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ANALYSIS OF THE EFFECTS OF REMOVING NOSE BALLAST FROM THE F-15 EAGLE

THESIS

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THESIS

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Preface

The research in this thesis investigated the feasibility of removing lead ballast from the nose section of the F-15 Eagle. This research was pursued to provide the engineering background needed to verify that the ballast could indeed be safely removed, with the goal of improved aircraft nose authority being the primary objective.

In performing the analysis and writing this thesis I have had a great deal of help from others. I am indebted to my thesis advisor, Major Curtis Mracek, for supporting my needs in an unusual thesis topic. I also wish to thank Mr. Jeff Preim from the F-15 SPO for his help in locating current F-15 center of gravity data, and Mr. Dave Potts from ASD/SCES for his immense help in running the computer simulations. Finally, I wish to thank my wife Laura for putting up with my hectic schedule these past 18 months and for tending to all of the needs of our happy family.

Richard L. Bennett

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Abstract

This study investigated the results of removing lead ballast from the nose section of the F-15 Air Superiority fighter. The goal of the investigation was to determine if aircraft handling qualities remained acceptable with the ballast removed, and also to determine what improvements in aircraft nose pointing authority resulted. Actual F-15 weight reports were used to calculate the worst case aft center of gravity location shift due to the ballast removal. Several configurations with different center of gravity locations (based on various amounts of lead weights removed) were used for comparison to the baseline aircraft. Moments of inertia were calculated for each configuration, which in turn were used in a 6 degree of freedom computer simulation of the F-15. Simulation test points were then examined throughout the flight envelope of the F-15. Simulation results and better aircraft weight management results support removing (on average) approximately 200 pounds of lead ballast from the nose section of the single seat Air Superiority F-15 Eagle, with a resulting 3 percent increase in pitch rate. A suggested flight test profile is presented for flight verification of the simulation results.

ANALYSIS OF THE EFFECTS OF REMOVING NOSE BALLAST FROM THE F-15 EAGLE

I. Introduction

During this time of shrinking defense budget dollars, the need to be able to do more with existing hardware is quickly becoming a requirement for maintaining our national defense. The potential to make a small improvement in performance in the Air Superiority F-15 Eagle exists through a minor hardware change that will cost virtually nothing: namely, removal of lead ballast from the nose section of the aircraft. Originally placed in the aircraft to maintain strict center of gravity (c.g.) location requirements, a large portion of the lead ballast is no longer needed due to avionics updates in the F-15 since the aircraft was fielded.

In addition to these changes in aircraft weight distribution, due to newly incorporated avionics, this thesis looks into the effects of loosening the established c.g. requirements for the F-15 with the goal of possibly eliminating even more of the lead ballast. These effects will primarily be measured against changes in aircraft handling qualities that will result from the shift in c.g. location. Removal of the lead ballast will not only reduce overall aircraft weight, but it should also enhance the nose pointing authority of the aircraft at airspeeds below corner velocity (the highest velocity at which full aft stick will just reach placarded g limits), a significant tactical advantage when employing current technology 'point-and-shoot' close range missiles.

As with most engineering changes in high performance (tightly-designed) aircraft, tradeoffs will occur in either capabilities or performance. In this particular design change, the major tradeoff due to an aft c.g. shift will occur in the area of aircraft stability versus maneuverability. The more aft the c.g., the more maneuverable the fighter will be (theoretically). However, the aircraft will also be less stable since the static margin will be smaller. As long as the c.g. shift is not significant (in classical aircraft with positive static stability, at least), the static margin will remain positive with the result being a most noticeable effect in fine tracking aircraft handling qualities instead of gross acquisition handling qualities problems or even loss of positive aircraft static stability. With that premise in mind, the analysis in this thesis starts by looking at small incremental changes in c.g. location, beginning with the nominal aircraft c.g. location for the worst case air superiority configuration (external wing tanks and pylons, a centerline pylon, 4 Aim-7's loaded, 1100 pounds of fuel remaining, and expended 20 millimeter ammunition) and incrementally removes the lead ballast until all the ballast has been removed (6:12). Each incremental change was examined in a 6 degree of freedom F-15 simulation and compared with the baseline aircraft for differences in both fine tracking tasks (low g-command step inputs) and gross acquisition maneuvers (high g-command step inputs) throughout the entire F-15 flight envelope. Military Standard 1797A, Flying Qualities of Piloted Aircraft, was used as the primary source for evaluating the changes in flying qualities between the different c.g. locations (4).

Although the simulation should prove to be a useful tool in identifying potential limitations and problem areas, the results of this study will obviously not be complete and ready for release to the F-15 fleet for possible incorporation until the results have been verified through actual F-15 flight testing. Although the flight test portion is beyond the scope of this thesis, a recommended flight test profile is included as a final chapter to this study that will summarize potential problem areas that need to be examined.

II. Center of Gravity Calculations

Analysis of the Air Superiority F-15 c.g. location took place in two parts. The first part examined actual weight reports of A and C models of the F-15 to verify that current c.g. calculations were based on current avionics packages as well as determining average amounts of ballast in the various model aircraft. The second part took the weights of a generic F-15C model aircraft in the critical aft c.g. configuration and calculated how much further aft the c.g. shifted as the lead ballast was removed.

Updated Avionics Effects

The F-15 has been in the active inventory since the early 1970's, and has had many avionics changes that affect c.g. location. Since the first block of aircraft was delivered, modifications to the basic airframe (more internal fuel in C and D models, for example) and avionics packages have drastically changed the aircrafts mass distribution and c.g. location. Unfortunately, the reference point with which the critical aft c.g. balance calculation for the F-15 is calculated has not kept pace with the changing avionics configurations that are actually being flown in the field. The most significant changes in avionics that have occurred in the F-15 fleet have been the incorporation of an internal countermeasures set (ICS) at a weight addition of 323 pounds, and the addition of a signal data recorder (SDR) at a weight addition of 42 pounds (6:11).

When the first F-15A was fielded, neither the ICS nor the SDR were planned additions to the aircraft. Hence, the critical aft c.g. balance calculations did not reflect the equipment. When the F-15C model came out, both the ICS and SDR were planned in the aircraft, but due to shortages of the equipment, not all of the C models were delivered from the factory with the black boxes. Therefore, the worst case aft c.g. would still occur and be flown in the aircraft that did not have the equipment yet delivered. Several years went by before the entire F-15C fleet all had the ICS and SDR on board, and it would be several more years before the F-15A's would be depot modified and equipped with the ICS and SDR. (The modification may still be on-going for the F-15A fleet.) Unfortunately, and for unknown reasons, the incorporation of this new heavy equipment located forward of the aircraft c.g. (the SDR in the aircraft mid-section, the ICS located in bay 5 behind the ejection seat) was never included in the critical aft c.g. balance calculation.

The specific effects of including the ICS and the SDR equipment in the F-15A/C were surprising. (The F-15B/D are the two seat trainer variants of the F-15A/C. Since the aft seat compartment is located in bay 5, it precludes installation of the ICS. This study concentrated on the combat version of the Air Superiority F-15, the single seat F-15A/C.) Table 1 shows the center of gravity limits for both the F-15A/C (6:1). Appendix A shows the aircraft reference datum for the F-15 and provides the conversion equations for calculating the c.g. in percent mean aerodynamic chord. (% MAC)

TABLE 1

Center of Gravity Limits in % MAC

Forward Limits	Gear Up	Gear Down
Without Wing Pylons		22.0
With Wing Pylons		23.0
Aft Limits		
Without Wing Pylons	29.9	
With Inboard Wing Pylons	29.0	
With Outboard Wing Pylons	29.4	

For the critical aft c.g. balance calculation, which includes external wing tanks loaded on inboard wing pylons, Table 1 shows that the allowable c.g. range is between 23.0 and 29.0 % MAC. For example, F-15A serial number 74-094, a block 11 A model, has a c.g. location at 28.3 % MAC in the critical aft c.g. configuration (5:15). The aircraft also has 356 pounds of lead ballast located between fuselage stations 208 and 228 (5:4). By removing 108 pounds of ballast located between fuselage stations 219 and 227, the c.g. location shifts back to 28.95 % MAC and is just within allowable limits. (See Appendix A) By including the ICS and SDR in the critical aft c.g. balance calculation, an additional 232 pounds of lead ballast can be removed to bring the c.g. location back to 29.0 % MAC. (See Appendix A)

This removes a total of 340 pounds of lead ballast while remaining within allowable limits, leaving only 16 pounds of lead ballast in the nose of the aircraft. Although this is a specific aircraft and similar calculations would have to be accomplished for each and every F-15 in the fleet, it can be considered representative for most block 11 aircraft. Table 2 on the next page shows a representative cross-section of various F-15A/C blocks of aircraft, including current ballast loads and c.g. locations, how much ballast can be removed to bring the aft c.g. limit right to 29.0 % MAC, and how much additional ballast could be removed if the ICS and SDR were included in the critical aft c.g. balance calculation.

TABLE 2

Effects of Including ICS and SDR Equipment in Critical Aft C.G. Calculations (Gear Up)

Model-Block	Current Ballast (Lbs)	Current C.G. (% MAC)	Ballast Removed (Lbs) To Reach 29% MAC	Ballast Removed (Lbs) With ICS/SDR For 29% MAC	Remaining Ballast (Lbs)
F15A-7	476	27.6	240	445	31
F15A-8	434	27.5	245	434	0
F15A-11	314	28.3	112	314	0
F15A-17	277	28.8	38	239	38
F15C-24	243	28.9	18	225	18

Table 2 shows that including the ICS and SDR equipment in the critical aft c.g. balance calculation substantially reduces the amount of lead nose ballast required in each aircraft to maintain established c.g. limits. Currently, the USAF does not include this equipment in the critical aft c.g. balance calculation. (Current as of 13 September, 1991, per a telephone conversation with the 1st TFW/QA office, Langley AFB, Va.) Operational maintenance effects will be discussed in the conclusions and recommendations section of this report.

C.G. Shift Due to Lead Ballast Removal

The second part of the c.g. analysis did not include the effects of the new avionics equipment. This section took the mass distribution of a generic F-15C from a

McDonnell Douglas Mass and Inertia report and calculated the effects on the c.g. location and moments/products of inertia of the aircraft as the lead ballast was incrementally removed (7:3.13). The new c.g. location and the new inertia information were then used in the computer simulation work, as explained in section III of this report. Appendix B shows the tabulation of the moments and products of inertia of the generic F-15C. The baseline aircraft is in the critical aft c.g. configuration, with external wing tanks and 3 external pylons, 4 Aim-7's, 1100 pounds of fuel remaining and expended 20 millimeter ammunition casings. Table 3 provides a summary of the various configurations used by the computer simulation.

TABLE 3

Summary of C.G. Configurations Used in Simulation (Gear Up)

	C.G. (%MAC)	Ballast Removed (Lbs)	Ix	Iy	Iz	Ixz	Aircraft Weight (Lbs)
Base (B)*	28.7	-	27209	165597	187520	-1208	33506
CNF1 (1)	29.0	54	27201	164695	186631	-1161	33452
CNF2 (2)	29.5	144	27184	161859	183811	-1217	33362
CNF3 (3)	30.0	240	27166	160918	182889	-1216	33266
CNF4 (4)	29.2	98	27217	164052	185972	-1115	33408
CNF5 (5)	29.4	133	27186	162837	184787	-1146	33373

* Letter/numbers in parenthesis correspond to the configurations labeled on simulation result graphs given in Appendix D.
All moments/products of inertia are in units of slug-ft^2.

The baseline aircraft has a critical aft c.g. location at 28.7% MAC. The lead ballast, with an average fuselage station at 211 inches, was removed in increments so that the c.g. would fall at 29.0, 29.5, and 30.0% MAC (configurations 1, 2, and 3, respectively) for analysis purposes. As the focus narrowed during the analysis, configurations 4 and 5 were added at 29.2 and 29.4% MAC, respectively.

III. Computer Simulation

A 6 degree of freedom F-15E computer simulation was used to examine the effects of moving the aircraft c.g. aft. A more detailed description of the simulation program is given in Appendix C. Although the simulation is currently designed around the F-15E, the basic program evolved from use in the Air Superiority F-15 flight test program. Response of flight control characteristics are virtually identical in the areas that were examined in this thesis between the F-15C and the F-15E. The external aerodynamic configuration was the major difference between the two models. However, the program was designed to allow changes in the external aerodynamic model. For all of the simulation runs in this report, the aero model included the single seat canopy design, no LANTIRN pods or conformal fuel tanks (CFT's), external pylons on aircraft stations 2, 5, and 8, wing tanks on stations 2 and 8, and 4 Aim-7's on fuselage stations 3, 4, 6 and 7. The single seat canopy and lack of LANTIRN pods and CFT's turned the F-15E into an F-15C aero model. The remaining external hardware put the F-15C into the proper critical aft c.g. configuration from an aero modeling and drag count standpoint for the testing. The remaining critical aft c.g. requirements (1100 pounds of fuel remaining and spent ammunition casings) were accounted for in the c.g. and inertia moments calculations, as shown in Appendix B.

The major limitation of this simulation from an analysis standpoint was that the individual test runs had to start from a trimmable aircraft condition. In other words, the aircraft had to start the run from a straight and wings level position with enough flying airspeed such that engine thrust capabilities could sustain the initial conditions. Although a minor limitation for most considerations, this requirement, nonetheless, prevented any analysis from being done in examining reduced nose down pitching authority due to the aft c.g. location in extremely high angle of attack (AOA) or spin conditions of flight. Another limitation of this requirement was that as maneuvers were accomplished and thrust could not be changed from the unloaded trim settings, airspeed effects occurred as the aircraft slowed down in the maneuver.

With these limitations in mind, the following test plan was formulated for accomplishing the computer simulation analysis. Analyzing the effects of a change in c.g. location on an aircraft are very similar to analyzing the effects of adding a store or munition to the airplane. The only real difference is that external drag or external aerodynamic effects are not a factor for this c.g. shift since all changes are internal to the aircraft. Therefore, the store certification test process, as outlined in the USAF Test Pilot School curriculum, was used as a guide to accomplish this analysis in support of the follow-on flight test work (1:4.75). Obviously, any aspects dealing with interference drag or other non-applicable external aerodynamic effects as outlined in the store certification process were neglected.

The store certification process starts by picking a test point in the heart of the aircraft flight envelope and working from there out to the corners of the envelope. For our F-15C configuration, the aircraft speed was limited, since external tanks were loaded, to 660 knots calibrated airspeed (KCAS) or Mach 1.5, whichever is lower (3:5-15). The central starting point was chosen to be trim conditions for 20,000 feet pressure altitude (PA) and 300 KCAS. From there, the aircraft envelope was sampled at 1,000 feet PA and 180 KCAS, then at 1,000 feet PA and 660 KCAS, then at 45,000 feet PA and Mach 1.5, and then the analysis concentrated on a small transonic area starting at 15,000 feet PA and .85 Mach and sampled results at various altitudes and Mach numbers around that point. (The F-15 dash 1 points toward this area as being an F-15 pitch sensitive area (3:6-1).) As noted, each test point was started from trimmed conditions. Then, at 1 second elapsed time, a step input was made for a low 'g' commanded input (fine tracking task simulation) at the baseline c.g. configuration of 28.7% MAC. The test point was then repeated at 29.0, 29.5, and 30.0% MAC c.g. locations (Configurations 1, 2, and 3, respectively) and the results plotted on the same graph for direct comparison to baseline. Typical plots were either pitch rate or normal 'g' versus time. Next, all 4 configurations were run again with a high 'g' (approximately 7) commanded step input to simulate a gross acquisition maneuver. And finally, both the low and high 'g' step inputs were repeated for all 4 configurations with the F-15 control augmentation system (CAS) turned off. Turning the CAS off is the worst case flight control situation for the aircraft. Therefore, all 4 c.g. configurations were examined at each test point under 4 different test conditions for a total of 16 computer runs per test point.

The next part of the simulation work examined the transonic region between .7 and 1.0 Mach in the flight envelope where aircraft pitch authority may be sensitive to abrupt changes (3:6-2). Altitudes were varied between 5,000 and 30,000 feet PA and Mach numbers between .6 and .95 for the analysis. Airspeeds above .95 Mach were not attainable at the lower altitudes due to thrust limitations. The results and analysis section of this report will discuss the various conditions that were examined and their rational for selection.

Finally, the computer simulation work looked at potential improvements in nose authority due to the aft shift in c.g. location. At airspeeds above corner velocity, the aircraft is limited in pitch rate by the aircraft placarded 'g' limits and will not benefit from any c.g. shift. Therefore, improvements in pitch rate were analyzed at airspeeds below corner velocity.

IV. Results and Analysis

As stated in the previous section, the initial computer simulation analysis started in the critical aft c.g. configuration at 20,000 feet PA and 300 KCAS and then expanded to the corners of the flight envelope. For this initial portion of the analysis, 4 different c.g. locations were analyzed at each test point and included c.g. positions of 28.7, 29.0, 29.5, and 30.0% MAC. 28.7% MAC was the nominal c.g. of the baseline aircraft before removing any ballast. 29.0% MAC is the current allowable aft c.g. limit for the F-15A/C. Removing all of the 243 pounds of lead ballast in the nominal F-15C aircraft placed the c.g. location at 30.0% MAC, and 29.5% MAC was selected as an intermediate test point between the allowable and maximum aft location. The graphical data presented from this portion of the analysis and located in Appendix D is typically labeled with a "B" on the plot for the baseline configuration of 28.7% MAC, and a "3" on the plot refers to the 30.0% MAC configuration. 29.0 and 29.5% MAC results were always in increasing order of amplitude between "B" and "3" on the graphs. Several measurements were taken from the graphs, including the calculation of short period damping for each c.g. location, as shown in Figure 1.

