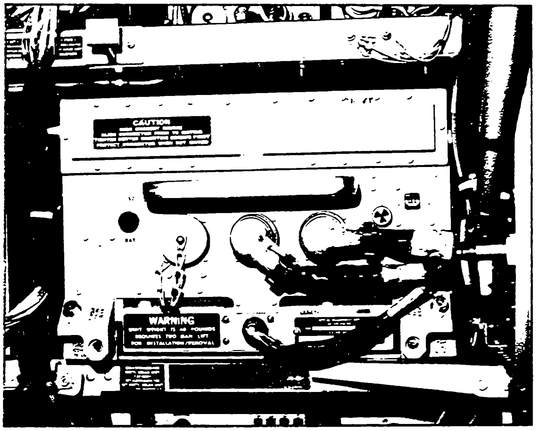
(PUBLISHED 1983)

Part I

By HARRY LYONS/*Field Service Engineer, Kadena AB, Japan* and CARMON D. THIEMS/*Senior Electronics Engineer, St. Louis*

The Eagle aircrew and maintenance manuals are pretty thick documents, and information on the INS (Inertial Navigation System) takes up its share of space in both the flight and shop T.O.s. Even so, there are many "fine points" about care and handling of some of the sensitive components, about the reasons behind some of the system precautions, and about some of the techniques prescribed in the T.O. procedures, that the official manuals just don't have room to get into.

Therefore, the 18th TFW Stan Eval section at Kadena AB, Japan recently asked MCAIR avionics rep Harry Lyons to prepare a discussion of the F-15 INS from a pilot's point of view. His material was published in the "BEAK & TALON" (wing ops bulletin), and was reprinted in similar bulletins at a few other F-15 bases. In the meantime, Harry sent his material in here to the St. Louis Home Office for our Avionics Department to review and expand upon from a design/engineering standpoint. MCAIR had recently conducted an extensive base-level study into the causes of excessive "CND" rates for several F-15 avionics components (including the INS), and reported the results to the Air Force in MDC Report A6512. Several meetings were also held on the subject between USAF, MCAIR, and Litton Systems Inc. (system vendor); and visits were made to various Air Force bases. Results of these investigations have been incorporated in this expanded discussion of the "AN/ASN-109 Inertial Navigation System" - the avionics system that tells the pilot where he is now, where he's going, and how to get there. Now in two parts because considerable information on maintenance and troubleshooting has been added to the original pilot-oriented data, Part I examines some mysterious CND problems with the IMU (Inertial Measurement Unit) of the INS; Part II will offer some cockpit INS alignment tips. Aircrews, flightline maintenance, and avionics shop personnel should all be interested in these two articles.



Inertial Measurement Unit (IMU) installed in Door 3R area. Note cautions and warnings.

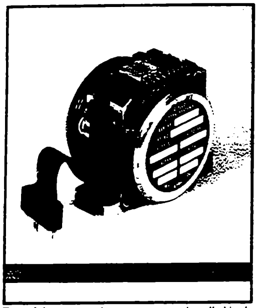
There are more CND's ("cannot duplicate") against the Inertial Measurement Unit, by both flightline and AIS maintenance, than any other system in the F-15. This certainly does not indicate any wholesale abuse of this component of the Inertial Navigation System; on the contrary, problems with the IMU are mostly inadvertent, which means that they are also mostly preventable with better understanding of the Eagle's automatic navigation system.

The CND rate for IMU's often exceeds 50%, and we have never been certain about the reasons for this, since most failures seem very legitimate at the time the LRU's are taken from the aircraft. Occasionally, pilot error or shotgun troubleshooting will cause workable units to be sent in for repair, but for the most part the failures are valid at the time they occur. In most instances, they are simply non-reoccurring failures. Therefore, the intent of this discussion is to dispense some general information on the care and use of the INS/IMU which might help reduce these mysterious CND actions.

COMPONENT SENSITIVITY

Even though the Inertial Measurement Unit weighs forty pounds and requires two persons to handle it, there are sensitive components in the INS which can be easily, if unintentionally, mistreated and damaged. It's also unfortunately true, especially in an airplane, that if something is sensitive, it's usually expensive. Take the gyroscopes for example.

There are two gyros in the F-15 INS, and they carry a price tag of about $14,000 each. A gyro weighs just 12 ounces and is so small you can hide one in your hand. They are made with extremely fine precision which requires them to be operated at a fixed temperature so that the moving parts expand to the tight tolerances needed.

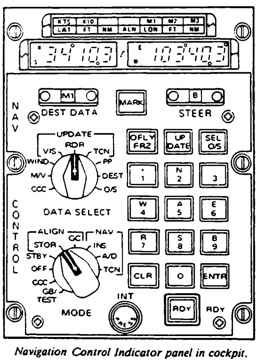


Two of these two-axis gyroscopes are installed in the Inertial Measurement Unit.

This temperature is 170°F and must be maintained within ±0.2°. The gyro is so sensitive to aircraft position change that one milliradian of angular change produces a 1.3 volt signal output. This signal is amplified and almost instantly drives a synchro motor that restores the platform to the level position, and the gyro with it. The flight manual instructs the pilot to wait 15 seconds after turning off the INS before cutting aircraft power. The reason behind such a requirement is that this much time is needed to electrically "brake" the gyro rotor, where rpm is being reduced from 22,500 to zero. It may hurt the IMU when the generators are cut off prior to turning the mode switch on the NCI (Navigation Control Indicator) panel to OFF.

If the ECS caution light comes on, the avionics equipment (including the IMU) will be operating with reduced or no cooling air. This does not mean an automatic shut-down of the IMU. If the aircraft is on ground power, the IMU will shut down. If the aircraft is operating on aircraft power with one or two engines, the IMU will stay on. While there is no spec requirement, the IMU (depending on ambient conditions) can operate approximately 30 minutes without cooling. If the ECS light is on, check the BIT control panel for the INS light, indicating a possible overtemp shutdown has occurred. When power is removed from the INS without turning the mode switch off (whether by turn off of the generators, unexplained power interruption, or over-temperature shut-down) and the battery has been depleted, the gyros continue spinning at high speed without benefit of the dynamic braking. With no power to hold the platform level, the platform can tumble. Movement of the aircraft in this position can cause gyro problems. The gyros, which are designed to operate within a few milliradians of turn, are subjected to a large motion and are slammed hard against their internal mechanical stops because of the high inertia they still retain from spinning rapidly. This could cause physical breakage, or a shift in mass balance (a change in gyro drift characteristics). This is no way to treat a sensitive, high precision instrument and repeating it will eventually cause the gyro to fail. After landing, it is best for the INS to leave it in the NAV mode until taxi is completed and the aircraft parked.

"There are more CND's against the Inertial Measurement Unit than any other system in the F-15."



Navigation Control Indicator panel in cockpit.

Similar damage can occur to the gyro when moving the IMU in an unpowered condition during maintenance or basic transportation. Transportation should be done in specially designed containers or as a minimum, on a thick pad to absorb shock.

After power is removed for any reason, turning the unit back on into the align mode too soon can cause similar physical problems. If the turn off is uncontrolled or it occurred in the first 60 seconds of the align mode, the IMU should not be turned back on for five minutes since the gyros must spin down without dynamic braking. If the turn off was an orderly shutdown by turning the mode switch off, it is only necessary to wait 15 seconds.

When aircraft power is applied or bus power transfer occurs, the IMU has a battery to power the IMU during the power interruptions. This battery provides power for two-second intervals while on the ground, and a minimum of seven seconds in the air. During flight, the battery will operate (if required) until power is depleted to a minimum voltage level. Depending on condition of the battery, this could be in the range of two minutes. If the battery is not working properly, these power interruptions can cause the IMU to dump.

Intermittent failures will most likely occur under stress, when the aircraft is pulling g's. Since g stresses cannot be duplicated during ground testing, inflight gyro failures usually set up a cycle of CND's between flightline and the AIS (Avionics Intermediate Shop). This in turn sets up more airborne INS failures before the IMU is recognized as a bad actor and goes NRTS back to Depot. AIS personnel might profitably NRTS the IMU after three failures with short time span between failures, rather than waiting for four or more per AIS procedures.

It takes about four hours to complete a check of a good IMU across the test bench in AIS - four hours which are wasted when the unit turns out to be CND. Since this bench is also used to check and repair many other F-15 avionics systems, and since about one-half of the IMU's tested on the bench turn out to be CND, you can see the negative effects of this situation on "productive" testing and repair. Accurate data inputs to the INS, accurate data recording of the flight results, and evaluation of data and testing at the flightline can help to reduce the number of IMU CND's and allow more efficient use of AIS test time to check units with valid failures.

Nothing is free today, and every aspect of the current excessive CND rate of IMU's adds to the Air Force's "cost of doing business," so to speak. Removal/installation time expended by flightline maintenance on CND-associated units adds to the cost. The four hours required to check a CND unit in the Avionics Intermediate Shop adds to the cost. And the increased number of spare units that must be purchased because of a high CND rate definitely adds to the cost. A recent study has shown a decline in MTBD (Mean Time Between Demand - the mean flight time between requirements for a piece of equipment from supply to replace equipment on the aircraft) for the IMU from 103 hours in Fiscal Year 1980 to 84 hours in Fiscal Year 1982. To support this decrease, an additional 69 spares are required, at a cost of several million dollars. Any reduction in IMU CND's obviously means improved operational capabilities at less cost, and isn't that the name of the game?

CONDITION CRITERIA

One of the easiest ways to improve maintenance evaluations of the IMU is simply to improve "communications" about the problem incurred. This includes communications between the pilot and debriefing/flightline maintenance, between debriefing and flightline maintenance, and between flightline maintenance and the Avionics Intermediate Shop. Periodic meetings with representatives from all groups concerned could do much to assure efficient operation, and cross-training between groups via OJT could help alert each group to the other's problems.

Some of the criteria for determining the condition of the IMU are the position error and ground speed at the end of the flight. An acceptable accuracy level output is dependent on the accuracy of the input data and the quality of the alignment performed. For a BATH alignment, accurate Magnetic Variation should be inserted in the STANDBY mode position. The value should be accurate for that earth location, including local magnetic distortions in the vicinity, such as buildings, other vehicles, etc. If the aircraft has not moved from the previous flight, the Mag Var stored in the CC is the most accurate and will be used if Mag Var is not inserted. Also for all align modes (BATH, STORED, RAPID, or GYROCOMPASS), the present position inserted in the align mode should be accurate for the location of the aircraft, not a base location that could be several thousand feet away. Desired accuracy to actual aircraft location is 0.1 arc minutes (600 feet).

Other elements in accurate IMU performance are the type of alignment and time in align mode. If a BATH alignment is performed, the flight results need to be evaluated within BATH alignment tolerances. The criteria for determining acceptable IMU performance should include type of alignment performed, time in the align mode, present position error and ground speed at the end of flight, and conditions which may have occurred in align or inflight such as aircraft movement in align, power interruption inflight, and requirement for INS update inflight. (NOTE: present position error can be read out directly in nautical miles on the NCI if the Steer switch is in "B" and "VIS UPDATE" is selected on the Data Select switch when the aircraft has stopped after return.)

If these criteria indicate a bad IMU, an additional aircraft level test should be performed. The impact of many of these items can be eliminated, if time permits, by performing INS-initiated BIT, a GC alignment, and drift run on the IMU while in the aircraft as defined in Job Guide T.O. 1F-15( )-2-34JG40-1. This test takes approximately 45 minutes. If the IMU meets the requirements of this test, it should stay in the aircraft. This procedure is detailed in the INS Fault Reporting T.O. 1F-15( )-2-00FR-00-1 and INS Fault Isolation T.O. 1F-15( )-2-34FI-00-2. Figure 1 (reproduced from the Fault Reporting T.O.) shows that it may take more than one flight to evaluate the INS - a one-flight pass criteria (I), a one flight rejection criteria (III), or a three consecutive flight rejection (II). As noted earlier, each CND which is avoided by test at the aircraft level will save a minimum of four hours test time at the Intermediate level.

In order to identify an undetectable or intermittent problem, the aforementioned data should be maintained and evaluated for several flights to identify repeat failures under similar conditions. When an IMU has a repeat failure, the IMU plus all the justifying data should be sent to the Intermediate or Depot level test station for evaluation and repair.

"One of the easiest ways to improve the IMU is simply to improve communications about the problem incurred."

OTHER FACTORS

While we have been discussing conditions associated with the IMU itself to this point, there are several related components and situations which also have important effects upon quality of INS performance or which can produce degraded or intermittent CND conditions.

IMU Mount — One part of the INS which is often overlooked is the IMU mount. When the mount is installed in the aircraft, it is precisely aligned to the aircraft axis by the boresighting procedure. If damage to the mount disturbs the boresighting, errors could be introduced which could affect attitude information used in weapon delivery. If similar errors are experienced with several IMU's in one aircraft, re-boresighting may be required. To identify this type of problem, history data must be maintained at the flightline level. Care must be taken with the mounting pads and with guide pin insertion when installing an IMU in its mount. Specified torquing procedures must be followed.

Navigation Control Indicator — The NCI continues to experience problems with water, even though sealing gaskets have been added. Extra effort to cover the cockpit electronics when the aircraft is parked will help reduce moisture problems. The bezel should have an RTV seal around the window edge, and if this seal is disturbed, water will reach the DRDs (Digital Readout Displays) and their circuit card systems. If the bezel is replaced or resealed, be sure the non-reflective surface is facing up.

IMU Battery — The IMU battery is to be removed from the aircraft and checked periodically. In addition to the voltage level, there is circuitry in the battery which controls temperature and charging of the battery. Malfunc-

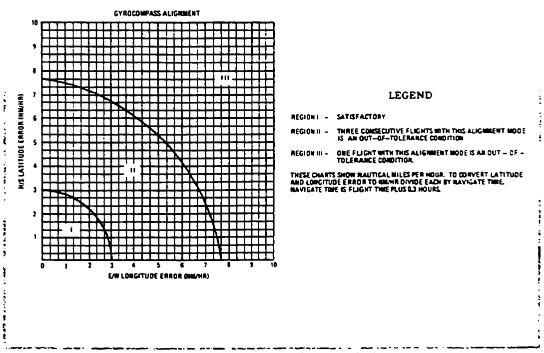


Figure 1 – Gyrocompass alignment criteria.

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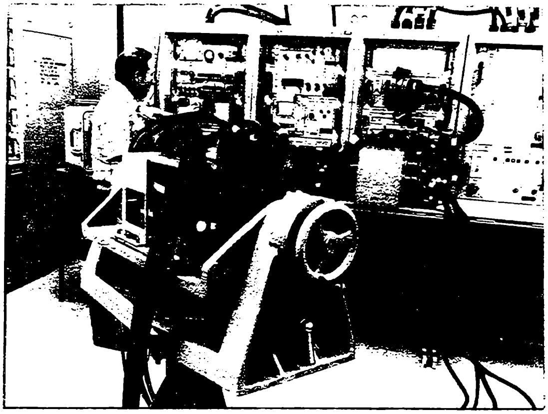
tion of this circuitry may result in the battery not charging properly and cause the IMU to dump on a power interruption or cause noise injecting extraneous pulses into the IMU circuitry, resulting in CND errors.

• Circuit Cards — Printed circuit cards in the INS should fit tight in the card guides. If they do not, an intermittent condition can occur because of a loose connection or because of overheating from poor heat transfer. Loose card guides should be tightened with the Birtcher card guide resizing tool. Although the test points brought to the edge of the board are not used at the Intermediate level, the contacts should be checked for contamination and kept clean.

• Electrical Arcing — When units are installed in the aircraft, electrical power should be disabled by the circuit breakers. This will prevent contamination and degrading of the power connectors from arcing. If power cannot be removed, extra care is necessary when attaching connectors to the unit. (This is also of concern with the IMU battery, which can have power to its connector when the INS is turned off. Any time power is applied to the aircraft, power is applied to the battery.)

• Electrostatic Discharge — Even though some of the IMU's are not marked with ESD labels, all are susceptible to electrostatic discharge. Components can be degraded or destroyed by ESD. Grounding protection should be provided whenever components, circuit cards, or connectors are handled. (MCAIR has prepared a series of video tapes on this subject, which are available through your local Rep.) ■



AIS computer test station used to test the INS – Inertial Measurement Unit mounted on attitude simulator in foreground.

"Don't Let the Cost of Freedom Go Up in Smoke..."

KADENA AB FOD POSTER CONTEST

The 18th Tactical Fighter Wing at Kadena AB, Japan, utilizes all of the standard USAF FOD prevention programs such as tool and hardware accountability, flightline walks, and x-ray analysis of hidden areas. However, the 18th MAQ office has also developed some other approaches including establishment of a Junior "FOD Council," a series of FOD newsletters (appropriately called "FOOT SHOTS"), and a variety of visual aids to get the word on FOD prevention out of all personnel. An example of the visual approach is the "Foreign Object Damage Prevention Poster Contest," sponsored by the wing FOD monitor, TSgt David Humphrey.

Winner of the most recent base-wide contest was SSgt Robert Godin of the Pacific Logistics Support Center, who created an impressive full-color picture of "your tax dollars" going up in flames because of foreign object damage. Colonel Philip M. Drew, 18th TFW commander, judged the entries and directed that copies of the top three posters be displayed on bulletin boards throughout Kadena. They were also publicized through base newspaper and Armed Forces TV releases, and submitted to AFISC at Norton AFB for consideration in an Air Force wide program. MCAIR's own in-plant foreign object control program also plans to display the winning posters in manufacturing and final assembly areas. ■

Second place winner A1C David Brush (18th EMS corrosion control specialist), top winner SSgt Godin, and Colonel Drew display winning FOD prevention contest posters. Honorable mention went to A1C Heidi Rhykerd, 15th TRS photo interpreter. (USAF photo by Sgt Carolyn Zephyr)

some practical pointers on the...

F-15 INERTIAL NAVIGATION SYSTEM (PUBLISHED 1984)

Part II

By HARRY LYONS/*Field Service Engineer, Kadena AB, Japan and* CARMON D. THIEMS/*Lead Engineer, Electronics, St. Louis*

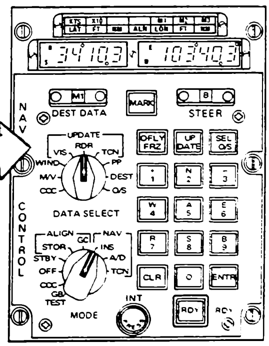
Part I of our F-15 Inertial Navigation System discussion analyzed some of the possible conditions and operations affecting the high CND rate being experienced by the Inertial Measurement Unit (IMU). We noted that one of the most important factors is a good alignment prior to takeoff, so this time we are going to look at the alignment in more detail.

The F-15 INS is a "wander azimuth" system, in which the platform is caged to the aircraft heading. True heading is then computed by determination of the wander angle (the angle between true north and the platform azimuth axis). The F-15 IMU does not require platform azimuth torquing to true north during gyrocompass alignment and therefore, can provide wide angle gyrocompass capability quickly. This is different from the "north oriented" platform system of earlier aircraft like the F-4. The F-4 system has to maintain its platform oriented to true north, which creates problems when the aircraft flies in the area of 70° latitude and above. In that area the platform must be constantly slewed to true north, and this platform motion unrelated to aircraft motion can cause errors rendering the system unreliable. This problem does not occur with the F-15 wander azimuth system which, if aligned at a latitude of 75° or less, will navigate through the polar region within the specified limits of accuracy.

TYPES OF UNITS

There are presently three versions of the IMU in the field. The part numbers are 683420-23, 683420-24, and 683420-25. The 683420-25 with improved align times is the present production version and is being delivered in production aircraft. F-15C 79-0049 and up and F-15D 79-0021 and up are being shipped with the −25 IMU in place. TCTO 5N1-3-24-512 provides the instructions for retrofitting the earlier versions (−23/−24) to the latest configuration (−25). This retrofit process is in progress at the IMU Depot (at this time retrofit is approximately 45% complete).

The three versions are interchangeable in the aircraft, but have different alignment characteristics as identified in Table I. From the cockpit, you cannot tell which version of the IMU is in the aircraft except that the latest version (−25) has two rates of flashing align light signal. The slow flashing for RAPID align occurs at a rate of 0.7 Hz and the fast flashing for STORED heading and GYROCOMPASS (GC) at a rate of 3 Hz. In the STORED mode posi-



F-15 Navigation Control Indicator (NCI) panel is located inboard on right console. NCI contains all controls needed for operation of Inertial Navigation System, and has its own power supply.

TABLE I – IMU ALIGNMENT CHARACTERISTICS

IMU PART NO[a]	MINI-BIAS	STORED		BEST AVAILABLE TRUE HEADING[b]		RAPID ALIGN		GC	
		INDICATION TIME	ACCURACY	INDICATION TIME	ACCURACY	INDICATION TIME	ACCURACY	INDICATION TIME	ACCURACY
683420-23/688775-1 (earliest version)	initiated at 8.5 minutes of GC align	FAST FLASH 3 minutes	5 nm/hr CEP	STEADY LIGHT 3 minutes	10 nm/hr CEP	N/A	N/A	FAST FLASH 10 minutes	3 nm/hr CEP
683420-24/688775-2	N/A	FAST FLASH 3 minutes	5 nm/hr CEP	STEADY LIGHT 3 minutes	10 nm/hr CEP	N/A	N/A	FAST FLASH 10 minutes	3 nm/hr CEP
683420-25/688775-3 (latest version)	N/A	FAST FLASH 3 minutes[d]	5 nm/hr CEP	STEADY LIGHT 3 minutes[c]	10 nm/hr CEP	SLOW FLASH 6 minutes[d]	4 nm/hr CEP	FAST FLASH 10 minutes	3 nm/hr CEP

General Notes:

1. Align times are based on 0°F (time to align increases with decrease in temperature).
2. Units are tested to an accuracy of 2 nm/hr for GC, 4 nm/hr for RAPID align and STORED, 7 nm/hr for BATH.
3. IMU alignment requires the aircraft to be parked in a relatively level area during the alignment period.

Column Notes:

(a) For maintainability/spares supply the 683420 unit is stocked as 688775 (IMU without a battery assembly) and 685409 battery assembly.

(b) BATH align assumes accurate magnetic heading and magnetic variation inputs from aircraft systems. An inaccurate magnetic heading or magnetic variation input will affect BATH accuracy and should not be cause for IMU removal. A ground check of the system prior to removal should be performed per T.O. procedures.

(c) The 3 minute BATH align of the latest version (–25) will be more accurate than the earlier versions (–23/–24) depending on the initial temperature of the unit.

(d) For the latest version (–25), STORED align can occur from 1.5 to 3 minutes and the RAPID align from 5 to 6 minutes depending on temperature of the unit at time alignment is initiated. Preheating could be done in STANDBY mode.

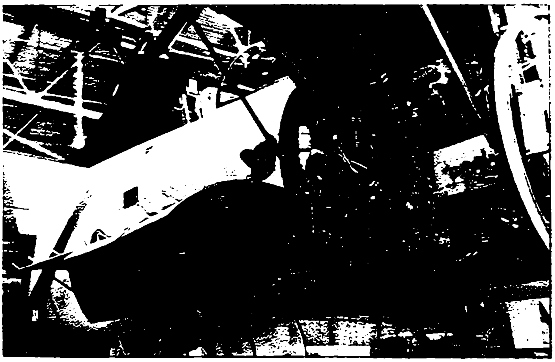
tion, the ALN light will go from OFF to flashing (fast flashing). In the GC mode on the earlier versions (–23/–24), the ALN light will go from OFF to steady ON to flashing (fast flashing). In the GC mode on the latest version (–25), the ALN light will go from OFF to steady ON to slow flashing to fast flashing.

The flashing align light indicates that alignment is complete, which is determined by an accurate and stable velocity reference. It does not mean that *optimum* navigational quality is achieved. This quality depends on an accurate latitude input (desired accuracy for latitude is 0.1 arc minutes/600 feet). A flashing align light can be achieved with a latitude input error of as much as 1.25 degrees but the IMU will provide degraded performance.

MINI-BIAS/AIRCRAFT MOVEMENT

The earliest version (–23) IMU has a mini-bias feature which, during the align mode at approximately 8.5 minutes and beyond, will adjust the bias to more accurately compensate for gyro drift. This has some impact on the current flight, but also improves alignments of future flights. This feature was deleted in the later versions (–24/–25) because of problems encountered by movement of the aircraft while in the mini-bias phase of the align mode, frequently resulting in extra maintenance actions.

"Movement of the aircraft" is defined as taxiing or other movement which would introduce a constant change to the reference heading of the aircraft. (Normal wind gust aircraft motion causing oscillation about the reference aircraft heading will not normally affect GYROCOMPASS align or mini-bias.) The effect of aircraft movement on alignment depends on the time of movement. A movement early in the alignment may only extend time of align completion, while a movement late in the alignment may prevent completing alignment or result in gyrocompass error and poor navigation. As a general rule, a realignment should be performed if the aircraft has been moved while in align. In the case of the earliest version (–23) IMU, if movement occurs after 8.5 minutes in the align mode, mini-bias correction is in operation and errors will be introduced in the basic gyro bias permanent memory compensation. This degrades gyro ability to keep the platform level. When this occurs the IMU must be removed from the aircraft and rebiased at the AIS.



Inertial Measurement Unit (IMU) is located behind door No. 3R in right side equipment bay No. 2. All three current versions of IMU are interchangeable in aircraft, but have different alignment characteristics as noted in Table I.

IMU ALIGNMENTS

There are two switch positions on the NCI mode switch for alignment. In the STOR position, the IMU will perform a STORED heading alignment. In the GC position, the IMU will perform and indicate a BATH, RAPID, or GYROCOMPASS alignment depending on the time in alignment. Alignment procedures are detailed in the flight manual.

Gyrocompass Alignment

The recommended and the most accurate alignment is the full GYROCOMPASS (GC) mode. This mode has a specification accuracy of 3 nm/hr CEP, but data taken at MCAIR shows a 1.2 nm/hr CEP for production acceptance flights. After switching to the GC align mode, accurate present position must be entered (desired accuracy 0.1 arc minutes/600 ft). Alignment completion is signaled by a flashing ALN light. For the earlier versions (−23/−24), the indication is a change from a steady light to a flashing light. For the latest version (−25), the indication is a change from a steady light to a slow flashing light followed by a fast flashing light. The indication shall occur in approximately 10 minutes.

Stored Heading Alignment

An accurate three minute alignment can be achieved if advanced planning is used and a STORED heading alignment used. The requirements are to park the aircraft where it will start from on the next flight, then turn OFF the INS for a minimum of 15 seconds, turn ON the INS, select GC align mode, enter present position, wait for a fast flashing align light (approximately 10 minutes) and turn the INS OFF. A minimum of 15 seconds later, aircraft power could be turned OFF (this allows normal gyro "braking" to occur).

As long as the aircraft is not moved, the next alignment can be a 3 minute STORED heading alignment with accuracy comparable to a GC alignment. The actual STORED heading alignment is accomplished by entering present postion and waiting for the ALN light to go from OFF to flashing at approximately 3 minutes for the earlier versions (−23/−24). The latest version (−25) can complete a STORED heading alignment in 1.5 to 3 minutes depending on initial platform temperature.

Rapid Alignment

An intermediate align mode, RAPID align, is now provided as part of the latest version (−25) IMU, which provides an alignment more accurate and reliable than the BATH. This mode specification is a 4 nm/hr CEP in 6 minutes (actual flight acceptance data shows an accuracy between 1 and 2 nm/hr CEP). This mode is obtained by performing a GYROCOMPASS align, but switching to Nav in about 6 minutes when the ALN light changes from steady ON to slow flashing.

BATH Alignment

BATH (Best Available True Heading) alignments can range from good to very bad, with the bad alignments almost always at no fault of the INS. The worst condition for a BATH alignment is to perform it while in the chocks, inserting the chart magnetic variation, and going to Nav as soon as the align light comes on steady. In the F-4, you can do this because, as noted earlier, that INS is a north-seeking system and will align itself to north during BATH with a reasonable accuracy if the magnetic variation has been nulled prior to turning off aircraft power following a good INS flight. The F-15 INS is a wander azimuth system, and does not align itself to north. It aligns itself to the nose of the aircraft and the initial heading of the aircraft becomes the platform heading from that time on until the next alignment. The F-15 BATH alignment is not inertially derived north alignment, but is a best available reference to north derived from magnetic heading and magnetic variation inputs. These inputs determine the initial wander angle value.

If there is not time for a full GC or RAPID alignment and a STORED heading alignment is not possible, a BATH alignment will have to be used. The following will make the BATH more accurate:

- Wait as long as possible prior to going to Nav. This allows some gyrocompassing to be done. In the latest version (−25), it is continuous from BATH align light ON (3 minutes); in the earlier versions (−23/−24), it starts after 6 minutes.

- If the aircraft has not been moved since the last good INS performance flight; if the CC has not been changed or had its magnetic variation changed; and if magnetic field distortions in the area have not been changed, then the CC stored magnetic variation will be more accurate, will provide a more accurate BATH alignment, and should be used instead of a manual input.

- If the aircraft has been moved or CC changed, the BATH will be more accurate if the alignment can be done in an area where distortions from buildings, power lines, etc., are minimum. This minimum distortion area could be near or at the end of the runway. If alignment can be done in that location, record a good present position and magnetic variation for that location from a good GC alignment for future use in BATH alignments.

If a BATH alignment accuracy is acceptable, the mode switch can be moved from OFF to Nav. After 3 minutes the IMU will automatically go to Nav, eliminating the need or concern of switching to Nav at a later time. Checking magnetic variation entry of present position, and non-movement of the aircraft are still required.

INS MANUAL INPUTS

Ability of the gyros to maintain the platform level depends upon having precise latitude and an accurate true heading during alignment. Initial latitude is inserted via the NCI at the beginning of the align mode. True heading is determined by using magnetic heading and magnetic variation. Magnetic variation can be checked while in the STANDBY mode for accuracy and can be changed to improve the value. STANDBY mode must be selected to insert a corrected magnetic variation into the INS. Magnetic variation inserted after switching to an align mode will only go to the CC and have no effect on alignment. In addition to earth variations, magnetic variation must include variations for effects of metal buildings, fuel trucks, power lines, power transformers, underground plates, cables, pipes, etc. Accurate magnetic heading and magnetic variation are important only if a BATH alignment is being used.

After the INS has been fully GC aligned and switched to Nav, the magnetic variation is continually computed by the CC from AHRS magnetic heading and INS true heading. This magnetic variation includes all distortions as previously mentioned, and is stored in the CC when the flight is terminated. If the aircraft is in the same position of the last flight, this CC stored magnetic variation combined with AHRS magnetic heading will give a very good approximation for true heading and could make BATH alignment reasonably accurate.

The effects of incorrect magnetic heading and magnetic variation inputs can be corrected by remaining in align after a steady light and doing a complete GYROCOMPASS alignment. However, there is no automatic correction to incorrect latitude inserted during initial present position entry. It is important that latitude be as accurate as possible, otherwise ground speed and position error result. Desired accuracy is 0.1 arc minutes (600 ft). Incorrect longitude is not as significant; it will remain a longitude error, but will not affect the alignment or cause performance errors. If you err on one of the present position coordinates, make it longitude. If a present position error is recognized early in the align cycle, the correct number can be entered and still get a good alignment.

If latitude is not inserted, the alignment cycle will hold, waiting for latitude entry, for 17.5 minutes after which the IMU will give a fail indica- ▶

tion. For the best and quickest alignment, verify or enter a good magnetic variation in STANDBY and enter the correct present position (latitude and longitude) when in an align mode. Present position entry should be after the NCI data readout displays indicate all "zeros" and before 1.5 to 3 minutes has elapsed in the align mode.

ALIGNMENT ACCURACY IMPROVEMENT

Accuracy of the BATH alignment can be improved by staying in the align mode beyond the initial 3 minutes. For earlier versions (−23/−24), align improvement does not begin until after 6 minutes. For the latest version (−25), align improvement is continuous from 3 minutes. Also increased accuracy is achieved if you remain in RAPID align after the slow flasher occurs. With both the BATH and RAPID align, a full GYROCOMPASS alignment is accomplished after 10 minutes in the align mode. If you remain in align beyond the GC fast flashing light there will still be improvement, but the gain level is so low that the amount of improvement is not significant. Preheating the gyros (which occurs in STANDBY mode) could shorten time in align modes (see Table 1).

