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ADVANCED ELECTROMECHANICAL ACTUATION SYSTEM (EMAS) FLIGHT TEST

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The Advanced Electromechanical Actuation System (EMAS) flight test project successfully demonstrated, for the first time, the electrical actuation of a primary flight control surface in flight. This test was a major step toward the realization of the All-Electric Airplane (AEA) concept. An electric actuator was installed in a modified C-141A aircraft to power the left aileron. Testing included ground and flight trials to ensure unchanged control system damping. Aircraft roll performance tests included maximum effort rolls, degraded system rolls, and autopilot rolls. Sideslip and trim test points were also performed. It was verified that EMAS performance was similar to the normal hydraulic actuator. Results include lessons on aircraft modification, general system characteristics, maintenance factors, and compatibility with other aircraft systems that may influence future installations. Keywords,

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SUMMARY

The purpose of this test was to demonstrate in flight for the first time the feasibility of powering a primary flight control surface with an electromechanical actuator (EMA) in place of the standard hydromechanical actuation system. An advanced development model model EMA drove the left aileron on a specially modified C-141A aircraft and was to duplicate the functions of the standard hydromechanical unit. Lockheed-Georgia Co. (GeLac) was the prime contractor with Sundstrand Corp. as the subcontractor.

The Electromechanical Actuation System (EMAS) consisted of a dual motor electric actuator mounted in the aileron actuator bay, and associated power supply and logic electronics constituting the dual channel Controller Electronics (CE) unit located in the cargo bay of the aircraft. Both channels were as physically and electrically isolated as possible. The system obtained its power from two separate electric buses aboard the aircraft. A single EMAS channel could power the actuator, though with reduced capability. The EMAS incorporates extensive built in test and fault detection circuitry that can automatically shut down a faulty channel(s). Operating procedures of the EMAS were identical to that of the normal system.

The test aircraft was modified at the 4950th Test Wing to accommodate the EMAS, including electrical and hydraulic changes necessitated principally by the requirement to maintain electrical power to the system or to select backup tab operation of the aileron under all conceivable emergency situations. EMAS and aircraft instrumentation permitted the monitoring of system performance, aircraft response, and a one-for-one comparison with the unmodified right aileron control system.

The testing included ground trials to ensure readiness for flight and a ground vibration test (GVT) to determine the dynamic frequency response of the system and to verify unchanged damping characteristic. The flight test began with a damping investigation to clear the test envelope. Subsequent trials consisted of roll performance and degraded systems tests.

Ground and flight tests showed that the damping of the modified system was unchanged from the baseline configuration. A system modification was required to eliminate a system instability experienced during the GVT. The instability was characterized by a neutrally-damped aileron oscillation which was attributed to an aileron control system characteristic.

During the switching from ground to aircraft power sources, a single EMAS channel often shut down. This may indicate a sensitivity to momentary losses of power. Also in ground trials, a channel repeatedly shut down during maximum command aileron control cycles because of the drifting of electrically set actuator travel limits. This drifting may have been caused by large ambient temperature variations or travel limit overshoot in the absence of airloads, and is undesirable. Rigging adjustment eliminated the shut downs.

The normal hydraulic actuator and the EMAS produced sudden aileron movement during initial power up when the surface deflection was not in the proper

position relative to the yoke. This created a potential hazard to maintenance personnel. The nature of the EMAS may provide the opportunity to eliminate this ground hazard. Higher backdrive forces for the inert EMA as compared with the inert hydraulic actuator increased the forces required to move the unpowered aileron and presented an increased maintenance task. It was possible to position the aileron by turning the ball nut that moved the EMAS actuator ram. This was advantageous to maintenance personnel. When temperature dropped to approximately below freezing, manual actuator movement was not possible.

Except for a few discrepancies, the flight test demonstrated that the EMAS duplicated the hydromechanical aileron actuator performance, within the measurement capability of the test instrumentation. Because of off-the-shelf component inadequacies, the EMAS could not meet the normal travel limits of the C-141 aileron, falling short by one degree at each end-of-travel. The EMAS end-of-travel slow-down feature resulted in the aileron requiring an average of 0.3 seconds more to reach the travel limit than the unmodified system. These two discrepancies resulted in different roll rates during maximum command rolls to the left and right and inability to match limited baseline performance data. These discrepancies were not detectable by the pilots.

Single channel operations were similar to full system operation and no aileron movement was experienced during channel deactivation and activation. Because of the higher forces required to move the aileron against an inert EMA, the unpowered left aileron floated to a smaller deflection than the unpowered hydraulically-actuated right surface, although backup tab operable performance was not reduced from baseline.

A maximum current draw by the EMAS of 12.5 amps was observed, well within the 50 amp excess capacity of the electrical buses. The EMAS was observed to be more sensitive to inputs than the hydraulic actuator, making more and larger fine adjustments during autopilot trim maneuvers and reproducing small control system aberrations. The EMAS responded to autopilot inputs without difficulty. No motor or actuator bay temperature gradients were observed at any time during ground and flight operations

85% of the planned flight test was successfully completed. The testing was terminated on the sixth sortie because of an EMAS failure. One channel repeatedly disengaged in flight because of a current imbalance between the motors, and the aileron exceeding the normal displacement limits. During subsequent ground testing, the actuator exhibited similar faults and also executed uncommanded deflections at higher than normal maximum travel rate. The last of these uncommanded excursions passed the normal travel and electrical limits, and resulted in the mechanical stops being damaged (not designed to withstand such a load). The cause of these faults could not be determined and corrected in time to complete the flight test within schedule constraints.

The EMAS flight test demonstrated the feasibility of powering a primary flight control surface with an electromechanical actuator and tailoring it to specific performance requirements. It showed that the installation of such a system into a pre-existing airframe can be effected without structural modification and with only minor electrical changes.

PREFACE

This flight test constituted the first use of a electromechanical actuator for a primary flight control surface, and represented a major step toward the realization of the all-electric airplane concept.

This flight test was performed under project number 240306TP. The test was part of a program sponsored by the Air Force Wright Aeronautical Laboratories (AFWAL) in response to an unsolicited proposal from the Lockheed-Georgia Co., with the Sundstrand Corp. acting as the subcontractor. The test aircraft, NC-141A, serial number 61-2775, was modified by the 4950th Test Wing and flown for 12.9 hours in 6 sorties, all of which were launched and recovered at Wright-Patterson AFB, Ohio.

The author wishes to acknowledge the contributions of the aircrew; Project Pilot Capt Samuel Kinard, Test Pilot Capt Larry Schultz, Pilot Maj Wayne Stanberry, and Flight Engineers MSgt Peter Van Havermat, TSgt Kenneth Hauprich, MSgt Joseph Keck, SSgt Duane Smith, and CMSgt Donald Turner. Also, a special note of thanks for test planning and modification assistance is extended to Flight Engineer TSgt Stephen Broander. Thanks are extended to the instrumentation personnel, Mr Bill Benedict and SSgt David Jankowski, and for data reduction in the person of Mr Hobart Drum. A special thanks to Ms Lydia Flaugher and all the fine engineers at the Special Programs Division of the Directorate of Aircraft Modification. A note is due Mr Faustino Zapata for his advice on structural analysis. Thanks to the contractor personnel who spent many long days on the project; Ken Thompson, Ralph Alden, and Mark Bailey of Lockheed-Georgia, Brent Kaiser, Fenton Reese, and Graham Bradbury of Sundstrand. Lastly, thanks to the Program Manager, Capt Larry Hunter, who suffered through it all from the beginning. Accolades also to all the other unnamed contributors in the Program Office, Test Wing, and contractors organizations.

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INTRODUCTION

1. Background

a. The Advanced Electromechanical Actuation System (EMAS) flight test program demonstrated for the first time the performance of an electromechanical actuator (EMA) while powering a primary flight control surface on a C-141A aircraft. The EMA had been designed to replace the existing hydromechanical actuator system with a power-by-wire system. When combined with its natural adjunct, the fly-by-wire technology, this will be a critical step toward realization of the all-electric airplane (AEA) concept. The principal benefits of EMAS and an all-electric airplane are anticipated to be reduced component weight, greater reliability, improved maintainability, reduced logistics, better redundancy management, increased safety, and a significant reduction in life cycle costs. Among factors that influenced the selection of the C-141 aircraft as the testbed airframe was that there is sufficient space within the C-141 actuator bay to accommodate the EMA unit. The C-141 has a backup tab operable control system for aileron actuation which enhanced flight safety in the event of a total EMAS failure. Operational incidents have demonstrated that the aircraft can be safely recovered and landed with an aileron hardover failure. And, the test aircraft was readily available.

b. NC-141A, serial number 61-2775, was modified by the 4950th Test Wing for the EMAS flight test between July and September 1985 at Wright-Patterson AFB, Ohio. Testing included a ground phase to ensure flight safety and system preparedness followed by the flight phase. The flight phase consisted of an airworthiness portion followed by roll performance tests. The aircraft was flown for 12.9 hours during six sorties in February 1986. All phases of the test were conducted at Wright-Patterson AFB. The 4950th Test Wing project number was 240306TP.

2. Objectives

This project was to demonstrate, in-flight, the feasibility of powering and controlling a primary flight control surface with a dual channel redundant electromechanical actuator, document the performance of the actuator, and confirm that the EMAS installation could be implemented safely and efficiently, consistent with the performance requirements of the control surface.