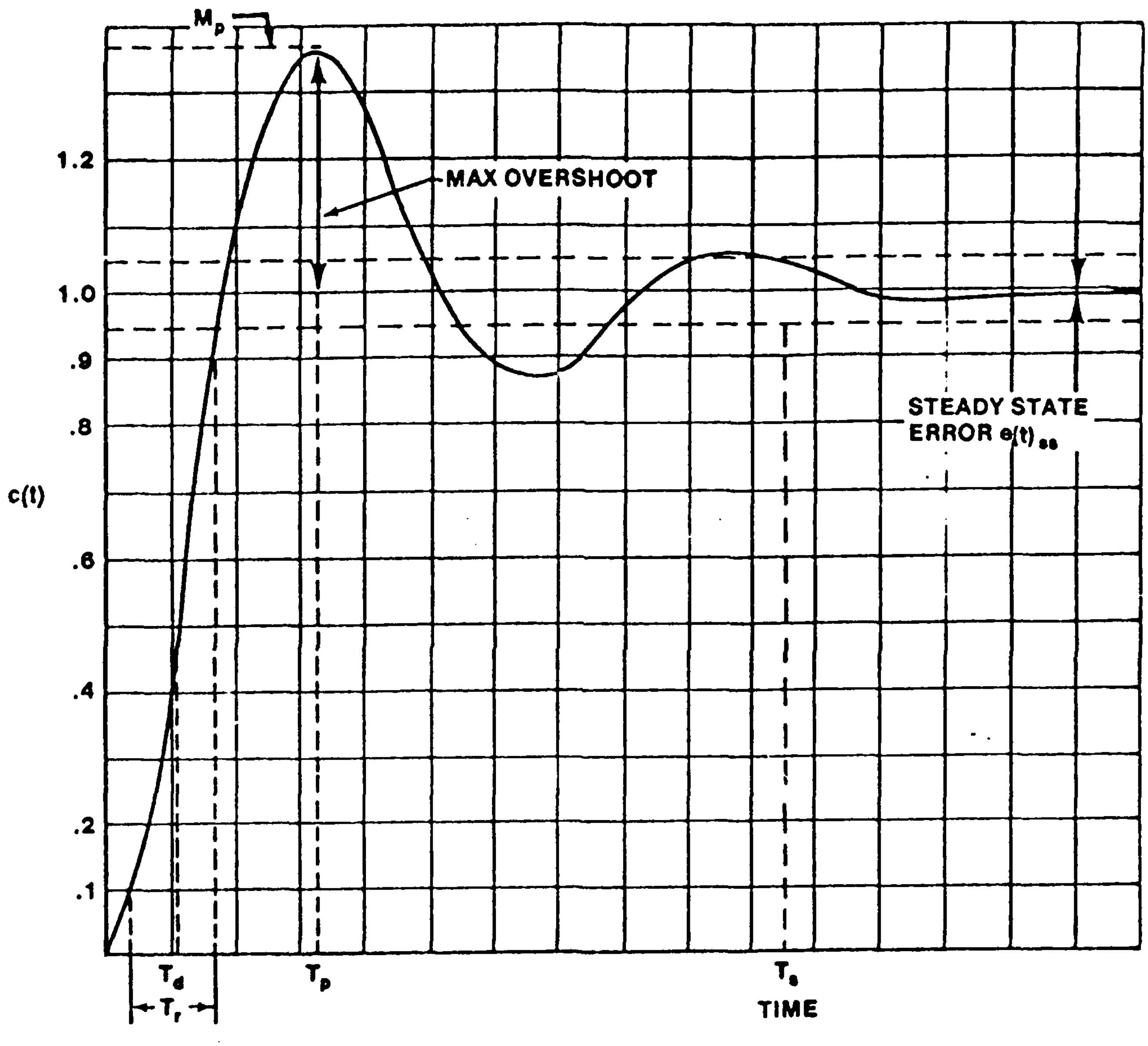


Figure 1. Time Domain Specifications (2:13.22)

In order to find the damping ratio, the maximum peak overshoot, Mp, must first be converted to ratio form with respect to the steady state response value to the step input, as shown in Figure 1. Then, the damping ratio, ζ, can be calculated from the equation:

$$Mp = 1 + \exp(-\zeta/(1-\zeta)^{1/2}) \qquad (1)$$

by solving iteratively for ζ.

A limitation of this measurement is that it assumes a second order (or equivalent second order) system response. Very few of the test points demonstrated second order equivalent responses. And of the few test points that did demonstrate adequate second order responses, a majority of those points fell prey to simulation limitations due to loss of airspeed from trimmed conditions during the maneuver, in which a steady-state response could not be identified. (As airspeed bled off during the constant step input, the pitch rate would steadily increase, preventing identification of the nominal steady-state response value.) Table 4 on the next page provides a summary of the damping ratios that could be calculated from the computer simulations. As can be noted on the table, all of the points are high energy (high airspeed) test conditions, which were not affected as much by the maneuver, and generated a fairly constant steady-state response.

TABLE 4

Changes in Damping Ratio Due to C.G. Shift

Test Point*	Damping Ratio at: 28.7%MAC	29.0%MAC	29.5%MAC	30.0%MAC
(1) 1000/660/5/ON	.43	.43	.43	.43
(2) 1000/660/20/ON	.33	.34	.35	.37
(3) 15000/465/4/OFF	.37	.41	.49	-

*Parameters in PA(feet)/airspeed(KCAS)/step input force (Lbs)/CAS on or off

Figures 3, 4, and 5 in Appendix D correspond to test points 1, 2, and 3 in Table 4. Test point 3 did not accomplish the maneuver at 30.0% MAC c.g. location, because it had been determined (as will be discussed later in this section) that this c.g. location was unstable. (This particular test point was accomplished later during the analysis in comparison to the first 2 test points listed in Table 4.) Because the measurement of damping ratio was so limited, it was not a major consideration for analysis purposes in this report.

Another measurement used for comparison of the c.g. locations is outlined on pages 217 and 218 of MIL-STD-1797A (4:217). This measurement is specifically designed to examine pitch rate responses to step inputs for both fine tracking and gross acquisition maneuvers. It accomplishes this by examining the transient peak ratio, equivalent time delay, and effective rise time as defined in Figure 2 on the next page.

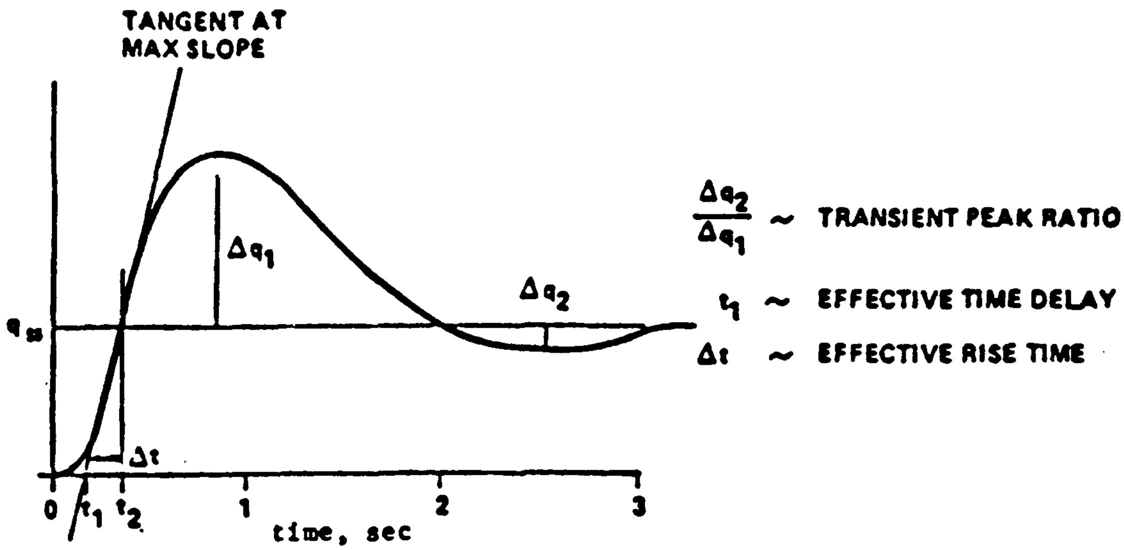


Figure 2. Pitch Rate Response Measurements from MIL-STD-1797A (4:218)

Table 5 lists the requirements for these parameters to meet specified flying qualities levels as outlined in MIL-STD-1797A.

TABLE 5

Flying Qualities Requirements (4:218)

Level	Equivalent Time Delay	Transient Peak Ratio	Rise Time *
1	≤ .12 sec	≤ .30	9/V ≤ t ≤ 500/V
2	≤ .17 sec	≤ .60	3.2/V ≤ t ≤ 1600/V
3	≤ .21 sec	≤ .85	N/A

*Nonterminal flight phases, V is true airspeed in ft/sec.

This portion of the analysis examined the corners and center of the F-15 flight envelope looking for degradations in flying qualities levels based on the measurements of the transient peak ratio, equivalent time delay, and effective rise time.

20,000 Feet PA, 300 KCAS

No significant findings or degradations were discovered at the center of the envelope test point with either the CAS on or off at both low and high 'g' commanded step inputs. Transient peak ratio and equivalent time delay were found to be level 1 for all 4 c.g. locations. Effective rise time was unmeasurable due to airspeed effects and the lack of a good steady-state pitch rate value. Figure 6 in Appendix D shows that the only difference is an increase in overshoot amplitude in pitch rate of .3 degrees/second, which is not enough to change flying qualities levels.

1,000 Feet PA, 180 KCAS

This test point was accomplished with only the low force step input, due to the slow starting airspeed. Figure 7 in Appendix D shows that the equivalent time delay of .04 seconds meets level 1 requirements for all 4 c.g. locations. No other information was measurable with regards to MIL-STD-1797A requirements due to lack of a steady-state pitch rate value. However, a significant increase in pitch rate of 1.2

degrees/second, or 19% higher for configuration 3 over baseline, was observed with the CAS off and with no other apparent degradations in flying qualities. Stability at the gear lowering airspeeds (below 250 KCAS) is critical, since the c.g. shifts .5% MAC further aft when the gear is extended and the wheels swing aft. This particular run showed that no problems should occur at gear up c.g.'s forward of 29.5% MAC.

1,000 Feet PA, 660 KCAS

This particular test point represents the lower right corner of the flight envelope, or the high dynamic pressure point. Table 6 summarizes the response measurements for all 4 test conditions (low and high 'g' commands, CAS on and off) at the baseline c.g. of 28.7% MAC and at 30.0% MAC.

TABLE 6

Summary of Results at 1,000 ft. PA, 660 KCAS

	CAS on 28.7%/30.0%		CAS off 28.7%/30.0%		Level 1
Measurement	Low G	High G	Low G	High G	Requirement
Transient Peak Ratio	0/0	.05/.11	.15/.20	.1/0	≤ .3
Equivalent Time Delay	.03/.03	.04/.04	.05/.05	.04/.04	≤ .12
Effective Rise Time	.1/.12	.09/.11	.1/.03	.16/-	.008≤ t ≤.45

Figure 8 in Appendix D shows the CAS off low 'g' command step input of 5 pounds of aft stick force. It was assumed that the steady-state pitch rate response occurred between 2.4 and 2.6 seconds, and that beyond 2.6 seconds, airspeed change effects caused the decrease in pitch rate. The increase between baseline and 30.0% MAC c.g. location in peak pitch rate amplitude was approximately .5 degrees/second, which equated to about .3 normal g's/second. This was not considered a significant difference, and all values of the baseline as well as the 30.0% MAC aircraft were level 1. No problems were discovered in this corner of the flight envelope.

45,000 Feet PA, Mach 1.5

No significant degradation in aircraft response was discovered at this test point. Although the low force command step input of 4 pounds with the CAS off was found to be level 3 for the transient peak ratio at 30.0% MAC, it was level 3 for the baseline aircraft as well. Figure 9 in Appendix D shows that there is very little difference in system response across all 4 c.g. locations, and this was the worst case test point of the 4 variations examined at this test condition. Most likely due to the shift of the aerodynamic center from quarter to half chord as speeds increase from subsonic to supersonic, the effects of an aft c.g. shift at supersonic speeds were negligible.

This completed the "corners of the flight envelope" analysis. Although some degradation occurred via increased overshoot amplitudes in pitch rate due to the aft shift in c.g. location, none of the problems were significant enough to suspect a drop in predicted levels of flying qualities ratings. Therefore, the next phase of the analysis was started, which investigated the pitch sensitive region of the F-15.

F-15 Pitch Sensitive Region Analysis

The first point that was analyzed was close to the center of the pitch sensitivity region, which was at 15,000 feet PA and .85 Mach (437 KCAS). CAS on responses showed almost no degradation as the c.g. was moved aft to 30.0% MAC at both low and high force step inputs. Peak normal 'g' overshoot for a commanded 6 'g' pitch up was only .25 g's higher for the 30.0% MAC c.g. location than for the baseline aircraft. Otherwise, the time history traces of pitch rate response were virtually identical over all 4 c.g. locations. However, CAS off responses were a different story. Figures 10 and 11 in Appendix D show the pitch rate (in degrees/second) and normal 'g' versus time for a 4 pound low force step input. As can be seen, at approximately 7 seconds after applying the step input, configuration 3 (30.0% MAC) becomes unstable and exceeds negative 'g' structural aircraft limits on the third peak overshoot (F-15 negative limits are -3.0 g's). This test point is obviously a show stopper for the 30.0% MAC c.g. location. The CAS off high force step input had some unusual results also, as shown in Figure 12 in Appendix D. Although apparently stable at all c.g. locations, the initial transient peak experienced a quite noticeable dip, probably due to passage of the horizontal stabilator through a disturbed flow region. At the higher commanded turn rates, though, the stabilator probably stays above the disturbed flow and does not experience the same pitch reversals that the low force step input sees. Because of the unstable response of this test point at the 30.0% MAC c.g. location, the remainder of this analysis was limited to looking at c.g. locations between baseline and 29.5% MAC.

The next step in the analysis was to verify that the worst case response was indeed at this test condition of 15,000 feet PA and .85 Mach. In order to sample a large region around this unstable area, altitudes were varied between 5,000 feet and 30,000 feet PA at 5,000 foot increments, and Mach numbers were varied between .6 and .95 Mach at increments of .05 Mach. Changes in Mach number affected the results dramatically at all altitudes. Below

.7 Mach, differences in peak overshoots were typically less than 1 degree/second in pitch rate between baseline and 29.5% MAC, and beyond the third or fourth overshoot, differences between the traces were negligible. Conversely, above .85 Mach, increases in peak overshoot were typically less than .5 degrees/second in difference between configurations and quickly damped out to no real differences after the second or third overshoot. Figure 13 in Appendix D at 15,000 feet PA and .95 Mach typified high subsonic (.9 Mach or above) responses at all altitudes between 5,000 and 30,000 feet PA. (The last 2 seconds of response on this graph can be ignored- the variations in pitch rate are due to problems in the simulation when the aircraft passes through vertical flight parameters.) Therefore, the analysis concentrated on Mach numbers between .7 and .85 Mach while searching for the worst case test condition.

Variations in altitude also quickly defined boundary conditions for the worst case test condition. At or below 10,000 feet PA, the higher dynamic pressures damped out the overshoots quickly at all the different c.g. locations. And above 25,000 feet PA, Mach effects (above .9 Mach) took over at much lower indicated airspeeds. Below the Mach effects, indicated airspeeds were so low that the simulation showed no real problems caused by the c.g. shift as well.

Therefore, altitudes were limited in further analysis to between 15,000 and 25,000 feet PA.

15,000, 20,000, and 25,000 feet PA were examined at c.g. locations between baseline and 29.5% MAC at .7, .75, .8, and .85 Mach. Figure 14 in Appendix D shows the worst case pitch rate response of this group, which occurred at 15,000 feet PA and .75 Mach. With the c.g. located at 29.5% MAC (labeled configuration 2 on the graph) the pitch response is very close to being neutrally stable. Figure 15 in Appendix D, which is the normal 'g' response of the same test condition, shows this a little more clearly. After the initial transient peak occurs from the 4 pound step input at 1 second elapsed time, the baseline configuration damps out very quickly to a minor oscillation around positive 2 g's (2.05 $\pm$ .05). However, configuration 2 shows positive peak values that are not decreasing, with peak readings of 2.95 g's at system response times of 9.3 and 16.5 seconds. Unfortunately, decreasing airspeed during the maneuver is causing a slow increase in the steady-state commanded g, which prevents getting an accurate reading of system damping from the data. However, to meet even level 3 requirements, MIL-STD-1797A requires that there be at least a 15% decrease in peak amplitudes between concurrent overshoots. (Refer to Table 5, page 19: level 3 transient peak ratio must be less than or equal to .85) Obviously, configuration 2 at 29.5%

MAC or configuration 5 at 29.4% MAC do not meet this requirement. But the lack of a steady-state pitch rate value precludes making any firm judgments for the other 3 configurations. Although the baseline configuration is the best responding system, the amplitude of the second peak (first negative overshoot) compared to the first overshoot puts it at or above level 3 requirements for transient peak ratio, depending on where an estimate of the system steady-state response is made. It is not until the third overshoot that the baseline response becomes significantly better. In comparison, configurations 1 and 4 (at 29.0% and 29.2% MAC, respectively) lie in an obviously less damped responding system. Although nothing can be definitively stated concerning estimates of flying qualities levels for the configurations out to 29.2% MAC, all 3 configurations <u>are</u> (at least according to computer simulation) <u>positively stable</u>, which means that they should be safely explorable through <u>flight test</u> to determine the actual (versus predicted) flying qualities levels.