PRESENT POSITION ERROR

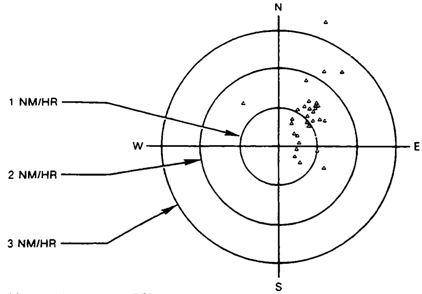
Present position error is evaluated by circular error probable (CEP). CEP is expressed in terms of the radius of a circle centered on the desired terminal point within which 50% of the terminal errors will fall (see Figure 1).

Evaluation of acceptance flights by company and Air Force pilots at McDonnell Douglas in St. Louis have shown a CEP of 1.2 to 1.3 nm/hr for 1336 flights between July 1979 and May 1982. These flights included 946 GYROCOMPASS flights, 259 STORED heading flights, and 131 RAPID align flights. It is significant to note that for the latest version, the RAPID align flight test data approaches that of full GC align results.

When the pilot, the debriefer, and flight line maintenance evaluate the INS at the end of a flight, the type of alignment, updates, and data from previous flights are important. If a BATH alignment was used, the accuracy of the INS cannot be evaluated with enough confidence to remove the IMU from the aircraft. If updates have been performed during the flight, the INS accuracy cannot easily be evaluated without consideration of the update reference accuracy and the amount and time of the update. As indicated in Part I and in the INS technical manuals, for an intermediate

FIGURE 1 — PRESENT POSITION CEP (30 TYPICAL FLIGHTS)

(CEP is 1.2 nm/hr)



Note:
CEP is derived from radial error rate (RER) for each flight −

$$RER = \frac{\sqrt{[(\text{LATITUDE ERROR})^2 + (\text{LONGITUDE ERROR})^2]}}{\text{NAVIGATION TIME}} \text{ nm/hr}$$

(error in nautical miles; navigation time in hours (time in NAV Mode))

band of present position error it takes three consecutive flights of the same type alignment to confirm an INS problem requiring removal of the IMU from the aircraft. "One flight rejections" which are within T.O. tolerances result in frequent AIS CND results.

An easy and accurate method of obtaining terminal present position error is to perform a VIS UPDATE upon return to base. The following procedure stores the base present position during align to use for comparison by VIS UPDATE after parking the aircraft.

1. At start of flight:
   MODE switch − STBY
   Check magnetic variation, correct if necessary
   MODE switch − align (STOR or GC)
   DEST DATA switch − B
   DATA SELECT switch − PP
   Enter present position (will be stored in 'B' location in the CC)

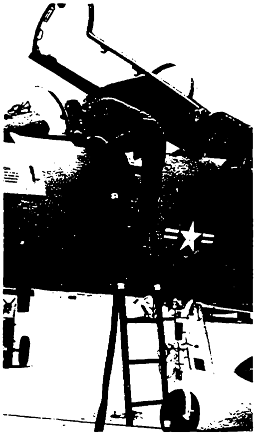
2. After alignment complete:
   MODE switch − NAV

3. At return of flight (assume the flight is terminated near the point of original alignment):
   STEER switch − B
   DATA SELECT switch − VIS UPDATE
   The CC will compute and display the latitude and longitude error (in nautical miles) between the initial and terminal present position

These two articles have been pretty heavy reading about a complex but vitally important part of the F-15 weapon system. However, if you have stuck with us to this point, you might be ready for a few "conclusions" that attempt to tie this whole package of advice together. Despite the internal technological sophistication of the Inertial Navigation System as an avionics component, there are some fundamentally simple and "workaday" approaches to operating, maintaining, and troubleshooting the INS that everybody can take to assure its successful utilization.

The first, most vital, and all-important requirement is to IMPROVE COMMUNICATIONS — all of you must talk to each other! Pilots, flight line maintenance, debriefers, and avionics shop specialists must all be in this together. The AN/ASN-109 Inertial Navigation System has demonstrated a very good 1.2 to 1.3 nm/hr CEP performance for GC, reduced GC (RAPID align), and STORED heading align modes. However, ineffective communications, incorrect operating procedures, or careless handling practices will continue to place needless demands upon supply and AIS because of invalid IMU removals. Therefore, starting with the need to improve communications, we have summarized most of the points made in Parts I and II into the single page of guidance at right. Regardless of where you fit in the overall picture, read them all for they mark the route to fewer INS CND's and successful navigation in the F-15 Eagle. ■

How to Minimize INS "CND" Problems



PILOT

- IMPROVE COMMUNICATIONS (with debrief and maintenance personnel)
- Insert accurate inputs (present position — 600 ft desired) (magnetic variation for BATH in STANDBY mode)
- Record complete data for maintenance evaluation (AFTO 241)*
- Allow aircraft movement in NAV mode only
- After IMU turn on, maintain power on IMU during aircraft movement
- Turn on and operation of INS with one engine operating is acceptable
- Use VIS UPDATE for present position error calculation at end of flight
- Do not use BATH flights to evaluate IMU accuracy or for equipment removal
- Turn INS off 15 seconds minimum before aircraft power shutdown
- Observe proper turn on and turn off of IMU for power interruption
- Do not delay when switching from STORED heading align to NAV
- On latest version of IMU (−25), note change from slow flashing to fast flashing for GC align

FLIGHT LINE MAINTENANCE

- IMPROVE COMMUNICATIONS (with pilot, debrief, and AIS personnel)
- Minimize IMU removals — verify problem on aircraft by test or second flight
- Prior to removing IMU, turn NCI mode switch off and open IMU circuit breakers
- Handle IMU with care — avoid physical shock to unit. Use thick pad or transport box when moving the IMU
- Maintain history file for identifying "bad actors" and marginal units (AFTO 241 and debriefing data)
- If updates are performed in flight, INS can not be evaluated without compensating flight data for update errors
- Minimize water on NCI (close canopy, cover equipment)
- Track IMU batteries to insure they are periodically checked by AIS
- Record complete data for AIS maintenance evaluation (AFTO 350)
- Prevent electrostatic discharge to connector pins
- Use accurate present position (same as pilot) for testing unit on aircraft
- Turn INS off 15 seconds minimum before ground power shutdown

(*) Information useful for INS evaluation: Type of alignment; time in align; problems during align; time in NAV; updates in flight (type, amount, and time); unusual occurrences/INS problems; present position and ground speed error at end of flight.

(#) Test option selection: Option 50 — suspect problem with unit when it is first turned on (cold start problem); Option 13 — problem undetermined (complete performance test); Option 41 — only gyro bias required.

DEBRIEFING

- IMPROVE COMMUNICATIONS (with pilot and maintenance personnel)
- Record detailed description of data and failures
- Record consecutive flights data — can take three flights to identify first-time failures (see Fault Reporting manual)
- Do not use BATH align flights for INS accuracy evaluation

AVIONICS INTERMEDIATE SHOP

- IMPROVE COMMUNICATIONS (with flight line and depot personnel)
- Evaluate flight line maintenance data to select appropriate test option #
- Evaluate IMU battery for control circuitry problems and corrosion
- Maintain tight card card guide fit
- Verify sealing gaskets and bezel window seal on NCI
- Handle components, cards, and unit with care regarding electrostatic discharge
- Maintain history file for identifying "bad actors" (put copy in spare card slot when sent to depot) ■

F-15 Hydraulic System

(PUBLISHED 1976)

By ROBERT S. ANDREWS/*Senior Engineer, Hydraulic Design*

The F-15 Hydraulic System incorporates some of the latest hydraulic design concepts from the standpoint of safety, survivability, and maintainability. McDonnell has been able to incorporate these many design improvements because of the high learning curve obtained from design and operation of the highly successful F-101 Voodoo and F-4 Phantom II.

The Hydraulic Systems consist of three independent systems: Power Control 1 (PC-1), Power Control 2 (PC-2), and Utility. PC-1 and PC-2 systems power the primary flight controls and the Utility system supplies all other requirements, plus back-up for stabilator longitudinal and roll control, aileron roll control, and rudder directional control. Hydraulic power is available to adequately and safely maintain control for flight and landing with any one of the three systems operational.

INTERFACE OF SYSTEMS

The block diagram shows the various subsystems in the "A" and "B" circuitry of the PC-1, PC-2, and Utility systems. In the Utility system, the "A" circuit lines are primarily on the left side of the aircraft and the "B" circuit is primarily on the right-hand side. This improves survivability from a gunfire standpoint.

Since any one of the three hydraulic systems can maintain a supply of hydraulic pressure to the control system, it is obvious, as you refer to the illustration, that the crisscross of hydraulic supply to the flight controls from left and right engine driven pumps, through RLS circuitry and switching valves, gives multiple redundancy of hydraulic supply to the F-15 primary flight control components. Here is what will happen during several emergency situations:

- When all electrical power is lost, control is maintained with ailerons and differential stabilator for roll, stabilator for pitch, and two rudders.
- When either PC hydraulic system plus the Utility hydraulic system, and all electrical power are lost, control is maintained with ailerons on one wing and differential stabilator for roll, stabilator for pitch, and one rudder. (If PC-2 and Utility are lost, the Control Stick Boost and Pitch Compensator will be inoperative.)
- When all mechanical controls are lost, control is maintained by the Control Augmentation System driving the differential stabilator for roll and pitch, and both rudders.
- When both PC-1 and PC-2 hydraulics are lost, control is maintained with the Utility hydraulic system supplying power to all primary flight controls.

HYDRAULIC PUMP

For ease of maintenance, the F-15 pump was designed as a plug-in type. The intake, outlet, and case drain fluid flows are directed to the spline-drive end of the pump where they pass through quick disconnect couplings. These connect the pump to an aircraft mounted manifold which has rigid tubing attached, allowing the pumps to be installed and removed without disconnecting hoses and tubes. Doing away with hoses eliminates the possibility of chafing and there are fewer leakage points. Self-sealing checks were incorporated to prevent line drainage during replacement. The pump also incorporates fast-response compensator shutoff to lower hydraulic system pressure spikes. Basic system accumulators found in most aircraft have been eliminated (these are high replacement items and can be responsible for introducing air into a hydraulic system).

FILTER PACKAGE

Each of the three systems (PC-1, PC-2, and Utility) has a single filter module which incorporates pressure and return filters, system relief valves, pressure switches, pressure transmitters, and pump outlet check valves. As a result, there is one module and one door per system, simplifying servicing. All pressure and return elements are non-collapsible at 4500 psi ΔP and are in one size and type (15 micron absolute with an approximate 8 gram dirt capacity) for commonality and good logistics control. The filters have self-sealing checks incorporated to prevent line drainage, and there are delta-P indicators at the bottom of the bowl to reveal a dirty element. The bowls must be removed to reset the indicator and the bowl cannot be replaced without an element inside. The bowls feature self-locking ratchets, and are non-interchangeable pressure-to-return to assure murphy-proof maintenance. The relief valve is a fast-response type backing up the fast-response pump compensator allowing elimination of accumulators.

The pressure filter is non-bypass while the return filters are dual purpose. They filter the system return oil (bypass) and the pump case drain (non-bypass). This allows the pump case drain (which carries particles from the hardest-working, most wear-producing component in the system) to have a large, high-dirt capacity filter with no danger of particle recirculation to accelerate pump wear. This also prevents wear particles from a failing Utility pump from contaminating the second Utility system pump.

RESERVOIRS

Each of the three F-15 bootstrap type reservoirs incorporates reservoir level sensing (RLS). RLS works on the principle that a leak developed in the aircraft will cause the reservoir level to sink. As the level decreases, RLS sensing mechanically operates a valve which shuts off half the system (designated "A"). If this stops the leak, the reservoir level will stop sinking and the other half of the system (designated "B") will be retained.

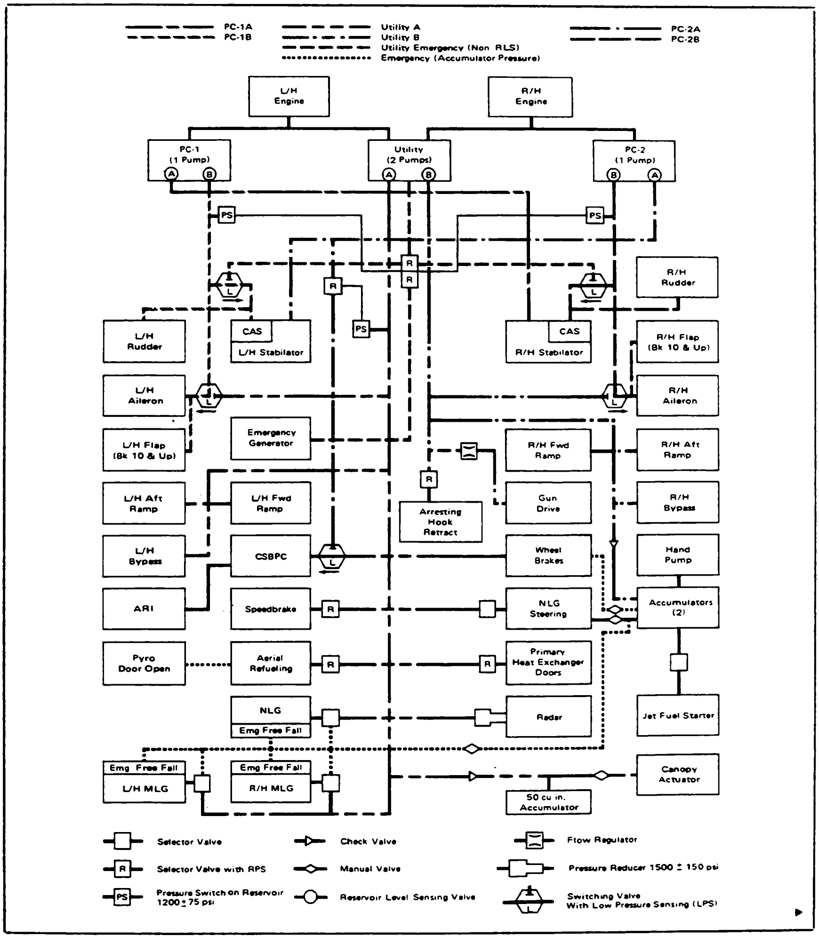
On the other hand, if the leak continues, the reservoir will continue to deplete until a second valve shuts off the "B" half of the system. When "B" shuts off, the "A" system returns, reactivating one-half of the system. This is accomplished by mechanical linkage between the "A" and "B" shutoff valves. Leaks in the pump or filter circuit are *not* protected by reservoir level sensing. However, as you can see, RLS improves the survivability of the aircraft.

The gaging system on the F-15 reservoirs is also unique as the gaging is temperature compensating to allow for volume increase or decrease due to oil temperature changes. Automatic overflow occurs if the reservoir is overfilled, preventing reservoir damage.

SWITCHING VALVES

Another new type of hydraulic component found in the F-15 is a "switching valve." Four of these are used to further improve the survivability of the primary flight control systems. Two switching valves are in the aileron circuit; two others are in the tandem stabilator/rudder circuits.

These valves allow the normal operating pressure from the "B" RLS circuits of both PC-1 and PC-2 to pass directly through the switching valves to the left and right ailerons, to one side of each tandem stabilator, and to each rudder. Should a "B" circuit lose pressure for any reason (leak, pump failure, etc.), the switching valves will move to a test position to assure that the system downstream of the switching valve is intact. If system integrity is verified, the Utility system will be switched into the downstream flight control actuators. This test position prevents loss of Utility oil should the break be downstream of the valve.



HYDRAULIC SELECTOR VALVES

In the F-15, the hydraulic selector valves have a design feature called return pressure sensing (RPS) which was incorporated to improve hydraulic system reliability. Selector valves with RPS will not operate if there is a leak in the selected lines or in the return line to the first check valve. This prevents the pilot from switching into a failed hydraulic circuit where the oil would be directed overboard, thus losing the entire system, or half a system if the failure was in one of the RLS branches.

Return pressure sensing blocks the pressure to the solenoid pilot-operated section of the selector valve. The block is achieved by sensing the loss of return line pressure in the subsystem lines which have failed. Subsystems which *must* be operated after failure have emergency back-up provisions.

In selector valves, care was also taken to design out "man traps" such as doors or surfaces that are hydraulically positioned open or closed upon removal of electrical power. An example is the F-15 speed brake valve which remains in a full trail position (both selected lines become common to return if electric power is removed from the aircraft).

F-15 check valves are designed so that they can be installed in only one direction. Therefore, it is impossible to install one backwards during maintenance. The secret lies in the different size end fittings.

The return check valves in each subsystem have been installed as far downstream as possible, just prior to entering the main return trunk line. This gives the maximum line protection against losing reservoir oil from back-flow into a leak in a return line.

Hoses and most swivels have been eliminated in the F-15 through use of coil tubes. Some of the common problems of the past (including chafing, installation in a twist which accelerates failures, cross-connection which is dangerous, and weepage through hose liner imperfections) have been avoided. In addition, swivels with rotating dynamic seals are at a minimum in the Eagle.

F-15 flight control and engine inlet components use dual external dynamic shaft seals. This design utilizes two seals in series with the center area vented to return through a restrictor which reduces system internal leakage in event of a first stage seal failure. The second stage atmospheric seal is normally subjected to return pressure but is capable of withstanding full pressure should the first stage seal fail. This allows increased seal life and component survivability as the first stage dynamic seal is lubricated on both sides. Should the first seal fail, the second seal can act as a back-up.

FITTINGS

The F-15 plumbing uses a new, permanently swedged fitting in some locations, eliminating many potential inline tube connector leak points. The tube connectors used at valves, and at remaining inline connectors, are of the latest design, stay tight, and require less maintenance. (The DIGEST took a closer look at the Dynatube fittings in Volume 22, Number 3, 1975.)

AIR PROBLEMS

Air in hydraulic systems is an age-old problem. The F-15 components have been specially designed to eliminate this possibility. The canopy accumulator is the only unit in the hydraulic systems where pressurized air leaking by a seal can enter the hydraulic system. In this case, space dictated the use of a smaller standard accumulator with a single dynamic seal.

Air problems such as overflow or bursting of reservoirs, excessive bleeding after emergency operations, and cavitated pumps with momentary loss of system pressure have been minimized during design of the Eagle.

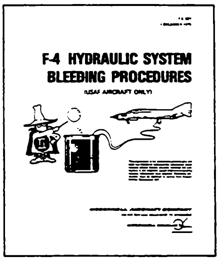
Here are some of the applications that minimize on-board air problems.

- Basic system accumulators have been eliminated.
- Dual vented seals are used in components which have air chambers. Typical of these are jet fuel start accumulators, arresting gear cylinder, and canopy counterbalance actuators. Dual seals allow the air to be vented overboard instead of into the hydraulic system should air leak by a dynamic seal.
- Emergency air systems have been eliminated. The landing gear, brakes, and steering emergency systems use oil from the jet fuel start accumulator. The aerial refueling emergency system uses a pyrotechnically operated system.

TO WRAP IT UP

With all these new features, we feel that the F-15 exhibits a giant step ahead in hydraulic system design. The results - improved system maintenance, reliability, and aircraft survivability. Things that make a product better, and a weapon more effective.

New Information Available



While you've just finished reading an introduction to the Eagle hydraulic system, we want to remind you that there are still quite a few Phantom hydraulic systems out there, too! And here's another in our informal series of booklets on that system, this time covering USAF (only) F-4 hydraulic system bleeding procedures. MDC Field Service Reps already have copies, or you can write us directly. Ask for P.S. 927.

VIDEO TAPE ON AIR IN HYDRAULIC SYSTEMS

For the past several years, our hydraulic design and test engineering staffs have been engaged in studies of air in hydraulic systems - how to discover it, remove it, and prevent it. Some interesting visual techniques were developed during testing which have provided new insight into understanding this old problem. These studies apply to both the F-4 and F-15 models, and have been condensed into a 25-minute 3/4 inch color video cassette, prepared by our Training Group and available for training or orientation by asking for Film No. C-5-128.

1000 HOURS–1st "OLD PRO" EAGLES

(PUBLISHED 1979)

For some reason, no doubt lost in aviation history, the accumulation of 1000 hours of flight time has become the first significant milestone in the life of both aircraft and aircrew. To reach that plateau is an honored event, but to be the *first* to reach it is one of the highest honors. To date no pilot has flown 1000 hours in the Eagle, however, three Eagles have passed that mark.

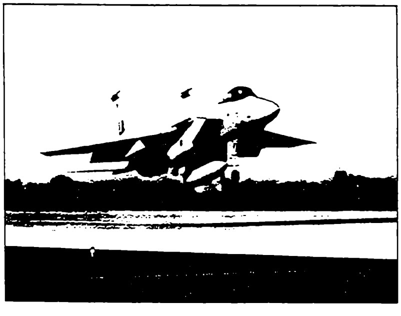
The first 1000-hour Eagle is Air Force serial number 71-291, or TF-2 (the Bicentennial Eagle) as known to most MCAIR folks. The second two-place F-15 and the eighth preproduction Eagle to roll off the line, TF-2 passed the mark way back on 21 October 1977. As of 30 May, TF-2 has 1428 hours and 1069 sorties and is still going strong.

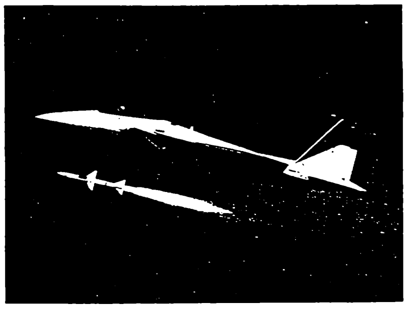
TF-2 has served in two primary roles: As a world good will traveler and as a testbed for most of the air-to-ground development programs. These world travels have included trips to 15 countries on four continents and numerous demonstration rides for dignitaries and military personnel. The less glamorous role is typified by the above photo taken during weapons release testing with the conformal tanks aboard. Notice the cameras installed under both the wing tips and the tail section to record the release sequences.

The second Eagle to fly 1000 hours was also the first production aircraft to do so. 73-090, the 24th "A" model and the 30th F-15, reached the mark on 3 April 1979 at Luke AFB, Arizona. Assigned to the 550th TFTS, 3090 is a real workhorse whose endurance was demonstrated during March when over 71 hours were flown while thundering down the stretch in the race for the F-15A 1000-hour honors.

The third Eagle to pass the 1000-hour mark was 71-285, doing so on 27 April at Edwards AFB, California. The sixth preproduction airplane, this Eagle leads a unique life. As the first fully avionics equipped F-15, it is used by the Air Force as the Radar and Weapons Systems evaluation aircraft. During this testing, 285 has fired numerous AIM-7/9 missiles against various targets, shooting down a total of 16 BQM-34, 14 QF-102, and 2 QF-86 drones. For this exceptional prowess in the air, 285 is dubbed "Killer."

To all three Eagles, their Keepers, and their Drivers, the *DIGEST* sends congratulations and "Good luck for the next several thousand hours!" ■





IFF

The F-15A/B aircraft "identification system" is composed of a transponder set and an interrogator set. Through this system, the Eagle pilot can identify other airborne aircraft and be himself identified to both ground and airborne locations. The identification system is not, electronically speaking, especially complicated, but there is a lot going on inside several black boxes whenever it is activated. Several months ago, some USAFE pilots expressed a desire for a little "chalkboarding" to back up the T.O.'s on the subject, and we turned to Fred Mueller — MCAIR's Comm/Nav/Ident rep at Bitburg AB, Germany. Fred outlines the IFF (Identification Friend or Foe) subsystem in this issue, and covers the AAI (Air-to-Air Interrogation) subsystem in the next. He writes primarily to aid system operators but also offers information of interest to maintenance people — his topics include general information, normal operation, indications of system failure, and finally, built-in test (BIT) functions.

(PUBLISHED 1978)

"HEY, I'M A GOOD GUY!"

By FRED C. MUELLER/*McDonnell Field Service Engineer, Bitburg AB, Germany*

"Hey, I'm a good guy!" exclaimed Major Van Sickle.

Major Lawrence J. Van Sickle is now Operations Officer of the 525th Tactical Fighter Squadron at Bitburg, but at the time of our conversation, he was commander of the 36th TFW Stan/Eval Group. At the time of our conversation, he was also sort of unhappy with certain parts of our Eagle. Our discussion involved some of the unexplainable problems the wing was encountering with the F-15 IFF/AAI Systems.

"That," the Major continued, "is what I would want to be absolutely certain I was telling the other friendly forces if we were at war and I was scrambled to fly an intercept mission. I would not like any of our people, either on the ground or in the air, not *knowing* I was one of them! I would also want to be able to pick out, with absolute certainty, other friendly aircraft."

"Well," I said, "of course your identification systems should do just that for you. After all, the F-15 has some very sophisticated IFF gear aboard."

"That's right, Fred, and they will if they are working correctly. Now, I realize these systems are almost always functioning as advertised. What I am really talking about are the problems we pilots face convincing ourselves before and during a flight that these systems are indeed working. We are confronted with a whole bunch of indicator lights, BIT lights, audio tones, and what have you, to say nothing of the switchology involved."

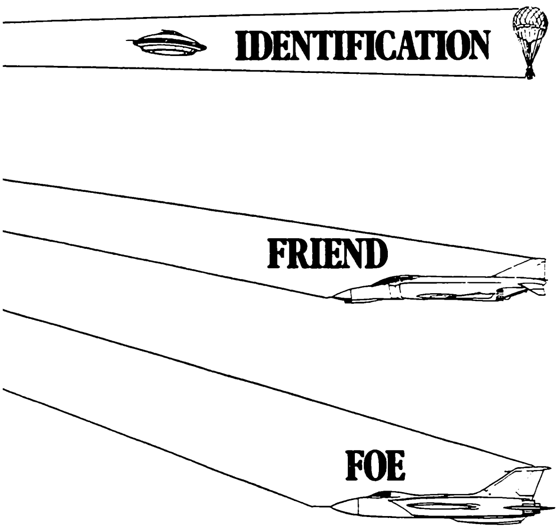
"You mean these systems are *that* complicated?" I asked, rather surprised.

"No, no, I don't mean that at all. The problem, as I see it, is that most of us pilots don't really understand all that is happening when we do certain things. Most pilots aren't interested in getting involved in the technical intricacies of complex electronic systems, so I don't mean that. I am referring to understanding what particular reaction should, or should not, occur when the pilot does something. For example, we may get a BIT light and then after an Initiated BIT is performed, the failure indication goes away. Is the system OK or not? Then there is this business of the Mode 4 codes dumping. Just how *does* that darned HOLD switch work? And Mode 4, well, I could write a book on the questions we have about that system. So all in all, I believe we pilots have developed a very low confidence level in these systems. That is the situation we are faced with."

"I see what you mean and I understand the problem. How about me writing up a brief explanation of the systems. It could be distributed to all pilots."

"Hey, that's a great idea! In fact we could print it in the BUSH RAG. That should really give it good coverage." (The BUSH RAG is a publication put out by the 36th TFW Stan/Eval Group to provide aircrews with information related to flying operations.)

Well, that's pretty much the way it happened. A quite similar conversation did take place between Major Van Sickle and myself. (I have taken some advantage of "literary license" in reporting the conversation, and for that I must apologize to the Major.) I have revised the original BUSH RAG article for presentation here in that there were some omissions that I felt should be included and some new information had to be added due to a Radar software change. I also expanded the article to include some more general information that is, nevertheless, still pertinent to the F-15 identification Systems.



PART I
IFF TRANSPONDER SET

The IFF portion of the F-15 Identification System is usually referred to as the "Transponder" Set. This is because it automatically "transmits" or "responds" whenever it is acceptably interrogated. "Acceptably Interrogated" means that the interrogation is correct and is in a mode that has been enabled by the pilot. If these conditions are met, the IFF Transponder will transmit a reply code that has been preset for that mode. The code contains intelligence pertinent to the aircraft's identity and position. This information is decoded and used by the interrogating station.

OPERATIONAL MODES

SELECTIVE IDENTIFICATION

SIF (Selective Identification Feature) modes are used to identify an aircraft and its position. They are Modes 1, 2, and 3A, and are selected or enabled with the Mode ON/OUT switches. The pilot may enable any one or all modes and the Transponder will reply in the mode that has been interrogated, providing the interrogation was valid.

ALTITUDE REPORTING

Mode C (sometimes called Mode 5) is the Altitude Reporting mode and is used to transmit aircraft altitude for use in air traffic control. This mode is enabled and functions the same as the SIF modes.

SPECIAL MODES

The IP (Identification of Position) mode is selected by momentarily pressing the IP button. The IP function will be enabled for approximately 20 seconds. An expanded reply will be transmitted in the SIF mode that has been interrogated. This provides a more positive identification of the aircraft position.

The Emergency mode is enabled by means of the LOW/NORM/EMERG switch. When the Emergency mode is selected, all of the SIF modes are enabled and a special Emergency code will be transmitted in any SIF mode that is interrogated. The Emergency mode will also be activated upon pilot ejection, provided any SIF mode has been selected.

The other two positions of the LOW/NORM/EMERG switch control the sensitivity of the receiver. The LOW position decreases the sensitivity of the receiver and is used when in a high interrogation environment. The Transponder can only handle a given number of interrogations so, when LOW sensitivity is selected, only close-in interrogations will be recognized.

MODE 4

Mode 4 is a classified mode used for positive identification of an aircraft by a ground station or another aircraft. It is divided into two sub-modes or programs - Mode 4A and 4B. Mode 4 is selected with the A/B/OUT switch. Selecting either A or B will enable Mode 4 operation. The correct A or B program must be selected. The Transponder will then respond by transmitting a reply code when interrogated. (The exception to this is a "listening" mode which will be more fully discussed later.)

CODES

The SIF mode codes are preset into the Transponder System by means of switches. The codes are discrete inputs and are not, normally, changed during a flight. Modes 1 and 3A are set into the system by the pilot and can be changed by him during flight. Mode 2 is preset by ground personnel and these code switches are not available to the pilot.

Mode C codes are generated in the Air Data Computer. The ADC translates the aircraft altitude into a code which is transmitted by the Transponder via Mode C.

The Mode 4 code is preset into the Transponder Computer (KIT) and is classified crypto. Unless the pilot takes action to hold the code, automatic zeroization or cancellation of this code takes place at the end of each flight. The code will automatically zeroize anytime electrical power is removed from the system once the Landing Gear handle has been in the UP position. In the event it is desirous to retain the code for another flight, the automatic zeroization can be bypassed. This is done by placing the Landing Gear handle in the DOWN position and then placing the HOLD/NORM/ZERO switch in the HOLD position. The switch is held in the HOLD position momentarily and then released. The pilot must wait 10 to 15 ▶

seconds before removing electrical power from the system. The 10 to 15 second pause is essential for the retention circuits in the KIT to set.

The HOLD/NORM/ZERO switch is in the NORM position for normal operation. Placing the switch in the ZERO position will zeroize or cancel the Mode 4 code set into the KIT regardless of other switchology.

It is sometimes necessary to cut electrical power during pre-taxi checks, etc. As long as the Landing Gear handle has not been moved out of the DOWN position, electrical power can be applied, removed, and reapplied without zeroizing the Mode 4 code.

ANTENNAS

The Antenna Select Switch selects either the upper, lower, or both of the IFF Antennas. When BOTH has been selected, the Transponder will receive interrogations on both antennas but will transmit a reply only on the antenna that received the strongest interrogation.

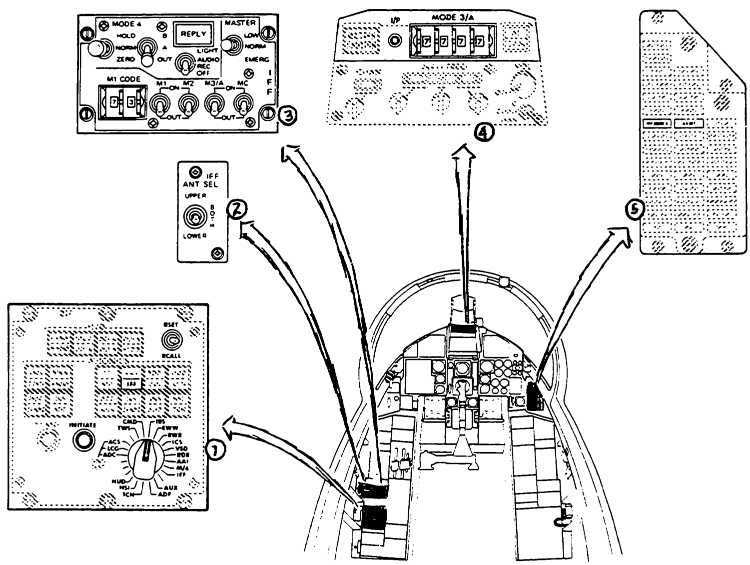
NORMAL OPERATION

The pilot has no indication resulting from the normal operation of the Transponder System in any of the SIF modes or Mode C. There is no indication available to him to show that the Transponder has been interrogated or that it has responded. These modes function completely automatically. Although the pilot has indication available to him of Mode 4 activity and how his Transponder is operating, the actual functioning of the Transponder System in Mode 4 is fully automatic as it is in the other modes.