TEST ITEM AND INSTALLATION

1. EMAS Technical Description

a. The advanced development model EMA (see Figure 2-1) weighed 65 pounds (lbs), six lbs heavier than the conventional hydraulic power control unit (PCU). This difference was caused primarily by an assembly that interfaces the single EMA ram with the existing dual ram connection, and the use of commercially available components and machined rather than cast hardware. The contractor predicted that a production version would weigh less than the hydromechanical unit. The EMAS dimensions are comparable to those of the PCU, fitting the actuator bay without structural modification. EMAS was designed to perform all of the functions of the original hydraulic actuator without altering the performance or stability and control of the aircraft. A listing of detailed EMAS technical data is given in Appendix D. The EMAS was a redundant dual channel unit (channel A and channel B) employing two electric motors, a gear train, and a linear ball screw assembly. A single motor is capable of effectively moving the aileron. The controller electronics (CE) provided the power to the EMA motors and processed all EMA signals, transmitted via a wire bundle between itself and the aileron actuator bay. The CE is contained in a double-bay Johnson rack (see Figure 2-2); however, a production unit would be reduced to a small box.

b. The CE required 115 VAC-3 phase and 28 VDC power. AC power was obtained from the aircraft number two main and number two essential buses, powering the A and B channels, respectively, and converted at the EMAS rack to direct current (up to 270 VDC) to power the actuator and other power for electronic requirements. The 28 VDC originated at the pilot overhead left aileron switching panel and serves to turn on the EMA channels when the switches are in the NORMAL position by holding closed a relay between the CE power supply and the actuator. These switches (one for each hydraulic or EMAS channel) have NORMAL (powered), OFF (unpowered), and TAB (tab operable) settings and served the same functions with the EMAS as they do for the hydraulic PCU.

c. Aileron deflection commands were transmitted to the actuator by way of the normal aircraft mechanical control system consisting of cables, pulleys, bellcranks, pushrods, and linkages (see Figure 2-3 and 2-4). The feedback linkage normally used to position the hydraulic actuator control valve displaced two input or position error Rotary Variable Differential Transformers (RVDT), one for each channel, through the override spring and input arm (see Figure 2-1). When displaced, the RVDT produced an electronic command interpreted by the CE. The CE supplied a corresponding amount of voltage necessary to produce the required motor torque (see Figure 2-5). The power is supplied until the surface reaches the commanded position which mechanically returns the RVDTs to the null position, thus stopping the power for motor torque.

d. Prior to the test, it was understood that deficiencies in off-the-shelf components resulted in the EMA being unable to meet expected maximum aileron travel limits. In order to provide a sufficient aileron over-travel safety margin above the normal C-141 aileron travel limits of 25 degrees (deg) trailing edge up and 15 deg down, travel was limited to 24 and 14 deg with the normal travel schedule maintained up to these points (see Figure 2-6). System damage

was prevented by electronically restricting deflections from exceeding this control surface travel limit by more than one degree, and with actuator command reduced to 10-15 percent of maximum deflection rate beginning at approximately two degrees from the limits. This slow-down feature also worked in reverse, reducing aileron motion as it came off of the limit. System disengagement occurred when the actuator reached the electrical stops. Mechanical stops are in the form of two pair of metal tabs that butt at maximum extension or retract deflection of 26 and 16 deg aileron travel. Additional aileron position RVDTs provided information for the over-travel limit function.

e. Both EMA motors operated simultaneously along a single drive screw, producing equal loads, unless either had been selected OFF or had been automatically shut down as a result of a fault sensed by the CE. One motor was sufficient to perform actuator functions, though with degraded roll rate because of limited load, deflection travel, and deflection rate capability. Motor torque was transmitted via the gear train and linear ball screw to produce surface deflection.

f. The EMAS incorporated fault detection and response functions for system safety. The CE shut down the affected channel when a thermal switch incorporated into the motor windings closed at 400 deg Fahrenheit (F) or when the motor drive power transistors' heat sink exceeded 250 deg F. The two EMA channels were monitored for force fight which would be reflected in a current difference. An excessive difference would result in channel A being disabled as an arbitrary selection to eliminate the imbalance. If channel B created the fault that produced the imbalance, that fault would then be detected and B shut down. In another fault detection, the affected channel is disengaged if a voltage of 200 VDC is supplied for two seconds or more (indication of a runaway motor) since only one second was required for the actuator to stroke from full retract to full extend, or vice versa. A disengagement of both EMA (or PCU) channels would result in the aileron floating to approximately 5-8 deg trailing edge up deflection, a freestream or zero hinge moment condition, prior to engagement of the tab operable system. 360 lbs of force was required to backdrive the screw against inert motors (both channels disengaged), equating to approximately 60-70 lbs of force at the aileron trailing edge. Other fault modes include component failure detection such as RVDT and motor resolver. Many of these faults were identified by indicators in the front panel of the CE. If a fault resulted in the channel being disengaged, only a power recycle (pilot switches) would reengage the unit, provided the fault was not still present. Each motor and CE channel was physically and electrically isolated to the greatest extent possible to ensure that a fault disabling one channel did not affect the other. The failure of one of the aircraft electrical sources or EMAS internal power supplies, such as a voltage surge to 325 VDC or sudden drop to 240 VDC, would result in the loss of the affected channel.

g. The EMA had undergone extensive laboratory testing and computer simulation and was certified as airworthy. As with the hydraulic PCU, a failure of either EMAS channel was indicated by a flashing MASTER CAUTION light and a failure light adjacent to the affected channel on the aileron power control switching panel. Power to the EMA was controlled by the aileron power control switches in the cockpit. Power to EMAS was controlled by two circuit breakers, one for each channel. In the event of a dual motor failure, EMA shut down, or a total aircraft electrical system failure the aileron could be operated by tab.

h. The EMAS had redundant fault monitoring for an aileron hardover failure. Such a failure would require a fault in both motors allowing runaway motors. A single motor runaway would be opposed by the second motor creating a force fight. This force fight would be detected as a current difference and result in an EMA shutdown. Travel limits and high motor drive voltage for an excessive period of time would also result in a shutdown prior to the aileron reaching the electrical or mechanical travel limits. Therefore, multiple faults had to occur simultaneously for a hardover to result. In the event of a hardover, a single motor was sufficient to return the aileron from the hardover position or be driven back by aerodynamic forces. Depowering EMAS and reverting to tab operable was the standard emergency procedure. Depowering would result in the aileron floating to a zero hinge moment condition.

2. Aircraft Modification

a. The EMA was designed as a "drop-in" replacement for the existing C-141 aileron actuator, requiring no structural modification to the aircraft (see Figure 2-7). It was compatible with the manual tab operable aileron system for emergency backup, and required no change to the autopilot, roll trim, or roll artificial feel systems. It utilized the existing flight control systems and monitoring functions and required no change to normal operating procedures. A modification profile is provided in Figure 2-8.

b. Sections of the unused PCU hydraulic lines in the actuator bay were removed and capped to provide room for the EMA. In order to permit backup tab operation of the left aileron in the event of total electrical system failure, the tab operable solenoid valve was connected to the Number 2 hydraulic system in place of the Number 3 system (see Figure 2-9). This valve ports hydraulic fluid to a tab operate cylinder, unlocking the tab lockout actuator and moving a tab input bellcrank from an over-center position to permit control commands to deploy the tab and assist in moving the aileron with aerodynamic forces. The Number 3 hydraulic system is electrically dependent whereas the Number 2 system is engine driven and would continuously provide pressure to the solenoid in the event of an electrical failure. In addition, a relay between the emergency DC bus and the isolated DC bus was installed to ensure an uninterrupted source for the 28 VDC power used to hold open the solenoid valve, thus ensuring that this function could be performed in the event of a total electrical system failure or if the crew should need to shut down the isolated bus as part of the electrical fire checklist procedure. A calibrated outside air temperature probe was added to the aircraft to provide a precise source for this parameter.

c. The EMA CE was placed in a double-bay "Johnson" rack. Electrical wire bundles from this rack and instrumentation wiring to the aileron actuator bays exited the pressurized cargo compartment through the life raft compartment inspection window apertures which were modified to provide a positive pressure seal. Life rafts were removed as a result of this modification.