One unusual test condition was discovered at 25,000 feet PA and .85 Mach, and is shown on Figure 16 in Appendix D. The test started with a 2 second aileron roll to bring the aircraft to approximately 60 degrees of bank, and then a 4 pound aft stick force was applied from 4 to 30 seconds of system time, all with the CAS off. The roll was added in an unsuccessful attempt to reduce airspeed effects in trying to yield a steady-state value of pitch rate. As can be seen on the graph, the pitch response of the 29.2% MAC c.g. location seems to be better damped than the currently approved for flight baseline configuration. In fact, the baseline configuration appears to be unstable at this test condition because the pitch rate amplitude is increasing with time, as can be seen in Figure 16.

Pitch Rate Improvements

The next portion of the analysis dealt with quantifying improvements in pitch control authority due to the lead ballast removal. Based on the previously stated results, the 30.0% MAC c.g. location is unstable (See Figure 10 in Appendix D) and the 29.4% and 29.5% MAC c.g. locations are approximately neutrally stable (See Figure 15 in Appendix D). Since the weight and balance data on any particular aircraft is only accurate to within .1% MAC, 29.2% MAC was assumed to be the practical aft limit for flight test (29.3% + .1% inaccuracy would reach the neutrally stable location). Hence, the baseline aircraft will be compared with configuration 4 (29.2% MAC) to determine pitch rate improvements. Since the aircraft is g-limited above the aircraft corner velocity, this analysis will concentrate on aircraft airspeeds below corner velocity for measuring improvements. The test conditions that were examined also assumed that the CAS would be on. This assumption was made based on the premise that a loss of the primary flight control system (CAS) would be grounds to return the aircraft to base for landing, even during a wartime scenario. Therefore, a prudent Eagle pilot would not (or rather, should not) find himself engaged in a slow speed dogfight without an operable CAS. Figures 17 and 18 in Appendix D show the difference in pitch rate between the baseline (B) aircraft and the 29.2% MAC c.g. location (4) aircraft at 10,000 feet PA and 200 and 250 KCAS, respectively. Although on average a modest 3 percent improvement is realized, a maximum improvement of 42 percent was realized at a system time of 15 seconds on Figure 17, and a maximum of 14 percent improvement at 17 seconds on Figure 18. In this particular part of the analysis, airspeed losses during the maneuver were actually beneficial. In Figure 17, for example, airspeed was 200 KCAS at the start of the maneuver and was at 120 KCAS at 15 seconds of system time. Therefore, the one maneuver actually presents a range of pitch authority improvement between 200 and 120 KCAS as the system time increases between 1 and 15 seconds. The largest improvements were typically encountered at airspeeds below 150 KCAS during the maneuvers. Unfortunately, this arena could not typically be explored because the aircraft could not be trimmed in level flight on the computer simulation at these slow airspeeds. A final point to be made in the area of pitch authority improvement can be made referring back to Figure 7 in Appendix D, at 1,000 feet PA, 180 KCAS, and CAS off. The F-15 dash 1 warns pilots to fly a faster 18 unit AOA approach instead of a normal 21 unit AOA approach for certain combinations of flight control malfunctions, including pitch CAS failure, due to reduced nose authority available at the normal, slower speeds (3:6-2). Figure 7 shows an improvement in generated pitch rates with the aft c.g. locations at these slower airspeeds with the CAS off, which should also improve the pitch authority problems experienced by the F-15 in this area.

Other Considerations

Two final areas were also examined during this analysis to ensure that all applicable topics of MIL-STD-1797A were addressed. One area, as outlined in MIL-STD-1797A, deals with roll coupling concerns that may occur when a c.g. shift is experienced (4:89). Due to the extremely high roll rates that can be generated in modern fighter aircraft, any change in AOA at fixed airspeed conditions (as would be experienced with a c.g. shift or weight change in the aircraft) should be checked to ensure that the change does not affect how AOA and sideslip angle interact during continuous, high-rate rolls. In other words, no additional kinematic coupling effects should occur as a result of a change. (The autoroll phenomenon in the F-15 is a good example of kinematic coupling.) As shown on Figure 19 in Appendix D, no unusual roll rate coupling occurred between the baseline aircraft and configuration 2 (29.5% MAC) at flight conditions that typically generate F-15 autorolls. Pitch rates were also stable, and no apparent kinematic coupling during continuous aileron rolls occurred between AOA and sideslip angle. Other considerations for c.g. shifts were listed in MIL-STD-1797A such as turbulence effects or loss of aircraft engine effects, but these considerations could not be examined in this particular computer simulation.

Another area of concern was examined at the worst case test condition. This dealt with recovery techniques for the aircraft from the areas of instability. It was assumed that the natural tendency of a pilot, when experiencing an instability brought on by an input to the control stick, would be to release the stick back to neutral. This was accomplished at the worst case test condition of 15,000 feet PA and .75 Mach. The stick was also released just prior to the first negative peak, which should also be the worst case time for the release. As Figure 20 in Appendix D shows, with the stick release occurring at 5 seconds, the first negative peak does increase quite dramatically. However, all configurations through configuration 2 at 29.5% MAC demonstrate positive static stability after the stick release is made.

V. Conclusions and Recommendations

The first point that needs to be emphasized is that this was a limited computer simulation analysis of the effects of removing lead ballast from the nose of the F-15 Eagle. It was limited in regards to the capabilities that the computer simulation could provide in response to the requirements laid out in MIL-STD-1797A, Flying Qualities of Piloted Aircraft. A few of the areas outlined in MIL-STD-1797A that the program could not simulate have already been mentioned in the Results and Analysis section, such as effects due to atmospheric turbulence or sudden loss of engine thrust. Several other areas were unable to be analyzed as well, making this an incomplete study at best. Also, problems resulting from introduced airspeed effects in the simulation made specific results difficult to compare directly with hard number requirements outlined in MIL-STD-1797A. However, it must also be noted that MIL-STD-1797A is a design guide and not a procurement regulation. (Were it a regulation, the F-15, long recognized as having one of the best designed flight control systems in the fighter force, might come under scrutiny as shown on Figure 16 in Appendix D where the baseline aircraft shows signs of stability problems.) Although an excellent reference guide, MIL-STD-1797A also has had 2 revisions in the past few years, pointing to the fact that the design industry continues to find better ways of assessing flying qualities performance. With that viewpoint in mind, this section of the report will summarize the information that was discovered during the analysis and hopefully provide some insight concerning its usefulness.

Information that was discovered while performing the background research for the c.g. analysis was very useful, particularly the data concerning avionics updates. It is easy to understand why the ICS and SDR were left out of the critical aft c.g. balance calculation. By leaving this equipment out, it would ensure that all models of the F-15 (including the 2 seat trainer variants) could use the <u>same</u> calculation sheet to ensure that their c.g.'s remained within limits. It would also ensure that if the avionics were pulled out for maintenance, the aircraft would not exceed flyable c.g. limits, nor would it require maintenance personnel to replace the black boxes with equal amounts of ballast. A little further research into the subject turned up an engineering change proposal (ECP) from the McDonnell Douglas Corporation, manufacturer of the F-15, concerning ICS ballast adjustment (McAir ECP 01469C1, approved 25 February 1982). This ECP basically recommends removal of 195 pounds of lead ballast from the nose of the aircraft when the ICS is installed. However, it makes no recommendation or any mention of including the equipment in the critical aft c.g. balance calculation. Since the critical aft c.g. balance calculation is the check that maintenance personnel use to ensure meeting specific aircraft c.g. limits, the ECP is apparently rendered ineffective. As an example, according to the 1st TFW/QA office at Langley AFB, VA., ICS equipment is removed from their F-15's before the aircraft are sent to depot for modifications due to the classified nature of the ICS. However, no ballast is required to be installed in its place for the delivery flight. (This would be in accordance with critical aft c.g. balance calculations, which do not include the ICS.) But it also means that when the ICS is installed, excess ballast must still be on the aircraft.

Another consideration that comes to mind is that the USAF seldom flies the F-15 configured such that it would ever experience the critical aft c.g. situation (although, nothing says that it could not be flown with just external wing tanks and Aim-7's loaded). It seems that the F-15 fleet is again paying a price in potential maneuvering performance to meet the extreme situation in which the aircraft may or may not find itself configured. This information led to the following 2 recommendations.

RECOMMENDATION 1. INCLUDE THE INTERNAL COUNTERMEASURES SET AND THE SIGNAL DATA RECORDER AS LOADED EQUIPMENT IN THE CRITICAL AFT CENTER OF GRAVITY BALANCE CALCULATION IN THE AIR SUPERIORITY F-15 EAGLE.

Two ramifications will result from this recommendation. The first will be that a change will result in maintenance requirements, such that whenever the ICS or SDR equipment is removed from the aircraft, an offsetting amount of lead ballast must be installed in its place to maintain c.g. limits. The second ramification is that in aircraft that do not have the equipment loaded, such as trainer models, the critical aft c.g. balance calculation must properly reflect that the equipment is not loaded.

RECOMMENDATION 2. MAXIMIZE PERFORMANCE OF THE F-15 THROUGH BETTER WEIGHT MANAGEMENT PROCEDURES AND BY OPTIMIZING CENTER OF GRAVITY LOCATION BASED ON THE ACTUAL AIRCRAFT CONFIGURATIONS BEING OPERATIONALLY FLOWN.

Actually, incorporation of this recommendation will cover the problem discussed under recommendation 1 as well. Basically, what this recommendation does is suggest that the critical aft c.g. balance calculation base its calculation on the worst case configuration that could result on a particular mission based on the aircraft's <u>takeoff</u> configuration, and not on some extreme, single configuration that may or may not be applicable. There probably is not a single F-15 pilot around that wants to sacrifice even the smallest amount of performance by carrying around an additional 50 (or whatever) pounds of lead ballast in the nose of his F-15 to ensure adequate handling qualities for 2 external wing tanks, particularly when he took off with only a single external centerline tank loaded. Now, obviously, the worst case decision tree must be used to determine the configuration. If an aircraft took off with 3 external fuel tanks loaded, there is always the possibility that it may end up with just 2 wing tanks left, and, therefore, must still use the standard critical aft c.g. configuration. But if the aircraft launches on a mission with only an external centerline fuel tank, then the critical aft c.g. balance calculation should be based on <u>that</u> configuration, and not on carrying external wing tanks. Another configuration example deals with AIM-9 missile launchers and adapters, which almost always are loaded on the aircraft. 4 launchers and adapters, which is the typical load, adds 308 pounds of weight and shifts the c.g. forward by .1% MAC (6:11). By including AIM-9 launchers and adapters in the critical aft c.g. balance calculation as well, another 20 pounds of lead ballast could be removed from the aircraft! The bottom line is that the F-15 should be optimized for c.g. location based on its operational configuration. And it should be optimized to just reach aft c.g. limits at missions end, and not be well forward of the limit.

Realistically, 10 years ago, this requirement would have probably kept a wings QA shop busy doing nothing but c.g. calculations. However, with the advent of small computers that can accompany units to battle zones today, a software program could probably generate that information in a matter of seconds. (Possibly a good future thesis topic.) Then, the information could quickly be relayed to a crew chief, who, armed with a speed wrench, could probably optimize the ballast load on his F-15 in a matter of minutes.

Turning towards the analysis of actually trying to change the aft c.g. limits for the F-15, the conclusions that can be drawn are not so straightforward. Obviously, a maximum aft limit exists according to simulation at 30.0% MAC c.g. location due to lack of positive aircraft stability. This means that the gear up limit can be at no greater than 29.5% MAC c.g. location, since when the landing gear is extended, the c.g. moves .5% MAC aft, thus putting the aircraft at the 30.0% MAC limit again. Further indications point towards a gear up limit of just under

29.5% MAC c.g. location, as several CAS off test points revealed close to neutral stability at this c.g. location. However, due to limitations in allowable starting conditions in the computer simulation, accurate predictions of aircraft flying qualities levels for c.g. locations forward of 29.2% MAC could not be measured. In fact, even the currently flown baseline c.g. location of 28.7% MAC showed instability at one test point. What can be said from the simulation work, though, is that the aircraft appears to be positively stable at all conditions sampled in the corners of the flight envelope as well as around the F-15 pitch sensitive region out to aft c.g. locations of 29.2% MAC in the worst case external configuration of the F-15. From this observation, the following recommendation is made.

RECOMMENDATION 3. THE F-15 SHOULD BE FLIGHT TESTED IN THE CRITICAL AFT CENTER OF GRAVITY EXTERNAL CONFIGURATION WITH A PROJECTED LANDING CENTER OF GRAVITY LOCATION OF 29.2% MEAN AERODYNAMIC CHORD TO DETERMINE ACTUAL AIRCRAFT FLYING QUALITIES RATINGS.

Since the aircraft showed positive stability throughout the flight envelope at this c.g. location, flight test work should be capable of being safely conducted. It was also verified that releasing the stick to neutral conditions during the worst case test condition will decrease aircraft instability and provide a suitable recovery technique for the test pilot to follow in the event that controllability problems develop. Therefore, the aircraft should be safely testable at these conditions in order to determine actual degradations in flying qualities, if any. The next section of this report will outline considerations for conducting this flight test verification.

It can also be concluded that a minor improvement (3% on average) in pitch rate will be experienced below corner velocity when the c.g. is moved aft to the 29.2% MAC location. In a slow speed dogfight, the model that this analysis tried to simulate, the pilot would tend to try to maximum perform his airplane. This means flying the F-15 in the light aircraft buffet region. Unfortunately, there was no way to tell when the computer simulation was encountering this buffet region. Therefore, the magnitude of the pitch step inputs were based on estimates from personal experience of how much aft stick force would reach those buffet intensities. Actual pitch rate improvements through flight test may show a higher average improvement rate than 3%. However, since 3% turned out to be a solid average over many (probably non-optimal) test conditions, it should prove to be a good minimum estimate of improvement. Also, the further aft c.g. location should help improve F-15 flare capabilities for landing with degraded pitch flight control anomalies.

VI. Proposed Flight Test Profile

In order to verify this analysis, the flight test work will require a minimum of 2 or possibly 3 F-15 sorties. Because of the tight tolerances on weight and balance requirements, it is possible that an F-15 that has not been modified for flight test work (no removal of avionics and no orange test wiring weight) will be required. Also, the single seat F-15A/C model should be used over the 2 seat models due to the difference in airflow effects over the single seat versus dual seat canopy designs. (Or, at least an analysis of the effects of the different canopies should be accomplished if a 2 seater must be used.)

A number of limitations are imposed by the F-15 flight manual that will affect the test profile set-up. As previously mentioned, the aircraft is limited to 660 KCAS or Mach 1.5 (whichever is lower) when carrying external tanks onboard. Also, with any CAS axis disengaged, the aircraft is limited to 600 KCAS, and is limited to a maximum continuous roll of 360 degrees (3:5-4). The flight manual also states that pilot induced oscillations (PIO) may occur if pitch CAS fails while supersonic above 600 KCAS (3:6-1).

For adequate comparison, a minimum of 2 sorties must be flown in this profile by the same test pilot in the same aircraft. The first sortie will be the control flight, where the baseline aircraft is flown in its currently operational normal c.g. location, with a standard load of lead ballast. The second sortie will be flown in the same aircraft with enough lead ballast removed to place the critical aft c.g. balance calculation at 29.2% MAC. It is also recommended that a third sortie be flown between the above mentioned sorties with the critical aft c.g. balance calculation located at 29.0% MAC. However, if funding or aircraft availability preclude this option, 2 sorties should still be adequate and safe.

The profile should follow established guidance as outlined in the Store Certification Guide from the USAF Test Pilot School. (Basically, this profile will follow the same test points as examined in this thesis, starting at the center and then working to the corners of the flight envelope, and then examining potential problem areas.) In addition, several other test conditions should be explored. Consideration should be given to flying the aircraft to 660 KCAS (external tank limits), turning the pitch CAS off, and then slowing the aircraft down below 600 KCAS. Although a prohibited maneuver, (being in excess of 600 KCAS with a CAS axis disengaged) it should be verified that losses of pitch CAS above 600 KCAS with the new c.g. location are safely recoverable. A buildup approach should be used, by disengaging the CAS first in straight and level flight, and then at increasingly elevated g-loadings. However, in accordance with the flight manual guidance, the aircraft should be immediately unloaded when the CAS axis is disengaged and slowed below 600 KCAS for each test point.