In Mode 4, the pilot has both aural and visual indication of IFF activity. An aural tone indicates the Transponder System has been acceptably interrogated in Mode 4. The frequency of the aural tone is a function of the intensity of Mode 4 activity. An illuminated REPLY light indicates the Transponder has transmitted a reply to an acceptable interrogation. The two indications can be utilized simultaneously, separately, or not at all.

The normal selection would be to place the LIGHT/AUDIO REC switch in the AUDIO REC position and the A/B/OUT switch in A or B. This gives normal Mode 4 operation and gives the pilot both aural tone and the reply light capability. Selecting the LIGHT position disables the aural tone but the system remains in full operation. Placing the switch in the AUDIO REC position and selecting Mode 4 OUT

PRIMARY IFF CONTROL AND INDICATOR PANELS



1	BIT Control Panel	4	Main Comm Control Panel
2	IFF Antenna Select Switch	5	Caution Lights Display Panel
3	IFF Control Panel		

will disable the reply capability of the system but the receive capability will be retained. This is the "listening" mode we mentioned earlier. It provides the pilot with a strictly passive system with which to "listen" for Mode 4 activity.

Placing the LIGHT/AUDIO REC/OFF switch to the OFF position, with Mode 4 selected, will disable both the aural tone and the reply light but full system operation will be retained. The pilot will have full Mode 4 activity. With the LIGHT/AUDIO REC/OFF switch in the OFF position, the only indication of Mode 4 activity is the IFF Mode 4 Caution light. However, illumination of the caution light in this condition only warns the pilot that he has not replied to a number of valid Mode 4 interrogations. If at the same time the Transponder is also replying to valid interrogations (possible during marginal Transponder operating conditions), the pilot will not be aware of these replies. Had the LIGHT/AUDIO REC/OFF switch been in the LIGHT or AUDIO REC position, the Reply light would have illuminated; and illumination of the Reply light inhibits illumination of the Mode 4 Caution light. Therefore, with the switch in LIGHT or AUDIO REC, the pilot would have had an indication of proper Mode 4 operation instead of a warning that he was not replying. (Another aspect of this situation to consider is that, since illumination of any individual caution light — including the IFF Mode 4 light — also causes the MASTER Caution light to come on, the pilot receives a "master alert" that the system doesn't really deserve.) To get the most complete indication of how the IFF system is operating in a Mode 4 environment, the LIGHT/AUDIO REC/OFF switch should not be left in the OFF position. I hope the above discussion lays to rest any misconceptions among pilots and maintenance people concerning Mode 4 normal operations.

Over the years, the DIGEST has frequently printed material first appearing in one of the many squadron/wing level safety publications - this article from the 36th TFW BUSH RAG on the F-15 IFF system being the latest. If you fly the Phantom, Eagle, or Harrier and publish a safety sheet about your experiences, we'd like to consider publishing material from it, too. Please put us on your distribution list.

FAILURE & WARNING INDICATIONS

IFF BIT LIGHT

The IFF BIT light, on the BIT Control Panel, will come on when a failure occurs in the Transponder, the Transponder Computer (KIT), either IFF Antenna, or the IFF Control Panel during normal system operation. These components are monitored by a continuous BIT (built-in test) which will cause the IFF BIT light to come on steady when a failure occurs. When the IFF BIT light comes on, the AV BIT light will also illuminate.

IFF MODE 4 LIGHT

This light, sometimes referred to as the Mode 4 Caution light, will be illuminated to warn the pilot he is not responding to valid Mode 4 interrogations. The light will also be illuminated if the Mode 4 code has been zeroized (or no code has been set into the KIT) or if the IFF System is turned off (all mode switches to OUT).

When the system is not replying to valid Mode 4 interrogations, the Mode 4 Caution light will be illuminated for a minimum of three seconds and should remain illuminated until the missed reply condition is corrected. If interrogations are terminated, the light should go out after three seconds. The Mode 4 Caution light under certain conditions may appear to flash. This is due to the interrogation rate. This flashing or cycling on and off is particularly noticeable if the LIGHT/AUDIO REC/OFF switch is in OFF.

Confusion arises at times between the indication given by the IFF Mode 4 (Caution) light and the Reply light. The Reply light indicates the aircraft has replied to an accepted interrogation while the IFF Mode 4 light indicates it has not. So, why both indicators? Well, the Reply light is a normal system indicator, as explained earlier. Under certain operating conditions, it may even be turned off. The IFF Mode 4 light tells the pilot his system has been interrogated but has not replied. This warning indication will occur regardless of the operational capability of the Reply light or the Aural Tone. As I said earlier, the IFF Mode 4 light is strictly a warning indicator.

In the event any of these failure or warning indicator lights are on, the first thing to do is check to make sure all switches are in the correct position. If the failure indication persists, the IFF System-Initiated BIT should be performed to further isolate the malfunction or remove the failure indication. The performance of the BIT check often corrects or removes the fault that has caused the failure indication.

INITIATE BIT CHECKS

The IFF System-Initiated BIT checks out the entire IFF System except for Mode 4 operation. The Mode 4 BIT is performed in conjunction with the AAI System and will be discussed in the next article along with that system.

To initiate IFF BIT, all SIF modes and Mode C should be ON. The IFF BIT test should be selected on the BIT Control panel and the BIT-Initiate button pressed and then released. The IFF BIT light should flash for approximately 2 seconds and then go out. This indicates a good system. If the system has a fault, the IFF BIT light will stop flashing and then stay on steady.

It is possible to establish which mode (or modes) is malfunctioning by placing the mode switches to the OFF position one at a time. ■

AAI INTERROGATOR SET

(PUBLISHED 1978)

PART II – F-15 "IDENTIFICATION SYSTEM"

By FRED C. MUELLER/*Field Service Engineer, Bitburg AB, Germany*

The AAI portion of the F-15 Identification System is usually referred to as the Interrogator Set. The system is used to interrogate another aircraft in order to establish its identity and position. The aircraft being interrogated automatically responds with a coded reply transmitted via its Transponder. This coded reply is received and processed by the AAI System of the interrogating aircraft, and the intelligence thus derived is displayed to the pilot on the VSD (Vertical Situation Display).

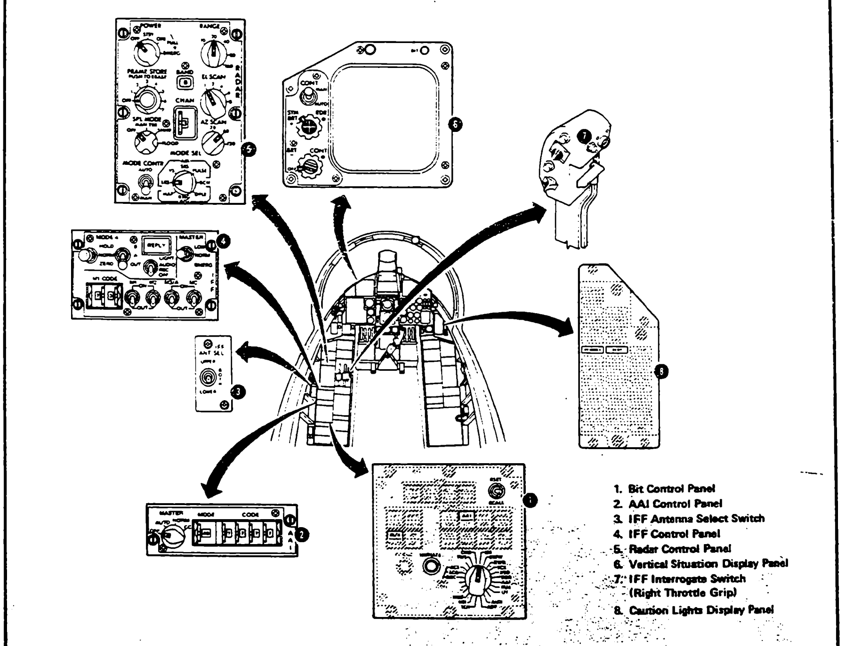
The AAI System operates in conjunction with the Radar System, which must be turned on in order to activate the AAI System. While the IFF Transponder System described in Part I of this series functions fully automatically, AAI System interrogation must be initiated by the pilot.

CHALLENGE CONDITIONS

There are three possible "challenge" conditions for the AAI System. These challenge conditions are selected by positioning the AAI Control Panel Master switch to the AUTO, NORM, or CC position. All interrogation modes (1, 2, 3, and 4A and 4B) are available in each of the above challenge conditions.

- AUTO – With the Master switch in AUTO, the AAI System is enabled to interrogate in a pre-determined "programmed sequence." In this challenge condition, the AAI System electronically steps through a pre-set sequence of interrogation modes.
- NORM – With the Master switch in NORM, the AAI System is enabled to interrogate in the system mode selected on the AAI Control Panel.

AAI CONTROL AND INDICATOR PANELS



• CC — With the Master switch in CC (Correct Code), the AAI System is enabled to interrogate in the mode selected on the AAI Control Panel. In SIF, only replies with the same code as the one set on the AAI Control Panel will be decoded.

The "programmed sequence" is controlled by selection of two ten-position switches located on the front of the IFF Reply Evaluator (IRE). These switches determine the interrogation mode sequence and the number of antenna scans the mode is repeated when in the AUTO challenge. In Normal or Correct Code operation, these switches determine the number of scans the selected mode is repeated. When the radar is in TRACK, these switches control the interrogation mode sequence during AUTO challenge but each mode is repeated for a fixed time period. In Normal or CC, this same fixed time period is used for the selected interrogation mode. The IRE program switches also set the identification threshold during Mode 4 interrogation.

NORMAL OPERATION

AAI interrogations are initiated by pressing the IFF Interrogate switch on the right throttle grip. The Interrogator transmits interrogations as long as the button is pressed or for the time period determined by the programmed sequence code. When the Radar System is in TRACK mode, the programmed sequence is repeated as long as the Interrogate button is pressed. This provides for continuous interrogation in RADAR TRACK mode. It should be noted here that in RADAR TRACK, when the AAI and radar have determined that the tracked target is a confirmed AAI target (target identified), AAI interrogation will be terminated. Further interrogation would obviously be pointless since you are only interested in identifying the target being radar tracked. With the new radar program, the interrogation cycle will be repeated approximately once a second. During each cycle when the target is identified, interrogations will cease. The AAI target will, however, be displayed on the VSD as long as the interrogation switch is held depressed. An "I" symbol will be displayed on the VSD during the interrogation period.

When the IFF Interrogate switch is pressed, a signal is sent to the radar. If the radar is in a compatible mode, the radar then commands the AAI System into operation and interrogation begins in the selected challenge condition. Radar modes of LONG RANGE SEARCH AND TRACK, SHORT RANGE SEARCH AND TRACK, PULSE, GROUND MAP, AND SNIFF are compatible with AAI operation. In all other radar modes, AAI operation is inhibited. In the hybrid modes of PULSE, GROUND MAP, and SNIFF, the radar commands the AAI into the Correct Code challenge condition.

In both AUTO and NORM, the SIF reply code brackets only are decoded. The actual code contained between the brackets is disregarded. This merely establishes that the reply is in the correct mode. Therefore, the AAI target must correlate with a radar target for positive position identification. In the event that there is no radar target with which the AAI target can be correlated, the Radar System commands the AAI System to check the reply for a correct code. In other words, the AAI System is switched to CC mode of operation for that particular target evaluation. If the reply has the correct code it will be displayed.

In the CC challenge condition, all replies must contain a correct code. In the SIF modes, the reply codes must match the code set on the AAI Control Panel. Only the replies with the correct code will be processed and displayed. No correlation with radar targets is necessary for positive identification. Mode 4 replies are automatically subjected to a correct code evaluation.

If acceptable responses are received from another aircraft, a diamond symbol will appear on the VSD. This symbol usually replaces a radar target. However, as we have stated, for correct code reply evaluation, a radar diamond symbol becomes a circle if, and only if, the AAI System accepts and identifies the response as a high confidence target. If, at any time, the reply evaluation circuits question the acceptability or confidence of a reply that has been accepted as high confidence, that target will immediately be down graded and the circle will be replaced with a diamond symbol.

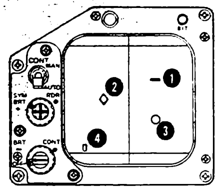
The "ID OFF" symbol will be displayed on the VSD if the AAI System is in the "overload" condition. This simply means that the target memory (storage), in the IRE, is full and there are more identified AAI targets than those displayed.

FAILURE INDICATIONS

AAI BIT LIGHT

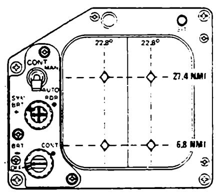
The AAI BIT light, on the BIT Control Panel (BCP), will come on steady when a fault occurs in the IFF

NORMAL AAI DISPLAYS

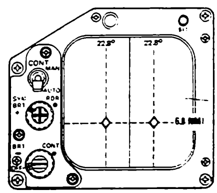


1. **Radar target, (IFF undetected)**
2. **IFF detected target (low confidence)**
3. **IFF identified target (high confidence)**
4. **Letter (I) displayed in character (1) position of Vertical Situation Display control panel BIT window whenever the interrogate switch is pressed or AAI BIT is initiated on BCP.**

AAI BIT DISPLAY WITH RADAR RANGE 40 SELECTED



AAI BIT DISPLAY WITH RADAR RANGE 20 SELECTED



Reply Evaluator, the AAI/Radar interface, or the Interrogator during normal system operation. These components are monitored by a continuous BIT which will cause the AAI BIT light to come on when a failure occurs. When the AAI BIT light comes on, the AV BIT light will also illuminate.

MODE 4 BIT LIGHT

The Mode 4 BIT light on the BCP will come on steady when a fault occurs in the Interrogator Computer (KIR) or the IRE Mode 4 video circuits. This light should not be confused with the IFF Mode 4 Caution Panel light. The Mode 4 BIT light on the BCP monitors the AAI Mode 4 system for failures while the IFF Mode 4 "Caution" light indicates the IFF System Transponder has not responded to good interrogations. The IFF Mode 4 Caution light tells the pilot the IFF Transponder System is not operating properly in Mode 4 and some action should be taken to determine why. The Mode 4 BIT light on the BCP is a more definite indication of an AAI Mode 4 system failure.

INITIATED BIT CHECKS

AAI

For initiated AAI BIT check, the following conditions are required to be pre-set:

I. RADAR
1. System on
2. Radar range set to 40
3. Azimuth scan set to 120 degrees
4. Long Range Search mode

(NOTE — If Radar Range is set to 10 or 20 or mode is SRS, only two targets will be displayed.)

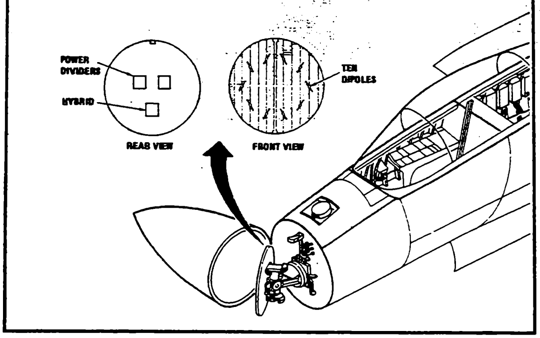
II. AAI
1. Master switch to AUTO, NORM, or CC
2. Code switch to 0000
3. Mode switch to any position

To initiate AAI BIT, select AAI BIT test on the BIT Control Panel and then press the BIT Initiate button. The AAI BIT light will flash for approximately five seconds and then go out. Four AAI target symbols (diamonds) are displayed on the VSD; two at 27.4 NM and two at 6.8 NM with an azimuth of +22.8 degrees. This indicates a good system.

The AAI BIT light will remain on steady after the BIT check period if a fault has been detected in the system. Should the AAI targets not be displayed, a failure is also indicated.

Failure to properly preset the system (both Radar and AAI) before initiating AAI BIT can cause the AAI BIT light to indicate a failure or cause

AAI SYSTEM ANTENNA LOCATIONS



the BIT target presentation to be incorrect, which is also a failure indication. Such a fault can (and should) be eliminated by properly presetting the system and then repeating the AAI BIT. If the fault indication persists, there is a system failure.

MODE4

The Initiated Mode 4 BIT checks out the Mode 4 operation of both the IFF (Transponder) System and the AAI (Interrogator) system. This is done by a "loop" check of the two systems, which simply inhibits the Receiver blanking signals and then interrogates the Transponder. The Transponder automatically transmits replies which are picked up and evaluated by the Interrogator. The whole operation is basically the same as normal Mode 4 activity; the only difference is that the AAI Mode 4 targets are not displayed on the VSD as they would be normally.

During this check, the pilot should have both an aural tone and a Reply light as he would have normally. Also, the pilot should observe the IFF BIT light and the AAI BIT light for failure indications resulting from the continuous BIT of these two systems. This is also a normal system function that becomes an integral part of the Mode 4 BIT.

Now let's see what actually happens in an Initiated Mode 4 BIT check. First of all, the following conditions are required to be pre-set:

I. RADAR
1. System on
2. Long Range Search Mode
3. Azimuth scan set to 120 degrees
4. Antenna Elevation centered or lower

(NOTE — It is possible for the interrogation signal transmitted through the AAI antennas mounted on the radar antenna to swamp the IFF Transponder receiver. This can give a BIT fail indication. The indication will be either an IFF BIT light or no Reply light. Therefore, should the BIT fail, lower (droop) the radar antenna with the elevation wheel on the throttle grip, and then repeat the BIT.)

II. AAI
1. Master switch to AUTO, NORM, or CC
2. Mode switch to the same Mode 4 program (4A or 4B) that is set on the IFF Control Panel.
3. Upper IFF Antenna selected

III. IFF
1. HOLD/NORM/ZERO switch in NORM
2. A/B/OUT switch to program 4A or 4B.
3. LIGHT/AUDIO REC switch in AUDIO REC

To initiate Mode 4 BIT, select Mode 4 BIT test on the BIT Control Panel and press the BIT Initiate button. The Mode 4 BIT light will flash for approximately five seconds and then go out. The diamond target symbol will not be displayed in Mode 4 BIT; however, the VSD may display the ID OFF symbol. This should be disregarded because it is not pertinent to Mode 4 BIT. The aural tone and the Reply light will indicate Mode 4 activity in normal manner. Finally, the pilot should observe the IFF BIT light and the AAI BIT light for any indication of Transponder or Interrogator System faults through their respective continuous BIT monitor circuits.

If the Mode 4 BIT light remains on after the BIT sequence has been completed or if it does not flash at all but stays on steady during and after the BIT, it is an indication of a failure. If either the IFF BIT light or the AAI BIT light comes on steady, it is also an indication of a failure.

So there you have it, that's how the Identification System works on the Eagle. I think you pilots should now have a pretty good idea of how the system works and what you can expect from it. I hope you will have the confidence in the system that it deserves. It will do the job for you.

NOTE: The radar contains a computer program (software) that is updated or changed periodically. Since these changes usually affect operation of the AAI System in some manner, some of the information contained in this article could change in the future. ■

(PUBLISHED 1980)

SEEING IS BELIEVING... LOOK AGAIN!

When we used the F-15 action photo from Bitburg AB, Germany for our 4/79 DIGEST front cover, and wondered whether the Eagles were on their way up or down, some response from you readers was expected, but certainly not the volume received.

It seems as though every opinion was backed by "indisputable" evidence to prove that the Eagles were landing or taking off. We heard from military personnel, MCAIR employees, and even some of the vendors. Some "evidence" was scientific and some preposterous, (and half of it had to be wrong) but it was all interesting. Here are some of the more creative responses:

Intelligence Officer, U.S. Naval Reserve - "Landing, the trees in the background have been blurred by the camera panning on the planes. It appears to be a slow pan. If it were a takeoff shot, the pan would have to be faster since the planes would be going faster."

Artist, MCAIR - "Takeoff, the nose wheel is rotating too fast to be a landing." (Individual claims he has an eye for detail.)

District Manager, Vendor - "Takeoff." He sent along a photo from later in a takeoff sequence to support claim. Good try, but it was a different Wing's aircraft.

Field Service Engineer, MCAIR - "It's obvious they are landing by looking at the photo. I'm surprised those stick jocks didn't notice it. If you look close, you can see the LOX indicator reflection in lead's canopy shows only 1/2 liter left!"

(Another) Field Service Engineer - "The second F-15 is taking off. Eagle claw (forward righthand AIM-7 retainer) is still perpendicular to the airstream. Normally with no AIM-7 aboard, these will tend to swivel into the airstream during flight and are turned back by the crew chief. Obvious to the most casual observer!"

To resolve the matter once and for all, we contacted the Bitburg folks, to find that the only person who could give us the answer was the photographer, SSGT Lopez. The sergeant confirmed that it was a landing shot. We thought for sure the case was closed until he added, "But I think they took right off again, you know, a touch and go!"

Enough is enough! One USAF Phantom Phlyer neatly solved the identification problem, at least for himself. "Who cares, they're only F-15s." ■

ASIP and the Eagle F-15 Fatigue Tracking Program

(PUBLISHED 1977)

By RICHARD E. PINCKERT/*Senior Technical Specialist and* RONALD A. MELLIERE/*Lead Engineer-Technology*

A lot of g's have clicked through the counters since the DIGEST first talked about structural fatigue, statistical accelerometers, and the F-4 Aircraft Structural Integrity Program(ASIP) back in 1973. During those four years a new Weapon System — the F-15 Eagle — has been accepted by Air Force organizations in quantity. Because the Eagle falls within the Aircraft Structural Integrity Program, and because life projections of the aircraft are dependent upon the fatigue tracking program, we have been asked to update the 1973 coverage. We hope the following article will answer any questions you might have about the ASIP/Eagle combination.

The F-15 fatigue tracking program is an important part of the overall F-15 ASIP. It is an on-going process for recording flight parameters for operational aircraft, converting those parameters into "airframe load forces" applied to the aircraft during flight, and calculating the percentage of structural fatigue life expended. Through ASIP, the customer can evaluate aircraft mission utilization and maintenance scheduling requirements.

Specifically, the F-15 ASIP is a comprehensive plan centering around four important objectives:

- To establish, evaluate, and substantiate airframe strength and durability (structural integrity).
- To assess continuously the in-service integrity of individual airplanes by utilizing operational usage data.
- To provide a basis to establish logistic support and aid in planning future aircraft utilization (maintenance, inspection, supplies, rotation of airplanes, and system phaseout).
- To collect usage data to aid in development of improved structural criteria and methods of design, evaluation and substantiation for future aircraft systems.

The first of the above objectives (structural integrity) was attained during the design, test, and development phase of the F-15. The F-15 Fatigue Tracking Program (as shown in the flow chart on page 23) contributes directly to the remaining three objectives.

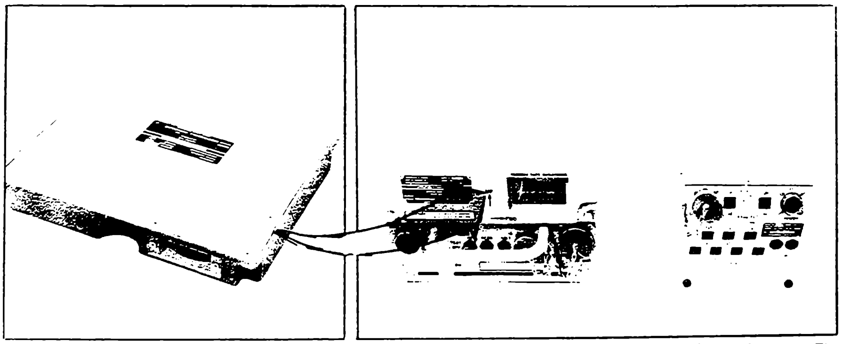
The tracking program consists of four phases: data collection, data reduction, fatigue damage analysis, and fleet management and future aircraft design.

DATA COLLECTION

The data collection phase is a crucial part of the fatigue tracking program since the remaining phases are only as reliable as the data collected. The responsibilities and procedures for collecting and reporting F-15 service usage data have been spelled out in Technical Order 1F-15A-2-2-4.

On the F-15, usage data are recorded automatically utilizing an Exceedance Counter Set, and a multichannel tape recorder/Signal Data Recording Set (SDRS). An Exceedance Counter Set is installed in every aircraft while the more sophisticated SDRS is factory installed in every fifth operational aircraft.

The SDRS automatically records significant flight parameters on a cassette tape from transducers installed in the aircraft. The SDRS (pictured below in center) located in Door 47L, consists of a Signal Data Recorder and a tape cassette with a 25-hour recording capacity. A total of 21 flight parameters, including altitude, true airspeed, angle-of-attack, and vertical load factor, are recorded continuously to provide a basis for accurately determining the damage accumulation rate.



Tape cassette (left) records twenty-one flight data points gathered by the multichannel Signal Data Recording Set/SDRS (center). The Exceedance Counter Set (right) records positive and negative "g" loads.

Documentary data (flight date, mission code, aircraft serial number, squadron number, and weapons identification) must be manually fed to the recorder tape at the beginning of each flight by the pilot and ground crew through the Navigational Control Indicator and Armament Control Panel. Proper insertion of the documentary data is mandatory if the data on the SDRS cassettes are to be usable.

When the magnetic tape in a cassette has been expended, as indicated on the Avionics Status Panel, the cassette is removed by the using command and shipped to Tinker AFB for data reduction. Since each aircraft containing an SDRS also contains an Exceedance Counter Set, an accurate determination of damage from SDRS data provides the means for determining damage from exceedance counter data which are available for all aircraft.

Each aircraft contains an Exceedance Counter Set (pictured to the right on page 22) to provide individual aircraft usage data. It consists of an accelerometer transducer located in the right main landing gear wheel well and a counter display unit located inside the aft end of Door 6R. The transducer continuously measures the aircraft vertical load factor while the counter display unit automatically records and displays the number of times the aircraft has been subjected to each of seven load factor levels: three negative (-2G, -1G, 0G) and four positive (+3G, +4.5G, +6G, and +7.5G).

Exceedance counter readings, together with flight log information, are recorded manually by the using command after each flight on AFTO Form 239 (as shown in the illustration on page 24). The forms have been designed in such a way that they can be read automatically through the use of optical scanning equipment. Completed forms are sent to Robins AFB for data reduction.

The mission type and aircraft gross weight affect the amount of damage caused by a given load factor occurrence. Therefore, flight log information recorded on the forms includes data necessary to associate load factor occurrences with mission type and average aircraft gross weight. The combination of load factor occurrences, aircraft gross weight, and mission type for a given flight is converted to fatigue damage during the "fatigue damage analysis phase" of the fatigue tracking program.

As major aircraft components are "changed" AFTO Form 238 is filled out (in accordance with Air Force Technical Order 1F-15A-2-2-4) by the using command and forwarded to Robins AFB for data reduction. The specific components for which AFTO Form 238 should be filled out are wing assemblies, stabilators, and serialized landing gear parts. This provides a way to monitor the fatigue damage on individual serialized aircraft components which were removed for repair or overhaul and installed on a "different" aircraft or on the "opposite side" of the same aircraft.

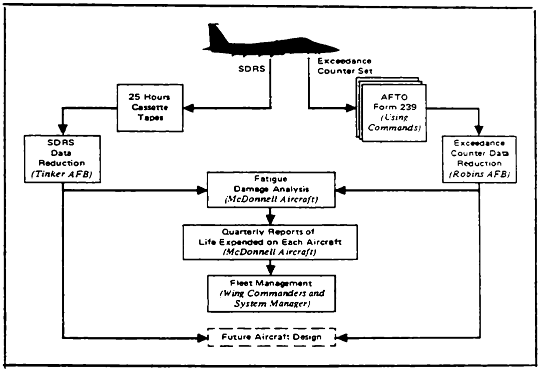
DATA REDUCTION

SDRS data are reduced by Tinker AFB personnel using ground playback equipment and computer programs. This is a multistep process which includes:

- Elimination of nonsignificant data such as straight-and-level flying.
- Deletion of erroneous data (for instance, from malfunctioning transducers).
- Conversion of remaining data to stress-time histories at fatigue-critical locations and to tables summarizing pertinent usage information.

Exceedance counter/flight log data and component tracking data collected on AFTO Forms 239 and 238 are reduced by Robins AFB personnel. Optical scanning equipment is used to extract data automatically from the forms. The exceedance counter/flight log data are then checked for validity and arranged into a flight-by-flight time sequence using a computer technique.

Periodically the SDRS data reduced by Tinker AFB, and the exceedance counter/flight log data and component tracking data reduced by Robins AFB, are sent here to McDonnell Aircraft Company for fatigue damage analysis.



F-15 Fatigue Tracking Program (ASIP)

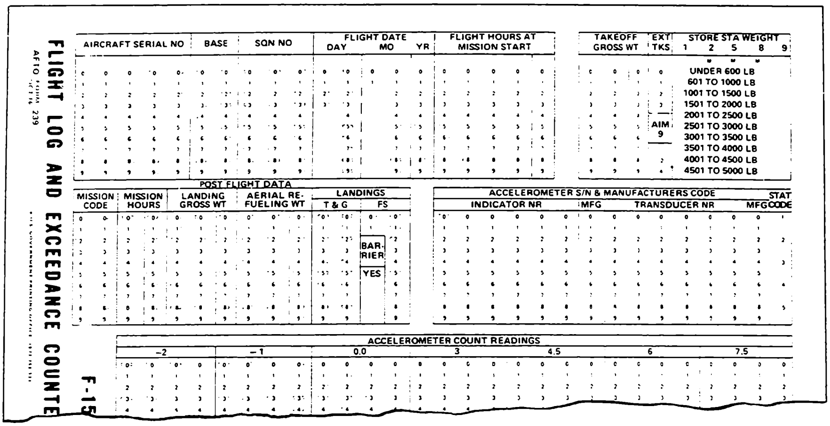
FATIGUE DAMAGE ANALYSIS

The F-15 fatigue damage analysis involves determining how much aircraft fatigue life has been expended by the wear and tear of day-to-day maneuvering. The fraction of fatigue life expended (the accumulated damage) is expressed as a damage index less than or equal to 1, where the limiting value of 1 indicates that fatigue cracks are predicted to have developed in the structure.

Damage estimates for all aircraft are made using a computer program which determines and totalizes the damage for each maneuver. The data fed into the program are a time sequence of stresses applied to each of several fatigue-critical locations. For aircraft containing an SDRS, the stress-time histories are directly available from data reduced by Tinker AFB.

For those aircraft containing only an Exceedance Counter Set, the sequence of load factors is determined on a flight-by-flight basis from the data reduced at Robins AFB. Each recorded load factor is converted to a stress for each critical location, based on the mission type and aircraft gross weight for the mission. The conversion from load factor to stress is based on the tabulated load factors and stresses associated with similar maneuvers on aircraft which contain an SDRS.

Damage for individual removable aircraft components is tracked with the aid of component tracking information (AFTO Form 238) reduced by Robins AFB. Damage accumulation estimates are updated and reported quarterly to the Air Force by McDonnell Aircraft Company for each operational F-15 aircraft. ►



The primary objective of the F-15 fatigue tracking program is to aid in fleet management. This will be accomplished in two ways:

• Usage data will provide the means to assess effects on remaining fatigue life when a new mission is dictated for all or part of the fleet.

• The quarterly fatigue damage report will identify to the Wing Commander and the System Manager (Robins AFB) by tail number how much fatigue life has been consumed by prior flight history. The Wing Commander, at his discretion, can schedule his aircraft missions to even out fatigue life consumption. This may involve reassignment of an aircraft to missions where the usage is less strenuous. The Wing Commander will also be able to schedule major TCTO compliance more effectively. Similarly, the System Manager can use the damage indices to schedule aircraft for Analytical Condition Inspections (ACI).

The usage data collected in the F-15 fatigue tracking program will contribute directly to the design of future aircraft systems. For instance, the speed, altitude, and gross weight at which load factor maneuvers are performed on the F-15 will provide a valuable data base for fatigue considerations in the design of future high performance fighter aircraft.