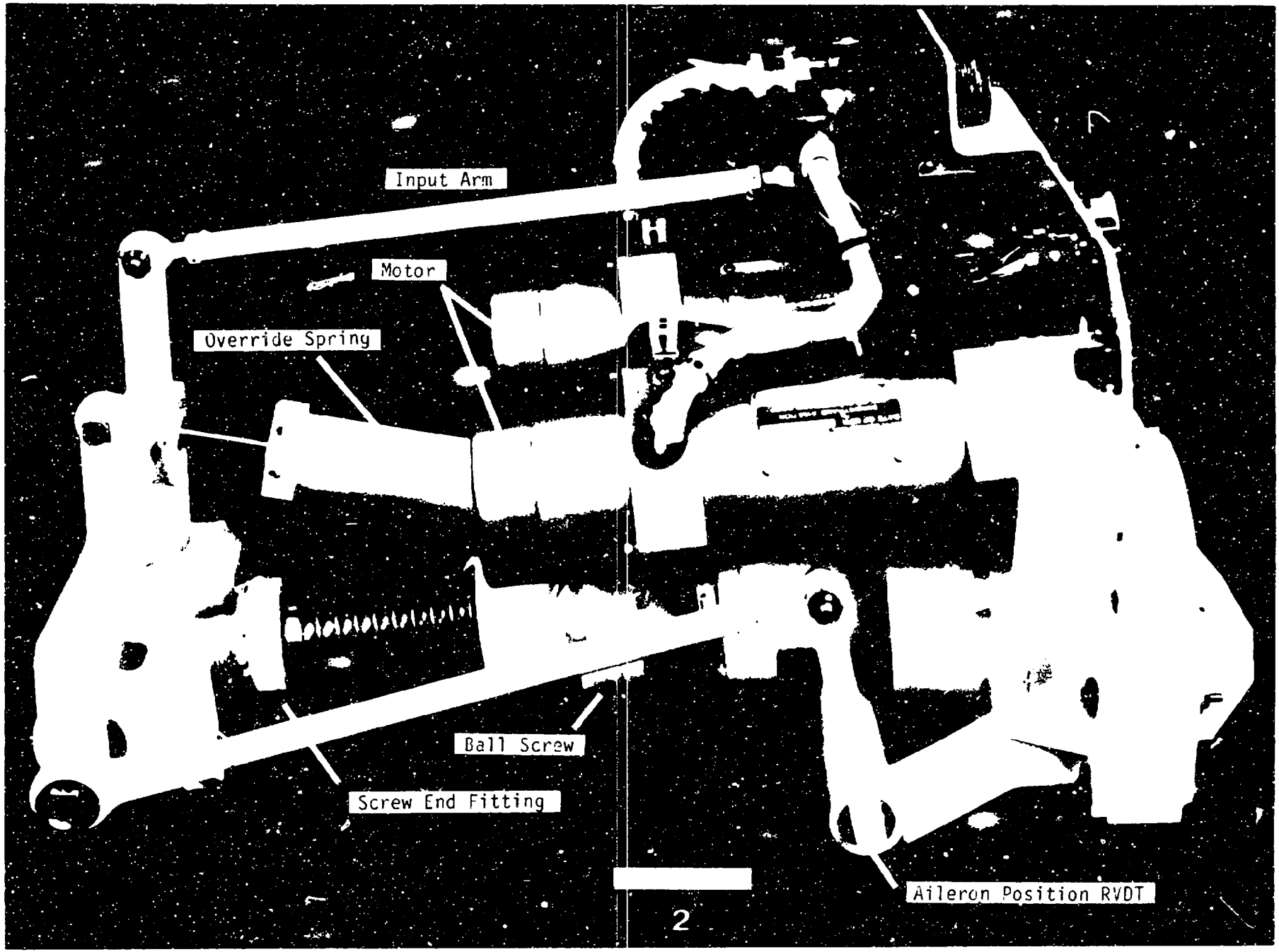


Figure 2-1 Electromechanical Actuator (EMA)