Simulating a wartime scenario, if an F-15 experiences a flight control malfunction (pitch CAS loss, in this case), the major consideration should be for safe egress from the battle zone and a return to base. Therefore, a majority of the CAS off handling tasks should explore cruise conditions at varying altitudes and at airspeeds up to 600 KCAS. Both low and high g turns should be performed at the various conditions, simulating navigation turns and reactions to hostile threats, respectively. Another area that requires CAS off exploring is aerial refueling, since the damaged aircraft may need extra gas to make it home from a distant mission. Altitudes should be explored throughout a typical refueling envelope (10,000 to 30,000 feet, say). This particular task will probably generate the worst results in handling qualities, since low force inputs under tight controls are typical of air refueling operations. Problem areas should be noted for possible inclusion in a caution statement in the F-15 flight manual. However, as long as a reasonable refueling envelope can be established, results can still be considered acceptable.

As far as CAS on maneuvers are concerned, a full range of handling tasks should be included. Fine tracking tasks should be examined, including gun tracking exercises as well as formation flying. Basic fighter maneuvers (BFM) should be examined also, although a 30 unit AOA limit exists with external wing tanks. Possibly, 2 additional sorties should be considered between the baseline and 29.2% MAC c.g. location aircraft with either a clean or external centerline tank only configuration. This will allow full maneuvering BFM to occur, with a chance to examine nose authority improvements between aircraft at high AOA conditions. And, finally, the last test condition to examine may be a mid-altitude, low g, CAS off turn at minimum allowable fuel, which will simulate the worst case test condition that this limited analysis discovered.

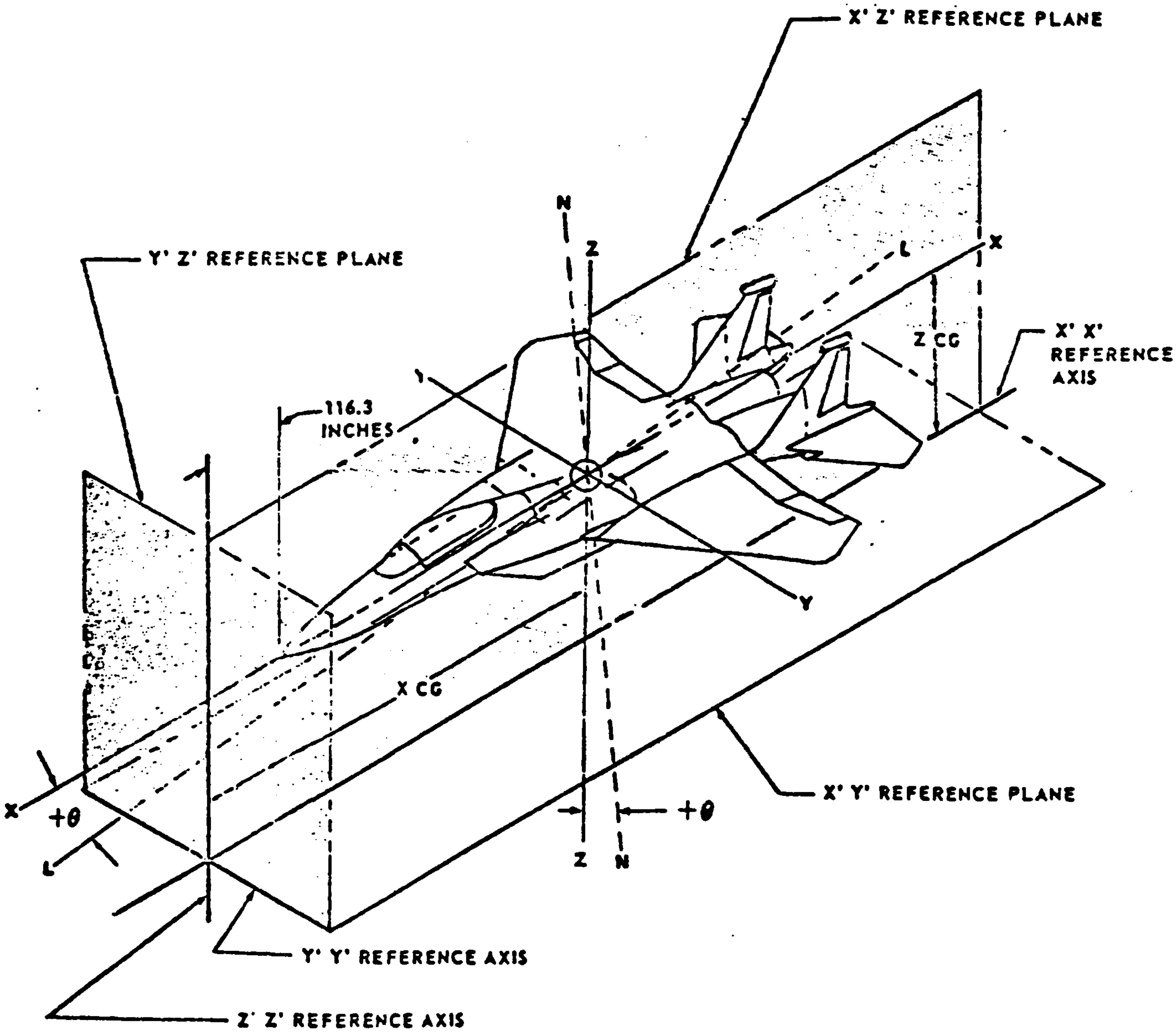
Appendix A: Aircraft Reference Datum and Center of Gravity Calculation Example

The F-15 Aircraft Reference Datum from the McDonnell Aircraft Company is shown in Figure 21 on the next page (7:vii). A majority of the lead ballast loaded on the aircraft is forward of the cockpit area in the avionics bays, between fuselage stations 207 and 227 inches. (Or, between 90.7 and 110.7 inches aft of the nose of the aircraft, since the nose is at fuselage station 116.3 inches.) To determine the effects on the aircraft c.g. location of removing the lead ballast, the ballast weight and fuselage station must be known, as well as the starting aircraft weight and c.g. location. For example, an F-15C in the critical aft c.g. configuration has a weight of 33,467 pounds and a c.g. location at 563.1 inches (6:12). To convert c.g. location in inches to percent mean aerodynamic chord, the following equation is used for all A through D models of the F-15:

$$\%\ MAC = \frac{c.g.\ (inches) - 508.1}{191.33} * 100 \qquad (3)$$

For our example,

$$\frac{563.1 - 508.1}{191.33} * 100 = 28.7\%\ MAC$$



REFERENCE PLANE LOCATIONS

X' Y' PLANE	WL 0.0	(APPROX 100 INCHES BELOW AIRPLANE)
Y' Z' PLANE	FS 0.0	(APPROX 116.3 INCHES FWD OF NOSE)
X' Z' PLANE	BL 0.0	(CENTERLINE OF AIRPLANE)

Figure 21. F-15 Aircraft Reference Datum (7:vii)

To calculate the effects of removing ballast, the resulting moment arm must also be known (weight * arm). For example, removing 100 pounds of ballast located at fuselage station 220 inches, the effect on c.g. location is determined as follows:

Item	Weight (Lbs)	Arm (in.)	Horizontal Moment (in-lbs)
Aircraft	33467	563.1	18845268
Ballast Removed	-100	220	-22000
Results	33367		18823268

New c.g. location is:

$$\frac{18823268}{33367} = 564.1 \text{ or } \frac{564.1-508.1}{191.33} = 29.2\% \text{ MAC}$$

If the ICS and the SDR are included in the calculation, the following results are found:

Item	Weight (Lbs)	Arm (in.)	Horizontal Moment (in-lbs)
Aircraft	33467	563.1	18845268
ICS	323	348.7	112630
SDR	42	484.6	20353
	33832		18978251

New c.g. location is:

$$\frac{18978251}{33832} = 560.96 \text{ in. or } \frac{560.96-508.1}{191.33} = 27.63\% \text{ MAC}$$

Appendix B: Moments of Inertia Calculations

This appendix will show how the numbers in Table 3 on page 9 were calculated for the F-15C moments and products of inertia. The numbers for the Basic Air Superiority Mission (BASM) F-15C came from the F-15 Stability Derivatives Mass and Inertia Characteristics Manual from the McDonnell Aircraft Company (7:3.13). The BASM numbers that are listed included 4 Aim-7's loaded, a full load of 20 millimeter ammunition, gear up, and 1100 pounds of fuel. The first requirement was to correct the aircraft configuration to the critical aft c.g. balance configuration. To do this, the 20 millimeter ammunition needed to be fired, a centerline pylon added, 2 wing pylons and tanks added, and a correction made for a lightweight pilot. The numbers for each of these changes as shown in Table 7 on the next page came from the same reference as above (7:3.15). The resulting moments were for the baseline configuration. After establishing baseline, configurations 1 through 5 were calculated by withdrawing varying amounts of lead ballast to reach the desired c.g. locations. All weights are in pounds, and all x, y, and z locations (see Aircraft Reference Datum in Appendix A) in inches. To convert the resulting moments from pounds-square inches to slugs-square feet, divide by 32.17 feet per second squared and then multiply by 144 square inches per square foot.

TABLE 7. F-15C Tabulation of Inertia Items (7:3.13)

Item	Wt(Lbs)	X	Y	Z	WX	WZ	$WX^2/1000$	$\frac{WY^2}{1000}$	$\frac{WZ^2}{1000}$	$\frac{Ix}{1000}$	$\frac{Iy}{1000}$	$\frac{Iz}{1000}$	$\frac{Wxz}{1000}$
BASM	32207	561.6	.5	115.5	18086100	3719153	10858458	77367	435369	24202	60914	74193	2082331
Fire Ammo	-282	495.0	-2.7	120.0	-139590	-33840	-69097	-2	-4061	-42	-42	-15	-16751
C/L Pylon	296	567.3	-	85.5	167922	25308	95261	--	2164	8	364	357	14357
Wg Pylons	690	579.9	-	114.1	400136	78724	232036	9173	8983	29	948	922	45655
Wg Tanks	640	564.3	-	90.0	361152	57600	203798	8508	5184	115	2171	2135	32504
Lt. Pilot	-45	299.0	-	140.8	-13455	-6336	-4023	--	-892	-3	-9	-6	-1894
	33506	563.0	.5	114.6	18862265	3840609	11316433	95046	446747	24309	64346	77586	2156202
		(28.7%)					-10620363	-17	-440040				-2161800
							696070	95029	6707	24309	64346	77586	-5598
	Ix= 27209 slug-ft^2				Iy= 165597 slug-ft^2					95029	696070	696070	
										6707	6707	95029	
	Iz= 187520 sluf-ft^2				Ixz= -1208 slug-ft^2					126045	767123	868685	-5598
CNF 1	-54	211.0	-	118.3	-11394	-6388	-2404	--	-756	--	--	--	-1348
	33452	563.5	.5	114.6	18850871	3834221	11314029	95046	445991	24309	64346	77586	2154854
		(29.0%)					-10622089	-8	-439330				-2160233
							691940	95038	6661	24309	64346	77586	-5379
	Ix= 27201 slug-ft^2				Iy= 164695 slug-ft^2					95038	691940	691940	
	Iz= 186631 slug-ft^2				Ixz= -1161 slug-ft^2					6661	6661	95038	
										126008	762947	864564	-5379
CNF 2	-144	211.0	-	118.3	-30384	-17035	-6411	--	-2015	--	--	--	-3594
	33362	564.5	.5	114.6	18831881	3823574	11310022	95046	444732	24309	64346	77586	2152608
		(29.5%)					-10631143	-8	-438148				-2158244
							678879	95038	6584	24309	64346	77586	-5636
	Ix= 27184 slug-ft^2				Iy= 161859 slug-ft^2					95038	678879	678879	
	Iz= 183811 slug-ft^2				Ixz= -1217 slug-ft^2					6584	6584	95038	
										125931	749809	851503	-5636

Appendix C: Simulation Program Description

A majority of the information in this appendix came from an interview with Mr. David Potts, an electronics engineer in the simulation division of ASD/SCES, Wright-Patterson AFB, Oh. The computer simulation that was used for this analysis was developed jointly by ASD/SCES, ENFTC, and the F-15 SPO at Wright-Patterson AFB, Ohio, for use in F-15 flight test work. The program uses a higher order, nonlinear model that is based upon wind tunnel data of the F-15 that was gathered by the McDonnell Douglas Corporation of St. Louis, Missouri, the manufacturer of the F-15. Any break points that occurred in the wind tunnel data were linearly interpolated between breaks to provide a continuous data file. Actuator models were second order, and included hinge rate effects (including blowback) as well as including position motion limits. Response rates were available at 80, 20, and 10 hertz.

The program itself is modeled in the z domain in ADSIM computer language, which is a continuous (versus discrete) domain language used mainly for aerodynamic simulation purposes, and is similar to the AXLE or MIMIC computer languages. The program runs on an AD100 computer (manufactured by the Applied Dynamics International Co. of

Ann Arbor, Michigan) which is a high speed, floating point scalar processing machine manufactured for real time simulations using 6 degrees of freedom. Detailed information about the program or simulation equipment is available thru ASD/SCES, Wright-Patterson AFB, Ohio.