IN SUMMARY . . . The service usage data collected on the F-15 Eagle will not only contribute directly to the well-being of the "Eagles of today" but also to the design of future fighter aircraft. The lead role played by the using commands in collecting and reporting "valid" data is vital to the F-15 Aircraft Structural Integrity Program. ■

While flying and fighting are always the watchwords of the USAF, even on the best of bases all work and no play makes Sgt Jackson a dull NCO. Here are a few scenes from some recent extra-curricular activities at Luke AFB which prove that there is always time for children, wives, and music. "We Could Make Sweet Music Together" might be an apt tune for the Air Force Band of the Southwest to play as it poses in front of a Luke AFB "Triple Nickel" Eagle.

F-15 FATIGUE TRACKING PROGRAM

(PUBLISHED 1981)

By RICHARD E. PINCKERT/*Section Chief - Technology*; and RONALD A. MELLIERE/*Unit Chief - Technology*

More than three years have gone by since the DIGEST last published an article about the fatigue tracking program on the F-15. During that time, a considerable amount of usage data has been collected from the Signal Data Recorder Set (SDRS) and Exceedance Counter Set located in the F-15 Eagles. Of the information collected, approximately 50% of the SDRS data and 90% of the Exceedance Counter data are valid. An improvement in the quantity of valid data is a goal that can be attained with your assistance.

In June 1980, the first "Service Aircraft Fatigue Estimate" (SAFE) Report was written by McDonnell Aircraft Company to inform the Air Force how much fatigue life had been expended on each F-15 aircraft. SAFE reports are updated every three months and are sent to the F-15 SPO, Robins AFB, TAC, USAFE, PACAF, and Tinker AFB. We felt that now was an appropriate time to explain how the Signal Data Recorder and Exceedance Counter information that you have been providing is being used.

The F-15 fatigue tracking program is a part of the F-15 "Aircraft Structural Integrity Program" (ASIP). In the fatigue tracking program, data are collected from F-15 airplanes and sent to Tinker AFB and Robins AFB for reduction (see Figure 1). From the reduced data, MCAIR then performs a fatigue damage analysis for each aircraft and publishes the results in quarterly SAFE reports. These reports aid in fleet management, and the usage information from the SDRS and Exceedance Counter Sets is valuable for solving in-service problems and for designing future aircraft. Currently eleven F-15 critical locations are monitored in the fatigue tracking program, as illustrated in Figure 2. The damage analysis approach employed for the F-15 utilizes the SDRS and Exceedance Counter data to determine the fraction of the crack initiation life expended (i.e. fatigue damage) at each location for each F-15 aircraft in the fleet. "Crack initiation life" is defined as the number of flight hours required to develop a 0.01 inch deep crack.

SAFE REPORTS

The fatigue damage values which are calculated for individual aircraft are presented as tables in the SAFE reports. The tables include the cumu-

FIGURE 1 - F-15 SERVICE AIRCRAFT FATIGUE PROGRAM

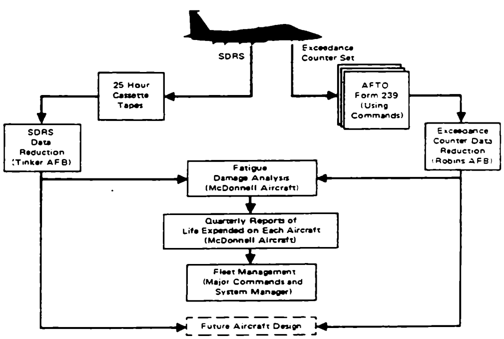
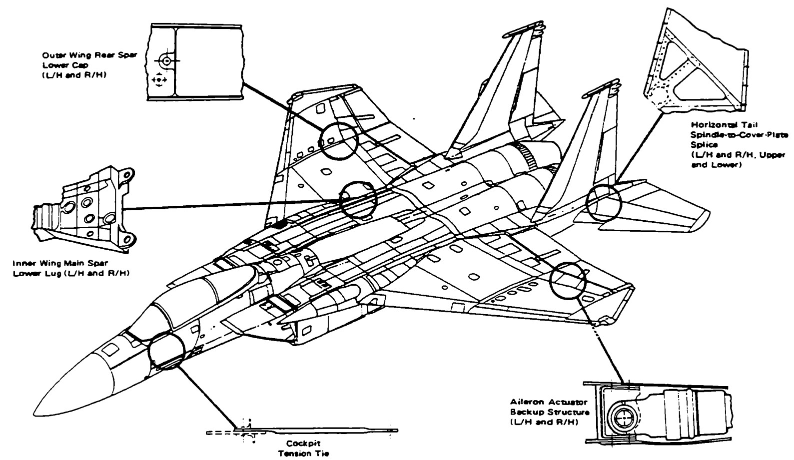


FIGURE 2 - FATIGUE CRITICAL LOCATIONS MONITORED IN SAFE REPORTS



lative fatigue damage to date for each of the eleven critical locations for each aircraft, and a projection of fatigue damage into the future to assist in maintenance scheduling and aircraft rotation planning. The fatigue damage projection is reported as the date at which the most critical location on the aircraft will reach its next "quarter life." For example, the high time operational aircraft reported in the 1st SAFE report was S/N 73-090. This aircraft had a cumulative damage of .09828 for the inner wing main spar lower lug, indicating that 9.8% of its crack initiation life had been expended. The aircraft is projected to reach its next quarter life (i.e. fatigue damage of .25) in late 1983. When the quarter life is reached, an inspection of the critical location will be performed. A typical page from a SAFE report is shown in Figure 3.

SOLUTION OF IN-SERVICE PROBLEMS

The SDRS data are also used frequently to assist in solving F-15 structural problems that occur in service. Based on SDRS data, engineering analyses can be performed to determine why cracking occurred in service, and to determine how a retrofit or design change should be implemented to prevent similar problems in the future.

The following examples of how SDRS data were used to solve in-service problems show how very important the information is in assessing fleet usage and in performing fatigue analyses of the F-15:

• Upper Inner Wing Skin Buckles - SDRS analyses of wing bending and torsion provided information to determine the cause of upper inner wing skin buckling. As a result, skins were beefed-up on F-15C/D models, and the USAF warned pilots about overloading the airplanes. An aural Overload Warning System has since been developed to warn the pilot of an approaching overload condition. (See previous article for a discussion of this new system.)

• Vertical Tail Buffet - SDRS data were used to define the high angle of attack buffet environment to which the vertical tail is subjected. After performing vibration and fatigue analyses and conducting tests, changes in the upper vertical tail design were made to reduce the probability of structural cracking.

• Upper Outer Wing Buffet - Analyses of upper outer wing skin cracking were performed using SDRS data as a basis. The primary cause appears to be buffet in the moderate angle of attack range. Design changes based on fatigue analyses and tests have since been made to the upper outer wing skin and some wing ribs.

CONTINUED SUCCESS DEPENDS ON YOU!

Continued success of the fatigue tracking program hinges on a reliable data collection effort. The quality and quantity of usage data incorporated in the tracking program are directly dependent on the USAF F-15 using commands. That is, a continuing and conscientious effort by USAF personnel in the field is required to record the proper data and ensure its reliability through timely mainten- ▶

FIGURE 3

FIGURE 3-1 (CONTINUED)

FLEET SUMMARY OF FATIGUE DAMAGE FOR INDIVIDUAL F-15 AIRCRAFT

LAST REPORTING DATE
AUGUST 1980

AIRCRAFT SERIAL NUMBER	TOTAL HOURS	FATIGUE DAMAGE FOR LOCATIONS 1 THROUGH 11 /1/ 1	2	3	4	5	6	7	8	9	10	11	PROJECTED DAMAGE DATE /2/ QTR YR	STP LIFE	CRIT LOC
780495	283.	0.00284 0.00912	0.00301 0.00947	0.00111 0.00377	0.00111 0.00371	0.00032 0.00136	0.00031 0.00127	0.00009 0.00036	0.00007 0.00028	0.00313 0.01354	0.00023 0.00093	0.00025 0.00099	2 1998	0.25	9
780496	227.	0.00228 0.00543	0.00240 0.00565	0.00098 0.00252	0.00097 0.00248	0.00036 0.00103	0.00035 0.00093	0.00010 0.00027	0.00008 0.00020	0.00345 0.01041	0.00026 0.00074	0.00028 0.00079	3 1998	0.25	9
780497	329.	0.00040 0.01043	0.00041 0.01065	0.00019 0.00476	0.00018 0.00465	0.00007 0.00161	0.00007 0.00152	0.00002 0.00044	0.00002 0.00034	0.00071 0.01557	0.00005 0.00096	0.00005 0.00102	2 1998	0.25	9

ance of the SDRS and Exceedance Counter Systems. The following list, though not inclusive, represents some of the contributions you can make toward a reliable data collection effort:

(1) Ensure that SDRS cassettes are replaced as soon as possible after they become filled. It is essential to continually record an adequate SDRS data sample from all types of usage, including data from special exercises. A "biased" (incomplete) data sample would lead to inaccurate fatigue predictions.

(2) Perform necessary repairs/replacements to malfunctioning SDRS and exceedance counter equipment as soon as possible so as to collect the largest possible amount of quality data. Repair and replacement delays cause the loss of valid data and/or retention of invalid information.

(3) Record flight log and exceedance counter data for each aircraft after each flight on AFTO Form 239. Proper recording of aircraft hours at mission start, exceedance counter readings, mission code, gross weight information, etc., are essential to an accurate data record.

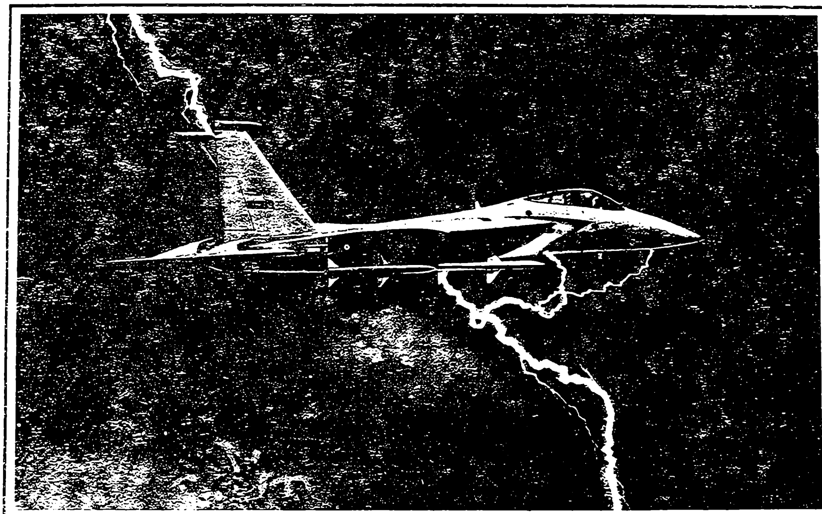
(4) Document wing and stabilator changes as they occur on AFTO Form 238, as specified in TO 1F-15A-2-2-4. Proper recording of component changes is essential to an accurate fatigue prediction for wing and stabilator components on each and every aircraft.

In summary, the F-15 fatigue tracking program has been successful to date. It has helped both MCAIR as contractor and the Air Force as customer to "know" all of these Eagles better. Continued success (which really means longer lasting, more useful aircraft) depends to a great degree on your continued contributions to a reliable data collection effort. ■

FOUR OF A KIND/FOUR AT A TIME

"Eagle Experts." On 2 March 1981, "Cowboy 1, 2, 3, and 4" took off from Luke AFB, Arizona. Some 1.3 hours later, the flight returned, and all four pilots had just logged their 1000th F-15 flight hour. From left, they are flight leader Major Mel Ezell and Captain Mark Beesley of 550th TFS; and Captains Steve Knight and Ralph Aguirre of 461st TFTS.

"Lightning is a force to be reckoned with, and aircraft and electrical engineers are constantly at work, trying to understand more fully how this phenomenon functions." This article, first in a series, is our way of telling you where we've been, where we are now, and where we think we're going in making aircraft less susceptible to lightning damage.



The Eagle Looks at Lightning

(PUBLISHED 1976)

By ROBERT ASTON/*Senior Engineer – Electronics*

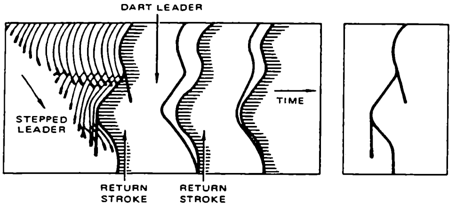
Ever since the F-15 was conceived, Eagle designers have "looked at lightning" with the aircraft in mind. The inevitable finally happened at 11:40 a.m., 30 April 1976; F-15A Serial Number 71-0289, piloted by Joint Task Force pilot Capt M. E. Durbin, was struck by lightning as it was flying a routine mission from Eglin AFB, Florida.

In the process, the radar warning receiver (RWR) system became inoperative, and although the damage was minor, the flight was quickly terminated. During the postflight inspection, it was found that the radome showed signs of lightning damage and there was a hole in the left vertical electronic warfare warning system (EWWS) antenna radome ring. There also were apparent lightning effects upon the IFF dipole antenna, the total temperature probe, and the right aft AIM-7 dummy missile.

This first F-15 incident reemphasized the unnerving and potentially destructive power of lightning. Man has discovered no way, up to this day, to prevent lightning, but man *does* have the knowledge to design aircraft in such a way that lightning strikes will cause only minor damage. Our continuing job is to assure that any skirmishes that Eagle pilots like Capt Durbin have with lightning remain in the "no" or "minor" damage category. There are many ways to provide this assurance, from fundamental shielding and bonding all the way to various sophisticated means of restricting lightning current to the outside of the aircraft. But before we look at ways to minimize its effects, let's take a look at lightning from a purely physical point of view.

(Incidentally, there have been many good articles published about lightning in the past; it would be well worth your while to browse through back issues of AIRSCOOP/Headquarters USAFE, AEROSPACE SAFETY/Air Force Inspection and Safety Center, TAC ATTACK/Headquarters Tactical Air Command, and APPROACH/Naval Safety Center. These publications have "looked at lightning" many times over the years. And if you have a file of back issues of the DIGEST, you will want to read Jack Sheehan's article,

FIGURE 1 – DEVELOPMENT OF A LIGHTNING STROKE



"The *Phantom* Looks at Lightning"/3rd Quarter 1970; it provides an excellent introduction and background for what we will be talking about in this series of articles.)

GETTING DOWN TO BASICS

Before we consider lightning as it applies to aircraft, let's review a few basic facts about this natural phenomenon. A lightning stroke begins with an invisible, downward-moving, traveling spark called a *stepped leader* (Figure 1). The stepped leader moves toward the ground in approximately 50 yard steps. Time between steps is in the order of 50 microseconds. A typical stepped leader has about five coulombs of charge distributed over its length.

When a stepped leader reaches the ground, the leader channel first becomes highly luminescent; this bright, visible channel is called the *return stroke*. The return stroke propagation time is approximately 100 microseconds. Additional charge can be made available to the channel top by the action of electrical discharges which move upward from the top of the previous return stroke into higher areas of the cloud. When this additional charge is available, a continuous lead, known as the *dart leader*, moves down the defunct return stroke channel. The dart leader then sets the stage for the second (or any subsequent) stroke. The dart leader takes a few milliseconds to reach the ground because of the time it takes the electrical field to break down the air (approximately 30,000 volts per centimeter at one atmosphere of pressure).

Figure 2 is a typical current waveform of a lightning strike which has two phases: a *high current phase*, and a *heavy coulomb phase*.

- The high current phase has a fast rise-time in the order of a few microseconds, lasting up to 100 microseconds with average current ranging from 10-20 thousand amperes to as high as 200 thousand amperes (the MIL Spec value). One to ten coulombs of charge can be experienced.
- The heavy coulomb phase offers one to five thousand amperes, in the millisecond range, with lower currents lasting up to one second. This phase develops from 100 to 200 coulombs of charge.

LIGHTNING AND AIRCRAFT

At this point, let's introduce an inflight aircraft; what will this aircraft have to do with lightning? An aircraft cannot *generate* lightning but it can *trigger* a stroke while flying in a potential source such as a charged cloud. Lightning not triggered by the aircraft would, at best, only be diverted; the total path of the lightning travel is not influenced by an aircraft. The lightning step leader attaches to some sharp extremity on the aircraft, leaving through another sharp extremity. When the step leader attaches to the aircraft, it may produce streamers from such things as static dischargers or points such as antennae or exhaust outlets, with possibility of attachment to them. However, the component itself will not trigger lightning. Static charges on an aircraft structure increase the lightning hazard but static dischargers tend to reduce the risks.

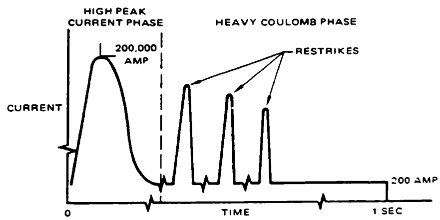
It takes approximately one millisecond for a step leader to reach the earth. Assuming that an aircraft is at 2500 feet, and is travelling at 300 knots (approximately 440 feet per second), the plane travels just six inches in the one millisecond it takes for the step leader to reach the earth. When the step leader reaches the earth, the return stroke returns to the cloud, via the aircraft. As you can see, so far as lightning is concerned, the aircraft is almost stationary.

VULNERABLE AREAS

Assuming that the aircraft has been struck by lightning, what is the likelihood of damage? Five aircraft areas are vulnerable to lightning damage: fuel system, lights, canopy, probes, and radomes. Let's consider each of these areas in some detail.

- There are four fuel areas which might be affected by lightning: external tanks, internal wing tanks, fuel vents, and dump masts. A lightning strike on any of these could do damage which could range from a small hole to complete loss of an aircraft through ignition of fuel vapor. If a volatile mixture is available at a dump line when lightning attaches, the lightning could follow the line into the tank and explode. Obviously, this is an area of major concern, and one upon which industry focuses much study and work. The F-15 has been designed and tested to assure minimum damage from a lightning strike

FIGURE 2 – COMPOSITE NATURAL LIGHTNING STRIKE CURRENT WAVEFORM



to the fuel area.

• Lightning has the potential to blow a light apart; at best a lens might be cracked, or some of the metal might be burnt away. Lightning might be coupled back into aircraft power systems through lights, causing other types of failures. During the F-15 program, tests were made using simulated lightning strikes to lights and no major damage was observed.

• It is believed that canopies can be punctured by lightning with the possibility of pilot shock; however, we have never seen reports of this being a problem. Figure 3 shows a laboratory experiment on a simulated canopy. In the F-15 program we have performed extensive tests on actual and simulated canopies using artificial lightning. Neither a thin canopy mockup or the actual F-15 canopy could be punctured with simulated canopy strikes using voltages as high as 1.4 megavolts and currents as high as 200 thousand amperes. There *have* been reports of flash-blindness from lightning (temporary in most cases).

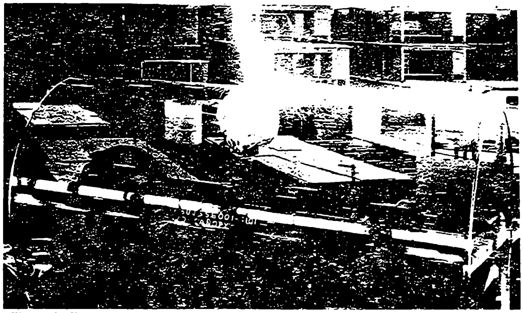


Figure 3. Voltages as high as 1.4 megavolts and currents as high as 200,000 amperes failed to puncture simulated canopies.

• As sharp pointed objects, angle-of-attack and pitot-static probes, and total temperature sensors, are prime targets of lightning which melts tips, deforms orifices, and freezes bearings. In addition, electrical currents can be coupled into the aircraft power system, causing additional problems. Figure 4 shows the melted tip of an angle-of-attack probe; when lightning hit this device the tip was melted and the internal bearings were frozen. Current moved from the tip, through the bearing, into the aircraft structure, disabling the sensing device. The F-15 has lightning arrestors on items such as these to prevent lightning currents from entering the aircraft.

• If a radome has a pitot tube, lightning can attach itself to the tip. Electrical currents can then pass down the probe, through the pitot heater wiring, into the aircraft power system. This may or may not destroy the radome. Additionally, lightning can weaken small pieces of the radome, or even blow them out. Where a radome does not have a pitot tube, lightning may attach to the tuning wire, attachment rings, or just go through the radome to the radar antenna.

In any case, lightning is a force to be reckoned with, and aircraft and electrical engineers are constantly at work trying to understand more fully how this phenomenon functions. As we progress in this series of articles, we'll look more closely at things that have been done to make aircraft less vulnerable to lightning. In the process, we'll be sharing what we have learned from the F-4, and what there is in the way of F-15 lightning protection.

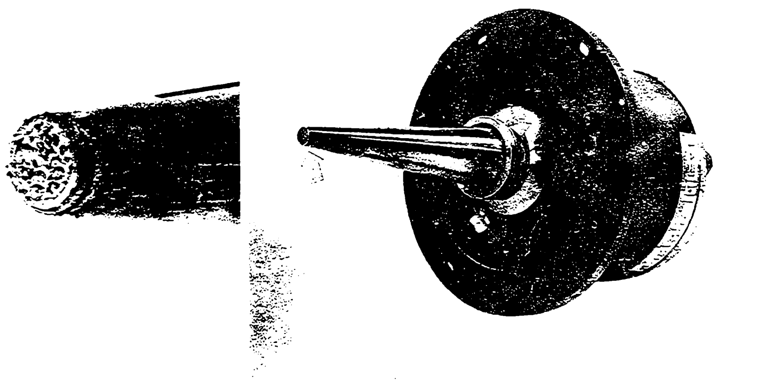
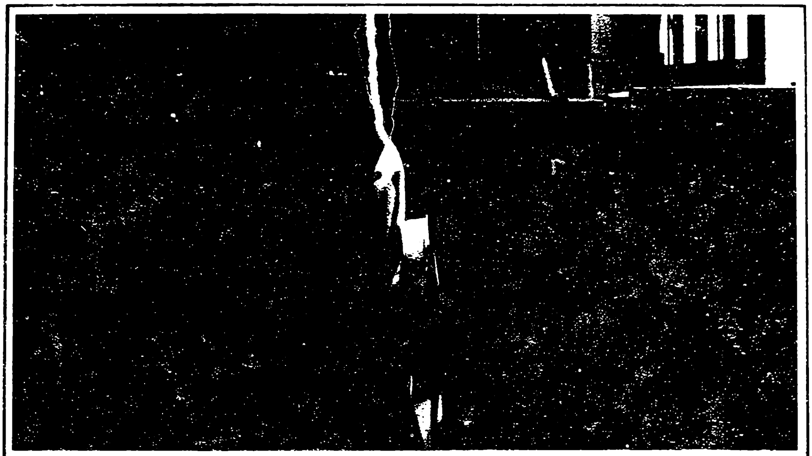


Figure 4. Evidence of a lightning strike is clearly shown on this F-4 angle-of-attack transmitter. On the right is an illustration showing three points where lightning attached to the unit. The illustration to the left shows an enlarged view of the damaged probe tip.

Part III (Part II of this series on F-4) (PUBLISHED 1977)



The Eagle Looks at Lightning

By ROBERT ASTON/*Senior Engineer - Electronics*

Having been introduced to the phenomena of lightning in the first article in this series, and having reviewed some F-4 lightning strike history in the second, let's look at how we determined what lightning protection was required for the F-15.

SCALE MODEL

The first thing we need to know is where lightning will attach on a new aircraft. Since this information can be obtained through use of model aircraft, we began F-15 tests with what is called the "attach point test" (once the "attach" points are known, we can design protection into the aircraft as required). Figure 1 shows a typical model mounted on a test stand as it was struck by simulated lightning (the model is 1/16th scale and is copper-plated to simulate the metal section of the aircraft). The test stand can be rotated through all axes, providing every possible strike point. The "lightning" voltage and current are in the order of two-million volts and four-hundred amperes and this power is produced by a Marx generator.

The F-15 model tests were performed at the Lightning and Transient Research Institute, Miami, Florida, in 1970. McDonnell Aircraft Company, since then, has developed its own lightning research facilities and the F-18 attach point tests were accomplished in St. Louis.

CAPTURE AREAS

The lightning strike points (or capture areas) identified during the tests in the vertical plane (zero yaw and with model rotation about the pitch axis), are indicated in Figure 2.

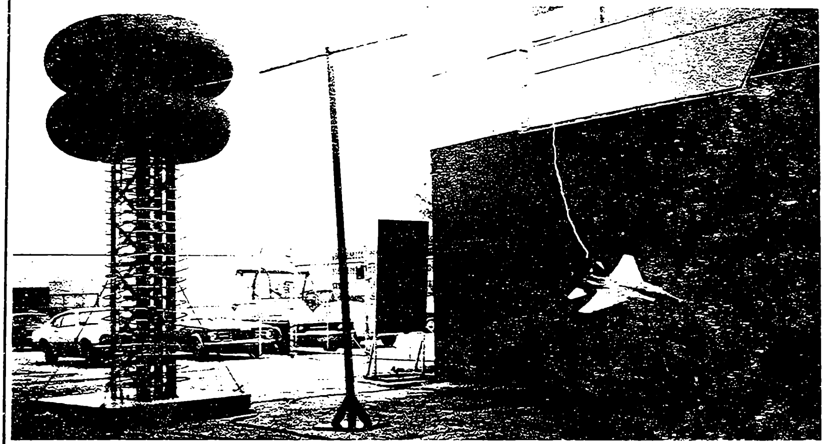
- The aircraft nose has the highest capture area (approximately 135 degrees).
- The horizontal stabilator was next, followed by the vertical fin, canopy, and center fuselage engine area which had the smallest capture area (approximately ten degrees).

Attach point tests are performed many times, and the model is changed to meet projected flight conditions including clean aircraft, with and without stores, and upright and inverted attitudes, until all strike points are known.

F-15 LIGHTNING PROTECTION

Having determined where the attach points are, lightning protection

FIGURE 1 – SCALE MODEL TEST



can be incorporated into the aircraft design. Let's look at some of the areas in the F-15 that received special attention.

• **Lights** - Since our primary lightning protection goal is to keep current flow outside the aircraft, and since MIL-B-5087 "Bonding, Electrical, and Lightning Protection for Aerospace Systems" states that "lightning discharge current carried between aircraft extremities shall not produce voltage within the vehicle in excess of 500 volts," lightning arrestors were installed on all the lights that might be struck and could couple currents into the aircraft. This requirement is based upon a lightning current waveform of 200 thousand amperes peak and a duration of 20 microseconds at the 50 percent point. Arrestors for the lights were installed on the wingtip and forward formation lights. Lights which were adequately protected by modifying their mounting shell include the wing and tail position lights, the tail anti-collision lights, and the inflight refueling light. Lights which were considered to be in a safe zone (that is, free from vulnerability to lightning strokes) are the formation lights on the aft fuselage and the anti-collision lights located on the leading edge of the wing near the root of the wing. Figure 3 shows a test upon F-15 wing tip lights.

• **Antennas** - As in the case of the lights, we want to keep the lightning current on the outside of the aircraft. Because of this, the F-15 antennas were designed so that a strike to an antenna would be diverted to the air-

FIGURE 2 – F-15 ATTACH POINTS

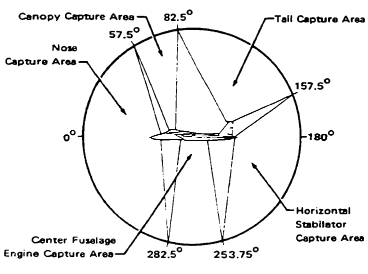
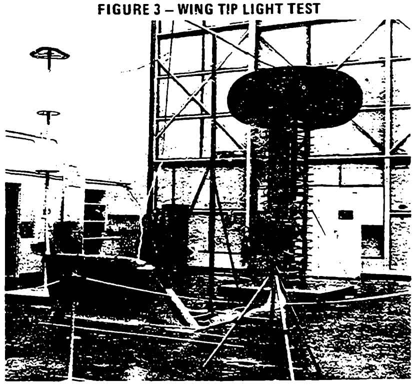


FIGURE 3 – WING TIP LIGHT TEST



▶

frame rather than through the coaxial cables within the aircraft. One such antenna, the UHF, is a grounded stub antenna.

• **Fuel System** - In the original design, a flame arrestor was installed in the fuel vent and dump line. Foam was installed in the wing fuel cells for explosion suppression because it offered the most effective lightning protection when considering cost and weight. The external fuel tank was designed with adequate skin thickness and bonding to exclude internal sparking. When we got into the qualification tests we found that the previously installed flame arrestor (Figure 4) had to be lengthened to suppress the high-velocity flame front resulting from a lightning strike. In addition, we found sparking in the interface between the aircraft and the tank (this was caused by an electrical path other than the designed lightning current path). To eliminate this, we investigated and put into use a plastic (Valox 310) air inlet and fuel outlet probe which eliminated the tank-to-aircraft interface sparking. Figure 5 shows the metal probes and arc points (arrows) as well as the Valox probes now in use.

• **Probes** - Various aircraft probes offer enticing points of contact for lightning. The angle-of-attack probe, pitot tube, and total temperature sensor are especially vulnerable. For the most part, these devices are relatively safe from lightning; however, there is a possibility that lightning could burn through the relatively thin probe tips, attaching to the heater wires. Once it attaches to a heater wire it could be conducted into the aircraft and on into the generator system causing greater damage. Because of this, lightning arrestors were installed on the heater lines of all probes.

• **Horizontal stabilator** - The horizontal stabilator, with a boron composite center section, was initially thought to require a conductive material coating on the composite section. As a result of extensive model tests and analyses, we found that no protection was required for the boron composite center section since the stabilators are not in a swept stroke zone. The metal surrounding the composite section could carry the lightning current which could pass through the shaft and bearing without causing any damage.

• **Electrical System** - The F-15 electrical system incorporates a split bus which provides greater protection for the generators than could be expected in a parallel-generator system. However, leak paths for lightning entrance into the aircraft *do* exist on the mold-line. These include lights, probes, and antennas for which protection has been provided. Because of this protection, we have eliminated all possible entry points, doing away with any requirement for protection within the split-generator system.

• **Radome** - Another part of MIL-B-5087, which establishes lightning protection requirements, specifies that "performance characteristics shall take precedence over a lightning protection requirement"; therefore, a study was made to see if there might be any effective protection that would not detract from radar performance. None was found that did not introduce some effect on the radar performance. Since flight safety is paramount, an investigation into the safety aspect of the radome was conducted. It was determined that a small radome hole, such as might be produced by a lightning strike, is not a flight safety item.

FIGURE 4 – FUEL VENT TEST SETUP

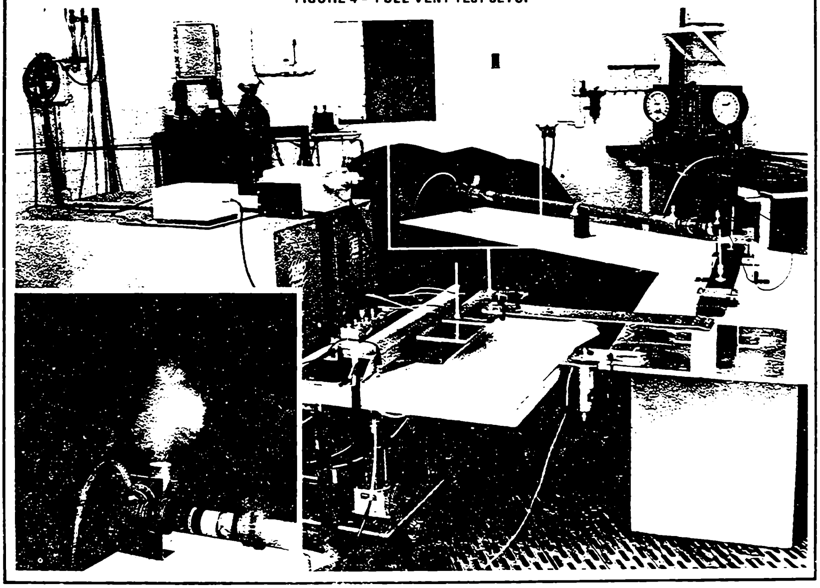
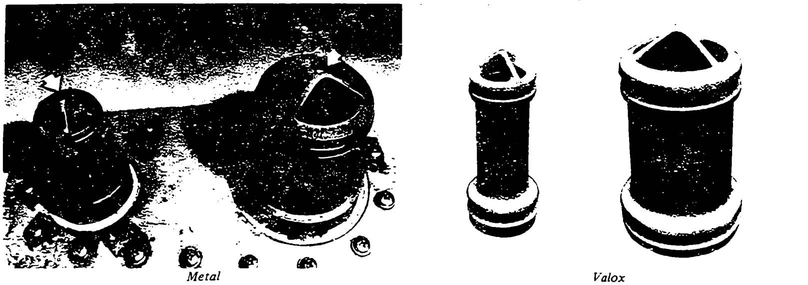


FIGURE 5 – EXTERNAL TANK FUEL AND AIR PROBES



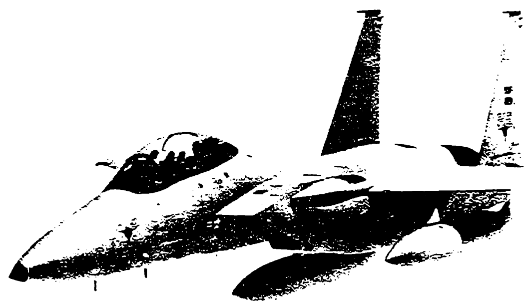
Metal *Valox*

IN CONCLUSION. . .