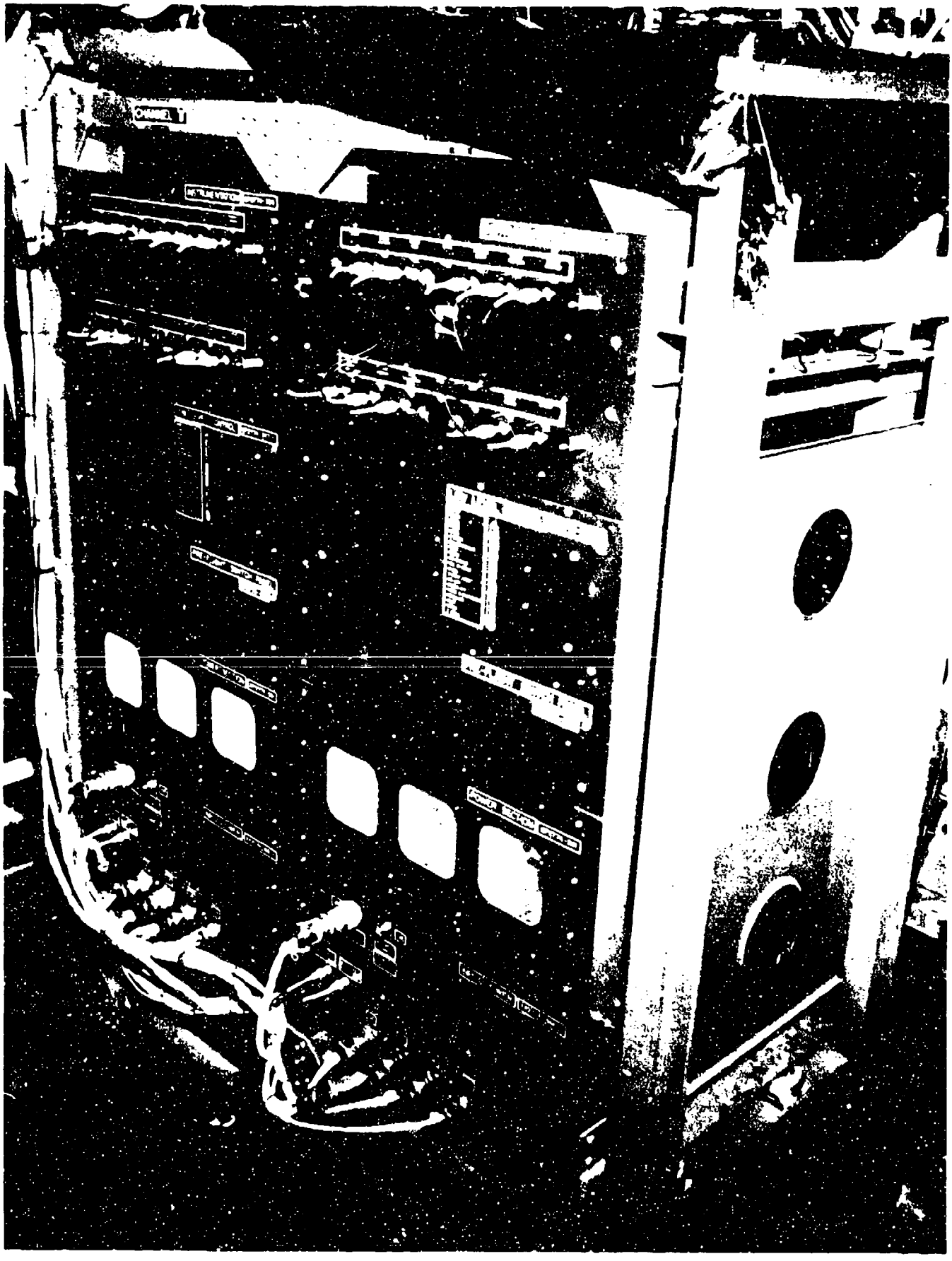


Figure 2-2 EMAS Controller Electronics

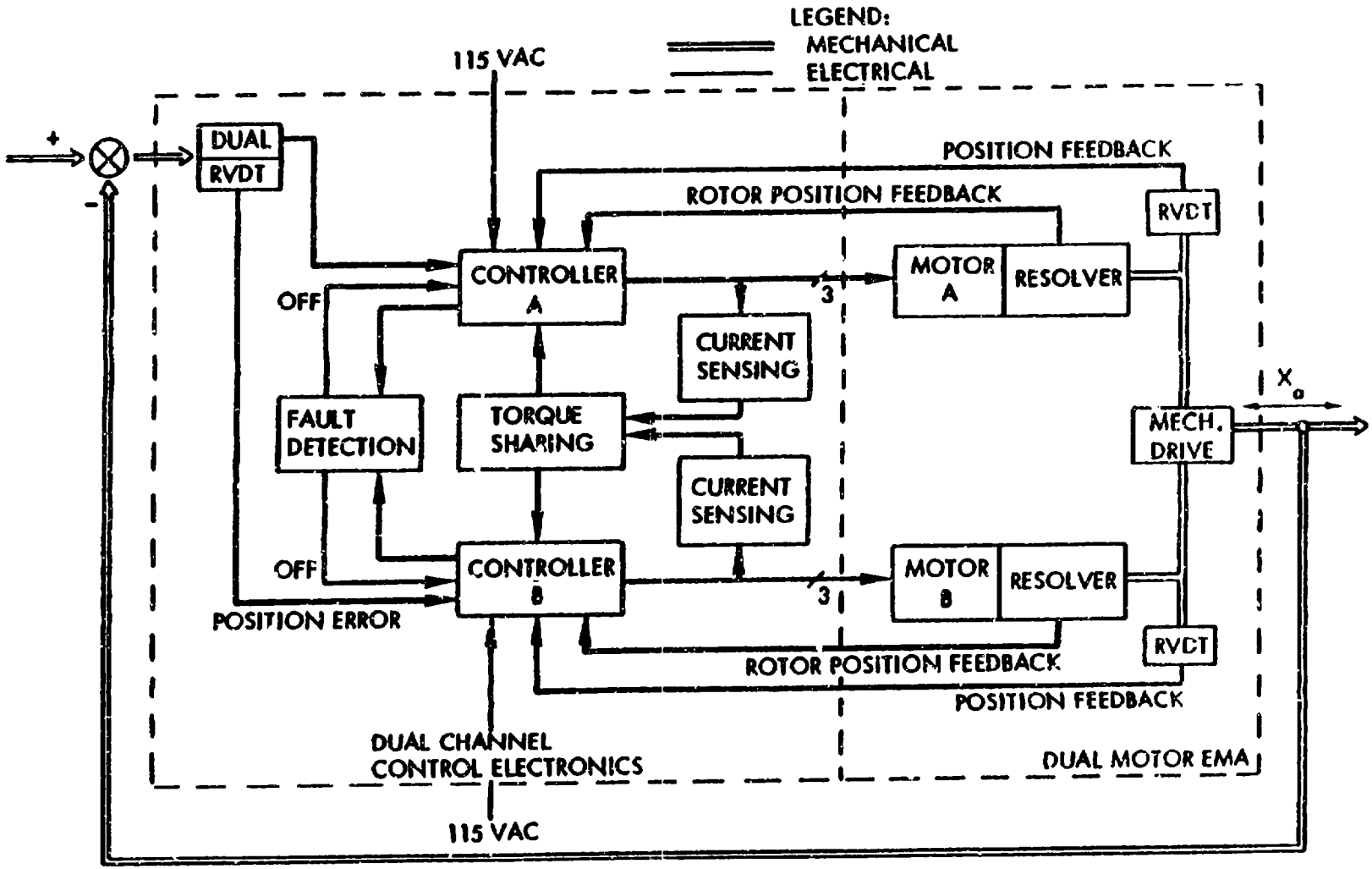
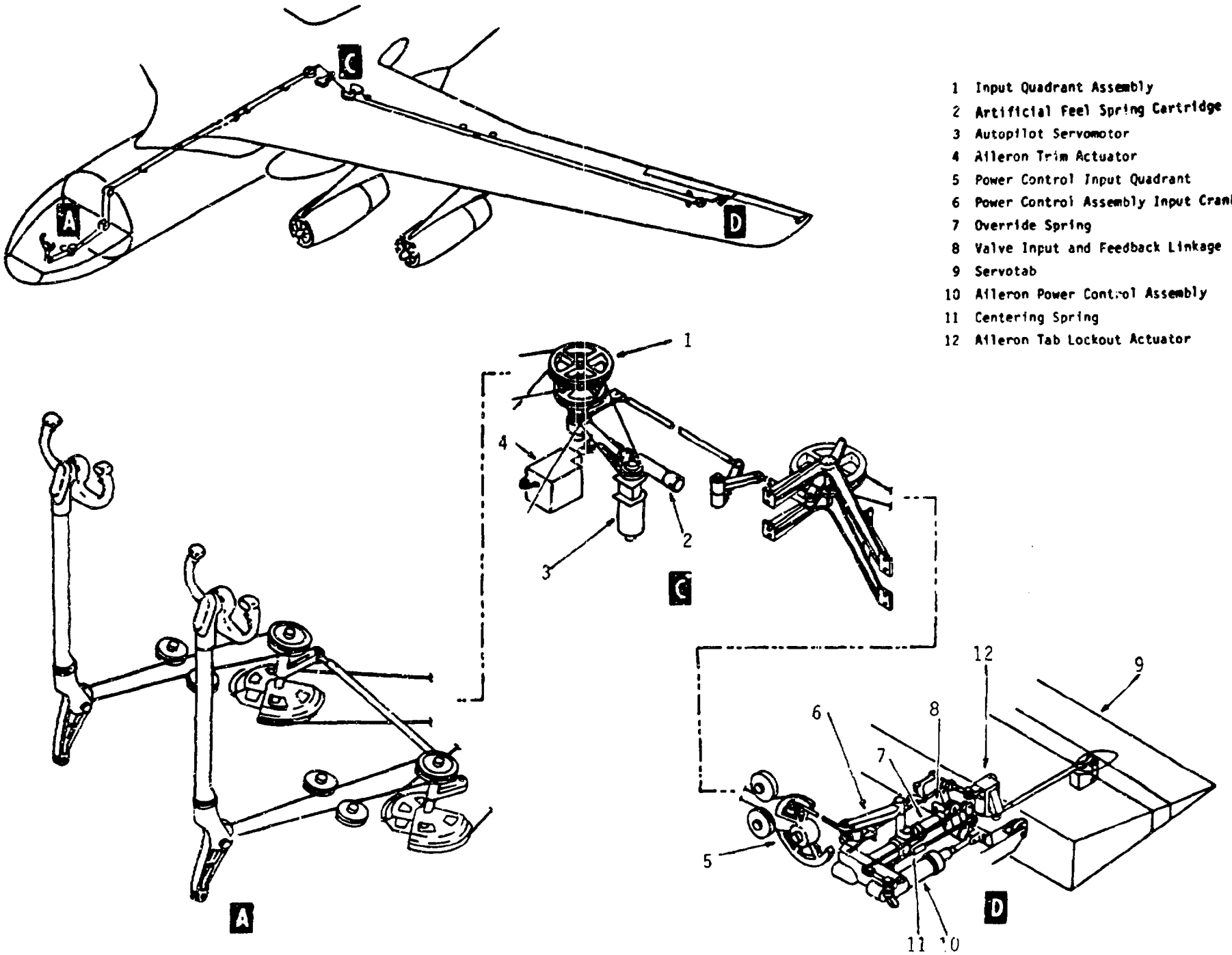


Figure 2-3 EMAS Block Diagram



1 Input Quadrant Assembly
2 Artificial Feel Spring Cartridge
3 Autopilot Servomotor
4 Aileron Trim Actuator
5 Power Control Input Quadrant