Appendix D: Simulation Results Graphs

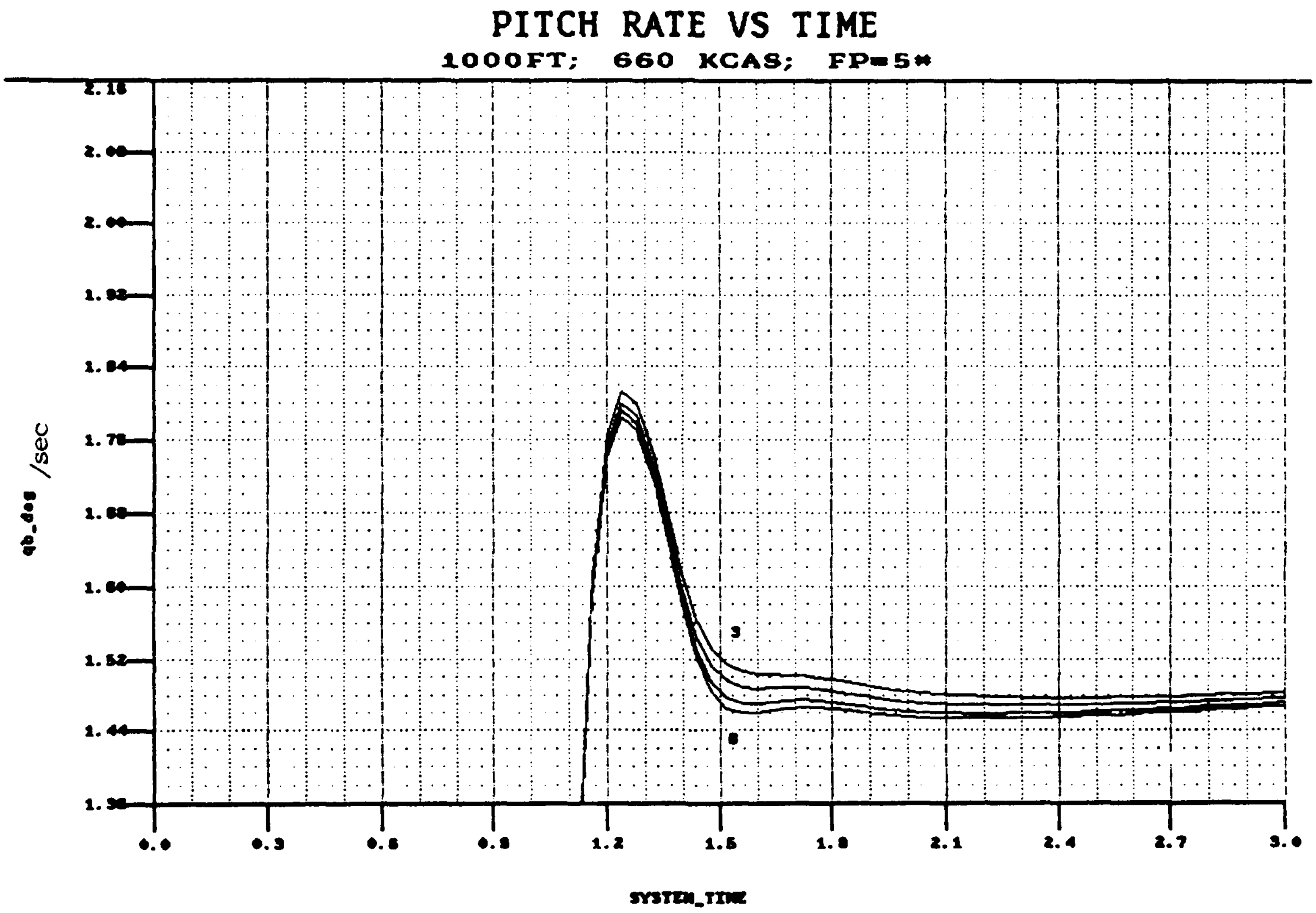


Figure 3. Pitch Rate vs Time at 1000 ft, 660 KCAS, Pitch Force=5 Pounds

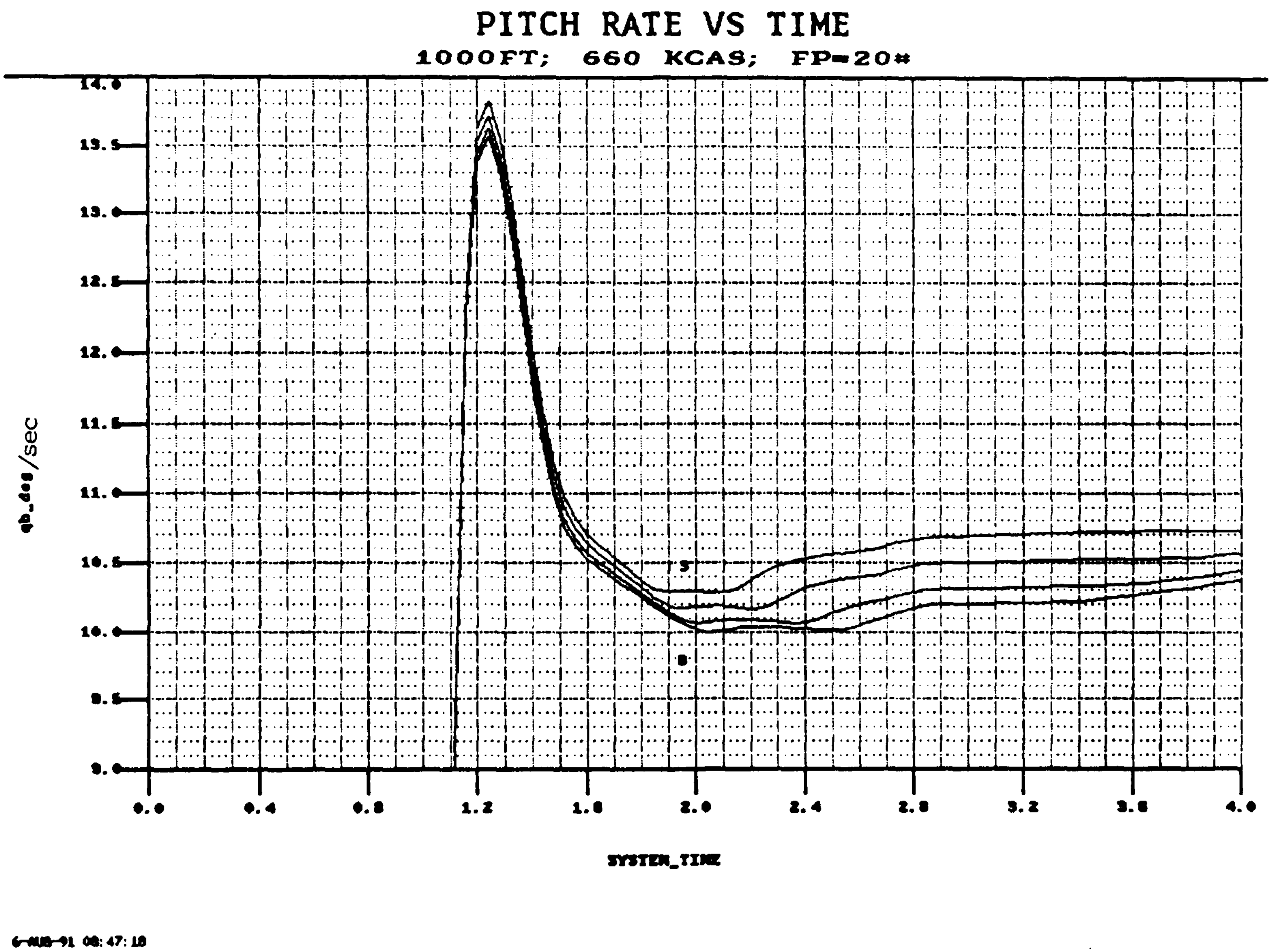


Figure 4. Pitch Rate vs Time at 1000 ft, 660 KCAS, Pitch Force=20 Pounds

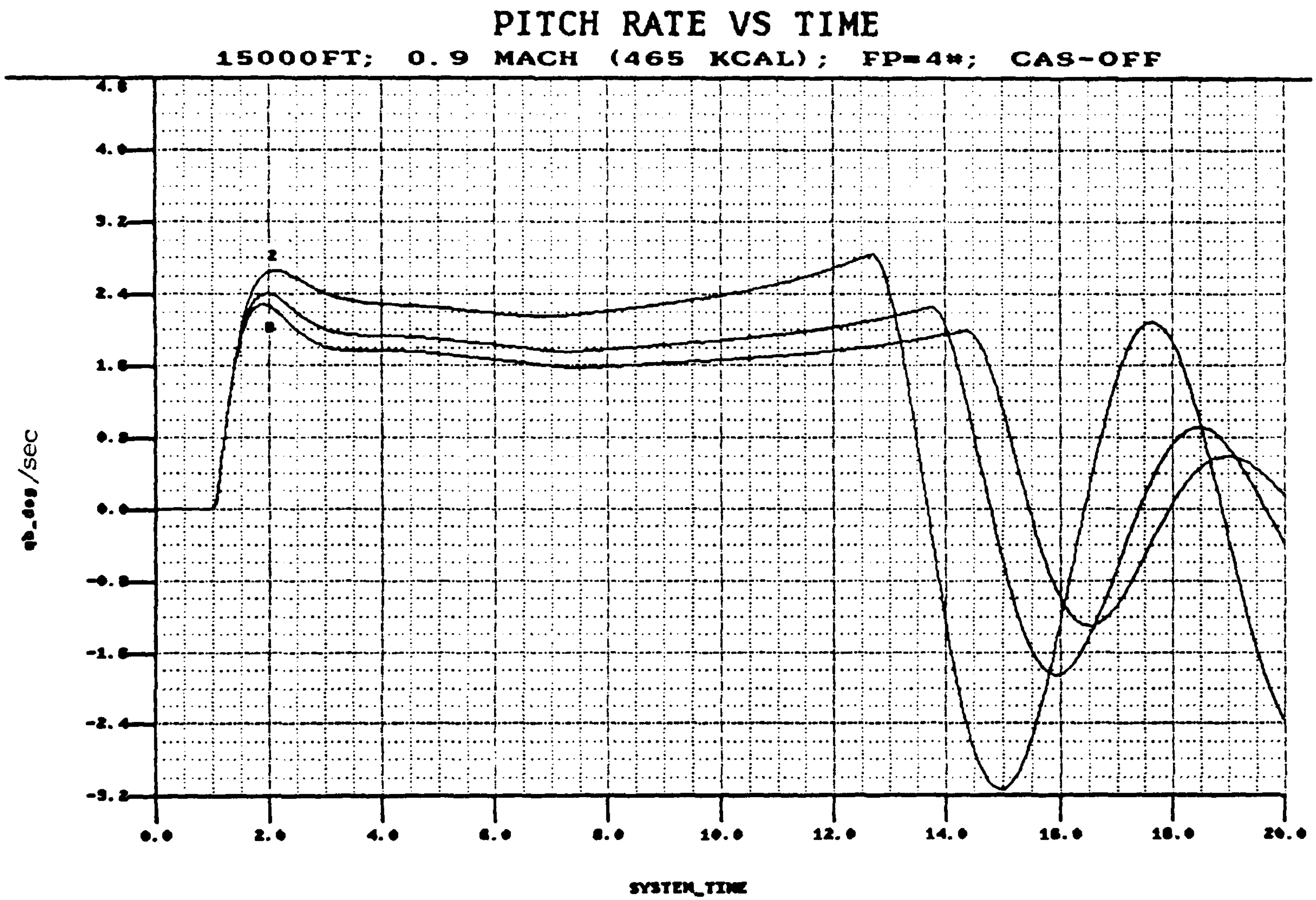


Figure 5. Pitch Rate vs Time at 15000 ft, .9 Mach, Pitch Force=4 Pounds, CAS off

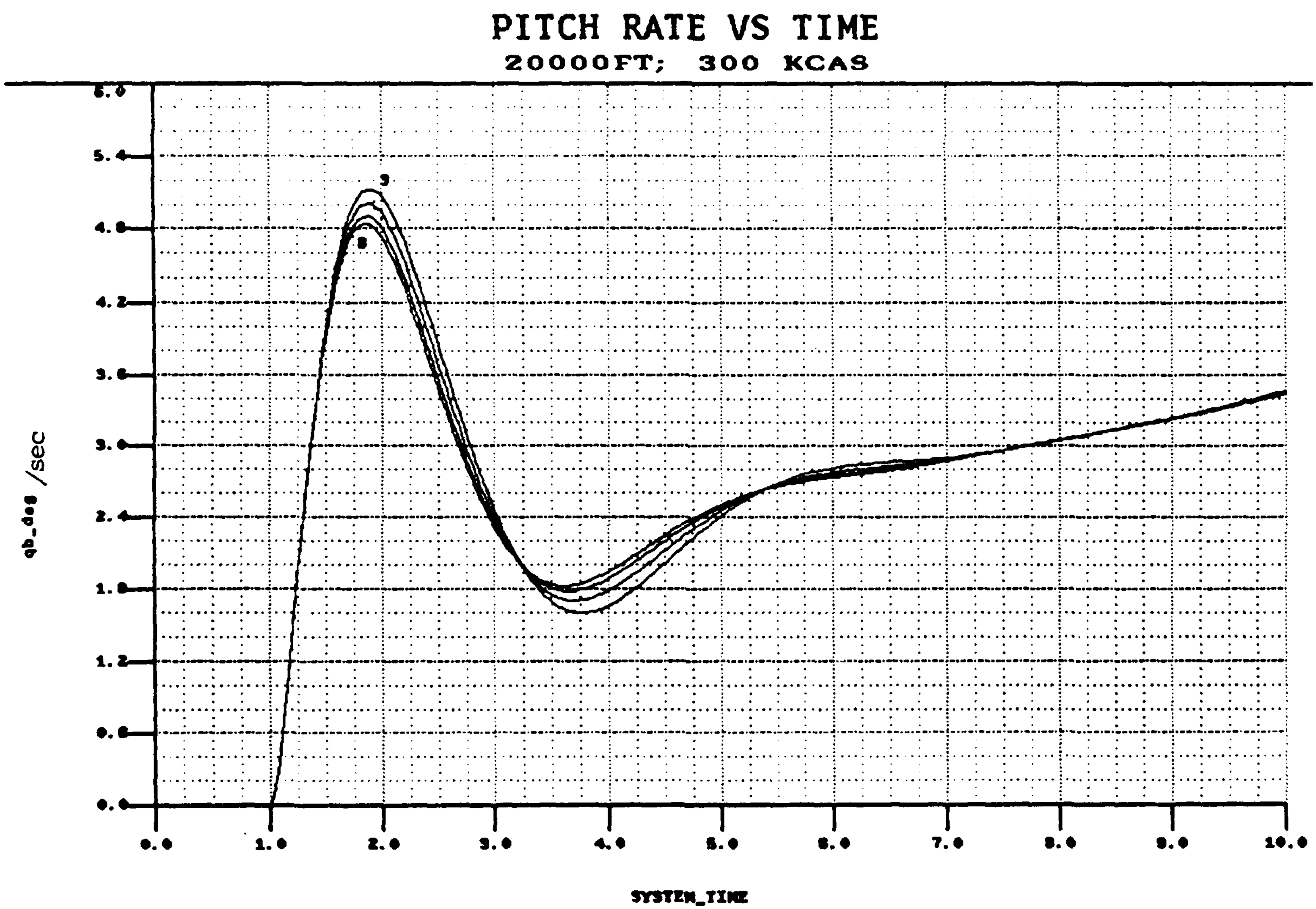


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Figure 6. Pitch Rate vs Time at 20000 ft, 300 KCAS