The F-15 lightning protection program has been one of the first total programs relating to a fighter aircraft. That is, we considered lightning protection from the very beginning (in the initial design) and then included necessary protection within qualification test and production aircraft.

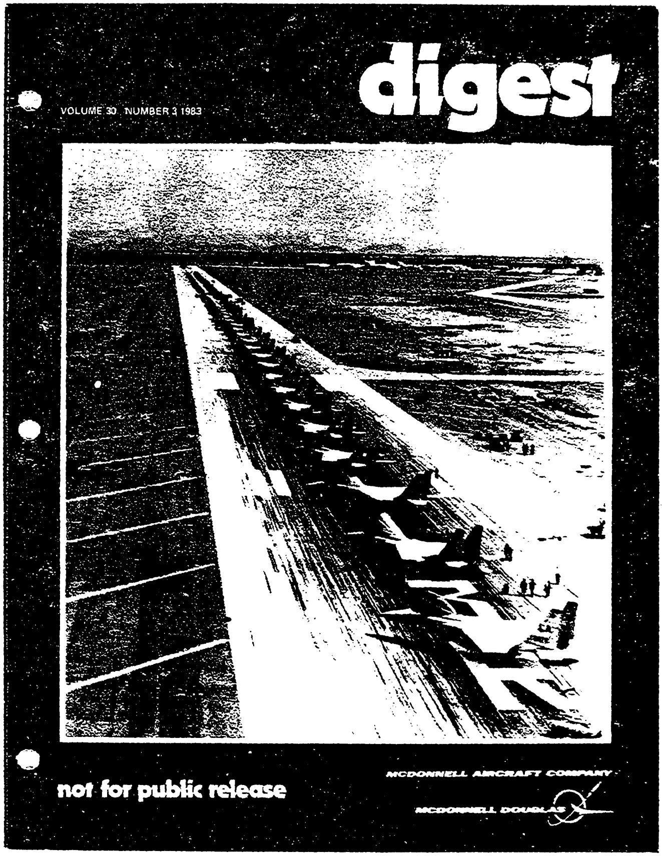
Though this article concludes the series by Bob Aston, you may want to see what Owen McBee says about lightning and advanced composites in the article beginning on page 22.

The author of this series wishes to extend his appreciation for technical support and advice from the following individuals: Mike Amason, Douglas Aircraft Company; Don Clifford, Section Chief Laboratory, and Ed Schulte, Senior Engineer - Laboratory (both of the McDonnell Lightning Laboratory); Jim Ketterer, Lead Engineer - Electronics; and G. L. Weinstock, Section Chief - Electronics. ■



Conformal fuel tanks now scheduled for delivery to USAF in June 1983, have received extensive evaluation and demonstration. Photograph above shows F-15B S/N 71-291 in 1974 during one of first test flights with tanks installed. Eagle improvement programs are continuous, and this particular airplane has been the "test bed" for many of them. It is flying today as the first F-15 converted to the "DRF" (Dual Role Fighter) configuration (see page 195).

SOME THINGS TO LOOK OUT FOR ►



F-15 ENGINE

(PUBLISHED 1977)

EXTERNAL EMERGENCY SHUTDOWN

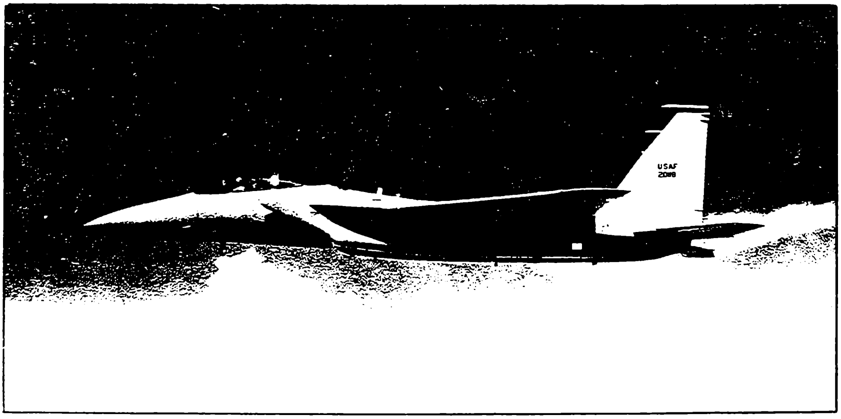
By JACK SHEEHAN/*Flight Safety Engineering*

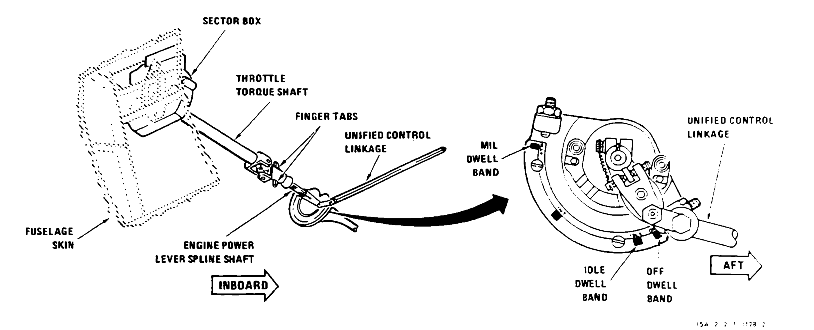
T.O. 00-105E-9 (USAF) and PS 952 (MCAIR) present crash rescue and firefighting information for the F-15A/B Eagle. However, neither of these documents indicates that there is an alternate way to shut down the F-100 engines in an emergency when access to the normal cockpit engine controls is prevented for some reason. This information is presented only in USAF T.O. 1F-15A-2-2-1 (Basic Maintenance Information). Instructions on emergency engine shutdown from the cockpit — by moving engine throttles to cutoff; turning engine master switches off; or by pushing the engine fire extinguisher button(s) — are covered in all three manuals; but we think everybody ought to be aware of an external shutdown method also. This method is not easy but it is possible, and conceivably could be the *only* way under certain circumstances, so it's well worth learning.

When the F-100 engines are running on the ground, the leading edges of the inlet ramps drop down and the left-hand ramp interferes with the manual cockpit entry handle. In a situation where the pilot (or engine operator) is not able to shut down, and where emergency rescue personnel cannot open the canopy to get at the cockpit controls, it's possible to shut the left-hand engine down with the UC (Unified Control) linkage lever which is located on the lower outboard side of the engine in the aft fuselage.

Normal access to this control is through Door 113L on the bottom of the aircraft, but assuming the door is closed (70 fasteners to open) or the airplane has landed gear up with the door in the dirt, the only way to get at the UC lever is breaking or cutting through the fuselage at the approximate spot circled on the photograph below. After gaining access, reach in, pull the finger tabs on the throttle torque shaft quick-disconnect toward you and turn upward to "open" (this disconnects the throttle shaft from the engine power lever spline shaft which allows the UC linkage to be moved). Then pull the UC linkage full aft and hold it there until the engine stops.

Simple, right? No, but it can be done. This method is obviously for an emergency situation, but "in extremis" is no time to be locating this spot on the airplane. If you happen to have an Eagle handy, take a look now, and fix the exact chopping spot in your mind (it's 25 to 30 inches aft of Fire Access Door 99L and just above the aft tip of the

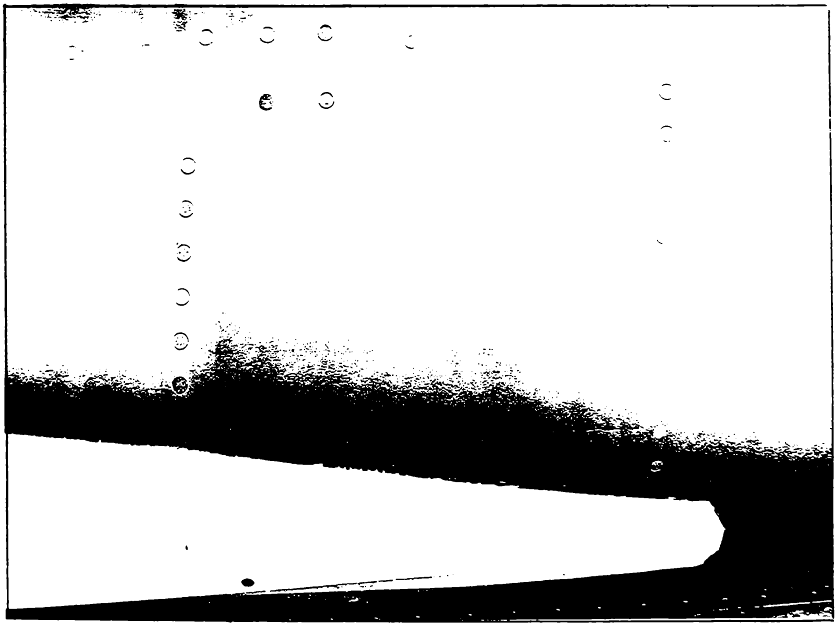




After cutting through fuselage in area colored below, this is what you'll find inside.

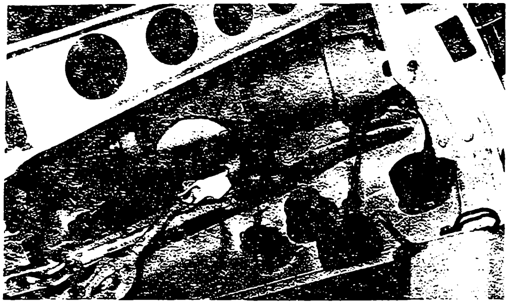
LAU-106A guided missile launcher — the close-up photo shows it best). Then make a close check of where the finger tabs and the UC actually are with respect to where you'd have to feel for them — maybe in the dark and surely in a hurry.

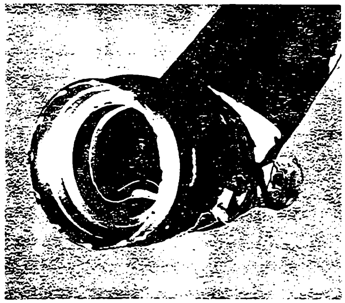
Incidentally, all these words hold equally well if you need to shut down the right-hand engine, aft of Door 99R and inside Door 113R. It's harder to get at the UC linkage, though, because the linkage is inboard when the F-100 engine is on the right side. ■



(PUBLISHED 1978)

Dust Covers





You tell us how that dangling dust cover could get up through this intricate network of linkage and actuator and then smash the fuel line in the area circled. That it did is obvious from the photo at right, of the actual fuel line damaged in this incident. Note that not only was the line mounting bracket broken, the line was ruptured internally.

Made to Protect Not Destroy

A recent report from the field cited an F-15 returned from flight with a fuel leak in the right hand main landing gear wheel well. Specialists investigating the problem found the 68A580983-2003 gravity flow fuel line from the internal right hand wing tank fuel cell 3A damaged between the attaching flange and the coupling. The flange was fractured and the line itself was ruptured (although this latter damage was not obvious until the part was removed from the airplane).

A careful analysis of the problem pinpointed the cause as being the Power Control II hydraulic quick-disconnect suction line dust cover. The cover had not been reconnected after ground servicing and was forced into the fuel line during landing gear retraction. If you take a look in this area, you'll probably agree that the odds against a dangling cover being able to find its way up through the maze of gear rods and plumbing in the few seconds it takes for the gear to retract and door to close are fantastic. But this dust cover beat the odds and broke the pipe, as proven by imprint marks from the cover on the self-sealing coating surrounding the line.

A similar incident occurred on an F-4 several months ago, resulting in replacement of a $1,475 inboard landing gear door (Product Support Digest Number 5/1977). However, because the incident on the F-15 produced a fuel leak, it could have been much more costly. We're all aware that fuel leaks can result in fires and inflight fires are the last thing a pilot needs.

With two incidents caused by dangling hydraulic quick-disconnect dust covers in so short a period of time, we feel a few words about these protectors of your hydraulic systems are in order. There are several of them on the airplane and they are, as their name implies, designed to keep solid contaminants out of the hydraulic system whenever a hydraulic test stand (mule) is not hooked into the system. On the F-4, each is secured to the airframe by a chain covered with vinyl tubing; on the F-15's by a stainless steel swedged cable. Purpose of the chain or cable is to prevent cover loss when disengaged from the disconnect. When threaded onto the quick disconnect (using hand pressure only, no water-pump pliers please) self-contained teeth on the cover engage with a spring-loaded self-locking device on the male portion of the disconnect. When properly secured, the covers will not vibrate loose. However, if other than hand pressure is used to tighten the covers, the locking device can be damaged to the point where it becomes ineffective. (Note that this was not the case with the incident described here, in which somebody just plain forgot to reattach the cover.)

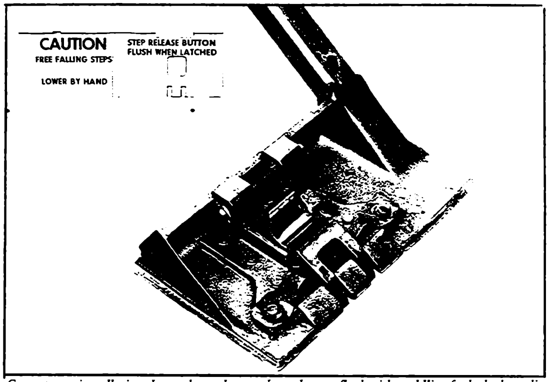
As a preventive measure against recurrence of the incident that took place on the F-15, MCAIR has initiated a Class II Change (production only), effective on Block 20 and up aircraft. Holes securing the stainless steel cables to the airframe will be relocated and cables will be shortened. For pressure cover cable, the hole will be .50 inch above horizontal web; for suction cover cable, .50 inch below horizontal web. Both holes will be 1.20 inches outboard of web wall; hole diameter is .116 plus .005 minus .000. Cable for the pressure cover is to be shortened from 14 to 8 inches; suction cover cable from 12 to 7 inches.

Notwithstanding any change, it is still your responsibility - crew chiefs, quick-check crews, and aircrews - to see that the dust covers are threaded onto their quick disconnects prior to flight.

Checking the wheel wells is part of the Preflight and Quick Check Inspection. Aircrew members check the tires for condition and inflation and gear struts for extension during the Preflight Walkaround Inspection. While you're checking out the wheel well area, cast a quick glance up at the quick disconnects to assure the dust covers are not swinging in the breeze. That little glance could save you from headache and heartache. ■

(PUBLISHED 1979)

Boarding Steps



Correct step installation. Inset photo shows release button flush with moldline for locked condition. Larger photo is internal view showing staked hinge pin, release button spring sandwiched between button and step housing, and button forced back against stop indicating an unlocked condition. Compare with lower photo which shows all the problems mentioned in article.

The aircraft internal mounted boarding steps continue to give problems. In recent months, six inadvertent extensions have occurred, five inflight and one during taxi out for flight. These extensions have occurred with the steps modified in accordance with TCTO 1F-15-511, which was intended to assure a positive latching of the step assembly. Obviously there are still some gremlins in the system!

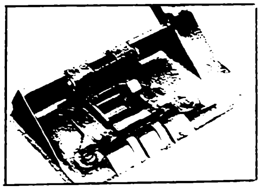
The first reported problems with the new steps showed up shortly after delivery, when two step assemblies extended. These were written off by USAF as improper lock-up by maintenance personnel. A few months later, another inadvertent extension occurred during a return flight to an east coast base. During postflight inspection, the release button hinge pin was discovered missing. Further investigation revealed that the release button was installed on the wrong side of the release spring. A one-time inspection at this location by USAF Quality Control personnel discovered one additional release button hinge pin not staked. This potential problem generated a check of all F-15s with new steps by MCAIR Field Service Engineers. A total of 25 improperly staked hinge pins and one improperly installed release button were discovered.

These deficiencies were corrected and everything looked good again for a few months, until suddenly three inflight extensions were reported within a matter of weeks. Investigation of the first incident revealed that the hat-shaped release button spring was deformed (flattened), permitting the release button to remain flush with the exterior moldline at all times. The hazard here is that with the button remaining flush, the steps would indicate up and locked even if they were not caught by the uplock hook. Investigation of the other two inflight extensions showed that the release button springs were again on the wrong side of the release button and the hinge pins had backed out and were not staked.

The latest reported problem is with cracked release button springs. The cracks are showing up in the radius of the hat section near the button. One base also reports numerous occasions when they could not get steps to extend without added force (added force being a screwdriver to pry the steps out of the moldline well and then using the lower telescopic strut extension as a slide hammer to force remaining struts to full extension).

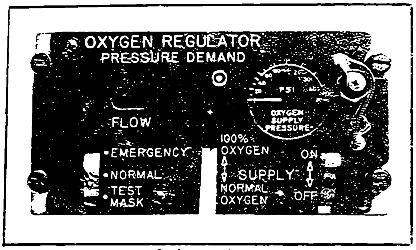
As you can see there are four clear-cut problems with the boarding steps — (1) hinge pin not staked; (2) release button installed improperly; (3) release spring deformed and cracked; and (4) telescopic struts jammed. You are probably now saying to yourself, here come the design changes, but this will have to wait for a later installment of the DIGEST since MCAIR Engineering is in the process of a complete design review of the boarding steps. In the meantime, we suggest careful attention to the four problem areas during maintenance and inspection.

A Final Reminder — A properly functioning spring and release button is the locked or unlocked indicator. ■

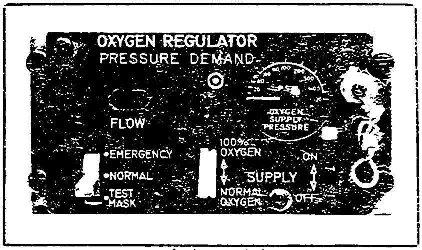


(PUBLISHED 1977)

Anyway you look at...



Look at repair cost



Look at repair time

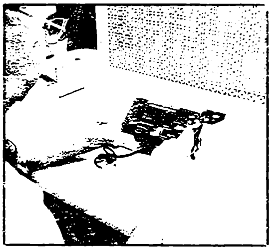
COCKPIT INSTRUMENT PANEL DAMAGE

...it's expensive

There are about a hundred different control panels on the side consoles of the F-15 cockpits. Most of them have knobs, switches, or buttons for operation; many of them have glass cover plates for protection; and all of them appear to be suffering severely at the hands and feet — and test equipment and tools — of Eagle Drivers and Eagle Keepers.

We don't know of a single panel that has totally escaped this destructive dinging, although certain ones seem to take more than their share. For an unhappy example, let's take a look at the Oxygen Regulator Panel... which is one of a thousand components covered under "PROJECT PACER WEB."

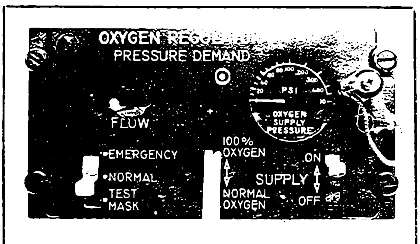
PACER WEB is a contract MCAIR has with the USAF to repair various F-15 components determined to be not within the repair capabilities of the Air Force. It's a time and materials contract — we've got 30 days to fix the item and you provide the spare parts. Since the Oxygen Panel is one of the items covered under this contract, theoretically we should care less how many busted panels you send us. You break; we fix; you pay.



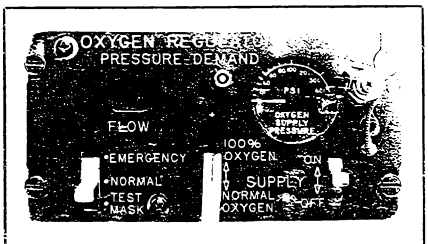
This man is not looking for work. O_2 panel repair is tedious and time-consuming; his efforts could be better used elsewhere.

But we find it impossible to look at this situation from a purely theoretical viewpoint, and so should you. Because "you" is us and everybody else in the long list of American taxpayers. Obviously, there is money to be made in the repair of repairables, but we think there is far more money to be saved than made. And we'd much rather sell you more "whole" airplanes than repaired "pieces."

Take a good look at the four panels pictured on this page — picked at random from the sorry lot on hand during a typical recent week; two cracked face plates, two broken switches, and some damage at the corners. We're at a loss to figure how all this is happening — the oxygen panel is not in a console position subject to that much daily wear and tear. Can *you* tell us? And then can you let up a little on all this assault and battery? There are normal and expected amounts of repair work on any airplane, and we're happy to do it; but *nobody* can afford this much. ■



Look at removal/reinstallation effort



Look at packaging/shipping expense

(PUBLISHED 1975)

Cockpit Console Damage

TO 1F-15A-1

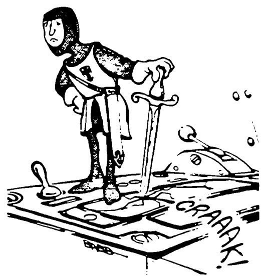
An oxygen hose stowage fitting is provided above and outboard of the right console. The oxygen hose should be stowed in this fitting at all times when not in use to prevent hose contamination and DAMAGE TO THE CONSOLE BY A FLAILING HOSE.

According to my history book, glass was first used as a protective cover on the 23rd of June, 575. King Arthur stuck a piece of it on top of an old round table in the great dining hall of his castle. Early the next morning, less than 24 hours after this historic moment, another historic moment took place — some knight propped his feet up on the table after breakfast, and busted the fragile piece of glistening material all to hell. Knights, swords, boots, belts, helmets, and other things have ever since been employed to reduce glass plate to glass shards on every type of vehicle from two-wheel carts to twin-engine jets.

Today, fourteen hundred years after our anonymous and heavy-footed knight broke the first glass cover, we are receiving discouragingly regular reports of instrument face glass being broken on the F-15 cockpit control consoles. Not so much now by a pilot's boots, but by other items of equipment equally peculiar to his trade — by for instance his oxygen hose connector or his seat belt buckle. Prithee, goode knights of the rectangular cockpit, we beg thee — cease and desist this practyce most foul! (We beg the same of all those in liege to you knights, like ejection seat technicians, instrumentation men, and anybody else whose daily duty takes him into this knightly area.)

Glass possesses two unique characteristics, one good and one not so good — You can see through it clearly and you can break through it easily. Nobody has yet found a way to let you do the first without risking the second. On the Eagle, for example, various alternative materials — acrylics, polycarbons, laminates, special tempers, overlays, etc., have been evaluated in attempts to solve this long-standing and serious problem. For one good reason or another, visibility prime among them, nothing better than good old clear-viewing, easy-breaking fused silica has yet developed.

The quotation from Dash One reminds everybody about damage the oxygen hose can do. The same applies to other equipment, gear, tools, feet, etc., belonging to either air or ground crewmen. Legend has it that King Arthur wasn't too pleased about his broken table glass. Reports have it that General Dixon feels the same way about his cockpit glass.



■

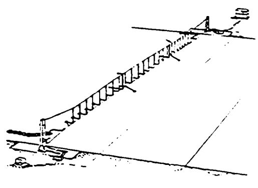
(PUBLISHED 1979)

Eagle vs MA-1A Arresting Gear

Some inquiries have been received regarding compatibility of the F-15 with the MA-1A barrier. There is concern over the problems an Eagle with full centerline tank might have when engaging this type barrier.

The MA-1A is the old chain-dragging type gear and its design was conceived to arrest aircraft that are not equipped with an arresting hook. Arrestment is accomplished when the nose gear engages a nylon webbing which extends about four feet above the runway. The arresting cable is attached to the lower portion of the webbing and is propelled upward as the nose gear pulls the webbing forward. The cable will rise above the main wheel tires and engage the main gear struts to arrest the aircraft.

The F-15 has not used the MA-1A barrier nor has our Engineering Department conducted any analysis to determine the compatibility of the two. However, it does appear that engaging this barrier with a centerline tank could be hazardous. The arresting cable would most likely strike the centerline tank, creating a potential fire hazard. Also, the tank would not permit the cable to rise above the centerline of the wheels which means the cable would pass under the wheels and no engagement would take place. All in all, we would not recommend taking the Eagle into the MA-1A with centerline tank aboard. ■



BINDING CONTROLS

(PUBLISHED 1978)

High among the worst feelings a pilot can experience is that which accompanies jammed flight controls or throttles. Pucker factors of 9+ (on a 10 point scale) are common along with the question, "Why did I ever take up flying in the first place?"

Fortunately most of the problems will be only temporary and will rectify themselves. Others will require brute force to free the stick, pedal, or throttle, while still others will defy all attempts to correct; those are the ones that make you sorry your seat preflight was not more thorough.

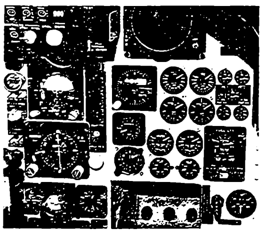
Combat damage aside, maintenance/quality control is the leading cause of control/throttle binding problems. How many times have you heard of wayward screws, fasteners, or tools jamming controls? How many loose dust covers have found their way into the control cables or linkages? "Murphy" will leave linkages unattached, cables unsecured, and forget safety wire and cotter pins. The number of these situations that Quality Control uncovers is unknown to us because the only ones that we ever hear of are the ones that were overlooked and produced an incident/accident.

Occasionally we hear of new binding problems that could have had far more ominous consequences than the reported incidents themselves. The following two cases, the first on an F-15, the second on an F-4, are provided as food for thought for both pilots and mechs to ponder.

CASE I — FROZEN PEDAL

"An inflight emergency was declared when binding of the left rudder pedal was experienced with approximately one-half input."

The culprit was the carrying handle from the RWR TEWS Display Unit; it was found lodged in the forward cockpit rudder pedal linkage. Vibrations had apparently caused the mounting screws to back out, as they were found inside the case. The other aircraft in the unit were inspected and more loose handles were discovered. As you can see from the photograph, the path from the top of the display unit to the rudder pedal linkage is a rather straight one.

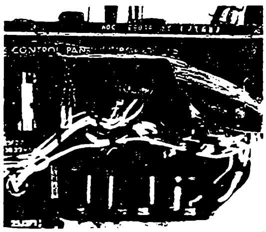


F-15A cockpit. RWR TEWS display at top, right rudder pedal at bottom. TEWS case handle jammed rudder pedal linkage behind lower center instrument panel.

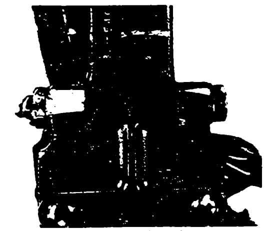
CASE II — FROZEN THROTTLE

"The starboard throttle froze in full afterburner for about two minutes before the pilot could free it."

The upper portion of the lower lever arm assembly of the right throttle had severely chafed the inboard engine control panel identification plate (see photos) and the castellated nut hung up under the identification plate flange at the point directly under the "P" in the word "panel." Other unit aircraft were inspected and similar, although not as severe, wear was noted on two-thirds of them. The MIM's/TO's require that during removal of the throttle quadrant, the number and position of the shims be recorded and the shims retained for reinstallation. Detailed installation/reinstallation instructions call for specific clearances between the throttle lever arm assemblies and the engine control panels. These clearances are obtained by the careful shimming of the throttle quadrant. Obviously these instructions were not followed very carefully as these clearances were non-existent, thus the resulting wear. Apparently neither Quality Control nor the pilots who flew the aircraft noticed anything strange about the throttles. Everyone had missed the error.



Severe chafing of inboard engine control panel nameplate. Right throttle hung up under nameplate flange at point just below letter "P" in the word "Panel."



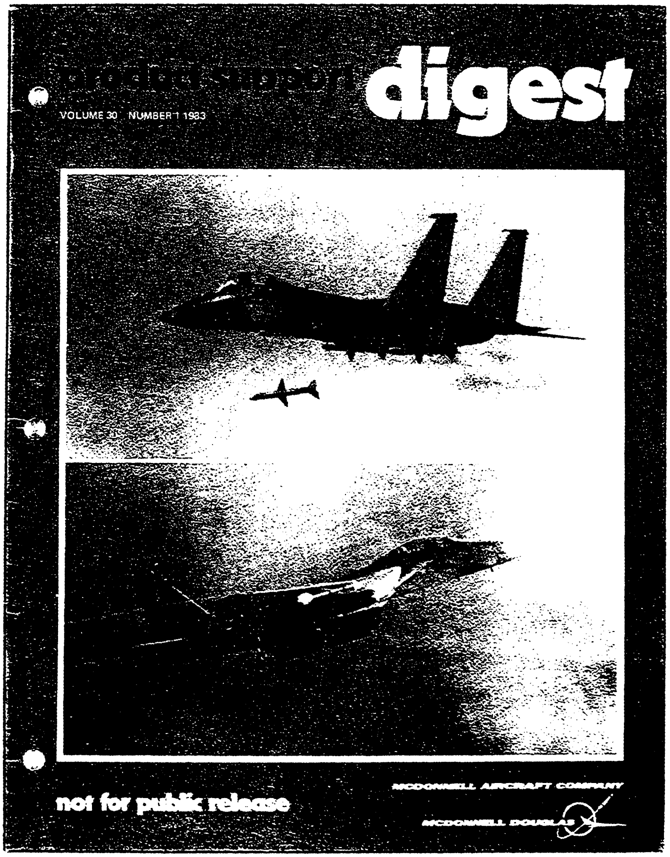
Lower lever arm assembly showing severe chafing. Castellated nut hung up under nameplate flange. (Also note poor cotter key installation.)

In both of these cases, more severe problems were averted by aircrew skill and professionalism; however, the possibility of other aircraft presenting their aircrews with the same problems, but with less fortunate results, is very real.

For you Eagle Keepers, TCTO 12P3-2ALR56-519 provided the fix for the TEWS Display Unit handle and directed the modification of the nameplate by the addition of a 1/16-inch "M" to indicate compliance; check to see if that TCTO has been complied with on all your Eagles.

You Phantom Phixers should visually inspect your birds for any signs of chafing in the throttle areas; clearance should also be checked and if necessary, the quadrant may have to be reshimmed. ■

F-15E "DUAL ROLE FIGHTER" (DRF) ►



product support digest

VOLUME 30 NUMBER 1 1983

Front cover photographs show F-15 test aircraft "DRF" in air/air and air/ground configurations. Back cover shows pilot and plane captain approaching F/A-18 Hornet for one of first flights by VMFA-314 at MCAS El Toro, California.

IRVING L. BURROWS/Vice President, Product Support; THOMAS L. PLEIN/Director, Product Services; HERMAN J. CORREALE/Director, Support Operations.

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(PUBLISHED 1983)

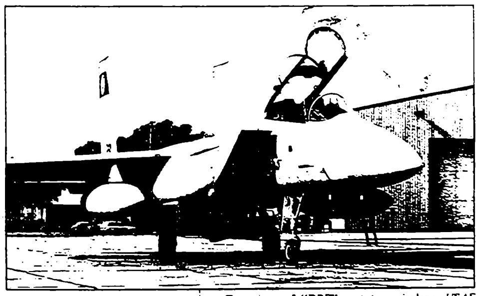
F-15 "DRF" Dual Role Fighter

Part I

While the era of the single-mission combat aircraft is far from over, there is a recognized US Air Force need for an aircraft which can fulfill equally well the distinct mission roles of air-to-air fighter and air-to-ground attack. While the McDonnell Douglas F-15 Eagle was originally designed for these dual capabilities, emphasis was placed during the 1970's upon its air-to-air mission and the aircraft became known as the free world's premier air superiority fighter. Only recently, because of its "designed-in" flexibility, have special attack modifications been underway for use in future Air Defense and Rapid Deployment Force missions. And looking still farther ahead as threats mount in various areas of the world, USAF needs are emerging for improved ways of destroying enemy armor and supply lines at night and in poor weather.