6 Power Control Assembly Input Crank
7 Override Spring
8 Valve Input and Feedback Linkage
9 Servotab
10 Aileron Power Control Assembly
11 Centering Spring
12 Aileron Tab Lockout Actuator

Figure 2-4 C-141 Aileron Control System

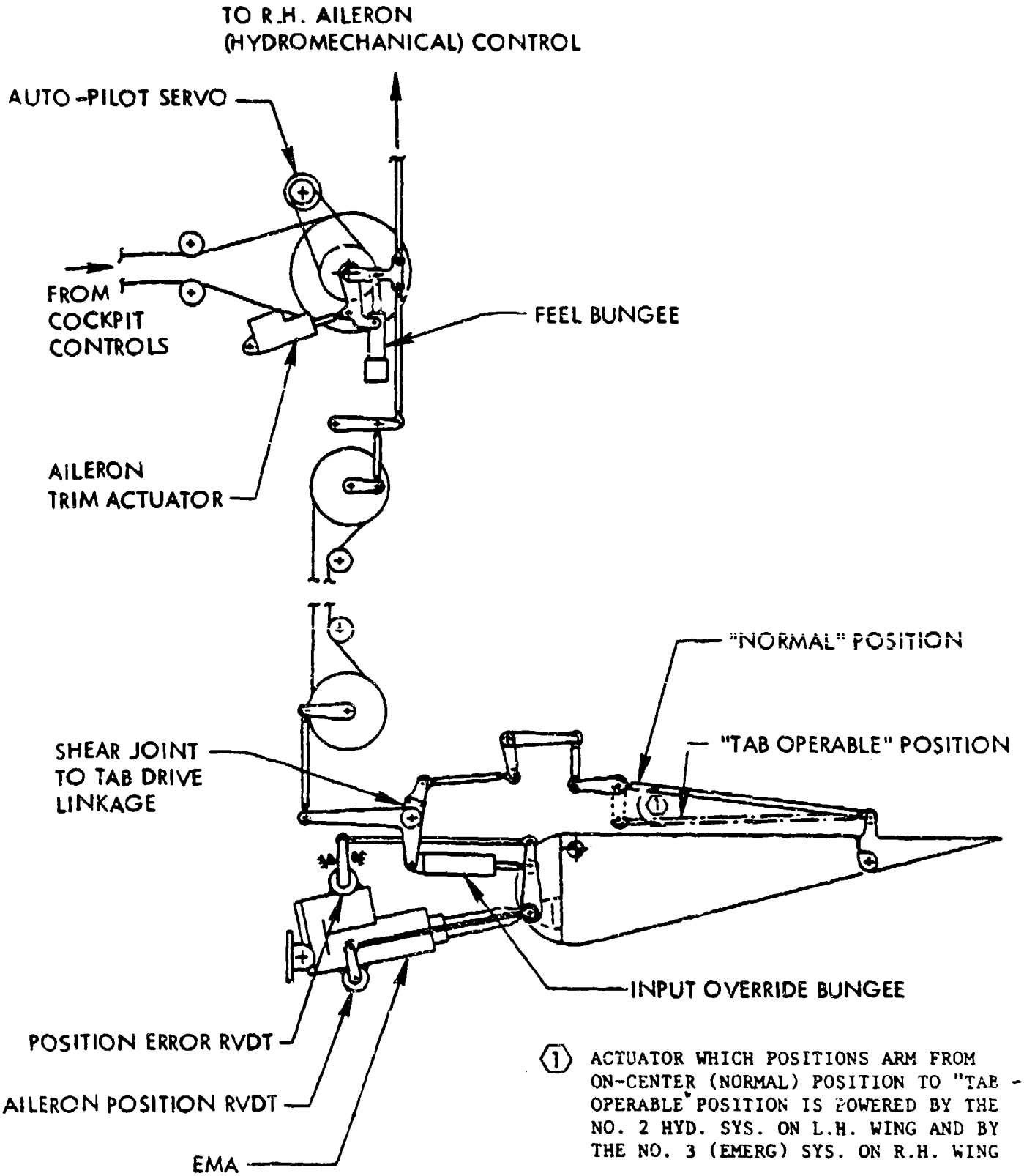
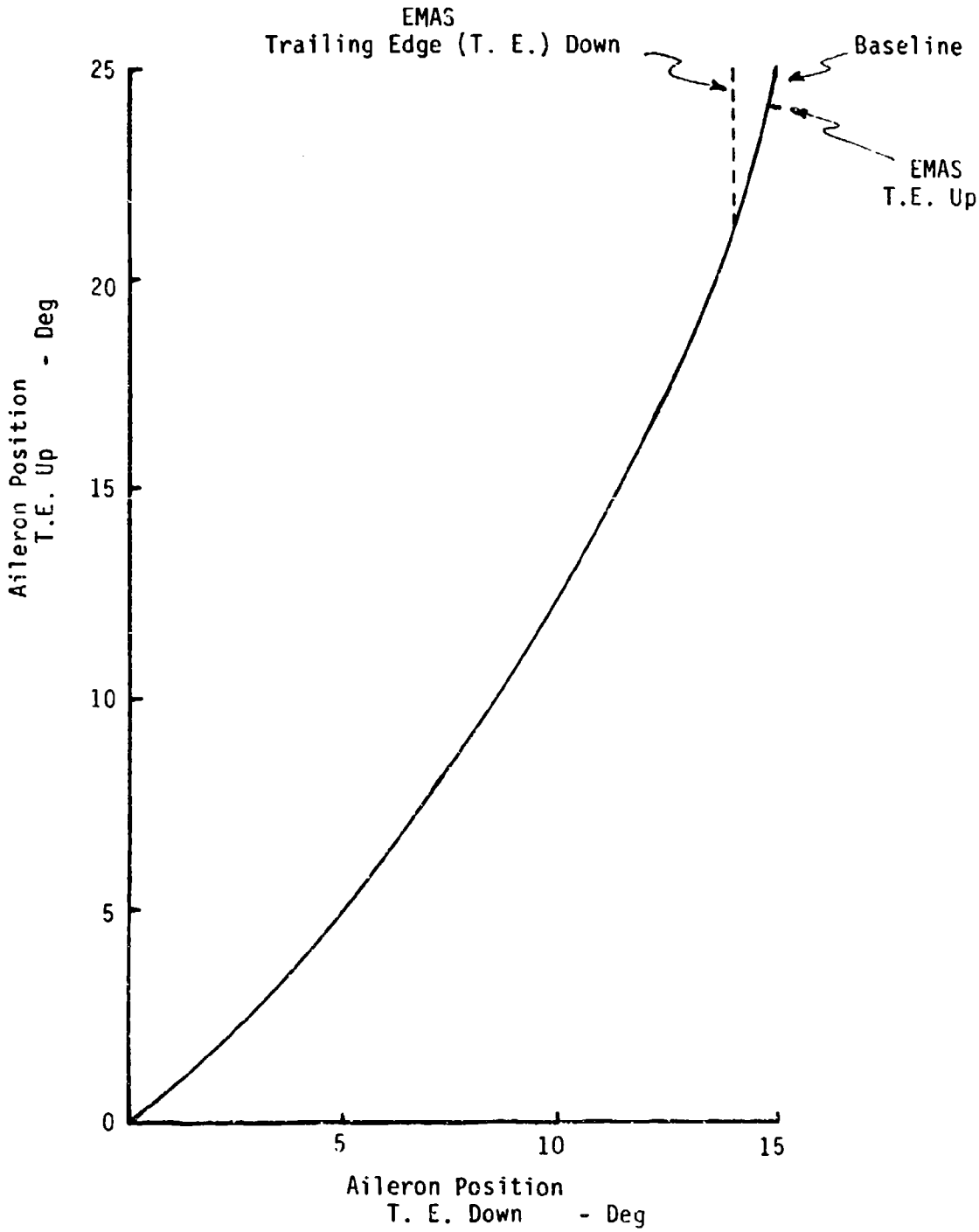