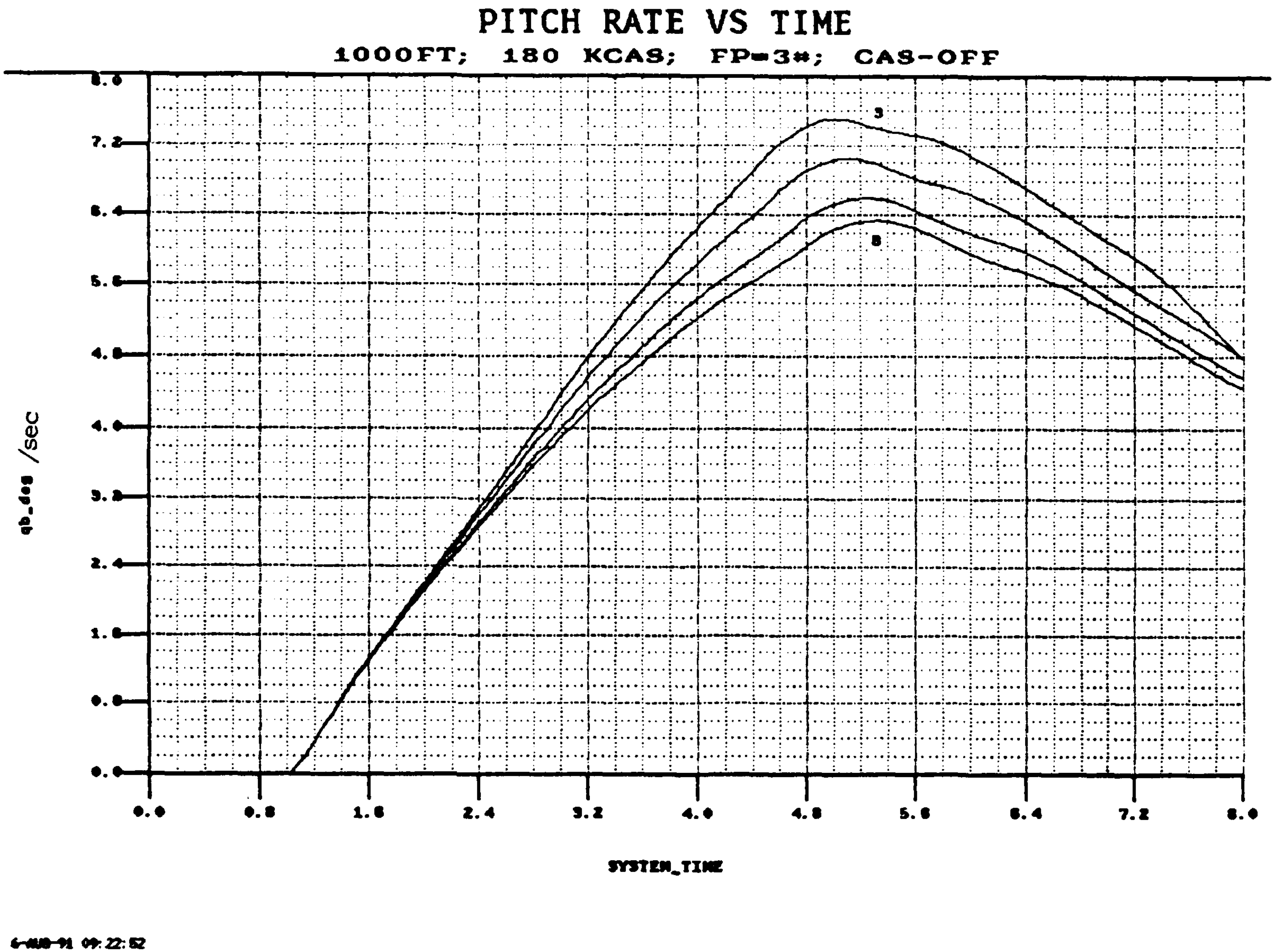

Figure 7. Pitch Rate vs Time at 1000 ft, 180 KCAS, Pitch Force=3 Pounds, CAS off

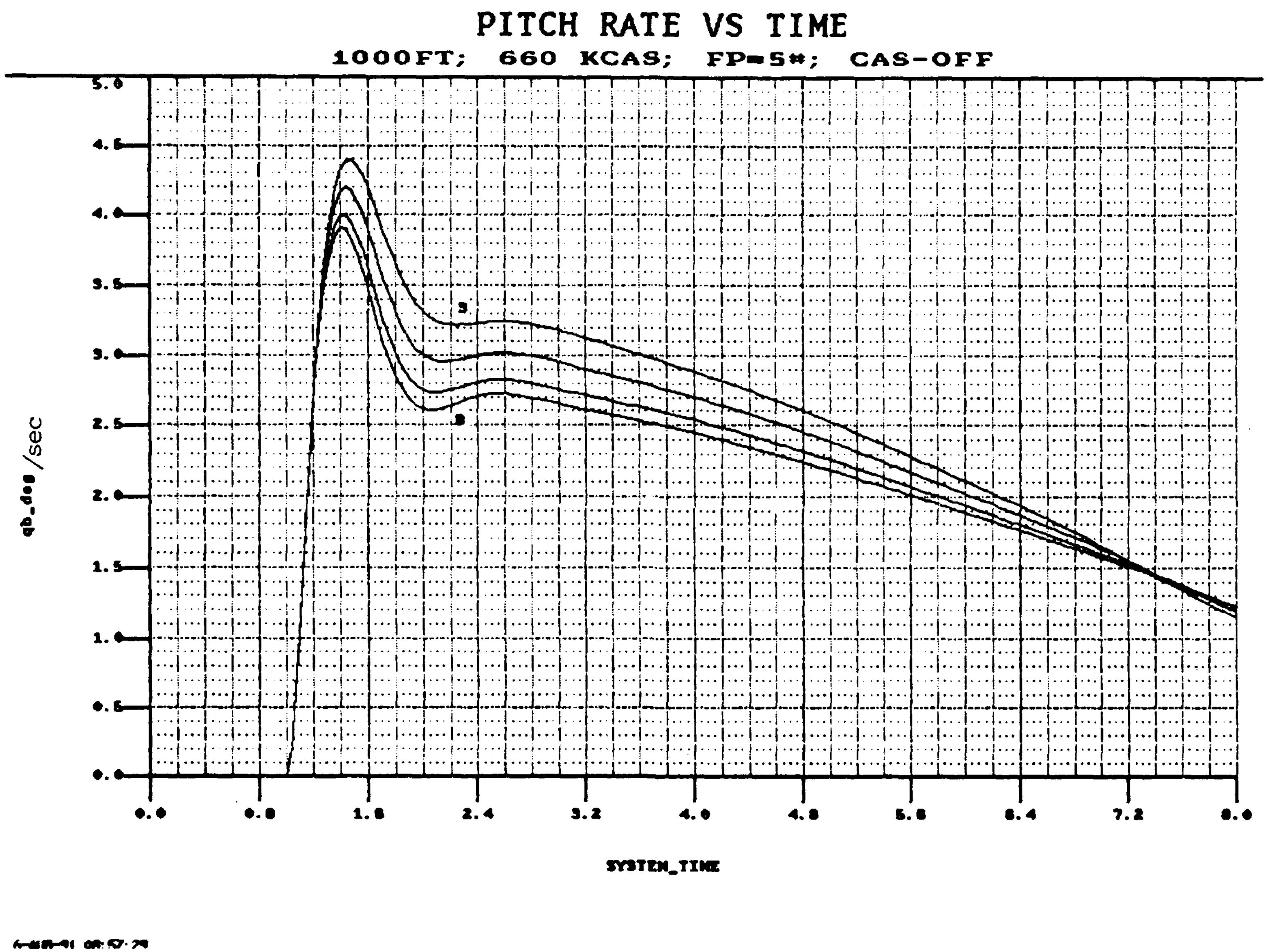

Figure 8. Pitch Rate vs Time at 1000 ft, 660 KCAS, Pitch Force=5 Pounds, CAS off

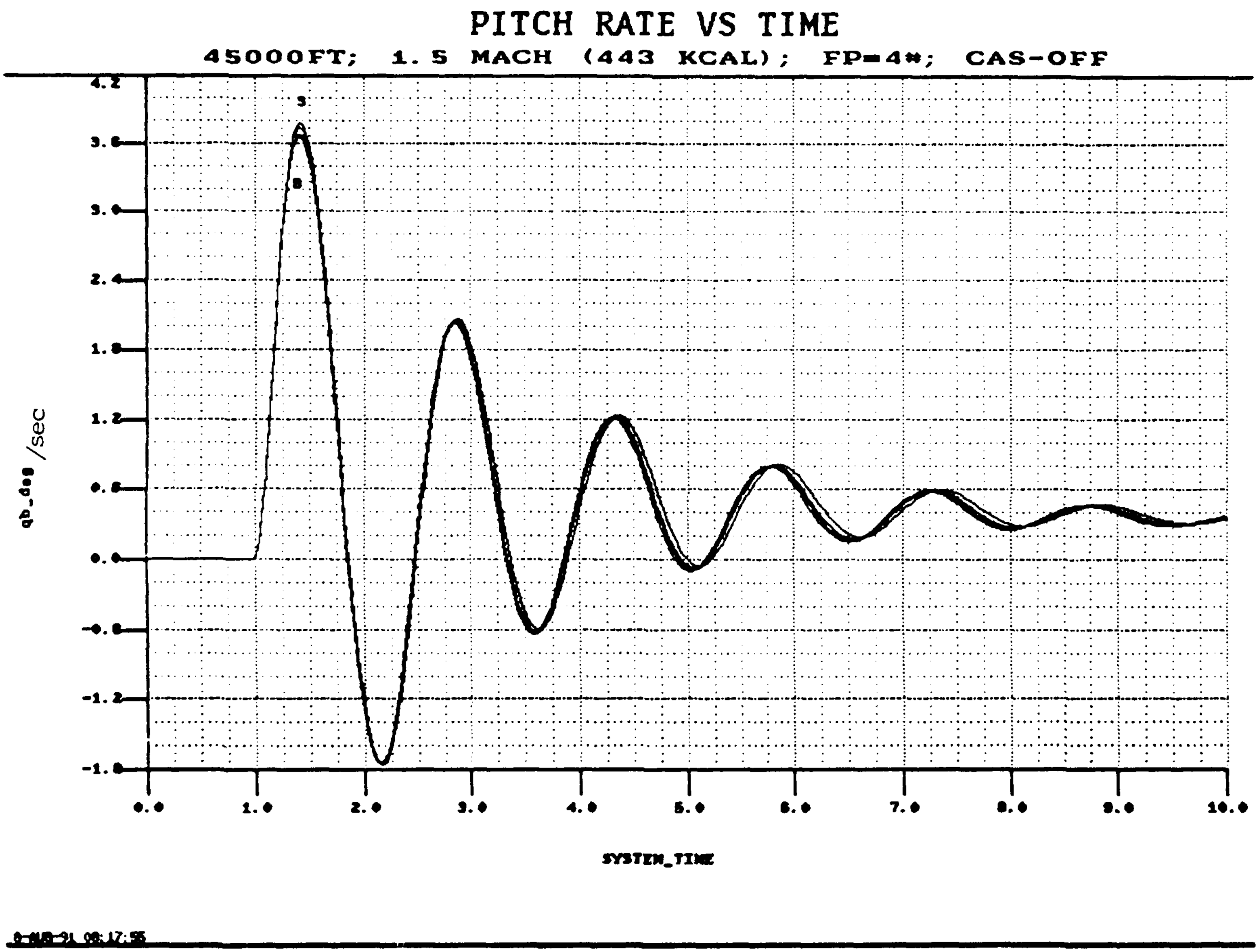

Figure 9. Pitch Rate vs Time at 45000 ft, 1.5 Mach, Pitch Force=4 Pounds, CAS off

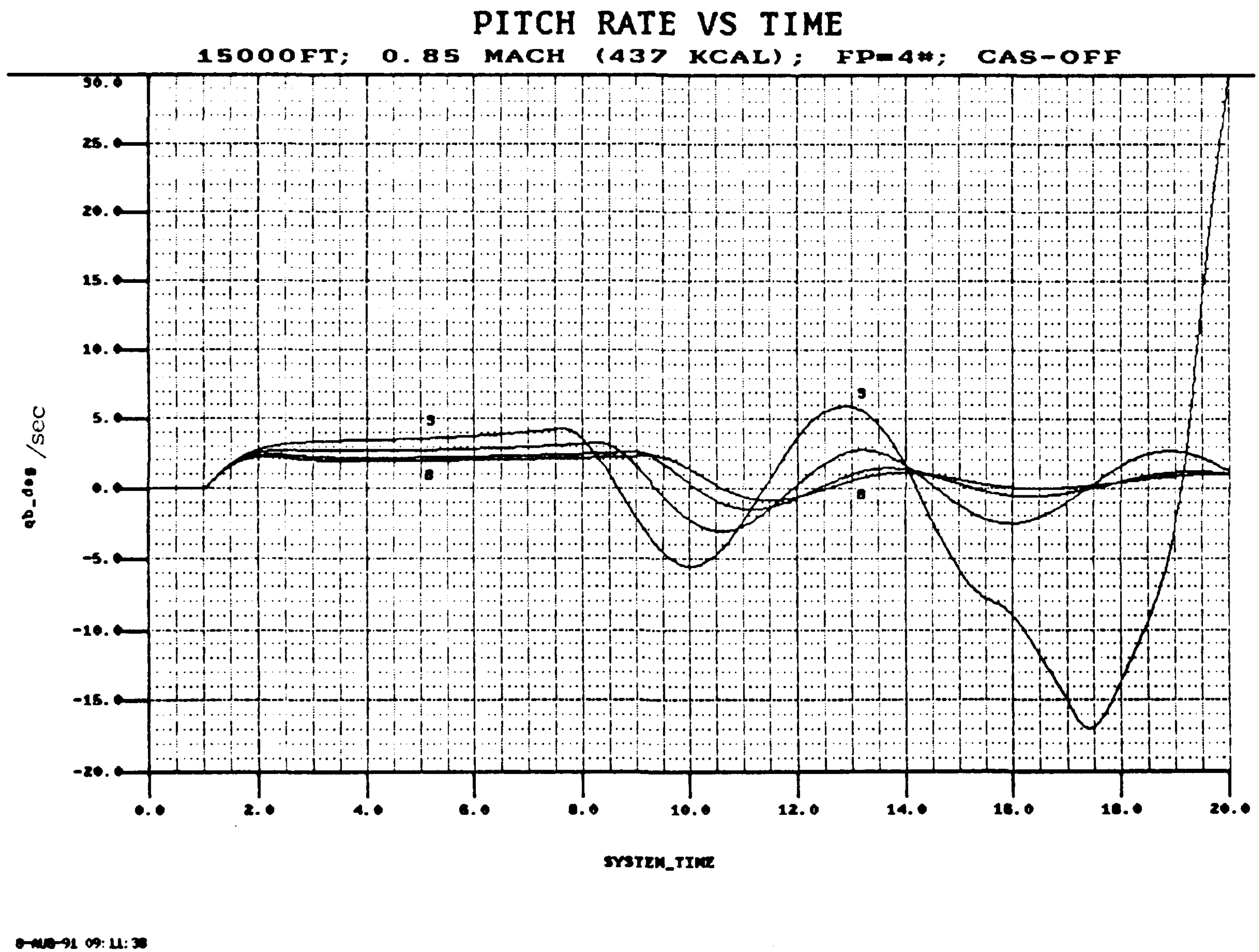

Figure 10. Pitch Rate vs Time at 15000 ft, .85 Mach, Pitch Force=4 Pounds, CAS off

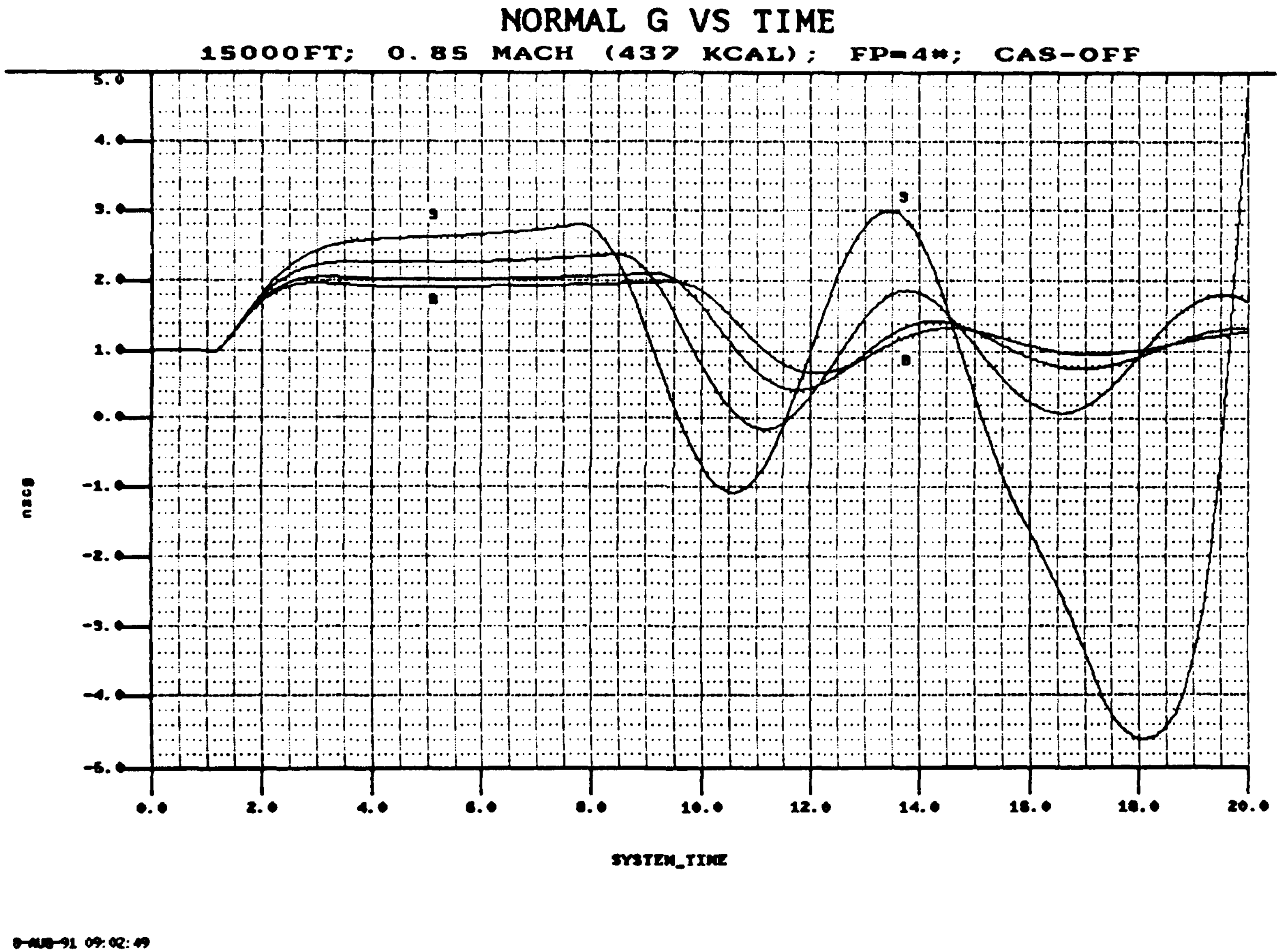

Figure 11. Normal G vs Time at 15000 ft, .85 Mach, Pitch Force=4 Pounds, CAS off

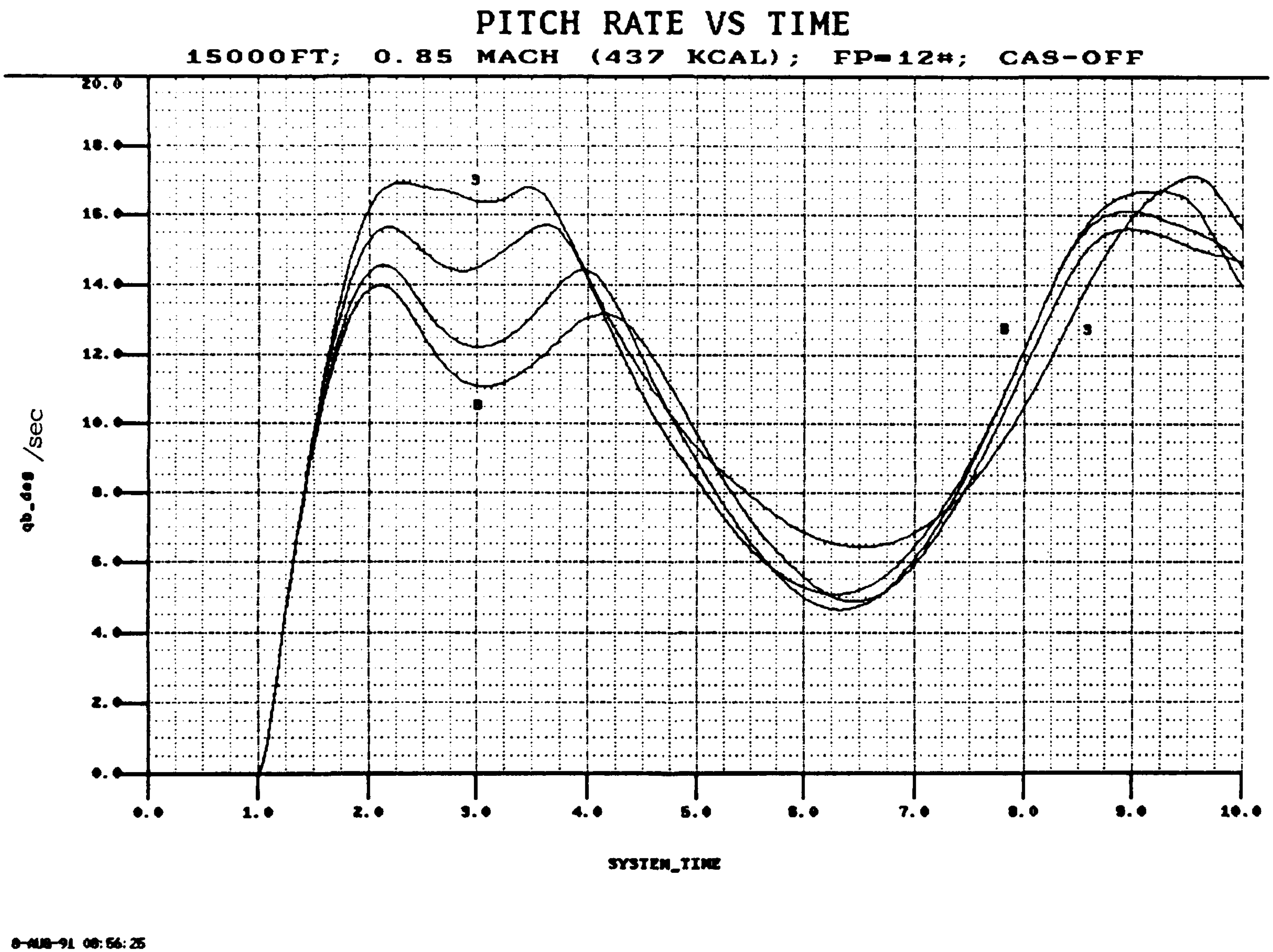

Figure 12. Pitch Rate vs Time at 15000 ft, .85 Mach, Pitch Force=12 Pounds, CAS off