The combat proven F-15, enhanced with easily-incorporated improvements, is the quickest, best, and most cost-effective way to meet these Air Force needs in the 1980's. Since 1977, McDonnell Douglas Corporation and Hughes Aircraft Company have been working on new technology for an Eagle configuration which today carries the designation of F-15 "DRF" (Dual Role Fighter). With only minor modifications to the radar and a redesigned aft crew station, the inherent range and payload capabilities of the current F-15 have been capitalized into the "DRF."

Known in the early stages of this development program as the "Advanced Fighter Capability Demonstrator," USAF F-15B S/N 71-291 contains the dual role improvements and has spent the past two years in flight



Two views of "DRF" prototype airplane. USAF 71-291 has been flying dual role evaluations for over two years.

evaluations and demonstrations of DRF capabilities. It has met all basic requirements of the program, and is even now being used to explore significant additional options for the future. After an introductory look at some of the basic characteristics of this newest Eagle configuration, we'll take you behind the scenes for an engineering summary of the design concepts and considerations which governed MDC's approach to a dual-role USAF fighter, and then into the cockpits for an aircrew's evaluation of the F-15 "DRF"...

To begin with, the "DRF Eagle" pilot will not be flying alone — one of the primary features of this configuration is the two-seat design standard. Where only every ninth original F-15 contained provisions for a two-man crew (primarily for training purposes, although the B/D Eagle is completely combat-capable), the Dual Role Fighter is dual cockpit all the way. The aft cockpit includes four multi-purpose display screens and two hand controllers for improved navigation and weapons delivery. The four CRT's (cathrode ray tubes), integrally connected with the aircraft computer and modified APG-63 radar, allow the crew to simultaneously monitor aircraft weapon status and threat defenses while using sensors for navigation and target acquisition. The hand controllers permit the aft crewman to focus his attention on the visual display screens, which may be utilized in many different ways.

There are two four-inch and two seven-inch diameter CRT displays.* The left four-inch screen is a so-called "menu," from which the crewman can choose the displays desired on each screen. The left seven-inch screen offers an electronic moving map for navigational purposes, which provides aircraft orientation and threat status/location. This screen can also be used for system status and operation, weapons display, and electronic warfare. The right seven-inch screen, used for targeting information, includes a ground moving-target detection mode and high resolution radar (McDonnell Douglas option), and a forward looking infrared sensor (FLIR). The right four-inch screen provides a duplicate of the head-up display (HUD) in the forward cockpit

Two additional features will enhance DRF capability for penetrating enemy territory and improve even more the already impressive survivability characteristics of the basic F-15 design. The terrain following/terrain avoidance system allows all-weather, low altitude penetration to avoid detection by the enemy; and internal countermeasures, such as a radar warning and homing system, active jamming systems, and an automatic flare and chaff dispenser, further shield the aircraft.

DRF precision air-to-ground radar modes provide continually updated, photographic-quality images of a target area from as far as 150 nautical miles and in any weather. As the aircraft nears the target site, the radar display has an 8.5 foot resolution which enables the crew to distinguish small tactical targets and even moving targets such as tanks and trucks. Also, in day/night, clear weather conditions, the FLIR/laser designator pod provides a close-up video view of the target; and a cueing mode allows precise tracking of stationary and moving targets.

The DRF configuration offers a greatly expanded bomb carrying capability for a wide variety of ordnance, including guided weapons, general purpose and cluster munitions, and airfield attack weapons. A maximum of five air-to-ground weapon stations provides compatibility with any type of ordnance the Air Force requires. An advanced navigation/attack system and all-weather sensors permit weapon release comparable to aircraft which use only visual release systems.

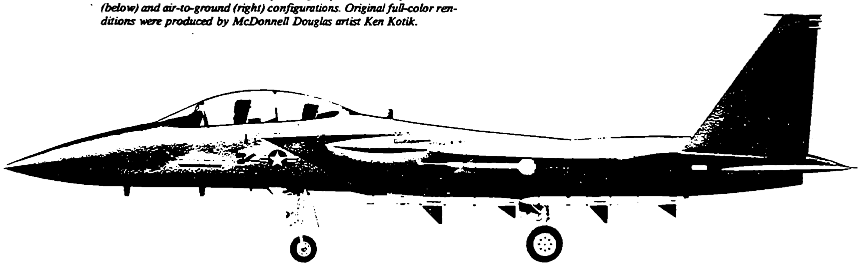
Ordnance is deliverable in three dimensional, high g maneuvers, enabling straight and level approaches, dive approaches, and an optional maneuvering attack system (MAS) which permits weapon delivery while in a turn and quick exit from the target area without overflying the target. MAS provides an exact bomb drop even though the aircraft is banking and turning away from the target. In addition, DRF has a full complement of manual and automatic release systems for weapons delivery throughout the envelope.

Conformal fuel tanks provide an extra 9750 pounds of fuel (with no increase in subsonic drag) for increased range and maneuverability. Mach 2+ speed, high thrust-to-weight ratio, and a 1000-mile mission range demonstrate that the DRF configuration sacrifices none of the original Eagle's fighter capabilities. It retains all-aspect, look down/shoot down radar and beyond-visual range missiles (four Sparrow radar missiles and four Sidewinder infrared missiles), and will also be compatible with the AMRAAM missile. It has the same internal growth capacity that characterized the basic F-15 design, with potential for such mission possibilities as anti-satellite (program now underway), Reconnaissance, Wild Weasel, and Sea Lane Control.

Because our dual role fighter program has been in existence for more than five years, the F-15 DRF offers ear-

**CRT display screen sizes for production aircraft are five inches and six inches.*

F-15 "DRF" is shown here in paintings representative of air-to-air (below) and air-to-ground (right) configurations. Original full-color renditions were produced by McDonnell Douglas artist Ken Kotik.



ly availability at minimal risk. And because its parentage is the well-tested A/B/C/D line of more than 750 production aircraft, there is demonstrated assurance of the same reliability and maintainability that produced the highest full mission capability rate for any US fighter in 1981. F-15 DRF survivability predictions are equally confident; based upon the loss ratio figures currently tabulated for the twin-engine Eagles now flying — only 4.4 aircraft have been lost for each 100,000 flying hours, which establishes the F-15 series as the safest USAF fighter of all time.

After five years of engineering and two years of testing, plus ten years of precedent, it is our considered opinion that the F-15 Dual Role Fighter is ready to meet current and future US Air Force needs. For some facts behind this opinion, let's turn now to the individuals who have been closest to the "DRF Eagle". . .

Dual Role Fighter Engineering Program

(PUBLISHED 1983)

By DON KOZLOWSKI /*Chief Program Engineer*

The F-15 might be considered as a "case history" example in engineering evolution of a fighter aircraft design. When initially introduced, it offered superior performance to beat then current and expected threats. As threat performance improved, so did the Eagle through its A, B, C, and D configurations, and the F-15 "DRF" now offers a weapon system capability that provides the same threat superiority as its predecessor configurations. Starting from the basic air superiority mission, we have added capabilities for a second crew member and significant all-weather attack improvements to permit engagements of the enemy around the clock around the world.

The "Dual Role Fighter" configuration is a logical yet significant step in the growth capability of the F-15 Eagle. When the US Air Force and McDonnell Aircraft Company were laying down the fundamental lines of this machine in the late 1960's, they were aiming at a very ambitious and immediate goal of a 40,000 pound fighter. At the same time, they also had the design foresight to include sufficient growth capability to allow an easy transition of the basic aircraft into other roles. That inherent growth is still being fully exploited, some ten years after the F-15 made its initial flight. What started out as a 40,000 pound airplane is still in that weight class in its air superiority role. However, with avionic systems improvements, addition of conformal fuel tanks, and external payload increases, we can now field takeoff gross weights approaching 75,000 pounds in the attack role! Even with 20,000 pounds of ordnance to deliver over high value enemy targets, the F-15 DRF is competitive in terms of takeoff and landing distance and maneuverability with the primary attack aircraft in the world today.

The Eagle is being enhanced through "Pre-Planned Product Improvements" (P^3I) to stay ahead of the threat in air-to-air and to expand capability for the attack role. Currently planned changes (all part of "MSIP" — Multi-Stage Improvement Program) include:

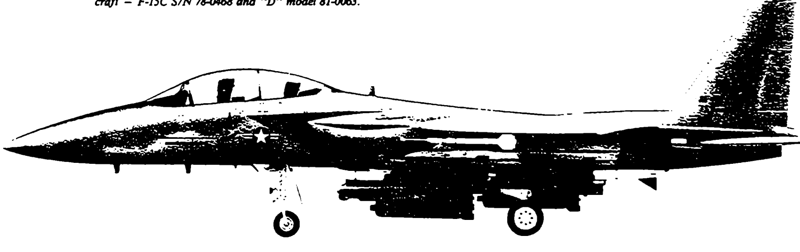
- Fire Control — Improved radar, new central computer, programmable armament control.
- Countermeasures — Countermeasures dispenser, improved radar warning, updated internal countermeasures (internal countermeasures set will be installed in the ammunition bay, with a reduction in ammo capacity).
- Weapons — AMRAAM compatibility, AIM-9/AIM-7 improvements, additional A/G weapons.

MSIP changes are keyed to improved air superiority and to expand vital capabilities of the F-15 for utilization by the RDF (Rapid Deployment Forces). They also provide the foundation for the DRF configuration, which includes:

- Missionized Cockpits — Improved displays, wide field-of-view HUD, upfront control improvements.
- Navigation and Targeting — High resolution radar mapping, moving target indication, improved inertial navigation, LANTIRN (low altitude navigation targeting infrared night) pod compability.
- Flight Controls — Automatic terrain following, fail-operate CAS, pilot relief modes, built-in test.
- Weapon Delivery — Additional weapons, nuclear compatibility, maneuvering attack.

For many years, the US Air Force has had an outstanding need for an aircraft offering these capabilities. The primary system for this role today is the F-111 which emphasizes attack against fixed

F-15B S/N 71-291 was first Eagle modified to dual-role fighter configuration. Program has since been expanded to include two more aircraft — F-15C S/N 78-0468 and "D" model 81-0063.

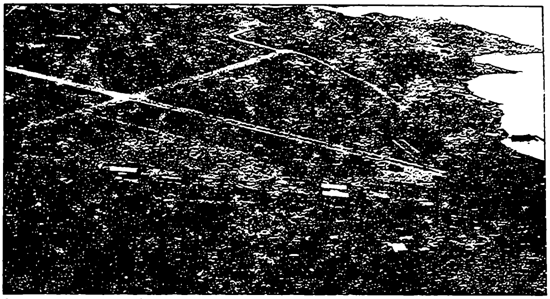


high-value targets such as on a nuclear mission or relatively large tactical targets in a conventional role. Today with the F-15 Dual Role Fighter, we are capable of providing all-weather electronic eyes that permit detection of even small tactical targets at very long ranges. This capability, provided by high resolution radar mapping modes, is not new technologically, but the capability of packaging it as a lightweight, low-volume system for installation in a high-performance fighter airframe is relatively new.

The APG-63 radar is being improved by Hughes Aircraft Company for its fundamental air superiority role, but those same improvements also provide the basic flexibility to do high resolution mapping as well as moving target indication (MTI). Improvements are also being made in mission computer, communications, tactical electronic warfare systems, and ordnance-carrying capability, but let's use the radar system as an example.

Evaluations of the high resolution mapping modes were completed in the "Advanced Fighter Capability (AFC) Demonstrator" program, under the direction of Ira Pope, MCAIR Chief Electronics Engineer. During this program, radar resolution of 8 1/2 feet was demonstrated at ranges of up to ten miles from the target. Of equal significance is that the radar pictures are achievable from very low altitudes, or "grazing angles" as they are called. The radar can literally see where the eye cannot. For example, imagine yourself in an airplane looking out at the horizon toward something ten miles away while you are only a few hundred feet off the ground. It is extremely difficult for the unaided eye to see anything, even on a clear day, at those ranges. The F-15 radar not only sees at those ranges, but it presents a picture, a true picture, of the target area to the crew. The scale and resolution of the picture are constant regardless of the range. Obviously, when you can see from long ranges, you have sufficient time to locate and designate targets — and most important — make a successful first-pass attack. Direct target attack flying over target defenses can be avoided.

The high resolution radar modes go far beyond the ten miles. Major target areas can be located with pinpoint precision from 100 miles away. This is a great advantage compared to current systems in terms of both accuracy and range with today's modern ordnance that provides long range standoff delivery. The F-15 DRF can accurately

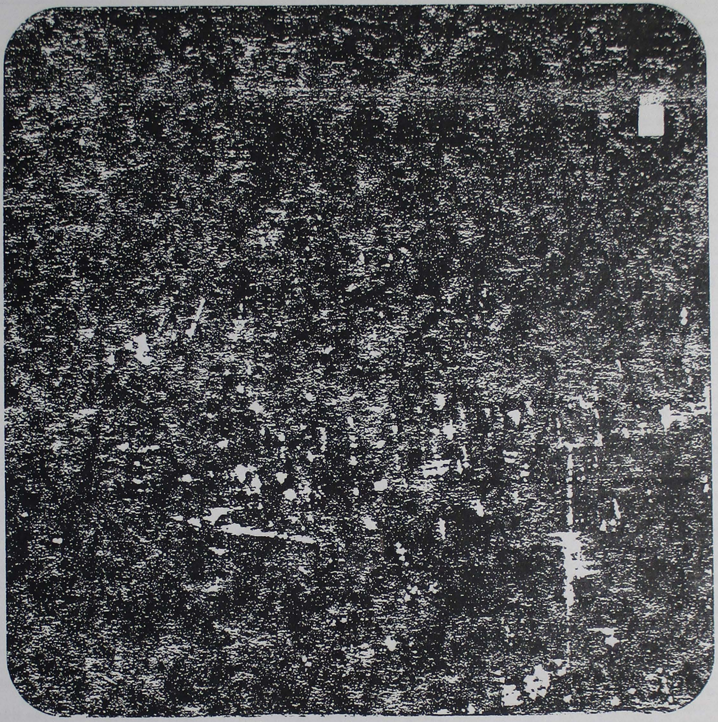


Same approximate areas of Langley AFB, Virginia as seen by normal aerial camera from directly over base (above) and by F-15 DRF high resolution radar from five miles away at 3,400 feet altitude (right). Object resolution is 8.5 feet. Sample radar image on page 5 is printed approximately the same size as viewed by aircrew during flight, but neither quality or color of reproduction matches actual cockpit presentations.

attack targets deep in enemy territory without ever approaching target area defenses. The series of maps shown here are of Langley AFB, Va., at various resolutions and ranges. The 8.5 ft resolution map provides sufficient detail to count aircraft on the ramp (in this case, most of them are Eagles from the 1st TFW). The image shown is comparable to the actual display in the cockpit. Image maps of an armored target array and a tank column are also shown to illustrate low grazing angle capability of the DRF radar. Nothing can produce this quality of data at such low grazing angles except radar, and staying low means survival.

Weapon delivery accuracy is the ultimate measure of aircraft attack capability. Long-range location, specific identification, and least-risk approach to a target are vital but preliminary to the final purpose of an attack aircraft — dispensing ordnance with a minimum CEP. The F-15 DRF provides night/adverse weather bombing accuracies equal to or better than many previous day-visual only systems, as indicated in the chart below. The F-111 does well today against larger targets, but the F-15 with high resolution radar provides accuracy, in weather, against *tactical* size targets. Weapon delivery accuracies of less than 100 feet were demonstrated with the AFC aircraft, and the radar can also

23 n.m; 9,900 ft alt; 42 ft resolution



High resolution radar image (right) of armored column and tank array (left) shows scene presented to F-15 DRF aircrew from range of eight nautical miles and altitude of 273 feet. Resulting "grazing angle" is 0.3 degree.

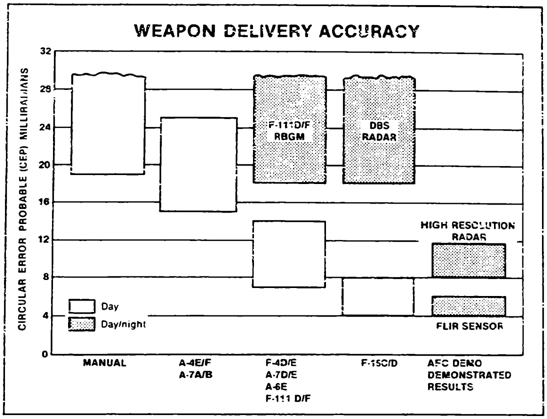
be used with the FLIR for even better performance, weather permitting.

The results of the engineering program on the Dual Role Fighter are most obvious in the cockpits, where improvements to the crew stations have been made to permit maximum utility and minimum work loads. Improved displays have been added to both front and back cockpits; and all DRF operation is either by hands on stick and throttle or with a very limited number of controls. Imagery from the high resolution sensors or a variety of other command and control communication links can be presented to the front seat crewman on any of three cathode ray tubes (CRT) and a wide field-of-view head-up display or to the back seat crewman on any of four CRT's.

Current position and target locations, sensor footprints, threat locations, etc., are all portrayed on a digital moving map display presentable in either cockpit at a variety of scales. The crew can instantly appraise the tactical situation, see the coverage of the sensors, select a run-in heading, locate other aircraft formations, etc., all in the context of a "map" background. When an image of a selected area is wanted, a point is simply designated on the tactical situation display, the radar or FLIR (forward looking infrared) is commanded, and the image is brought up on an adjacent display. A target can then be designated on that display for subsequent attack. There is also an option on the radar to designate a point and select a FLIR image of that point and then compare the radar and the FLIR images side-by-side. This is extremely useful on critical targets where visual confirmation is required or for such things as bomb damage assessment.

The same capabilities also have an obvious application to real-time reconnaissance and inflight targeting in support operations for other aircraft. We have also been demonstrating the capability to hold a frame of imagery, designate a selected area, and then transmit that image to another aircraft or to a ground station. Eventually, these systems will have the capability to both transmit and receive — either between aircraft or between ground stations and aircraft, thus permitting rapid dissemination of targeting information throughout the force.

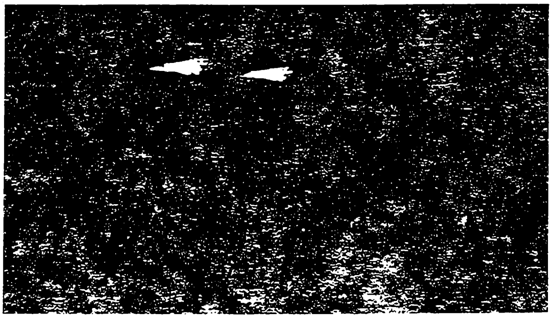
For the Dual Role Fighter, we will also be adding terrain following capability and a high resolution FLIR sensor for clear day/night operations. Terrain following radar and the FLIR sensor are provided by the LANTIRN program. Two pods are mounted on



dedicated hard points on the under side of the fuselage, for the low altitude penetration mission. The high resolution radar complements the FLIR in that precise target areas can be located from long ranges and the FLIR is automatically cued to look at the same target. As the aircraft reaches closer ranges, the high resolution infrared image provides sufficient detail for target recognition. The targeting pod also contains a laser designator/ranger for marking targets and for providing precise guidance for laser-guided bombs. While the technology is still a few years away, the LANTIRN system will eventually have an automatic target recognition system that will detect tactical targets and automatically hand off their locations to Maverick missiles.

This article has stressed the air-to-ground side of the new F-15 dual role configuration, but I want to conclude by emphasizing that these advanced capabilities have come at no degradation of the air superiority mission. Conformal fuel tanks are a standard part of the DRF, but the expanding roles for this aircraft are all accomplished without major changes to the airframe or the propulsion system. We even retain the "one-man operability" in the front seat for air-to-air. And the Dual Role Fighter configuration provides the foundation for future growth into the Reconnaissance and Wild Weasel roles. We are working toward that capability now.

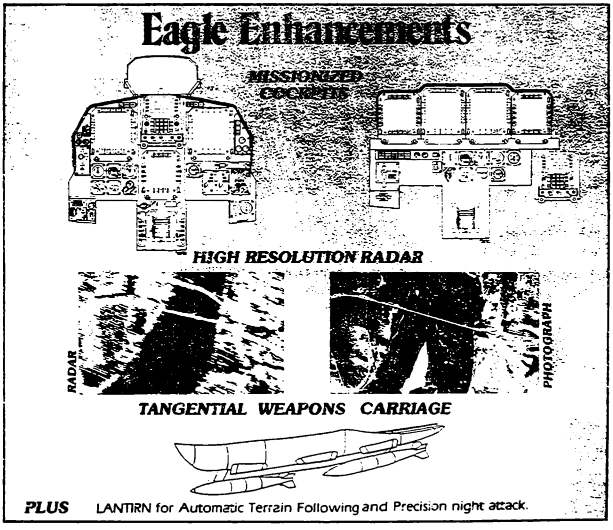
Gary Jennings and Wayne Wight, the primary MCAIR flight test aircrew for the programs described by Mr. Kozlowski, will continue this presentation in the next issue of the DIGEST.



F-15E Dual Role Eagle

On February 24, 1984, the U.S. Air Force announced plans to procure the F-15E Dual Role Fighter for the deep interdiction and air superiority missions. In an evaluation, the F-15E Dual Role Eagle demonstrated its excellent survivability, air superiority performance, range/payload capability and ability to detect and attack tactical size ground targets at night and in bad weather.

The F-15E will have a crew of two; its missionized cockpits feature cathode ray tube multi-purpose displays, a wide field-of-view HUD and up-front controls. It also has high resolution radar ground mapping capability, the LANTIRN system, survivability enhancements, flight control improvements, and the capability to carry and deliver an extensive array of weapons. The addition of conformal fuel tanks fitted with low drag tangential weapons carriage pylons gives the F-15E excellent range and payload capabilities.



(PUBLISHED 1983)

product support digest

VOLUME 30 NUMBER 2 1983

(front and back) Cover photograph shows four Hornets preparing to launch from USS CONSTELLATION (CV-64) during recent operational evaluation tests. Picture by MCAIR Field Service Engineer Art Mester.

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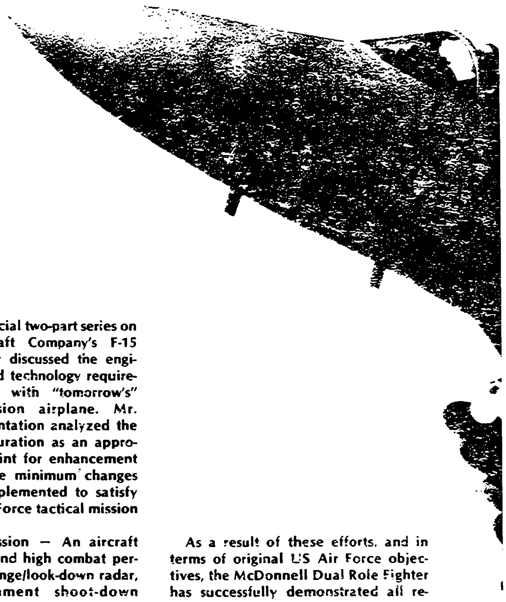


Part I of this special two-part series on McDonnell Aircraft Company's F-15 Dual Role Fighter discussed the engineering design and technology requirements associated with "tomorrow's" USAF multi-mission airplane. Mr. Kozlowski's presentation analyzed the basic F-15 configuration as an appropriate starting point for enhancement and described the minimum changes and additions implemented to satisfy the following Air Force tactical mission requirements:

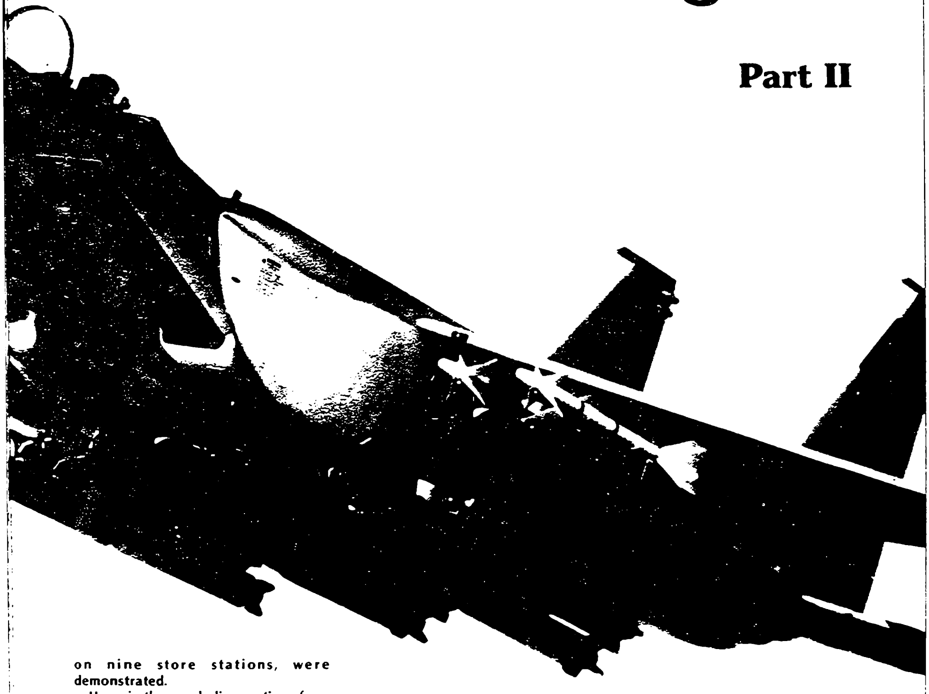
- Air-to-air mission — An aircraft with long range and high combat performance, long range/look-down radar, and all-environment shoot-down missile capability.
- Air-to-ground mission — An aircraft with long range, large payload, accurate and automated all-weather systems, high speed at low altitude, adequate avionics volume, high combat performance, and two-place missionized cockpits.

The F-15 Eagle as it exists today has demonstrated many of these characteristics, and with the enhancements described in these two articles, the F-15 "DRF" can achieve all of them. The concept of and configuration for our dual role fighter has evolved through an extensive company-funded program and USAF evaluations which have utilized four F-15 aircraft and two simulators; and required more than 2200 simulator hours, 600 flight hours, and 400 flights from St. Louis and Air Force bases around the country.

As a result of these efforts, and in terms of original US Air Force objectives, the McDonnell Dual Role Fighter has successfully demonstrated all requirements. It is a two-place, multi-mission aircraft with long range/all-environment radar; extended flying range and increased payload; high speed/low-altitude mission capabilities; improved survivability; and accurate weapons release. The F-15 DRF has demonstrated real-time, high resolution (8.5 feet), long-range (155 nm) radar mapping capability, with grazing angles as low as 0.3°. The cockpits provide the crew with highly integrated control and display concepts, including an electronic moving map and color displays, front and back seat display compatibility and interchangeability, and easy-to-learn, logically-arranged controls. In a recently concluded USAF flight evaluation at Edwards AFB, California, aircraft takeoff gross weights up to 75,000 pounds, with sixteen different payload configurations

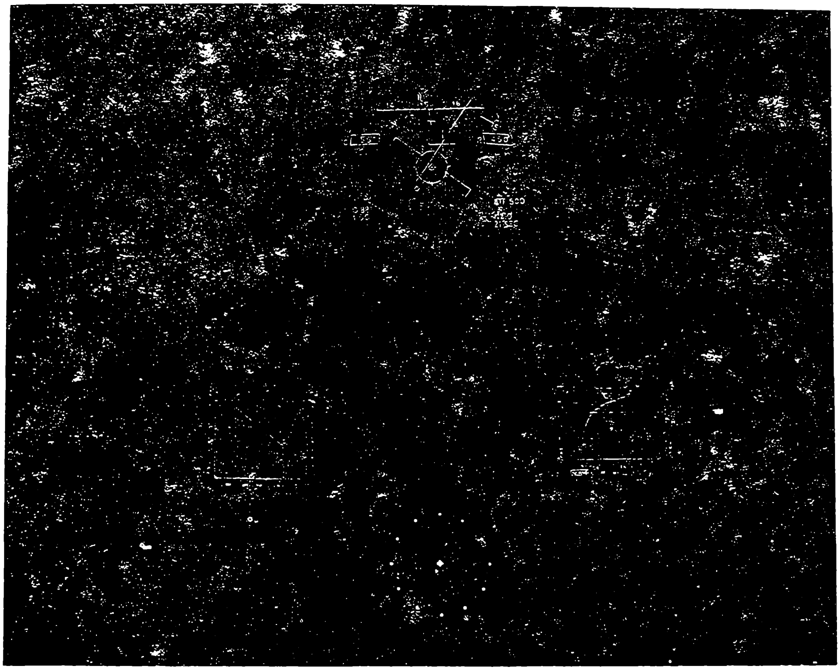
F-15 "DRF" Dual Role Fighter

Part II



on nine store stations, were demonstrated.

Here, in the concluding portion of our special presentation on the F-15 DRF, MCAIR project pilot Gary Jennings and weapon systems operator Wayne Wight take you "inside" the cockpits for a look at the controls and displays, the actual simulator and flight programs, and the tactical refinements which have resulted in a configuration which satisfies future USAF Air Defense and Rapid Deployment Force mission requirements.



(PUBLISHED 1983)

Aircraft Operation

By GARY L. JENNINGS/*Senior Experimental Test Pilot*

Fighter units that I have belonged to in the past considered "night flying" an unnecessary evil, and "night weather flying" an emergency procedure. No more! The F-15 Dual Role Fighter will join the gallant ranks of F-111 units in the ability to deny an enemy the sanctuary of moving under the cover of poor weather, day or night.

The DRF Eagle is designed to function and survive in an arena of high speed, very low altitude, and more often than not, in weather where the driver cannot see ten feet. Sure, you say! Well, take a look at the illustrations herein, which show the primary new equipment which is to be installed in the front and back offices of the F-15 DRF. Look closely, for they provide the capability for doing all of those things – high speed/low altitude/all-weather operations through MCAIR redesigned crew stations with advanced controls and displays. We are in the unique position of having the most advanced avionics-equipped fighter in production today – the F/A-18 Hornet, plus a large simulation facility to study new concepts. Thus the F-15 DRF has arrived by way of Hornet avionics through the MACS (Manned Air Combat Simulator) V. Simple, and simply beautiful capabilities.

Let's take a quick tour through the forward cockpit. Note first the obvious: CRT (cathode ray tube) displays, WFOV (wide field of view) HUD (head-up display), and expanded UFC (up front control). Note also the absence of a basic mechanical attitude indicator. The new HUD will be "holographic," which gives it the capability of displaying infrared video. Basics for the flight instruments (airspeed, altitude, and attitude) are presented in an easy-to-read and interpret format (Figure 1). Airspeed and altitude are large alphanumerics which allow both to be read quickly to one knot and ten feet respectively. The pitch attitude reference lines are tilted toward the horizon, with the magnitude of the tilt angle one-half the pitch attitude. These changes qualify the HUD as the primary flight instrument. If the HUD fails, the display can be called up on any of the CRTs; and for you old diehards, an electronic ADI can also be displayed on any CRT. Basic HUD symbology for A/A and A/G weapons delivery remains unchanged.

The front seat will have two six-inch monochromatic (green) and one five-inch color CRTs; the back seat two six-inch monochromatic and two five-inch color. CRTs are interchangeable front and back. (Incidentally, the question often comes up at program briefings as to "how big and how many displays does the aircrew need?" My answer is that you need as large a display as can be arranged within the physical constraints!) The DRF CRTs add a dynamic dimension to the display of aircrew information in that all displays now are free of mechanical devices and can be programmed for direct reading, easy-to-interpret sources of information. Figure 2 presents a menu display that shows all the options which can be called up on any of the CRTs in either cockpit.