Figure 2-5 Aileron Control System and EMAS Interface

Figure 2-6 C-141 Baseline/EMAS Aileron Schedule



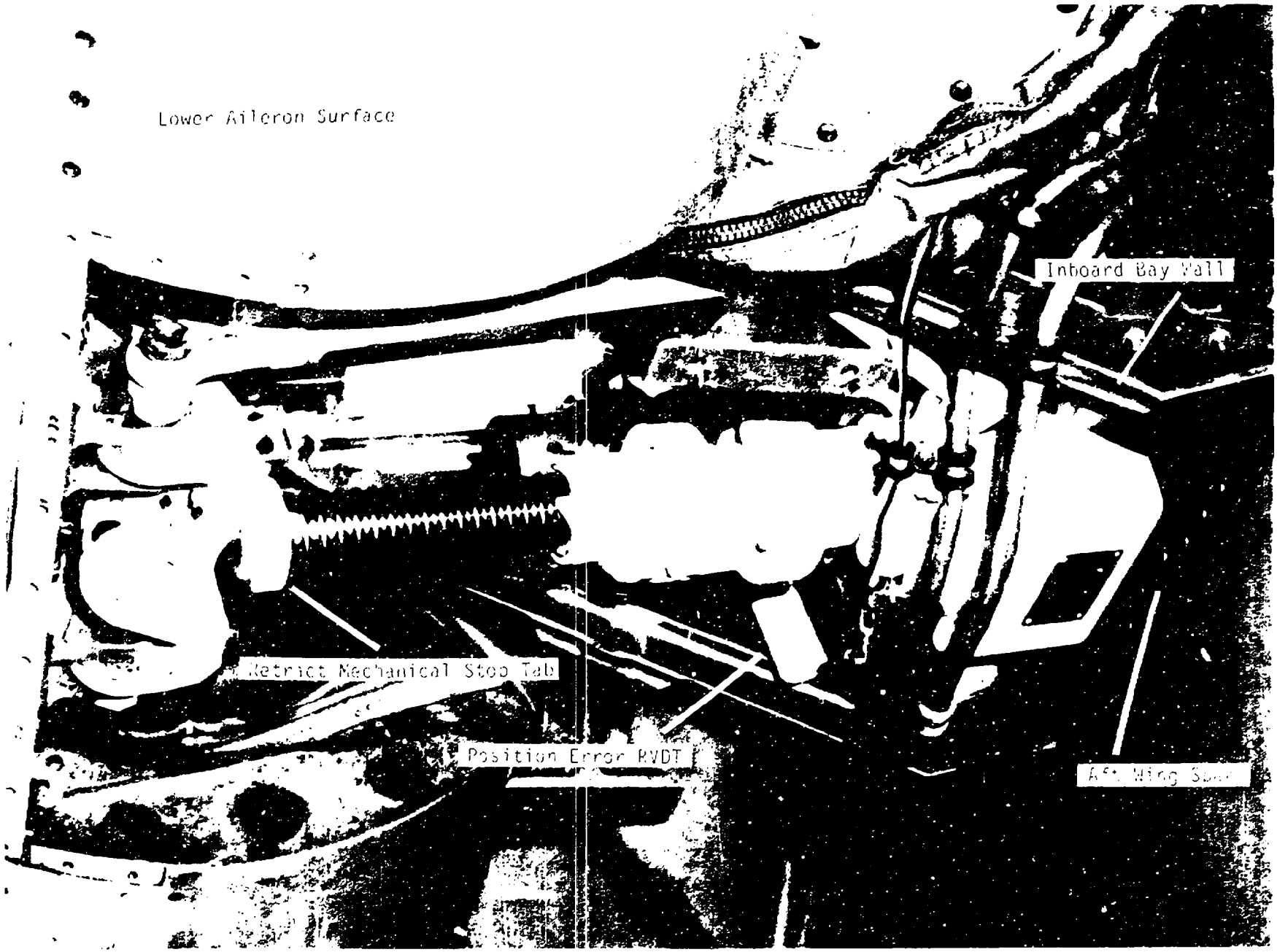


Figure 2-7 EMA Aircraft Installation

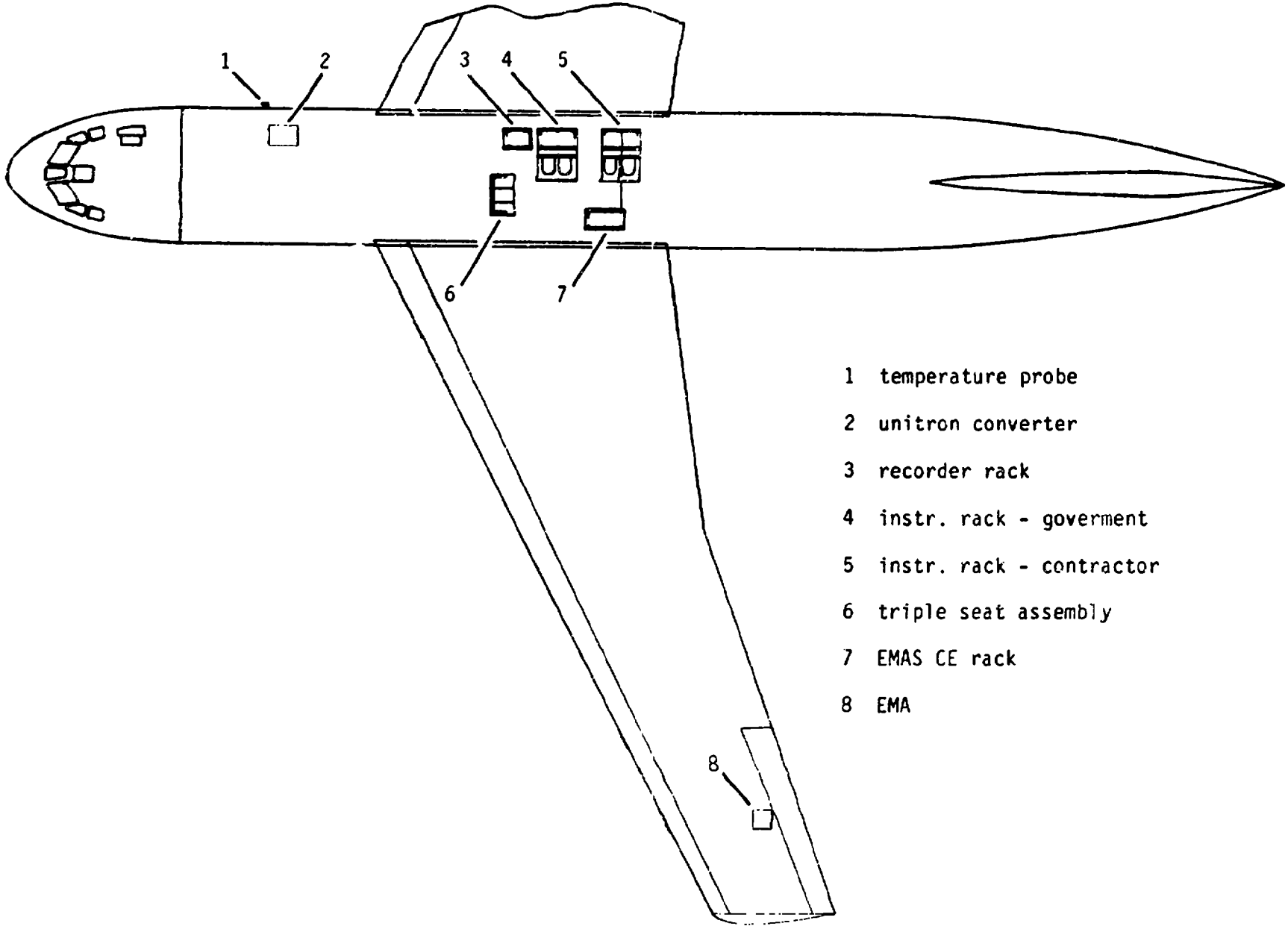


Figure 2-8 EMAS Modification Profile

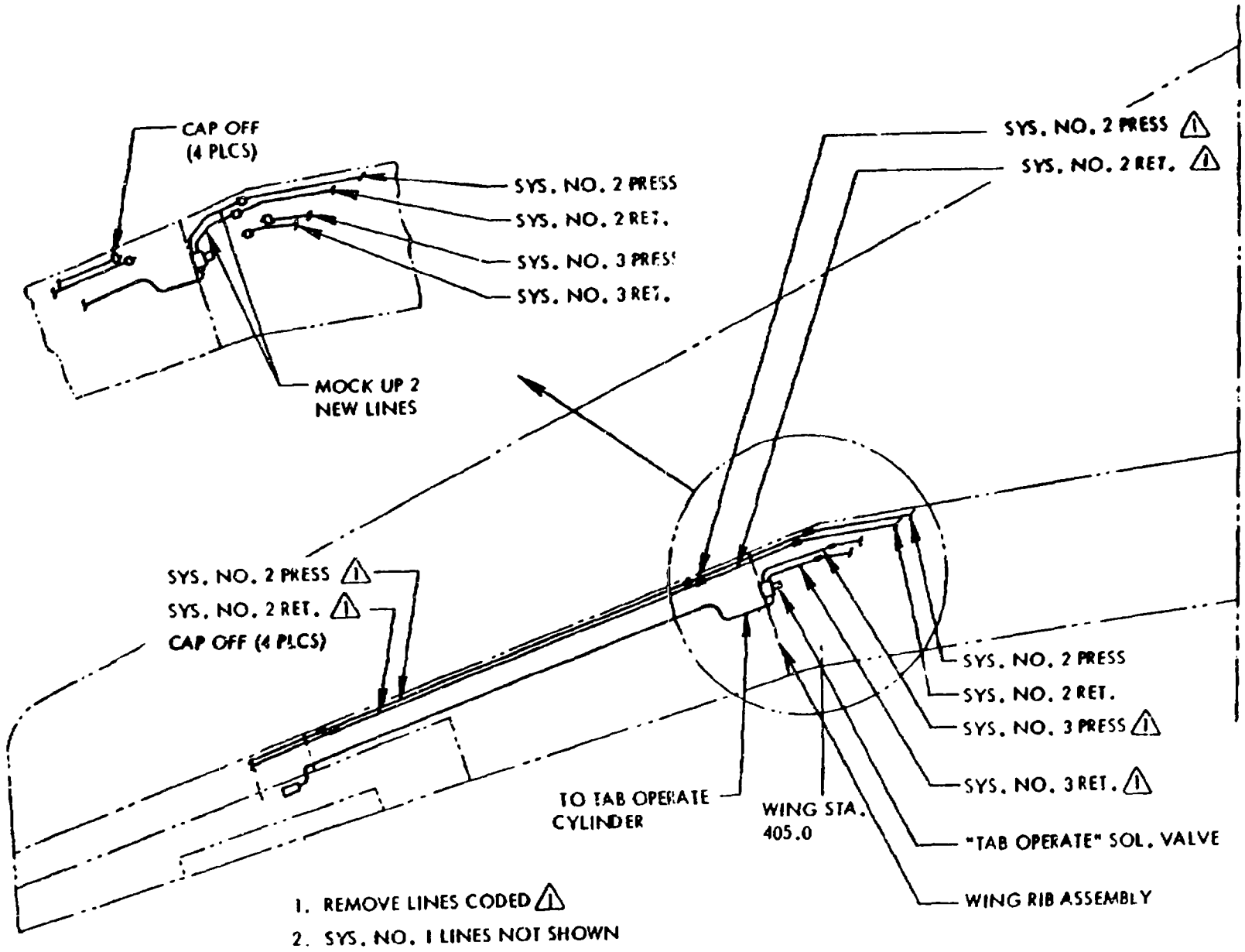


Figure 2-9 Hydraulic System Modification

INSTRUMENTATION AND TEST EQUIPMENT

1. The test EMA unit was instrumented by the contractor for performance evaluation. 19 parameters were selected. The asterisk indicates parameters displayed real-time aboard the test aircraft.

a. Power supply bridge voltage, one for each channel. *

b. Power supply bridge current, one for each channel. *

c. Motor temperature, one for each motor. *

d. Actuator output position, one for each channel. *

e. Actuator position error, one for each channel. *

f. Current imbalance. *

g. Voltage command, one for each channel. *

h. Controller ON indicator, one for each channel.

i. Actuator brake switch, one for each motor.

j. Motor speed, one for each channel. *

2. The test aircraft was instrumented to permit an evaluation of aircraft response to the EMA inputs. A total of 24 aircraft parameters complemented the EMAS parameters. A summary of the principal instrumentation used in the analysis of test results are provided in Table 3-1. The aircraft parameters are listed below. Asterisks indicate parameters displayed real-time aboard the test aircraft, although usually for A channel only.

a. Center of gravity vertical acceleration. *

b. Left and right wing vertical acceleration. *

c. Essential number 2 and main number 2 bus voltage.

d. Essential number 2 and main number 2 bus current.

e. Left and right aileron position. *

f. Rudder position (35 deg travel limit either side).

g. Wing center section aileron control quadrant position.

h. EMAS and PCU input quadrant position.

i. Time (slow and modulated code). *

j. Indicated airspeed.

k. Indicated pressure altitude.

l. Roll rate.

m. Yaw rate.

o. Bank angle.

p. Left and right aileron actuator bay temperature. *

q. Outside air temperature.

r. Autopilot aileron position command.

s. Event marker.

3. Primary test equipment included a magnetic tape recorder, stripchart recorders, D/Pad Three pulse code modulation (PCM) monitor, time code generator, digital temperature displays, and all supporting equipment.

4. The aileron position potentiometers had a relatively poor accuracy for the parameter measured. Although acceptable, this accuracy did not permit as fine a comparison of aileron positions relative to each other that would otherwise have been desirable. The relative low accuracy was the result of the small size of the instrument, dictated by the confined space available for mounting within the actuator bay. Aileron rigging also introduces an error into this parameter.

5. Wing accelerometers were mounted on brackets which were in turn mounted to the rotation axis of the power control assembly input crank. This placed the instruments within two inches of the inboard wall of the actuator bays. During the initial second of dynamic maneuvers, the mounting bracket vibrated at high frequency, as seen in the wing accelerometer response. This did not impact data reduction.

6. The autopilot command parameter is an autopilot computer syncro signal which is proportional to that produced by the aileron position transmitter. The relationship is:

$$\text{A/P Command (in volts)} = 11.8 \times \sin(2.06 \times \text{ail. def. (in deg)})$$

Table 3-1 EMAS Aircraft Instrumentation

Item	Manufacturer & Model	Purpose	Accuracy/ Resolution	Location
Rotary Potentiometer (4)	Veritron 105	Aileron System & Rudder Deflection	0.45/0.001 deg	Center Wing Input Quadrant Rudder & Aileron Hinge Lines
Rotary Potentiometer (2)	Veritron 2736	Aileron Input Bellcrank Deflection	0.95/0.001 deg	Left and Right Aileron Input Bellcrank Axis
Gyro	Lear-Siegler 1080Y	Roll Attitude	2.4/0.1 deg	Test Equipment Rack
3-Axis Rate Gyro	Northrop 50009-301	Roll & Yaw Rate	0.036/0.001 deg/sec	Test Equipment Rack
Accelerometer (3)	Sundstrand 303B	Wing & CG Vertical Accel	0.0008/0.001 g	Centerline at Approx. CG, Inboard Actuator Bay Wall
Pressure Transducer	Sundstrand 314D	Airspeed Determination	0.14/0.001 knots	Test Equipment Rack
Pressure Transducer	Sundstrand 314A	Altitude Determination	18/0.001 feet	Test Equipment Rack
Digital Thermometer (2)	Fluke 2160A	Aileron Actuator Bay Temperature	0.108/0.001 deg C	Left & Right Aileron Actuator Bay (Bi-Metallic K-Type)