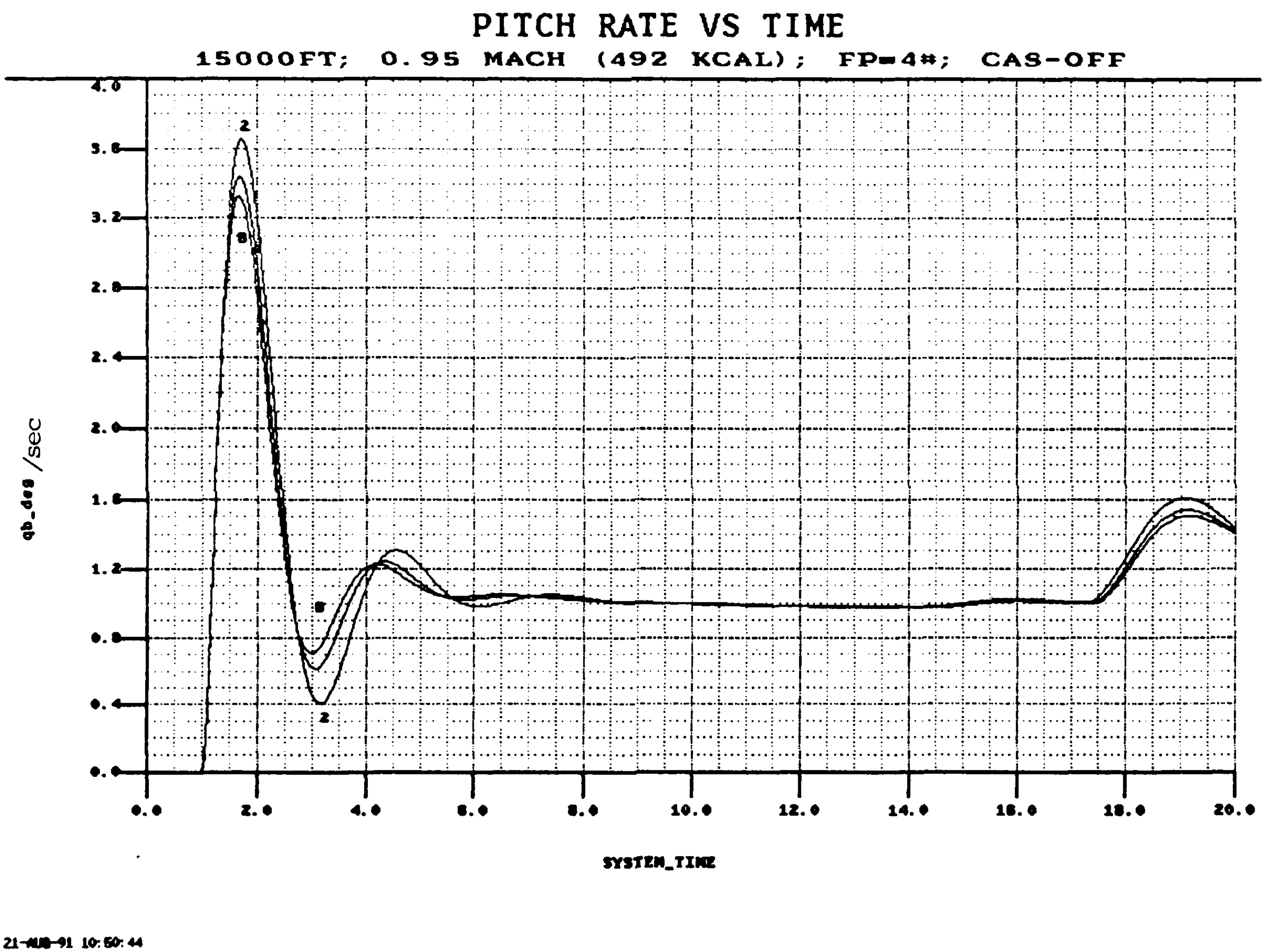

Figure 13. Pitch Rate vs Time at 15000 ft, .95 Mach, Pitch Force=4 Pounds, CAS off

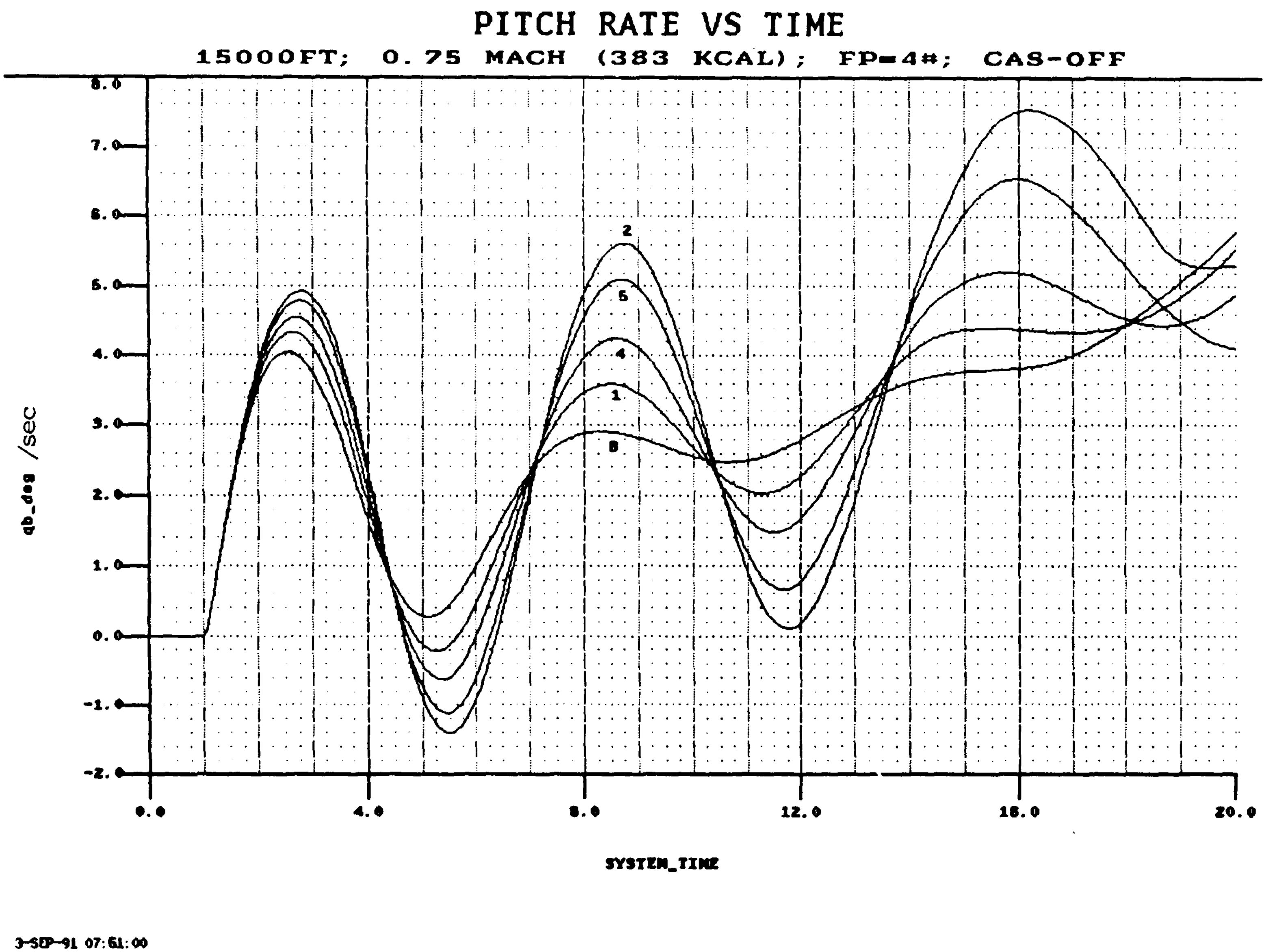

Figure 14. Pitch Rate vs Time at 15000 ft, .75 Mach, Pitch Force=4 Pounds, CAS off

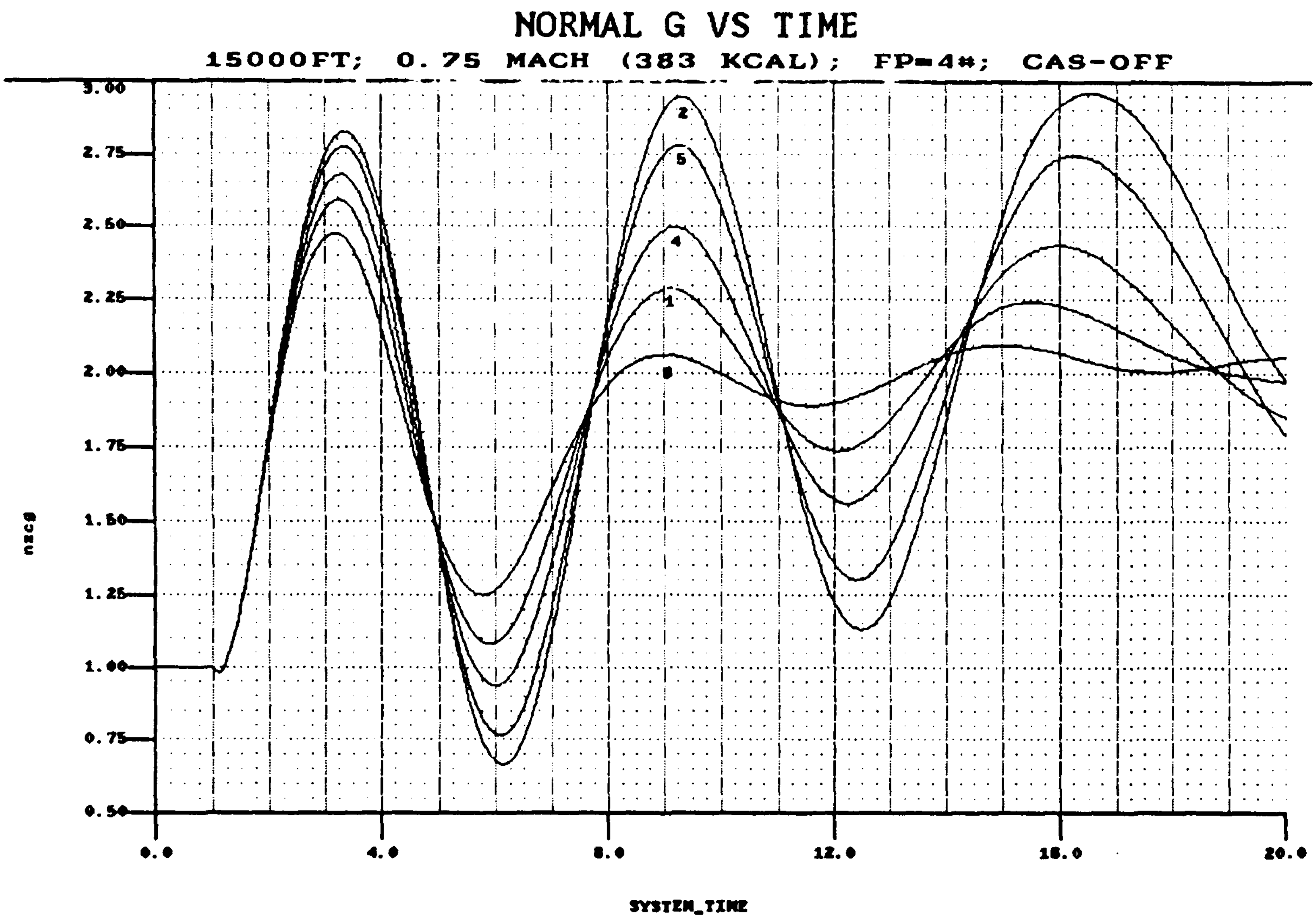

Figure 15. Normal G vs Time at 15000 ft, .75 Mach, Pitch Force=4 Pounds, CAS off

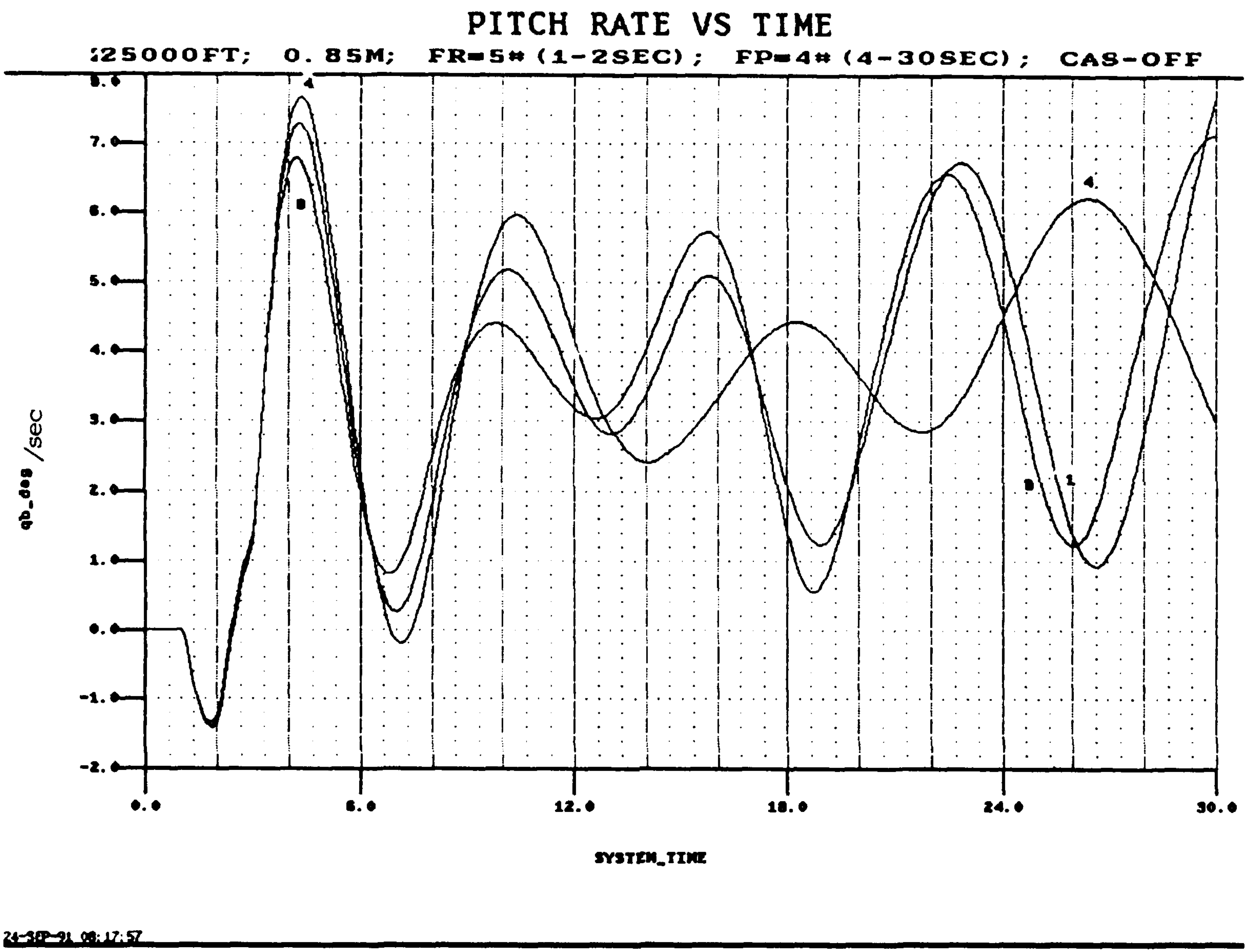

Figure 16. Pitch Rate vs Time at 25000 ft, .85 Mach, Roll Force=5 Pounds for 2 Seconds, Pitch Force=4 Pounds from 4 to 30 Seconds, CAS off