- BIT — Built-in test for all subsystems.
- DTM — Data transfer module control.
- ENG — Engine parameter readout. With digital electronic fuel control in the near future, the parameter list can match the maintenance trim stand.
- VTRS — Video tape recorder system. Will allow programming of which CRTs are recorder and on-board playback.
- RDR — Radar (A/A and A/G).
- TEWS — Tactical electronic warfare system.
- TF — Terrain following.
- TGT IR — Lantirn target pod infrared. Video and system control.
- WPN — Weapon video and control for TV and IR type weapons.
- ADI — Electronic attitude director indicator.
- ARMT — Armament control set for A/A and A/G weapon programming.
- HSI — Electronic horizontal situation display. Now Tacan and INS information can be displayed simultaneously in real world, God's eye view perspective.
- HUD — Head-up display symbology and TV video of the pilot's view out the front windscreen.
- TSD — Tactical situation display. Projected moving map with symbology overlay. The potential for this display is limited only by imagination and computer capacity. An example is threat environment that is dynamic with your flight conditions.
- PROG — Program display option. This feature has two separate programming functions:

  (a) With PROG selected, three options can be programmed. Note that RDR, HSI, and TGT IR pushbuttons have the numerals 1, 2, and 3 next to the display options respectively. This allows each crewmember to cycle the display on that CRT through these three options without taking hands off the stick and throttle or hand controllers.

  (b) Across the bottom of the display are the master modes with associated program options. In this case, if A/A master mode is selected, RDR will automatically be ▶

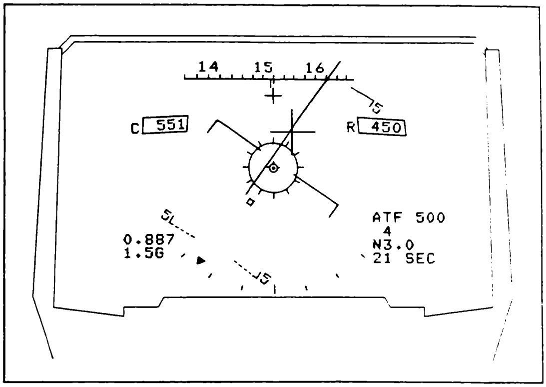


Figure 1 – Wide field-of-view HUD with improved graphics

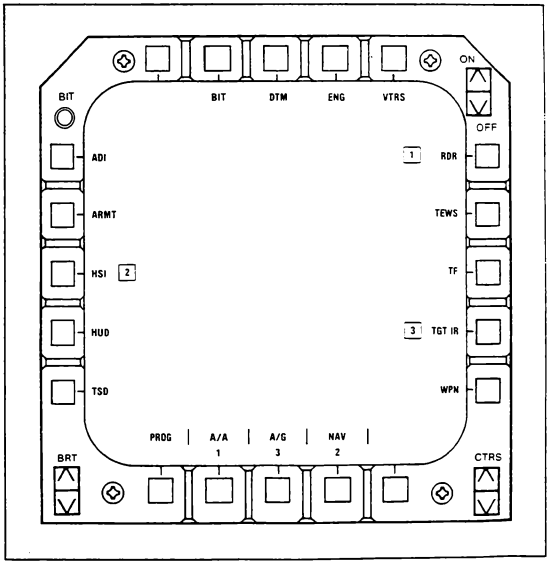


Figure 2 – Menu display showing subsystem options

displayed on this CRT: HSI with NAV master mode; and TGT IR with A/G master mode. This will allow each crewmember to program each CRT prior to takeoff for each major phase of flight.

The up front control shown in Figure 3 is the major interface unit for control of subsystems through digital avionics. If that sounds too much like engineer talk, here is a list of systems at your fingertips:

- INS — All NCI functions, including UTM coordinate entry.
- TACAN
- Autopilot — Altitude hold; command altitude; ground track hold; command ground track; heading set; Tacan, INS, ILS, and course line steering; and airborne ILS.
- TF
- IFF/SIF
- AAI
- UHF radios — Including ADF and KY-58.
- ILS
- JMC } — (Future growth associated
- JVC } with JTIDS.)
- Laser Code

All these systems are controlled by pushbuttons on the UFC status display. Operation is straightforward and quickly learned with hands-on experience. Flight safety will be greatly enhanced since the pilot now has only two control heads on the left console that will require occasional switch activation during flight — the automatic terrain following (ATF) engage and EW quick reaction.

A question that comes to everybody's mind is, "what if the UFC fails?" Well, there are two UFC's, and either controls all systems and each is driven by its own processor with paths to the other UFC. This arrangement provides a back-up to a UFC or processor failure.

Before discussing the new throttle and stick grips, let me reassure the fighter jocks that all the air superiority functions are being retained, including one-man operability. With the master arm switch in arm, the gun is still hot through the trigger. The weapon select switch on the throttle has not been changed. Switch aft to guns puts the radar into the A/A mode and the CRT displays change automatically. Subsequent movement to either missile position prepares the fire control for the appropriate shot.

Now what happens to the WSO when the front seater sets his fangs? Fortunately, his activity to that point in time will not be lost, i.e., his high resolution map will be retained and steering information to the last

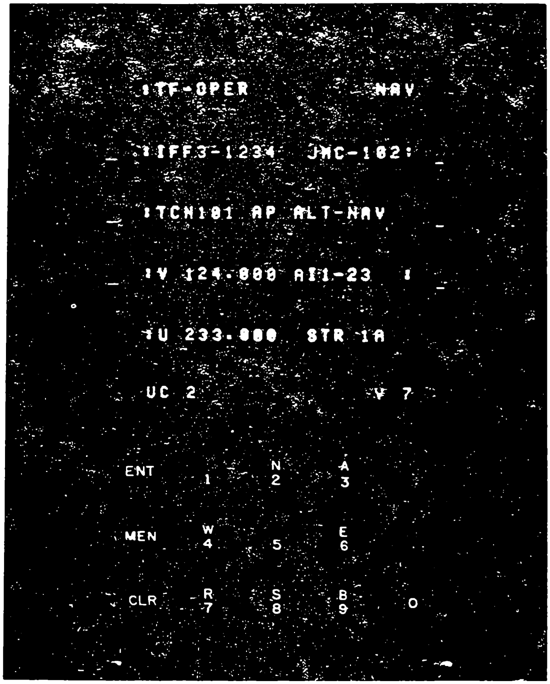


Figure 3 – UFC - advanced digital interface with avionics subsystems

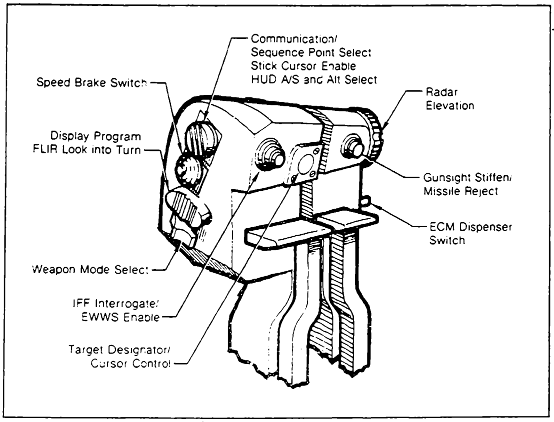


Figure 4 – Front seat throttles

designation will stay active. WSO operation of other sensors will not be affected.

The front seat throttles (Figure 4) are identical to the MSIP F-15 throttles and have the same number of switches as the current design. However, if you look a little closer, a few switches have added functions. The radio and IFF interrogate switches are both now four-position switches, and the EWWS enable button is now a position switch.

The front seat stick grip was adopted from the F-18; and as you can see in Figure 5, a four-position switch has been added. With four displays (three CRTs and the HUD), this switch tells the computer which display/system is to accept commands generated by the throttle switches.

The back seat hand controllers (Figure 6) are mirror images of each other. The left hand controller is tied to the left two CRTs; and obviously, the right controller works with the right two CRTs.

The front seat throttles and back seat hand controllers switch functions first appear to be complicated, but in general the switches do the same action on the display, and their function is then dependent on the system being controlled by that display. (After the DRF selection process is completed and controls and display integration finalized, a follow-up article will cover this subject.)

During the demonstration program and development of the high resolution ground mapping radar, it became apparent that synthetic aperture technology has changed the basic concept and use of radar as a sensor. In addition to in-the-weather capability, the HRR mode presents a totally different perspective to the operator. Those of you who have seen photographs or movies of high resolution radar can attest to the radar map appearing as though the picture was taken from directly above the target area regardless of actual distance. This startling capability gives the crew excellent target area orientation and long-range high resolution sensor video for accurate target designation. The high stress task of finding and attacking a target can now be accomplished much more effectively while at the same time allowing the crew to concentrate more on tactics and threat environment. All of this equates to a very effective weapon system that will enhance getting to, finding, and attacking the target on the first pass with a high probability of kill . . . which is the basic purpose of it all.

The rapidly expanding technology of programmable displays such as UFC's, and high speed digital computers will allow well designed weapon systems to stay current and capable of meeting new threats. Our current Eagle is unmatched in airframe capability, but needs the new DRF digital avionics and display CRTs to meet today's and future requirements.

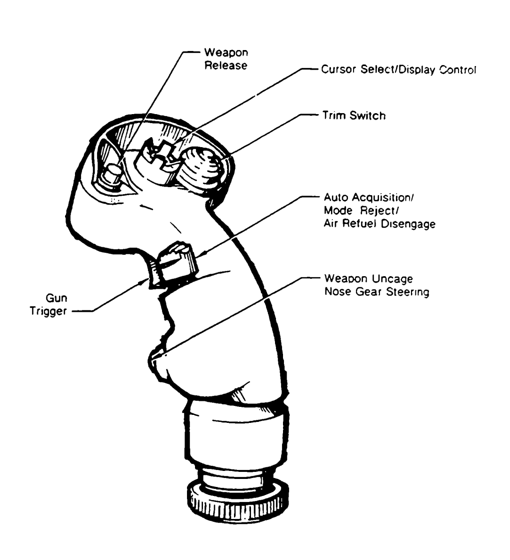


Figure 5 – Front seat stick grip

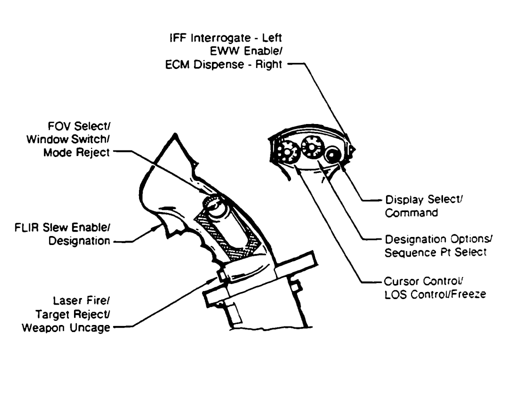
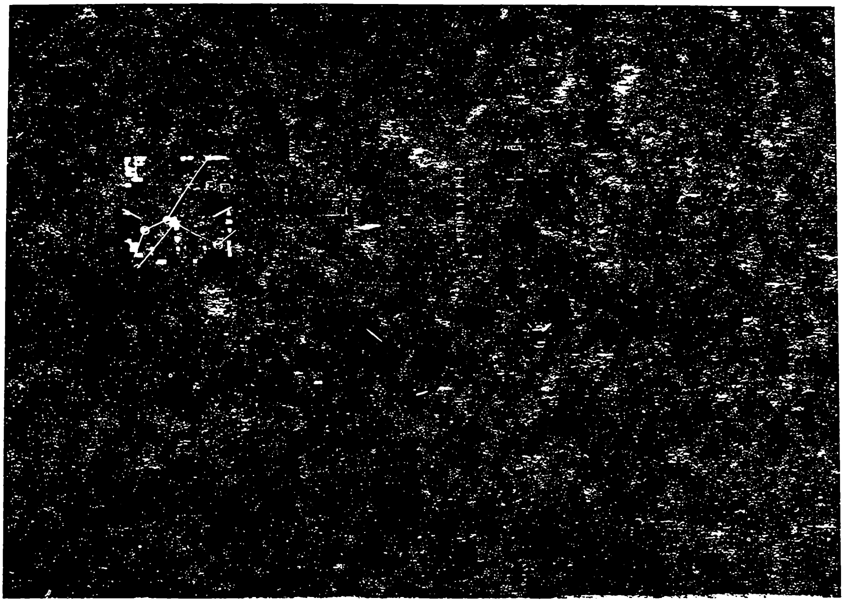


Figure 6 – Back seat hand controllers



(PUBLISHED 1983)

Weapon System Operation

By WAYNE WIGHT/*Chief Systems Operator*

The F-15 Dual Role Fighter development and test program has been an exceptionally interesting one from the standpoints of operational capabilities expansion and advancement, and the roles of the people in the cockpits. Therefore, I'd like to talk about where we started in this effort and how we got to where we are today, as illustrated in the aft cockpit simulator configuration shown above.

There have been a lot of lessons learned along the way of special relevance to WSO's, so if that's your particular field of interest, please stick with me. The Weapon Systems Operator may be considered as the air-to-ground expert in this dual-role/dual-cockpit airplane, while the pilot would continue to be the air-to-air expert. I'll finish by projecting what has been learned into what we visualize as the best package for handling the all-weather attack requirement.

While there have been a few twists and turns plus a dead end or two, as the individual responsible for much of the twizzling and tweaking that has gone on in the aft cockpit of this investigation into tomorrow I have been most impressed with the general continuity of our progress toward a true all-weather attack capability. The base F-15, the MACS IV simulator studies, the AFCD and DRF configurations installed in F-15 S/N 71-291, and the current MACS V simulator work have all been vital and informative steps in this cooperative engineering/operational effort.

PROGRAM REVIEW

It was back in 1977 when we first saw dual-role cockpit hardware emerge from paper, installed in a McDonnell simulator called the MACS (Manned Air Combat Simulator) IV. The new hardware consisted primarily of four display tubes, two hand controls, and a keyboard installed in the aft cockpit of an F-15B simulation. The front cockpit was essentially unchanged from the production configuration.

This aft cockpit was a multi-mission configuration, able to handle any mission then envisioned for the F-15. The primary missions then under consideration were Reconnaissance, All-Weather Attack, and Wild Weasel, but initial simulation efforts were directed toward solving the real-time recce problem. We thought there was a definite need in TAC for real-time reconnaissance, and that the F-15 configured as such would be the best solution. We talked to operational RF-4C crews to get a feel for their needs and began to put together a package in MACS IV that included all capability technically feasible at that time. The biggest player in this package was real-time "SAR" — synthetic aperture radar.

SAR itself was nothing new; in the early 60's, I was involved in flight testing of side looking radar (SLR) using the principles of SAR in RF-4B/C aircraft. However, SAR, up to now, was not "real time." Side looking radars, using SAR principles, stored the information needed to construct the image on film. This film then had to be transported to a ground correlator for processing which took several hours at best. Around 1972, MCAIR began delivering the RF-4E, which carried a side looking radar pod that could data link SLR information to a ground station. The ground station processed this information and put the image out on film in about five to ten minutes. It was still not possible to display SLR imagery in the cockpit.

By 1977, we were ready to move SAR into the near real-time world by constructing a map and displaying it in the cockpit in five to ten seconds, depending on conditions during map construction. This was not real time, like real-beam mapping, but close to it. Map construction time would vary, depending on ground speed, antenna azimuth angle, range, patch size, and number of looks for a given map. We also added simulated ground moving target detect & track and terrain following/terrain avoidance modes to the radar. This was all integrated with a targeting FLIR and a camera package to round out the sensor suite necessary to handle the complete recce mission. (Incidentally, when integration progressed to the point where we could "fly" profiles of various scenarios, it seemed that my box time began to exceed my flying time!)

We have a simulated terrain board which incorporates the types of targets expected to be encountered in a typical European land or sea conflict. We flew profiles against all types of targets to determine when we could detect, identify, and photograph them. More importantly, we also found out when we could *not* detect targets due to shadows from trees, structures, terrain, etc. All sensors that I know of are "line-of-sight"; that is, they don't see through rocks, trees, or anything else that's between you and the target. Radar is the only sensor that looks through clouds. We found out in a hurry that the low altitude environment is a tough situation to handle because you're dealing with a world out there that is obscured by shadows.

As simulation efforts continued, TAC indicated a need to solve the all-

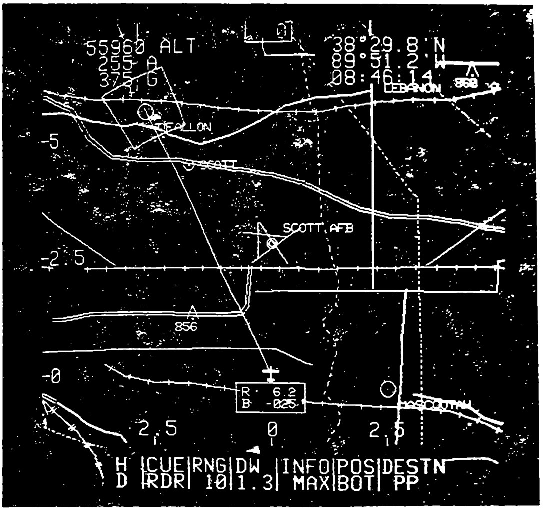


Figure 1 — Electronic map of Scott AFB, Illinois area

weather attack problem and the decision was made at MCAIR to proceed with a contractor-funded program in response to this need. The result was the "AFC Demo" program, and our multi-mission F-15 made its first change of role. F-15B No. 2 (USAF S/N 71-291) was selected for modification into the "Advanced Fighter Capability Demonstrator" configuration, and the back cockpit used in MACS IV was installed in B2 with very few changes.

The intent of the demonstrator program was to investigate the capabilities of synthetic aperture radar by using its imagery to detect tactical size targets, deliver dumb bombs on operator-selected targets, cue narrow field-of-view sensors (Pave Tack system), and to deliver dumb bombs using that system. Sensors necessary for in- and under-the-weather penetration, such as NAV FLIR and TF/TA radar, were beyond the scope of this demonstration. We did the program in VFR conditions with the pilot looking out the window.

As one might expect, progress came painfully slow in the beginning. Quite a few flights were made before we saw any eye-watering high resolution maps, but when this finally did occur, we began to call the SAR a "high resolution" radar (HRR) and certain facts of life began to emerge.

HRR maps are displayed to the operator as if he were observing the scene from directly above, even though the actual aircraft position may be miles away. While this is no different than it was 20 years ago with the SLR, when the image is presented in real time this characteristic really jumps out and grabs you! Orientation is simplified because of this feature, and that's a very important consideration when you're going warp nine on the deck over unfamiliar territory. No other sensor does this, including the eyeball looking through the canopy at low grazing angles. The shadows presented on a low grazing angle map tend to give it a third dimension of depth. These characteristics of radar hold true, regardless of range to the target, as long as the grazing angle is sufficient. Non-ranging sensors, such as eyeballs-out-of-the-cockpit and infrared sensors, approach the two dimensional (azimuth and elevation) when range increases to the point where grazing angle becomes small. We made HRR maps at less than 1° grazing angle, in some cases as low as 0.3°, and still had sufficient video to detect tactical size targets.

We did not need to fly this radar to find out whether it could determine if a vehicle was tracked or wheeled; we knew it would take higher resolution than 10 feet for that. We did, however ▶

want to determine just what could be identified. The answer in general terms is this — any fixed, man-made object larger than a vehicle can be identified with a high probability of success if some prior knowledge such as maps or photos is available. Large man-made objects and prominent natural terrain features can be identified with no prior knowledge by simply referring to the tactical situation display (TSD) on board the aircraft.

We verified once again that radar is truly an around-the-clock sensor. It looks through any kind of air, with the exception of heavy rain, with no noticeable degradation. A large thunderstorm, which an aircrew would not penetrate if they had the option, appears on an HRR map with a shadow behind it. Small showers caused some attenuation on the highest resolution maps, but only slightly degraded the map quality.

The HRR map also turned out to be an excellent cueing device. HRR itself may be considered a narrow field-of-view (NFOV) sensor in the highest resolution modes. Narrow FOV sensors such as targeting IR can identify tactical size targets in favorable weather conditions, but require cueing from an external source to initially bring the target into the field of view. The object (target) is detected on the HRR map, the map is frozen, then a narrow FOV sensor is cued from the map for identification or weapon delivery.

It became evident early in the program that high quality radar maps and accurate blind weapon delivery require an accurate and reliable inertial navigation system (INS). You have to know where you are within one mile, preferably a half mile, and onboard velocity error needs to be a half knot or less if you're going to be very effective in this business. Excessive velocity errors mean the radar will have to be used in its precision velocity update (PVU) mode to correct for INS velocity errors. If the INS position drift is excessive, frequent updates will be required to enable the WSO to find planned targets. When INS errors are small, maps can be frozen for long periods of time before another is required for a position update. This capability enables the crew to minimize RF transmissions when they are in bad guy country or to dedicate more time to the air-to-air mode before they need to return to HRR for an update map.

A TSD is required to keep the crew aware of the tactical situation. We used an in-house invention called an electronic (digital) map (Figure 1). Operating areas where we operated were digitized from aeronautical charts and the information stored in a map storage unit. The capacity of this map storage unit was less than desired, so the electronic map suffered from a lack of detail. However, it exhibited no noticeable registration error, a feature important in this business and an area where other types of TSD's, such as projected maps or remote map readers, don't do as well. When INS position errors were small, the electronic map proved to be an excellent cueing device. Unfortunately, the data base required for electronic map coverage of the world will not be available until many years after the DRF hits the squadron. The DRF will most likely leave the factory with a TSD that is a remote map reader.

ADVANCEMENTS

As you can tell, I am beginning to talk about the future now. We definitely learned some things that we would do differently the next time around.

- Doppler mapping is tough to do near your ground track, and it's impossible right on your ground track. The first time around, we did not attempt map construction closer than 8° from the ground track and left this area black if maps were commanded there. In the production DRF, we are going to fill in this zone with the best map available. Obviously, you still won't see much where you are going so you will still have to map territory offset from your track or deviate from your course occasionally to get that good quality map. We did not find this shortcoming to be much of a problem during our program and feel that the operational crew can also live with it.
- We flew this program with the real-beam PPI mode disabled. In the DRF, there will be a real-beam mode with its quality improved over that now in production F-15's. A real-beam map can give the crew an idea of the shadows present over a point of interest and whether or not they should attempt a high resolution map there. We are going to display inertial points such as steer points, aim points, and target points on the PPI map to give the crew added situational awareness and a means from which to cue high resolution maps.
- The DRF will have the inherent capability to extend the maximum range of its high resolution maps. This feature will enable the operational crew to obtain their map, freeze it, and get down in the weeds farther from the target. Which brings up another point; if you are going to drive into a target with a frozen map constructed a long way out, you had better have very small onboard velocity errors or you won't do much of a job delivering a weapon or cueing a NFOV sensor. The DRF will have a more accurate and reliable INS than we had in our program. It will also have a continuous PVU interleaved with high resolution mapping and display the velocity error to the crew so they can update their velocity if they think the error is excessive.
- In the demonstration program, we could detect ground *moving* targets on a high resolution map, but their position was displaced due to their doppler shift compared to that of the fixed terrain comprising the rest of the map. In the DRF, we are going to develop a ground moving target (GMT) mode which shows only "movers" and not the rest of the map. These movers would be displayed in their correct position. The DRF will also have closed loop track capability on these movers.
- We looked at ships at sea with the HRR mode. Detection of the ships was no problem, but ships out in the open water are generally moving along at a pretty good clip. The same problem occurs in this case as it does with ground movers in that their position is displaced due to their doppler shift. The DRF will have the inherent capability, by developing the appropriate software, of providing a mode optimized for Sea Surface Search (SSS). This would be a real-beam mode (non-doppler) in which ships would be presented as synthetic targets and in their correct position. Closed loop track on ships could also be mechanized.

There were some other interesting capabilities that we experimented briefly with during the later stages of the program. For example, we flew a couple of flights with the capability of zooming in with a 3X magnification on any point of interest on a frozen map. Figure 2 shows a frozen "mother" map of the Rolla (Missouri) National airport area. Figure 3 shows a zoom on a point of interest on that map. The magnification shows some detail not obvious on the mother map. The frozen mother map can be recalled after the zoom and another point of interest selected for a second zoom. This feature looks like it would have some potential for medium resolution maps and could be included in the DRF.

Figure 4 shows a down-linked map of a portion of Lambert St. Louis Airport. This map was viewed by people in the ground station seconds after it was seen by the aircrew. This map was also stored in an onboard bubble memory for retrieval after landing. A bit-by-bit comparison of the two maps showed them to be identical. Once map data is stored onboard or is down linked, all

Figure 2 – "Frozen" high resolution map of Rolla, Missouri airport

Figure 3 – 3X magnification of frozen map

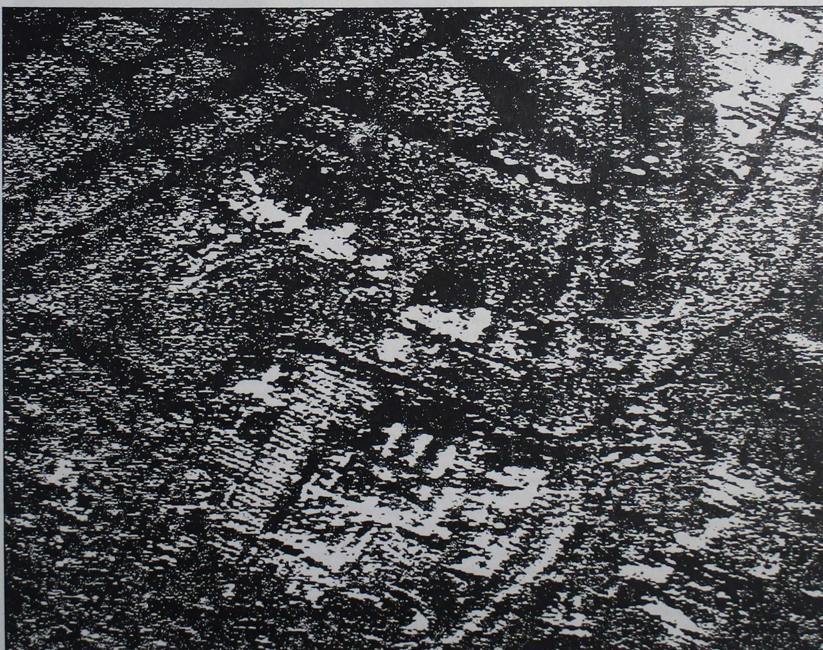


Figure 4 – Down-linked map of Lambert/St. Louis International Airport

kinds of computer enhancement games can be played to increase the utility of the original map data.

Detailed charts and photos are great, but they usually don't show the scene like radar does. Can you imagine how much easier it would be if you carried onboard, in bubble memory, high resolution maps of your target area taken from previous flights? Visualize taking off with 20 stored high resolution maps of your IP(s) and target(s), with some taken at three hour intervals to show movement. Overlayed on the map would be symbology produced by photo interpreters and intelligence personnel. This symbology would identify likely targets or other points of interest. When you've got your real-time map of the target displayed on one six-inch tube, you can call up a stored map of the target to be displayed on the adjacent six-inch tube for comparison!

If up-link capability is added along with the down-link, we have a whole new ballgame. The process mentioned in the previous paragraph could be accomplished in just a few minutes. Besides the two-way data link with the ground station, we could then send and receive high resolution maps to and from similarly equipped aircraft in the area. This same process could be done with IR video.

After reading this far, you jocks out there are probably wondering just how old or how long out of the service you will be when all of these fancy things come to pass. Well, let me assure you that the technology is here now; we could start flight development tomorrow!

F-15 B2, in its present back seat configuration, can do a creditable job in the All-Weather Attack role as far as the WSO's contribution goes. We have learned a great deal during these programs, and if this knowledge is applied, the F-15 will be just as capable in the air-to-ground role as it already is in the air-to-air role. ■

About the Authors

GARY L. JENNINGS

Gary L. Jennings has been an experimental test pilot in the McDonnell Flight Test Department since 1979. He is currently flying the F-15 Eagle and is involved in the "Strike Eagle" demonstration program and in advanced engineering/development for the F-15 Dual Role Fighter.

Gary has been flying fighter aircraft since 1967 when he completed training in the F-102, and has logged over 3700 flight hours in military and contractor flight operations. During his thirteen and one-half year US Air Force career, he became mission-ready qualified in the A-7 and the McDonnell F-101 and F-4, and served a combat tour in Southeast Asia in Phantoms with the 435th TFS (8th TFW) at Ubon AB, Thailand. He also served with the 62nd, 75th, and 445th Fighter Interceptor Squadrons, and with the Flight Test Acceptance Unit at OCAMA. He completed USAF Test Pilot School in 1974 and then served as a test pilot for the 475th Test Squadon; the Aerospace Test & Engineering Establishment at CFB Cold Lake, Canada; and as a TPS instructor.

Gary received a BSAE degree from the University of Illinois in 1966. In addition to flying with McDonnell, he is currently flying the F-4 as a Major in the 170th Tactical Fighter Wing of the Illinois Air National Guard at Springfield.

DON R. KOZLOWSKI

Don R. Kozlowski has been with McDonnell Aircraft Company since 1959 and is currently Chief Program Engineer for the F-15 Dual Role Fighter Program. He has previously been responsible for Advanced Tactical Fighter Studies and Advanced Supersonic V/STOL concepts. He has also served as Director, Offensive Air Support Mission Analysis for the USAF (ASD/XR) in a program culminating in a technology plan for close air support and battlefield interdiction.

Since 1973, Don has directed studies in design and operational effectiveness of attack aircraft, including ATF, ATCAF, and supersonic cruise for the USAF and VAMX for the USN. Prior to that he directed studies in target acquisition, advanced crew stations, and image processing for real time image enhancement, cueing, and target identification in the Reconnaissance and Exploitation Section of the McDonnell Reconnaissance Laboratory. His experience also includes work on radar, electro-optical sensors, countermeasures, command and control, and intelligence systems.

Don received a BSEE degree from St. Louis University in 1959 and an MSEE from Washington University in 1967. He is a member of AIAA and IEEE, and has received the USAF Meritorious Civilian Service medal.

WAYNE K. WIGHT

Wayne K. Wight has been a Weapon Systems Operator in the McDonnell Flight Test Department for nearly 20 years. He is currently Chief Systems Operator and involved in the F/A-18 test bed radar development program, after having served as Project Systems Operator on the F-4 Wild Weasel and F-15 AFCD projects and in the F-15 fire control radar test bed fly-off competition and subsequent development.

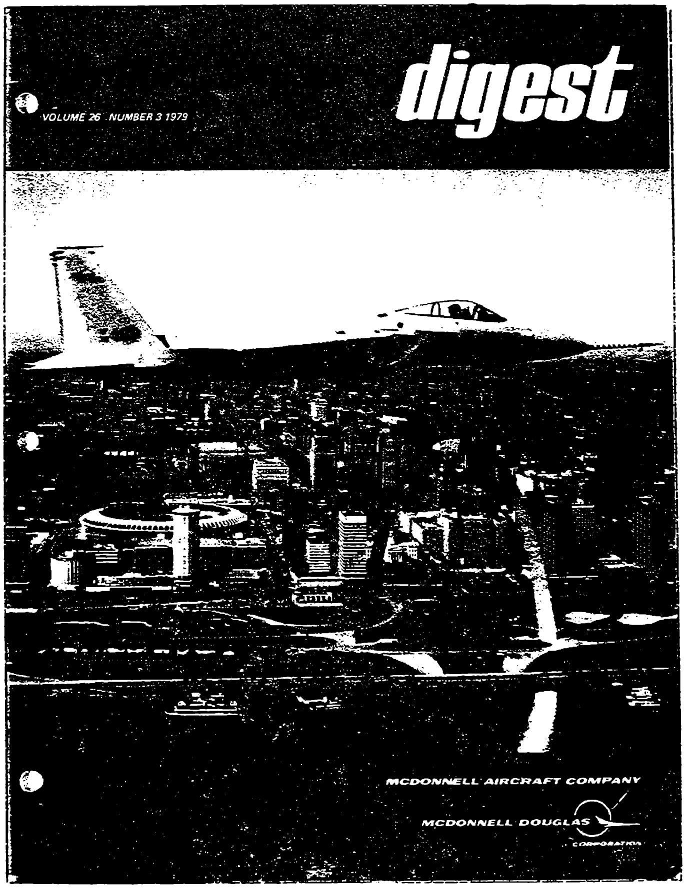
Wayne has logged more than 4800 flight hours, which includes 3500 hours in production and experimental flight testing of F/RF-4, F-15, and F/A-18 avionics systems. His F-4 work included the AWG-10 radar, weapon release computer set, LORAN, RHAW systems, and the Walleye, Shrike, and AIM-4D missiles. RF-4 assignments included the internal and podded side looking radars, infrared reconnaissance set, cameras, and forward looking radar development.