TEST PROCEDURES

1. Ground Tests

a. Rigging

The right aileron was checked for proper rigging prior to the flight test. The left aileron was rigged by moving the aileron to the normal travel limits, trailing edge up and trailing edge down, and adjusting the electronics to command a stop. This operation was performed for each channel separately. The same operation was performed for the electrical stops, ensuring that the channels shut down in this position. The mechanical stops were set by manually adjusting the screw end fittings which incorporate one half of the tab pair (see Figure 2-7). The aileron was moved by manually rotating the actuator ball nut and using an inclinometer for precise measurement of the surface deflection.

b. EMC/EMI

Immediately after modification, a system checkout was performed to ensure that the EMAS and the EMAS-to-aircraft interface were functioning properly. An electromagnetic compatibility/electromagnetic interference (EMC/EMI) test was performed on the modified aircraft prior to the first flight to locate any mutual interference problems between the EMAS and the aircraft. This test consisted of operating all aircraft systems in their various modes with the EMAS functioning.

c. Ground Vibration Test (GVT)

(1) A dynamic ground vibration test was performed by GeLac to reveal the dynamic frequency response characteristics of the control surfaces, system damping, and tendencies to couple with aircraft structural modes. This data was used to ensure that the structural damping margins had not changed appreciably. The EMAS had been designed to the same dynamic characteristics as the original hydraulic actuator. The C-141 aileron is statically and dynamically stable with the critical flutter parameters being tip weight and actuator stiffness. If either parameter were not present, the damping would still be sufficient within the aircraft normal operating envelope. Laboratory studies had shown that the stiffness or spring rate of the EMA was identical to the hydraulic unit. All else remaining unchanged, the structural damping of the modified aircraft was expected to be the same as that of the baseline aircraft. The baseline aircraft has a flutter margin more than 20% above the normal operating limitations of the C-141A, and well above the military specification (MILSPEC) requirements.

(2) The EMAS GVT consisted of three parts:

(a) Shakers were placed so as to induce excitation at the bottom of the ailerons near the trailing edges. Data was recorded during symmetric and antisymmetric excitations with the EMAS unpowered, fully powered, and with only a single channel powered. The hydromechanical PCU was both powered (single electrically driven channel) and unpowered. Most of the data was obtained with the surfaces tested individually. Frequency sweeps were made from 2 to 40 Hertz (Hz) with accelerometers placed on top of each aileron aft of the actuator and

near the trailing edge.

(b) Excitation was made through the copilot yoke by a shaker attached to the end of a bar spanning the yoke and producing a rotational displacement. Three sweeps were made from 2 to 40 Hz with approximately one, two, and three inches of yoke displacement. Accelerometers were placed as before with an additional sensor on the bar 8.5 inches left of center. The EMAS was fully powered and the PCU had a single channel powered.

(c) Autopilot/EMAS compatibility was investigated using a servo-scope function generator to input an oscillatory signal to the autopilot servo-amplifier. Accelerometers were located as described in paragraph (a). Frequency sweeps were made from .02 to 35 Hz with input voltages of 0.25, 0.50, 0.75, and 1.125 V. These inputs correspond to 33, 67, 100 and 150% of the voltage required to obtain full aileron travel at 0.05 Hz, respectively.

2. Flight Tests

a. Airworthiness Tests

The initial flight of the modified aircraft had the two-fold objective of verifying that the EMAS had not degraded the structural dynamic characteristics of the aircraft throughout the flight test envelope, and to gather preliminary data on the EMAS performance. As a follow-on to the GVT, this flight included a brief structural damping investigation. Critical damping parameters, consisting of left and right hand aileron deflection and wing tip accelerations, were displayed real-time aboard the aircraft. A direct comparison was made between the response of the two ailerons. An abort criterion of six overshoots, a conservative estimate of 0.03 aileron damping coefficient (minimum MILSPEC requirement), or an undamped or divergent oscillation, was maintained. Test points consisted of stick-free stick raps or sharp pilot-induced aileron pulses performed up to a high altitude and to near the operating limit of the aircraft where damping is the least effective. These raps were abrupt, maximum command aileron inputs, followed by an immediate release of the yoke. The raps were performed at 20,000 ft pressure altitude (PA) and 200, 250, and 340 knots calibrated airspeed (KCAS); and at 35,000 and 41,000 ft PA at the best endurance airspeed (Ve), 0.74 Mach (M), 0.78M, and 0.81M. These points served to clear the flight test envelope for the subsequent EMAS tests. Rolls were also performed on this initial flight to ensure proper EMA and instrumentation functioning. All maneuvers during the first mission were flown by a test pilot, and included a T-39 safety chase for the initial test points and a T-38 for later high altitude and high airspeed points.

b. Performance Tests

EMAS performance and aircraft response data was recorded and selectively displayed during the EMAS evaluation test points. The maneuvers consisted primarily of level turns with and without backup systems and autopilot, precision aileron maneuvers with and without autopilot (A/P), and steady-heading sideslips. Aileron trim was maintained at neutral for the entire flight test.

(1) These maneuvers were to achieve the following specific objectives:

(a) Measure the responsiveness of the EMA to both precise and rapid commands of the pilot.

(b) Compare the responsiveness of the EMA to that of the baseline hydromechanical and tab operable control in various scenarios of control system degradation.

(c) Subject the EMA to sudden control inputs producing rapid load buildup on the aileron and thus the actuator, and produce sustained loads requiring a constant motor torque.

(2) The individual flights consisted of similar maneuvers but performed at different conditions, and modified to suit these conditions. Test points were performed in a buildup fashion for pilot proficiency and safety purposes. The following points were performed at 10,000 and 20,000 ft PA, respectively, and at 200, 250, and 340 KCAS:

(a) Trim shots with autopilot ON and OFF.

(b) 30 and 45 deg bank angle rolls in both directions.

(c) Maximum autopilot command rolls to the left and right.

(d) Maximum command, rudder coordinated, full yoke throw step aileron input to 30 or 45 deg bank angle in both directions.

(e) Degraded system performance with various combinations of EMA, tab operable, and right hydraulic PCU channels on and off, executing 30 and 45 deg rolls in both directions.

(f) Maximum command, rudder coordinated, full wheel throw step aileron input from 30 deg bank angle to 30 deg in the opposite direction (bank-to-bank). Repeated with 45 deg bank angle.

(g) 30 second maximum rudder deflection steady-heading sideslip in both directions. This produced a sustained aileron deflection to counteract the rolling moment created by the sideslip and thus induced a sustained electrical and actuator load.

(h) Standard instrument landing system (ILS) approach with a 45 deg intercept to the localizer. At approximately ten miles from the threshold, the pilot deviated one dot to the left of course and flew the course deviation indicator to reestablish on course by the final approach fix. This was repeated with a right deviation. An autocoupled approach, deviating with control wheel steering (CWS), could not be accomplished due to A/P system malfunctions.

(3) The same test points were repeated, with minor changes, at 35,000 and 41,000 ft PA, respectively, and at Ve, 0.74M, and at 0.81M. Tab operable was not engaged outside of the envelope defined by 150 - 250 KCAS and 10,000 - 20,000 ft P.A. The 45 deg bank angle rolls and sideslips were not performed above 20,000 ft P.A (see Table 4-1 for summary of test points).

(4) The yaw damper was ON except for the steady-heading sideslips. All test points were performed with the autopilot OFF except where specified. The airspeeds and bank angles were selected across the range of the normal operating conditions of the aircraft. The degraded systems points were done only at the end point airspeeds, not at the middle speed.

Table 4-1 Flight Test Matrix

Maneuver	Altitude (feet) -	10,000			20,000			35,000				41,000			
	Airspeed (knots) -	200	250	340	200	250	340	Ve	.74M	.78M	.81M	Ve	.74M	.78M	.81M
30 deg Left Roll		x	x	x	x	x	x	x	x		x	x	x		x*
30 deg Right Roll		x	x	x	x	x	x	x	x		x	x	x		x*
45 deg Left Roll		x	x	x	x	x	x								
45 deg Right Roll		x	x	x	x	x	x								
Max Left 30 deg & Max Back		x	x	x	x	x	x	x	x		x	x	x*		x*
Max Right 30 deg & Max Back		x	x	x	x	x	x	x	x		x	x	x*		x*
Max Left 45 deg & Max Back		x	x	x	x	x	x								
Max Right 45 deg & Max Back		x	x	x	x	x	x								
30 deg Left Bank-to-Bank		x	x	x	x	x	x	x	x		x	x	x*		x*
30 deg Right Bank-to-Bank		x	x	x	x	x	x	x	x		x	x	x*		x*
45 deg Left Bank-to-Bank		x	x	x	x	x	x								
45 deg Right Bank-to-Bank		x	x	x	x	x	x								
Left Aileron Pulse					x	x	x	x	x	x	x	x	x	x*	x*
Right Aileron Pulse					x	x	x	x	x	x	x	x	x	x*	x*
30 sec Max Rudder Left Sideslip		x	x	x	x	x	x								
30 sec Max Rudder Right Sideslip		x	x	x	x	x	x								
EMAS A & PCU 1 Off, 30 deg Left Roll		x		x	x		x	x			x	x			x*
EMAS A & PCU 1 Off, 30 deg Right Roll		x		x	x		x	x			x	x			x*
EMAS A & PCU 1 Off, 45 deg Left Roll		x		x	x		x								
EMAS A & PCU 1 Off, 45 deg Right Roll		x		x	x		x								
EMAS B & PCU 2 Off, 30 deg Left Roll		x		x	x		x	x			x	x			x*
EMAS B & PCU 2 Off, 30 deg Right Roll		x		x	x		x	x			x	x			x*
EMAS B & PCU 2 Off, 45 deg Left Roll		x		x	x		x								
EMAS B & PCU 2 Off, 45 deg Right Roll		x		x	x		x								
Left Tab 30 deg Left Roll		x			x										
Left Tab 30 deg Right Roll		x			x										
Left Tab 45 deg Left Roll		x			x										
Left Tab 45 deg Right Roll		x			x										
Right Tab 30 deg Left Roll		x			x										
Right Tab 30 deg Right Roll		x			x										
Right Tab 45 deg Left Roll		x			x										
Right Tab 45 deg Right Roll		x			x										
2 min Trim Shot, A/P Off		x	x	x	x	x	x	x	x		x	x	x*		x*
2 min Trim Shot, A/P On		x	x	x	x	x	x	x	x		x	x	x*		x*