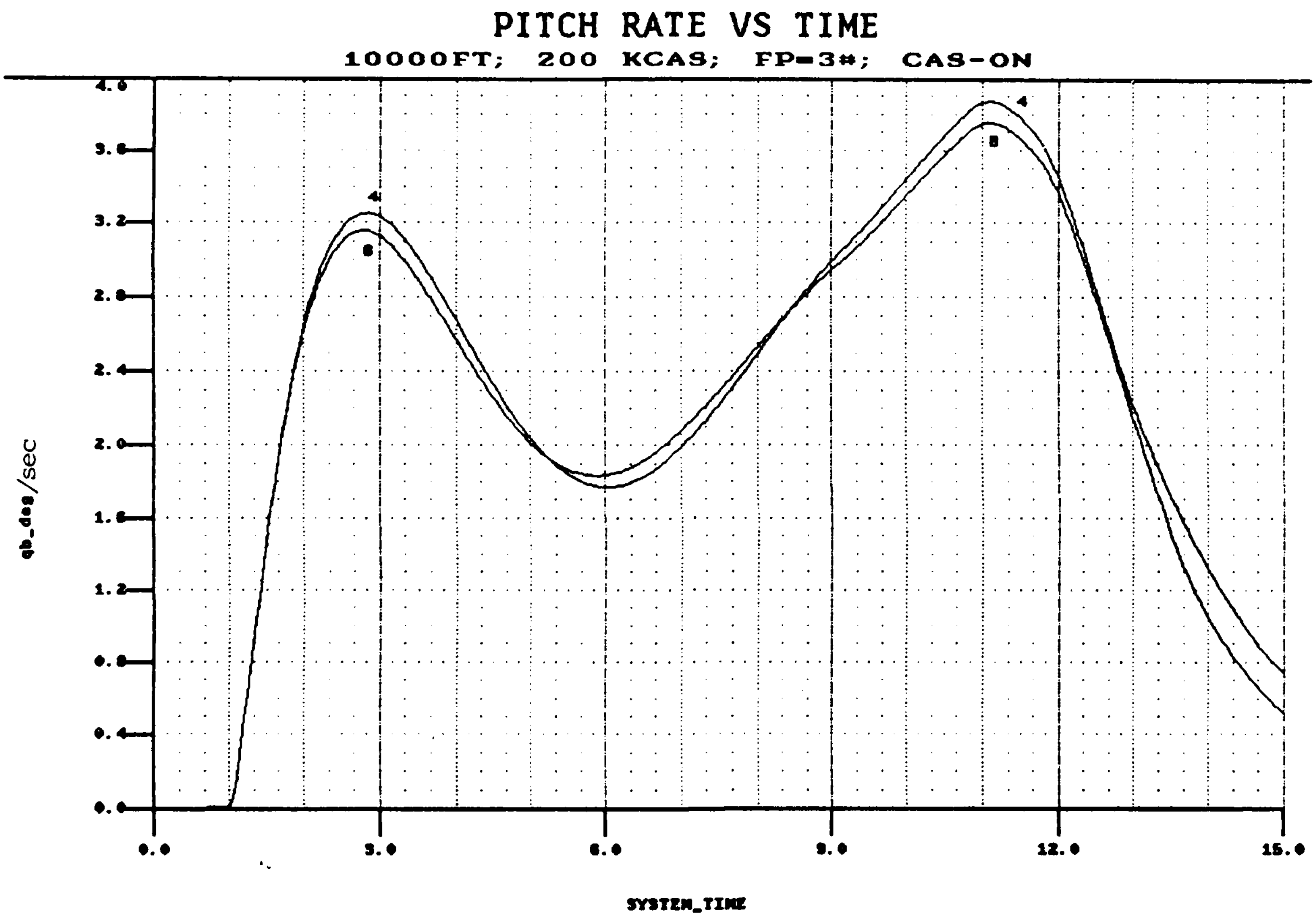

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Figure 17. Pitch Rate vs Time at 10000 ft, 200 KCAS, Pitch Force=3 Pounds

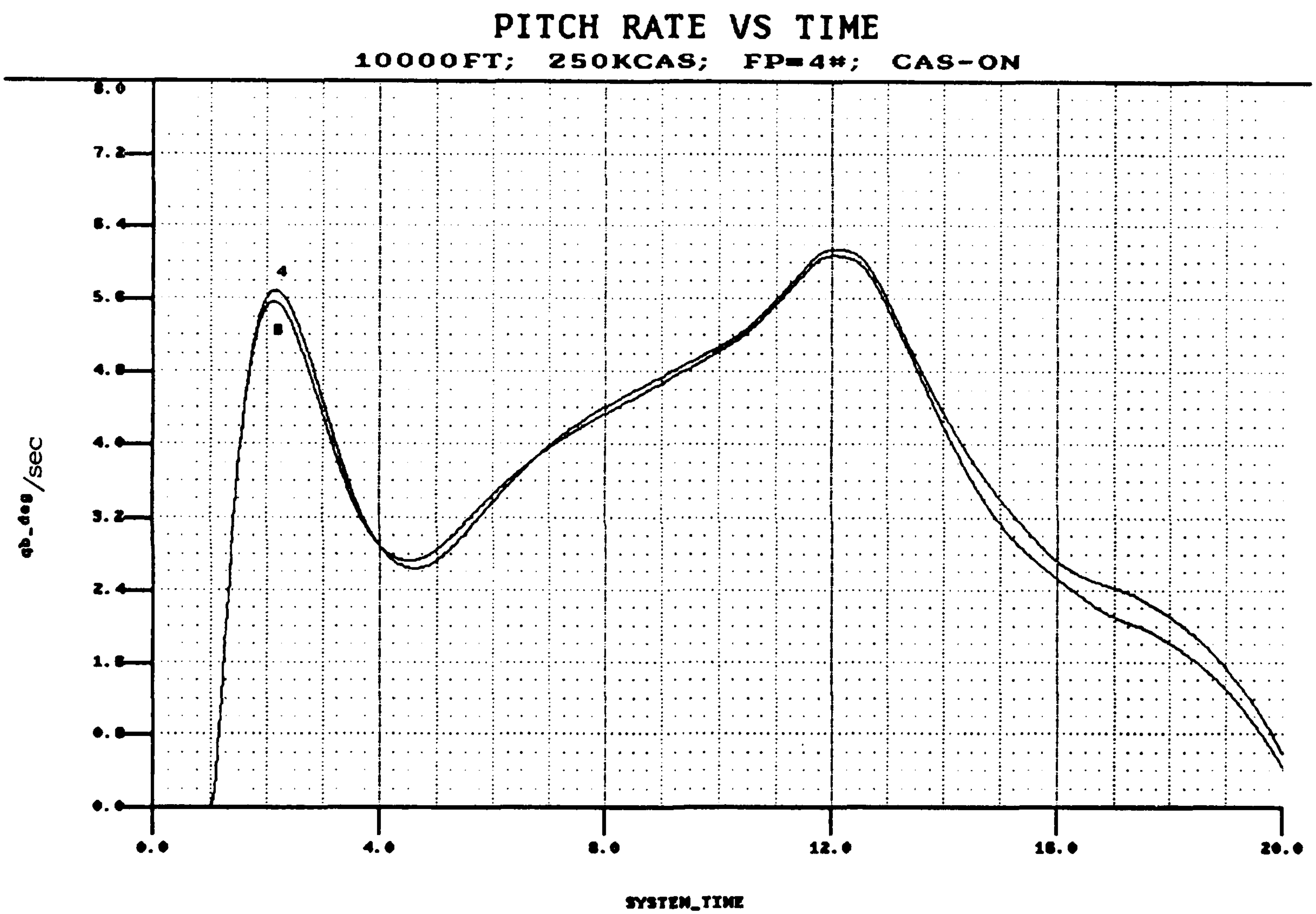

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Figure 18. Pitch Rate vs Time at 10000 ft, 250 KCAS, Pitch Force=4 Pounds

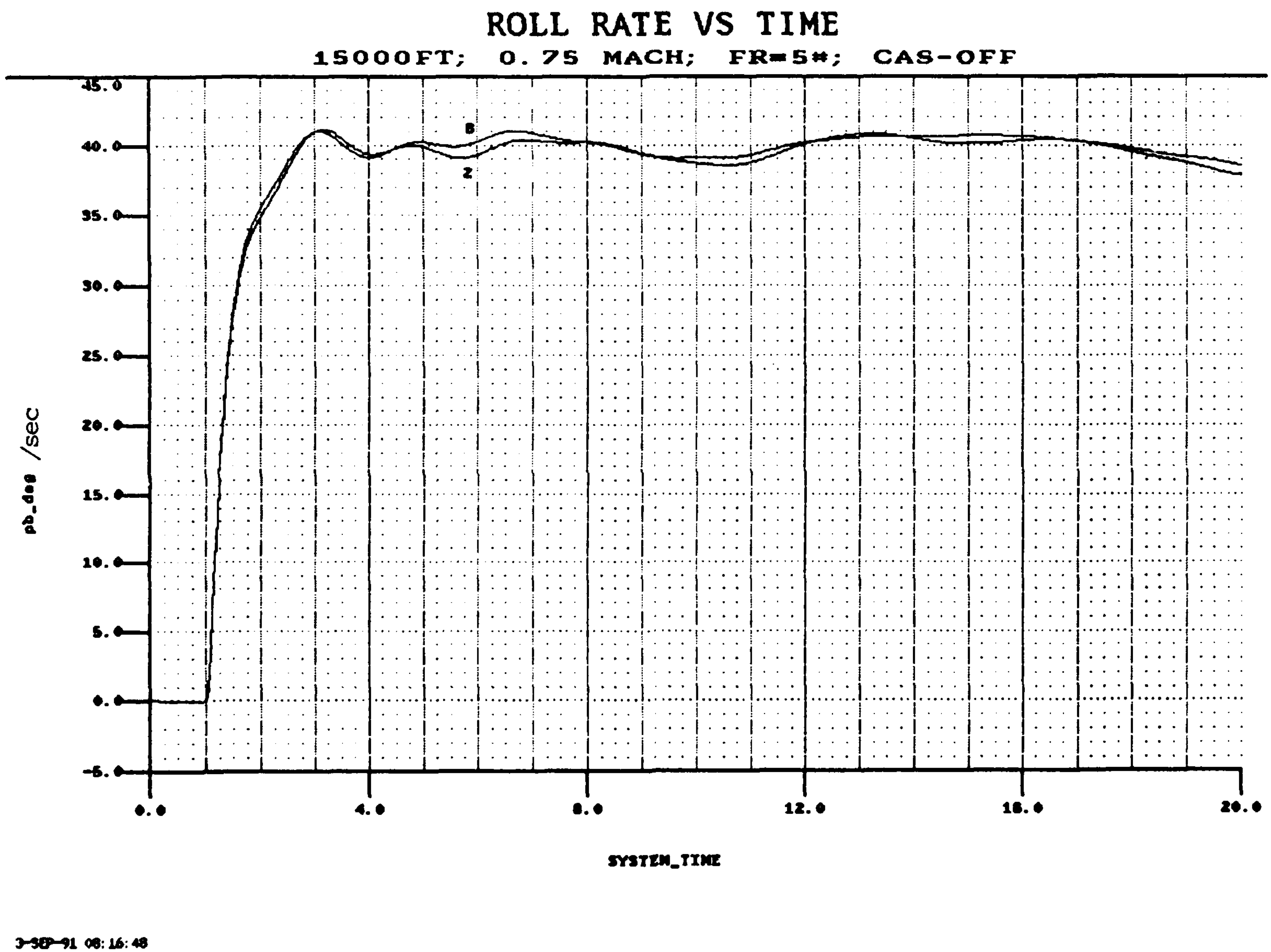

Figure 19. Roll Rate vs Time at 15000 ft, .75 Mach, Roll Force=5 Pounds, CAS off

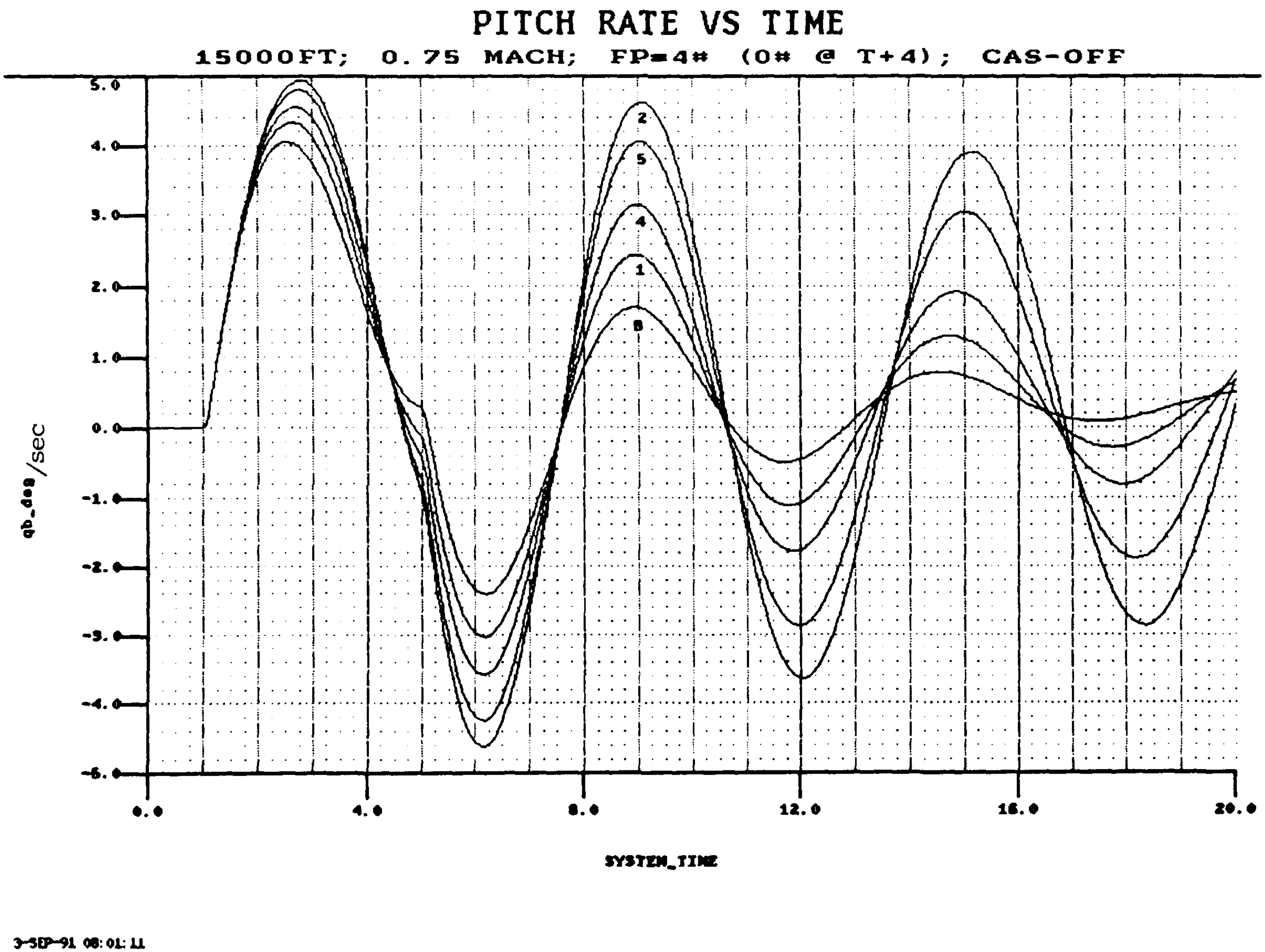

Figure 20. Pitch Rate vs Time at 15000 ft, .75 Mach, Pitch Force=4 Pounds for 4 Seconds

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Vita

Major Richard L. Bennett was born on 29 May 1957 in Irwin, Pennsylvania, and graduated from Norwin High School there in 1975. He then graduated from the U.S. Air Force Academy in 1979 with a Bachelor of Science degree in Astronautical Engineering, and a second major in Engineering Sciences. Receiving a regular commission in the USAF, his first assignment was pilot training. Upon completion of Undergraduate Pilot Training in June 1980 at Columbus AFB, Mississippi, he was assigned as an F-15 aircraft commander to the 1st TFW, Langley AFB, Virginia. Following that assignment, he became an F-15 RTU Instructor Pilot with the 461 TFTS at Luke AFB, Arizona in 1983. He was then selected to attend the USAF Test Pilot School at Edwards AFB, California, in June 1985. Upon completion of the Test Pilot School in 1986, he was assigned as an experimental test pilot to the 3246 Test Wing at Eglin AFB, Florida, where he flew F-15, F-15E, and A-10 aircraft. He was also the chief F-15 test pilot for the Advanced Medium Range Air-to-Air Missile (AMRAAM) from July 1987 to May 1990. In May 1990 he entered the School of Engineering at the Air Force Institute of Technology.

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13. ABSTRACT *(Maximum 200 words)*
This study investigated the results of removing lead ballast from the nose section of the F-15 Air Superiority fighter. The goal of the investigation was to determine if aircraft handling qualities remained acceptable with the ballast removed, and also to determine what improvements in aircraft nose pointing authority resulted. Actual F-15 weight reports were used to calculate the worst case aft center of gravity location shift due to the ballast removal. Several configurations with different center of gravity locations (based on various amounts of lead weights removed) were used for comparison to the baseline aircraft. Moments of inertia were calculated for each configuration, which in turn were used in a 6 degree of freedom computer simulation of the F-15. Simulation test points were then examined throughout the flight envelope of the F-15. Simulation results and better aircraft weight management results support removing (on average) approximately 200 pounds of lead ballast from the nose section of the single seat Air Superiority F-15 Eagle, with a resulting 3 percent increase in pitch rate. A suggested flight test profile is presented for flight verification of the simulation results.

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