Wayne received a BSME degree from South Dakota State University in 1960, after a college career that was interrupted for two years at Air Force Navigator and Radar Observer schools through the South Dakota Air National Guard. He flew with the Guard while employed as a Titan II inertial guidance systems engineer with General Motors Corporation until joining McDonnell Aircraft Company in 1963.

NOTES

FAMOUS AIRPLANE OVER A FAMOUS SKYLINE

F-15C Eagle No. 1 during a production test flight early in 1979 over downtown St. Louis. Among the many historic and civic structures visible in this striking photograph are the Saarinen Arch ("Gateway to the West"); Busch Stadium (home of the baseball and football Cardinals); Old St. Louis Cathedral (1831 - first church west of the Mississippi); Old Courthouse (begun in 1839 and scene of the famous Dred Scott trials in the late 1840's); and several city, state, and federal government buildings. Arch reflects in the Mississippi River, between some of the old-time paddlewheelers that still ply the river and maintain the spirit of an earlier St. Louis. Photograph was taken by MCAIR company photographer Bob Williams from another F-15.



"EAGLE TALK". . . a crewman's multi-volume history of the McDonnell F-15 airplane

(reprints from the MCAIR PRODUCT SUPPORT DIGEST)

The F-15 Eagle became operational on the 14th of November 1974, at Luke Air Force Base, Arizona. As of this point in time (1983), more than 800 F-15A, B, C, and D model aircraft have been produced for the air forces of the United States, Japan, Israel, and Saudi Arabia. Only speculation is possible regarding an ultimate number of aircraft and the Eagle's ultimate position in the history of aviation and the world, but its position thus far is both secure and spectacular. The McDonnell technical support publication — PRODUCT SUPPORT DIGEST — has documented this "progress of the Eagle" from the very beginning in articles and reports by flight test and engineering personnel. Prepared exclusively for our military customers, these articles offer both a fascinating, informal history of the F-15 program and contemporary technical discussions of aircrew techniques and procedures. Regardless of one's level of experience with or degree of exposure to the Eagle, information of the type published in PSD is worth reading and preserving. However, it is the nature of magazines to be temporary and disposable, to "disappear" in time with the consequent loss of valuable data to personnel newly assigned to our airplane. Therefore, as on previous aircraft such as the Demon, Voodoo, and Phantom, MCAIR preserves this hard-won expertise in the form of periodic collections of previously-printed articles. This is Volume I of the Eagle collection and is composed of general-interest material arranged in chronological order; if you are interested in how the F-15 got to where it is today — test programs, simulators, milestone events, etc., it's all here in this volume, in authoritative articles written by the pilots as they were performing the tests. Volume II contains the more technically-oriented aircrew articles, arranged by subject, from the past 10 years. Volumes III and up will be published as the accumulated information warrants.

There is a tremendous amount of information packed into these slender volumes of talk about Eagles, but there are two points to bear in mind when reading, one concerning the "currency" of the material; one its "applicability" —

• Articles published herein were up-to-date and valid technically as of the time of original publication (indicated in the table of contents and on each article). However, the F-15 Eagle as it is coming off the assembly line today contains many differences from the earlier (and earliest) configurations. Ship No. 1 and Ship No. XXX (latest to fly) may look alike on the outside but, from both system and operational standpoints, they are not alike. If you read something in these articles that does not resemble the cockpit or system as you know it today, please "check six" to see where the information is coming from — its date of publication. It would have been too difficult and time consuming on the part of our pilot/authors to review every past article for current validity (especially since some crewmen are no longer flying or are flying other airplanes). Therefore, we suggest you use these volumes for background and general information on aircraft systems, techniques, and procedures. EAGLE TALK contains a wealth of wise words, but only your DASH ONE is guaranteed to have the latest, and the official, ones. Which leads directly into the second point.

• Please be sure you understand the "type" of information provided in these volumes (and in the PRODUCT SUPPORT DIGEST from which they were reprinted) so you won't be looking for advice that isn't there and thus get disappointed. Our publications do not discuss F-15 "tactics." How to utilize the aircraft in combat is the subject of official military documentation; our only objective is to inform you about F-15 "capabilities." The theory behind this is that the more information you learn in our publications, the better you should be able to apply the information in yours.

Since this page deals with the "philosophy" behind EAGLE TALK and the PRODUCT SUPPORT DIGEST, it would be appropriate to end with a quote from an individual who has provided much of the information in both. In one of his articles on Eagle driving, Pat Henry, (current) MCAIR Chief Experimental Test Pilot, wrote. . . "As with most philosophical discussions, no decisions are made for you, so the monkey is still on your back to handle any given (soggy) situation. That's the responsibility that accompanies the pride of professional flying."

Thus, on behalf of the people at McDonnell Aircraft Company who have contributed to these publications, our wish is that, when the monkeys begin to climb up your back in some future (soggy) situation, you will recall some of the discussions herein and that all of your Eagle flying will be responsible, proud, and professional!

VOLUME I — TABLE OF CONTENTS

not for public release

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MCDONNELL DOUGLAS

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The before-takeoff engine runups are normally uneventful; however, it is important that the pilot pay close attention to the nozzle actions. Proper nozzle operation during an idle to military throttle slam on the ground goes like this: The nozzle closes rapidly to around 10% open as the N2 and FTIT are climbing rapidly toward representative military power numbers. As the EEC puts a final trim on the engine, the nozzle will stabilize somewhere between 0 and 10%. If the nozzle appears to be opening rapidly and is in fact opening past 30%, the throttle should be reduced to idle immediately, since the engine may overspeed, overtemp, and self-destruct. History, so far, is that three engines have failed by this route. The problem has been traced to faulty EEC's, which have since been modified.

When the engines pass the run-up checks, the ensuing takeoff with two of these hummers in max A/B is best described as fun! With the blessing of the local air traffic controllers, we take the new Eagles almost straight up and out of the airport traffic area.

We rarely see engine problems in cruise or in the landing pattern. About the only interesting items in cruise and landing are the nozzles. In cruise, the nozzles are normally controlled to about 10% open at military power and below. The nozzles do open up at high speed/low altitude even though at mil power in the later engines. In the landing configuration, the nozzles are programmed with throttle angle and are nearly closed at mil and are open at idle. That's why the nozzles go open when you put the gear down in the pitch out.

We make engine checks aloft as you would on an FCF. Specifically, we check idle to max A/B slams, mil to max A/B slams, airstarts, and engine operation throughout the airplane envelope.

Guess where we have the most engine-related problems? You're right - A/B lighting anomalies in the high altitude/low speed regime. The "funnies" we see include A/B no-lights, early segment (1-3) A/B blowouts, late segment (4-5) A/B blowouts, fan stalls that recover with no pilot action, and fan stalls that recover only if the pilot places the throttle in cut-off. That's the bad news. The good news is that the engine-related problems we see are less frequent now than in the past, and that the new airplanes are being delivered with engines that pass the required tests.

Reports from the field indicate that some of the same problems exist with you operators. About once every 200 aircraft flight hours, a stall/stagnation is reported. About 80 percent of the stall/stagnations occur as a result of A/B operation; the rest are caused by mistrim or hardware failure. The mechanics of a stall/stagnation usually go something like this:

1. The A/B blows out and relights.
2. A pressure pulse travels up the fan duct and stalls the fan.
3. The distorted fan flow stalls the compressor of the core engine.
4. The combustor blows out.
5. The core engine spools down rapidly (N2) to subidle speed.
6. The engine restarts and tries to accelerate, but is in a low stall margin subidle speed range.
7. The engine either restarts and accelerates or degenerates into a low N2/high FTIT condition that will eventually overtemp the engine.

This is what we see on the data printouts after a typical stall/stagnation test flight. If watching the engine gages, the pilot may notice one of the following sequences of events:

1. If A/B is selected from below mil, the engine accelerates toward mil and then allows the A/B to light in segments. If the A/B lights/blows out/relights hard enough to cause a pressure pulse sufficient to stall the fan, the fan stall is usually noticed by the pilot as a pretty loud pop. N2 and FTIT will start down at the stall; but as the core relights, FTIT will level off and then start to climb as N2 continues downward. This is an example of a stall that stagnates and must be cleared by shutting the throttle off and then restarting when in the airstart envelope.
2. Whether the engine goes into a stagnation after a stall seems to depend on the rate of decay of N2 at the stall. Some stalls recover if the N2 hasn't dropped more than about 8% following the stall. Some end up in a steady state "rotating air hammer-sounding" stall that probably will result in a shutdown and restart for clearing. These can be in the N2 65%, FTIT 500°C range and seem to last forever.
3. If you don't hear the noise of the fan stall or if the engine stalls due to a problem not associated with an A/B anomaly, your first clue may be a master caution plus a generator-off light on the telelight panel. In this case, shutdown and restart are usually required. (Note the max FTIT for engine disposition purposes. A momentary overtemp may only require a borescope whereas a serious overtemp will require an engine replacement and teardown.)

The engine manufacturing company thinks it has fixes to the stall/stagnation problem. The fixes attempt to eliminate the stagnation part at any rate. One change opens the nozzle at stall; another selects minimum segment one A/B when a stall is detected. The pilot must cycle throttle to below mil to relight the A/B. Testing and service use will be the final judge of these modifications. The stall/stagnation testing did clear the airframe of any contributory fault; the inlets, ECS, and fuel system were instrumented and operated on design.

That about brings you up-to-date on F100 engine activities here at the balloon factory. We'll continue to give you status reports on new and interesting developments as they occur. ■

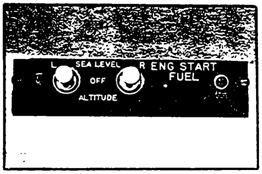
# of F-15 airstarts and things... (PUBLISHED 1976)

LIGHTING THE FIRES

By PAT HENRY/*Project Experimental Pilot*

After the traditional (but not considered mandatory by Stan-Eval) procedure of kicking the tire, further progress toward flight is in no small measure dependent upon successfully lighting the fire. With the Eagle, this lighting of the fires ceremony may require manipulation of a couple switches tucked way back on the rear corner of the right console, on a panel labeled ENGINE START FUEL.

The photograph shown at right proves that in some cases one picture is definitely not as good as a thousand words. In fact, the nomenclature on the Engine Start Fuel Panel fairly screams for further explanation. In a few more than 1,000 words from now, you'll have your explanation.



Granted, there's not a whole lot the pilot can do about immediately increasing his jet time when he's stuck with a bad starting engine on the ground; but understanding the system could help troubleshoot the problem and might even extend his jet time if an airstart is needed. Let's have a look.

Back in the Good Old (?) Days

For openers, how about a little history? The F100 engine has many good points - high thrust to weight, compact size, good response (for a fanjet), etc; but let's face it - starting is just not one of them. It's a problem that has been with us since before first flight; and although considerable progress has been made, it is recognized still as an area of desired improvement.

For what it's worth, the prototype engines were *horrible* at starting. We used a manually controlled bypass valve, either open or closed, and the amount of fuel bypassed was approximately 350 pph. Since this represented over half the total starting fuel flow, the potential for either no light at one setting or stagnated start at the other was tremendous. It was often necessary to close the bypass just to get a light-off, open it right back up to avoid stagnation, then cycle it as required to walk the line between a hung and a hot start. The exact amount of fuel bypassed had to be tailored to individual engines, in some cases by varying the bypass orifice size. Even P&W admitted this was not operationally suitable. Once at idle, opening the start fuel bypass would cause an N_2 rollback of from 3%-5%, which could put you in a sub-idle condition and looking at a potential off-idle stall if airborne.

The problem was not as straightforward as we mere mortal pilots tried to assume, due to the many variables involved. Furthermore, changing one variable for improved starting, such as fuel nozzles, could well lead to degraded performance elsewhere in the envelope. Among the key items to be juggled were fuel nozzles, combustor design, bypass ratio, JFS performance, start fuel flow schedule and repeatability, start bleed air volume, and start bleed strap closing point.

Rule 1: Set Attainable Goals

From day one, our design goal has been to build an engine that could be ground started under a wide range of ambient conditions without special starting techniques or devices. This ground starting envelope is bounded by sea level starts from −40°F to +125°F at one extreme to 10,000 feet field elevation starts at temperatures of −15°F to +85°F at the other. While testing the various engineering changes during the Category I program, it became apparent that we would still need some level of starting fuel derichment/enrichment in the production engines, for the time being anyway.

FIGURE 1		
Panel Name	Engine Start Fuel	Engine Start Fuel Flow
Switch Position	Sea Level Off Altitude	High Auto Low

Engine Start Fuel Panel Nomenclature – Old and New

Having bitten this bullet, the next logical step was to make as simple and reliable a starting derichment system as possible. Early analysis indicated that three different fuel flow rates would be required to assure successful starts throughout the envelope described above: 285 pph for extreme high altitude/hot day starts, 350 pph for most ambient conditions, and 450 pph for extreme low altitude/cold day starts. One mechanization considered would have made the middle fuel flow rate the base setting, with a manual pilot selection to either increase or decrease flow if necessary. Another consideration, which was also met with underwhelming enthusiasm, would have made the switchovers automatically as a function of ambient pressure and temperature. What finally evolved was a compromise - a reasonably uncomplicated two (vs three) flow rate system which can automatically handle the vast majority of starts. Given up in exchange was the upper right hand corner - extreme high field elevation, hot day starts.

The Mysterious Engine Start Fuel Panel

Referring again to the photo, you might ask, "Why did they label the switch positions that way?" Don't ask. The exact history of that nomenclature evolution has been mercifully lost. Suffice it to say that improved (we think) labeling is coming, presently scheduled to be effective with Aircraft F-120/TF-21 and up, plus provisions for retrofitting all earlier ships. Figure 1 compares the old and the new.

mum glide. That, in turn, has three distinct advantages over the continued high speed descent: time, range, and maintenance of good (i.e., slow) ejection speed. The time extension gives you a better chance to collect your thoughts, read the gauges, and hopefully make some of the best decisions of your aviation life. The differences between 220 kt and 350 kt glides are dramatic; as illustrated in Figure 2, descent rate at 350 kts is more than 2-1/2 times that at 220 kts.

Now or Later

As you approach the JFS altitude envelope, you're faced with the next big decision — whether to go for a JFS start at 25K, or delay until some given lower altitude, but with a higher restart probability. While it is our belief that the JFS should be good up to at least 25K, there doesn't appear to be much to gain by pushing the upper JFS altitude limit; and an initial try at 20,000 feet would seem to be a reasonable compromise, based upon the following facts —

- First, even if you get a good JFS start somewhere above 20,000 feet, the JFS may stay at idle due to insufficient control pressure to reset the fuel control. This will probably limit the engaged cranking speed to about 3%. Furthermore, there is probably inadequate JFS/Central Gear Box lubrication and cooling in this high altitude case, but you won't be in the area long enough to create any new problems. Finally, the JFS may have insufficient torque up there to provide adequate spool-up anyway. As you descend, the JFS fuel control will be able to reset to 100% speed, the torque will progressively increase, and the engine will spool-up accordingly.
- If the initial JFS start attempt is not successful, you'll have to give up 5000 feet or more waiting for the accumulator to recharge, based upon a nominal recharge time of 1-1/2 minutes. Since you don't know the status of the accumulator unless it's full up (no JFS Low Light) or totally depleted (handle just pulled), your best bet for a good light-off is to be inside the envelope, and then if the light-off is missed, wait out the recharge time. By trying the first start around 20,000 feet, you have a good chance of being ready for another first-bottle try around 15,000 feet, and a first and second bottle try around 10,000 feet.
- At very low altitudes, when you are well inside the envelope and Mother Earth is becoming a distraction, you could find yourself with an in-between situation: it's now or never for the JFS, but the JFS Low Light is still on. My recommendation here is to try the first bottle. If the partial charge was insufficient, which you'll know in 10 seconds ($\approx$500 feet), you have at least cleared the first bottle and can proceed to the second without fear of over-accelerating the JFS. While you're counting out the 10 second start window, why not peek at the JFS switch to make sure some gremlin hasn't turned it off. Personally, I fly with the switch on at all times.

We've talked about the altitude considerations of JFS operation; now let's look briefly at airspeed. You'll notice from Figures 1a and 1b that at 20,000 feet the 350 knot glide is inside the demonstrated envelope (for a new, prime condition JFS) but just outside the recommended envelope. Therefore, don't rush the JFS — let your transition to the 220 knot profile bring you toward the heart of the JFS speed envelope before the first handle pull.

BY THE WAY . . .

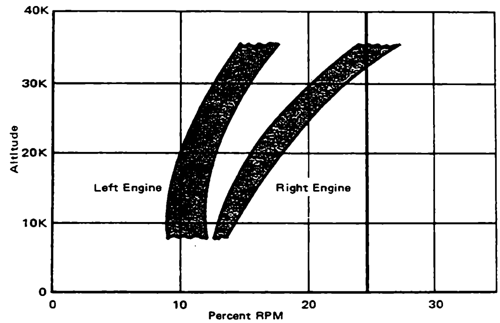
One final thought concerning the early spooldown airstart you may have had a chance to try on the way to the JFS envelope; if you were able to give it an honest 10-15 second try from the time the throttle goes to idle or above (and RPM $\approx$ 12%), with no apparent light-off, chances are it's down for the count. In this special case, I feel the best bet would be to go for a JFS assisted start on the backup engine (the one in stagnation) once the JFS is giving the green light. Here's the real moment of truth: you'll have to shut down that hydraulic power source — the stagnated engine — on faith that the JFS will engage and motor the engine of your choice. I think that's a reasonable assumption. Remember also that any time during a JFS assisted restart you need to discontinue cranking one engine in order to crank the other, just turn off that engine master switch to disengage the JFS. Of course, all of this has assumed a good Utility B circuit; you can quickly see the risks generated by ignoring a Utility B Light.

BACK TO BASICS

Without inflight JFS, of course, you've got no choice but to go for a high speed dive (350 knots minimum) to prevent the first airstart candidate from winding down below 12%. I have seen successful airstarts with as low as 9% indicated, but there is no guarantee down there. While you're screaming downhill with both engines out, having a hard time believing this is really happening to you, it's important to have an understanding of the utility hydraulic load-sharing between the engines. As you know, the L/H engine has the higher utility pressure output, as can be seen during start-up. If the L/H engine is shut down with the R/H running, you may notice an approximate switchover point as the L/H winds down, but don't be misled. In actual practice, the major utility load is progressively passed from left engine to right engine as the L/H spools down below approximately 20%. In the range between 20% and somewhere below minimum useable RPM (10%-12%), the utility load is shared by the two engines. So how does this affect your recovery technique?

First, the R/H engine spooldown rate is lower than the L/H for any given flight condition, and the difference is particularly noticeable in the airstart RPM range. This equates to more cool-down time, more time to accel the aircraft if approaching the restart from the slow side, and possibly lower torque required to achieve spool-up. Thus, the handbook guid-

FIGURE 3 — WINDMILL RPM AT 350 KCAS



ance to go for the R/H restart first, other factors being equal. (If the R/H stagnation is significantly hotter, we say then to attempt the L/H start first — an apparent contradiction. The assumption in that case is that the lower FTIT stagnation has the lower probability of internal damage, therefore the higher probability of restart... if you have to sacrifice one engine to save the airplane, so be it.)

Second, at approximately 13%-14% windmill RPM on the L/H, and the R/H at this RPM or higher, the utility load is being shared almost equally. So, if your glide speed is sufficient to maintain the L/H in windmill at or above this RPM, you no longer need to keep the R/H in stagnation; you may as well shut it down and let it start cooling off for its restart. Since both engines are supporting their own PC system, plus half the Utility, the R/H should then spool down no lower than the L/H. A possible key point to remember: a reasonably steady glide has been assumed, i.e., no large and/or rapid control inputs at this point. These RPMs are rock bottom for maintaining system pressures and normal flow demands; increased control inputs at this point will probably result in pressure drops and, therefore, slower than normal control surface movement. Also, as you descend to denser atmosphere, it will take slightly increasing airspeed to support a given windmill RPM, as shown in Figure 3.

Third, be aware that either engine alone must be windmilling at 20% or higher to totally support the flow demands of the Utility system. This is directly applicable to the R/H windmill, L/H in stagnation scenario. If the R/H has stabilized at or near this RPM, I would not glide along keeping the L/H cooking while waiting for the R/H to relight.

The big plus I see in all this knowledge is that it may give you a shot at starting both engines somewhat simultaneously, rather than hanging your hat on only one, possibly failed, engine.

I'm the first to admit that the situation can become painfully complex, and I hope that this article provides you with some insight to the possible solutions. The only real answer I see is a thorough understanding of the systems, combined with practice. If a realistic dual engine failure scenario is not available at your local training simulator, the only viable alternative is mental rehearsal — when you draw the short straw, you've got to be ready.

Until they manage to build the perfect airplane and engine, remember that you don't pick the failures — the emergency finds you. If you take the bird too much for granted, even the world's easiest flying and most forgiving airplane may find you unprepared.

COMING SOON

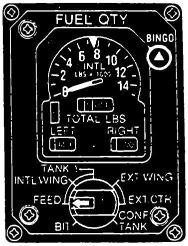
On the subject of airstarts, both Pratt and Whitney and McDonnell are continually studying the problem; and some improved guidance will reach you early next year via Revision D to the Dash One. In the meantime, the new theories are being verified by flight test, so are still subject to minor revisions. The sort of things we're looking at are the optimum and minimum RPM's at which to attempt a spooldown start; the maximum recommended FTIT at restart initiation (presently 500°C); and the best throttle position for stall free and rapid thrust recovery.

ANOTHER SITUATION

Since the following use of the JFS didn't really fit the scenario above, I've included it as a footnote. In some single engine emergency situations, you may want to utilize the JFS as a backup source of hydraulic power, assuming the dead engine will turn but not start. If the good engine generator and utility pump were inoperative — admittedly a remote failure combination — this could also return some electricity via the dead engine utility pump and the emergency generator. You folks have probably already thought about these back-up uses, but I wanted to pass along a caution. Extended use, either at idle or motoring an engine, and particularly at high altitude, could result in damage to, and loss of, the JFS. Therefore, it is advisable to use it sparingly, such as during final approach and landing only.

JFS OPERATION vs AN/ALQ 119 POD

Under present guidance, the AN/ALQ 119 pod is carried on the centerline station and jettison is prohibited, in fact jettison cartridges are not even installed. Until such time as this restriction is removed, the presence of the pod will lower the JFS envelope to < 200 knots at 20,000 feet and < 300 knots at 15,000 feet, as indicated in figure 1a. ■



(PUBLISHED 1984)

F-15 EAGLE

JP-4 is the primary fuel used in the F-15 F100-PW (Pratt & Whitney) engines. JP-5, JP-8, NATO F-43, or commercial JET A-1 and JET-B can be used as alternate fuels, which can be intermixed in any proportion with the primary fuel. However, before filling the Eagle with any alternate fuel, make sure you know its "additive content." Depending upon suppliers, alternate fuels may or may not contain corrosion and ice inhibitors (corrosion inhibitors provide added lubricity to the fuel). Operation of the F100 engine without a fuel system icing inhibitor is restricted to one flight. Operation without a corrosion inhibitor is restricted to ten consecutive hours. An alternate fuel may have a slower vaporizing rate than JP-4, and small leaks that might not be obvious when using JP-4 could show up when using an alternate fuel.

When using an alternate fuel it is not necessary to retrim the engine, and throttle handling limitations are the same as when operating with the primary fuel. Also, there is no significant change in engine operations. However, ground starts with a temperature below 6°F (−14°C) may produce more smoke and require a longer time for engine light-off. Ground starts should not be attempted with fuel temperature below −40°F (−40°C). Hot starts may occur during spooldown airstarts at airspeeds less than 350 knots at altitudes above 30,000 feet. Because freeze points vary according to the alternate fuel used, fuel in external tanks may not transfer after sustained operation (5 minutes or longer) below 200 knots above 25,000 feet or 250 knots above 45,000 feet.

At the present time there is no approved emergency fuel for the F100 engine. However, two fuels - JP-7 and JET-A - have been qualified by ground tests, and flight testing has been proposed. If successful, the engine manufacturer will propose changes to the flight manual to allow one flight with no postflight action required.

(PUBLISHED 1978)

MORE ON BALD EAGLES

By DAN DRAPP/*Senior Engineer - Design*

Well, we've been out conducting some more investigations of Eagle canopy incidents this past month and have come up with what we hope are some more helpful suggestions. (We promise to run out of suggestions just as soon as you run out of incidents!) Let's look at two of the most recent situations —

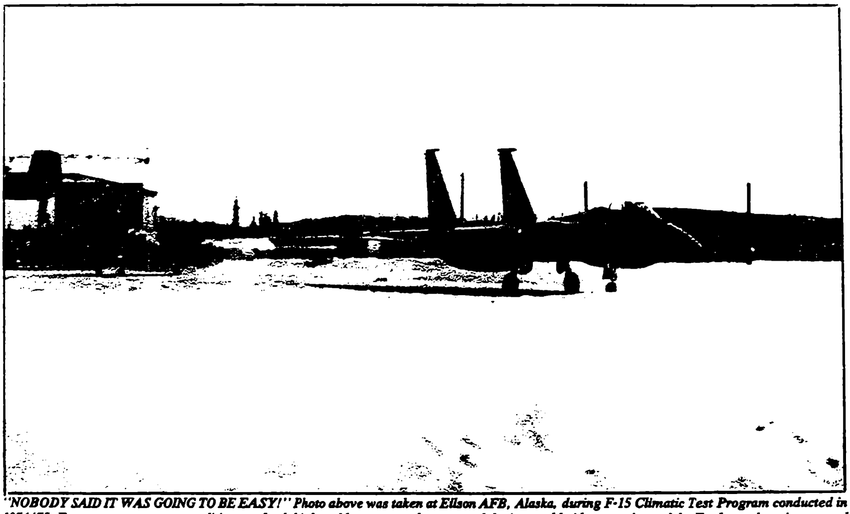
- An F-15A departed for a cross-country flight with a compacted load of cargo in Equipment Bay 5. The pilot noticed the canopy moving back just after liftoff and by the time he reached 180 knots it was gone. After dumping fuel he made an uneventful landing.
- Three days later, a pilot closed the canopy on another F-15A and attempted to taxi, but was unable to see due to frost on the windshield and canopy. He had to open the canopy back up to taxi and close it before takeoff. Not too long after takeoff, the canopy moved back .50-.75 inch. The pilot aborted the mission and returned to base.

What the investigations revealed —

- The first canopy loss was caused by cargo crammed into Equipment Bay 5, which activated the canopy jettison mechanism. There are no position switches on this part of the mechanism; hence no canopy unlocked light to warn the pilot that the linkage had been tripped.
- The near canopy loss on the second Eagle could not be duplicated on the ground. The canopy and locking mechanism were rig checked, and although minor discrepancies existed, no contributing factors were found. Both manual and hydraulic operation of the system were attempted and the system worked normally. Try as we may, it was not possible to defeat the interlock or warning light systems. The only conclusion possible is that the mechanism was not overcenter when the pilot selected lock and took off. The question remains whether or not he had a canopy unlocked light. The canopy lock position switch is now being analyzed for possible intermittent failure.

Our Suggestions —

- Extreme care should be taken if Equipment Bay 5 must be used as a cargo hold. The bay was not designed to carry uncontained cargo, and exposed areas where damage can occur are numerous:
  1. Relays, circuit breakers, and electronic equipment on the side panels.
  2. Control cables, control rods, hydraulic lines, and air supply lines routed across the floor.
  3. Canopy jettison mechanism on the under side of the canopy deck.
  4. Canopy mechanism on the forward and aft bulkheads.
- When closing the F-15A canopy,



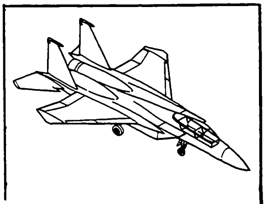
"NOBODY SAID IT WAS GOING TO BE EASY!" Photo above was taken at Eilson AFB, Alaska, during F-15 Climatic Test Program conducted in 1974/75. Extreme temperature conditions — both high and low — must be expected during worldwide operations of the Eagle, and environmental impact upon both airplane and people can be severe. However, these test programs showed that the F-15 can "take it" if you can, which means that such problems as frosted windshields and icy surfaces (not to mention sand, dirt, heat, and moisture) must be met and overcome to keep the force operational and safe.

moving the handle from down to lock shuts off hydraulic power to the locking mechanism at the selector valve; rotates an interlock into place; and opens the canopy lock position switch to turn out the canopy unlocked light. It is very important to observe what the —1 has written concerning this sequence:

1. *"Ensure canopy has completed movement and wait 10 seconds before moving handle to LOCKED position."* This allows the hydraulics sufficient time to lock the mechanism over-center.
2. *"Ensure canopy unlocked light is on with handle in DOWN and goes out with handle in LOCKED."* This checks the canopy lock position switch and interlock for proper function.

• Ground crews can help pilots avoid the possibility of mistakes caused by breaks in a normal routine. Anytime one is forced to do things out of the ordinary or to alter a normal pattern, he may not get back into the pattern right away. It's possible that the pilot in the second incident may have become so accustomed to the red light being on (it was illuminated the whole time he was taxiing) that he missed its warning on takeoff. If the frost had been cleared from his windshield and canopy, he would not have had to open the canopy to see; would not have had to alter his normal takeoff routine; would probably not have had to incur a mission abort.

We hope you can put this information to use; any one of the items discussed can rise up to bite the F-15 pilot. Let's see if we can't stamp out "Bald Eagles." ■



In the F-15B, it's the aft cockpit area and is mostly occupied by an ejection seat. In the F-15A, there is no seat and the area is identified as "Equipment Bay No. 5." While it may appear to be an inviting storage area, many serious problems can occur if uncontained and unsecured cargo is casually stuffed in this space.

The Breath of Life

(PUBLISHED 1975)

The oxygen system in the Eagle or Phantom is your "breath of life." Without it, your mission profile would be severely limited. While this limit could be due to a system malfunction over which you have no control, it could also be due to a failure to preflight the system correctly, over which you do have control!

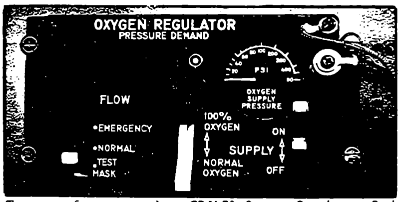
BIT (Built-In Test) is now a way of life in oxygen system analysis, as in so many other aircraft systems. The CRU-73 oxygen regulator has a BIT that will allow you to test the system for correct operation, and while we've referred to this capability in past articles, we want to make sure you are aware of and using the test.

Yes, "P-R-I-C-E" is a part of it, and in every visit to the altitude chamber you are reminded that PRICE stands for:

P - Pressure gage or quantity gage check to determine if adequate supply of oxygen is aboard aircraft.
R - Regulator - Check regulator supply switch is ON. Perform "blow back" check; pressure check mask for fit.
I - Indicator (Blinker)-Check for flow indication.
C - Connectors - Check all connections.
E - Emergency - Check emergency cylinder pressure and connections.

However, the BIT capability of the CRU-73 can be utilized *before* you make a PRICE check. A complete oxygen system preflight should begin by connecting the CRU-60/P and aircraft delivery hose to the mask. Then with the regulator in the "Off" position, test for leaks in the delivery system hoses and tubing by attempting to inhale. You should not be able to, first because there is no oxygen turned on and second, because the oxygen mixture toggle should have been automatically selected to the 100% position whenever the regulator is OFF. Internal valving isolates ambient air "normal oxygen" input and the crewmember cannot get a breath. This is your "BIT" check - if breathing is possible, a leak in the delivery system is evident and corrective steps are indicated prior to flight. The cause of the leak may be your mask, the oxygen delivery hose, or the regulator itself.

During production acceptance testing, we have determined malfunctions often enough using this technique that we thought it may be helpful to you in preventing a case of hypoxia. Remember, "BIT" now applies to the oxygen system, too; use it and assure yourself that "breath of life!"



Two manufacturers produce CRU-73 Oxygen Regulators. Both offer same BIT capability, but accomplish it in slightly different ways. With Bendix regulator (shown here), only a portion of OXYGEN mixture toggle moves to 100% oxygen when you turn SUPPLY toggle OFF. With ARO regulator, entire toggle moves. Incidentally, you can identify CRU-73 Regulator through the two test ports on either side of the pressure gage (CRU-68 does not have test ports, and SUPPLY toggle is safety-wired to ON position).

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