* not accomplished

DATA REDUCTION AND ANALYSIS

1. Immediate test results were available real-time aboard the test aircraft by display of limited parameters on two 8-channel stripchart recorders and display of all PCM test parameters on a non-recall basis with the D/Pad Three. All data was stored on magnetic tape for later playback and reduction. Stripchart playback of all parameters immediately after each flight permitted quick-look checks of results. Final reduction utilized a digital computer program for conversion to engineering units, printing, and plotting of the data.

2. Aircraft stripcharts were utilized for immediate postflight analysis and next-flight clearance. These recordings and the test event log were used to locate events on the stripouts of the test instrumentation tape containing all parameters. These stripouts permitted easy examination of data trends during events and comparison against all other parameters. Some rudimentary conclusions were drawn from the stripchart data; however, the data served largely to isolate time bands for digital printouts of the parameters from which more precise conclusions could be drawn. From this data, requests for plots in selected scales were made for final analysis and report presentation.

3. Time histories were selected as the best means of presenting the EMAS performance flight test data. Results of similar maneuvers performed in opposite directions are presented together in paired plots for ease of comparison. Bank-to-bank maneuvers are given as the primary roll performance maneuver. Principal areas for the comparison were similarity in maximum deflection angle, time to maximum deflection angle, deflection rate, roll rate, and yaw rate. The data was examined for uncommanded surface motion or motion that was not in concert with the opposite surface. Examination was also made for stability at surface positions and tendency to make more or less frequent small adjustments in position. This latter criterion was applied particularly to trim and approach data. Pilot technique can account for part of the differences in response between maneuvers in opposite directions. Results were also compared to baseline C-141A roll performance data gathered during prototype testing as reported in Reference 1.

4. Accelerometer data displays an offset from the anticipated 1.0 g in level flight. This is attributed to mounting alignment discrepancies, the inflight deck angle of the aircraft, and normal structural flexure. This offset must be considered when examining the accelerometer data. Wing accelerometer data has had any steady state vertical acceleration (obtained from the CG accelerometer) subtracted from it.

5. Yaw rates resulting from the rolls were taken between the time of the aileron input and the first evidence of a rudder input. A yaw rate often existed prior to the commencement of the roll. In this case only the difference of the roll-induced yaw rate and the intial yaw rate was used.

6. It was found that much of the data did not vary significantly between test conditions; therefore, only representative data is presented for many of the test maneuvers. Data obtained at the maximum dynamic pressure (300 kts at 10,000 ft P.A.) and the minimum dynamic pressure (Ve at 35,000 ft) are principally used for

this purposes.

7. Sign conventions for aircraft test parameters are as follows:

   a. Aileron trailing edge up is positive, trailing edge down negative.

   b. Rudder trailing edge right is positive, left negative.

   c. Right roll (bank angle and roll rate) is positive, left negative.

   d. Right yaw rate is positive, left negative.

   e. Vertical acceleration up is positive, down negative.

TEST RESULTS AND DISCUSSION

1. Aircraft Modification

a. The aircraft modification to incorporate the EMAS was designed to meet the enhanced safety requirements of a development test and evaluation project. It dealt with an engineering model item that did not represent a production article in terms of volume or weight. A production installation as a retrofit to a pre-existing airframe might choose other aircraft systems for the electrical interface and make different or no hydraulic changes for the tab capability.

b. A future installation of multiple EMAS units to an existing airframe would present the potential for an overload of the existing emergency generator in the event of an emergency where the electrical flight controls become a critical item to keep powered. A second or higher capacity generator may be required.

c. A dramatic failure of an EMAS system has the potential for producing unusual flight attitudes and motion rates which may make it difficult or impossible for aircrew members to reach system circuit breakers. The test installation had the EMAS circuit breakers located in the cargo bay of the aircraft. A production EMAS installation should feature system circuit breakers in the cockpit for easier aircrew access in the event of an emergency.

2. General Characteristics

a. During ground operation, it was discovered that the EMAS B channel would often drop off line when the power source to the number 2 essential bus was changed (such as switching from ground to aircraft power, or vice versa). This is considered to be an undesirable characteristic of the system, as an in-flight emergency procedure may include switching power sources and the disengagement of one channel would reduce aircraft roll performance until it was recycled. A dual channel disengagement would leave the aileron momentarily unpowered. This phenomenon could not be investigated in detail, and it is possible that the momentary power loss during a switching operation on the number 2 essential bus resulting in the channel disengagement is longer than the MILSPEC requirements to which the EMAS was designed.

b. Laboratory testing by the contractor demonstrated that temperature variations can result in drifting of the electronically set travel limits and electrical stops by as much as 0.3 deg. During early ground tests, the aircraft was often moved from a hangar to the outside winter environment. The travel limits and electrical stops drifted toward each other and caused one channel to repeatedly shut down when a full travel maximum aileron input was commanded. This problem was aggravated by each channel being adjusted separately, as part of the dual channel capability, with minor setting differences being inevitably introduced. It is possible that travel overshoot caused by the high inertias generated on the ground (no air loads to assist in damping) contributed to this fault. After re-rigging, with a greater spread between the two electrical settings, the problem did not recur. This is considered to be an undesirable characteristic of the system and may require redesign, greater spread between the

travel limits and electrical stops, or simply a caution against such abrupt inputs on the ground.

3. Maintenance Factors

a. It was found that if the left aileron and the yoke were in relative positions not coincident with the normal travel schedule (yoke deflection does not match aileron position) when the EMAS is initially powered, a sudden aileron movement will occur. This movement, if unanticipated, may create a hazard to unwary ground personnel. While this phenomenon is also a characteristic of the hydraulic actuator, the nature of the EMAS may permit a logic function to be designed that will eliminate or reduce this hazard (such as a slow movement at initial power-up).

b. It was possible to position the aileron by turning the ball screw by hand. This made the checking and adjustment of the end-of-travel, electrical stops, and mechanical stops during control surface rigging an easy matter. However, in temperatures below freezing, manual surface positioning was no longer possible without initially powering up and moving the actuator. It was also found that the force required to move the aileron against inert motors was considerably more than that required to move an inert hydraulic PCU powered aileron (approximately 60-70 lbs at the trailing edge versus 10-20 lbs). These characteristics will increase the difficulty of maintenance tasks.

4. EMC/EMI

a. The EMC/EMI ground test found no anomalies.

b. Current aircraft designs often feature a flux gate compass transmitter permitting the directional gyro to be slaved and thereby eliminate gyroscopic precession. This instrument is typically placed in the wing tip to remove the instrument from the effects of electrically generated magnetic fields generated throughout the aircraft. Placing electric motors such as those in the EMA in the outboard wing area may require a relocation of the flux gate or enhanced shielding of the instrument. This was not a consideration in the test aircraft because the flux gate was in the right wing.

5. GVT

a. The measured EMAS aileron rotational frequencies of 8.0 Hz for the fully powered configuration, 7.9 Hz for a single channel, and zero for unpowered, were compared with the hydraulic PCU aileron rotational frequencies of 8.5 Hz fully powered (from previous GeLac tests), and measured frequencies of 8.6 Hz with a single channel and 8.4 Hz unpowered. This comparison was considered to be satisfactory and the aircraft was cleared for flight. With autopilot inputs, aileron response was attenuated by about 20 decibels (dB) at 2.5 Hz and was essentially zero above 5 Hz, with both ailerons responding similarly. No servo instabilities were observed. This data was considered satisfactory to clear the autopilot for use in conjunction with the EMAS in-flight.

b. The GVT was interrupted by a system instability that manifested itself in a neutrally-damped 9.1 Hz oscillation of the control surface with approximately 3 deg of deflection. The oscillation could be excited by hand (longitudinal inputs to the input arm) with both channels on, but could not be sustained with a single channel though lightly damped. This response is attributed to a C-141 aileron

control system characteristic that is compensated for in the hydromechanical PCU by a viscous damper. The fault was eliminated by a system modification using a notch filter, and no further oscillations were observed despite considerable excitation during the GVT. This experience points up the importance of testing such systems on the aircraft in a ground environment prior to flight.

6. Flight Test

a. Current Draw

The highest current draw from the aircraft buses was a total of 12.5 amps. This represents the sum of the draw from both buses for dual channel operation or the draw from a single bus for single channel operation. This load was within the 50 amps excess capacity of each bus. This peak draw occurred during maximum roll command maneuvers at 10,000 ft PA and 340 knots, the highest dynamic pressure (q) or load condition. The steady-heading sideslips failed to produce any substantial sustained current draw as a consequence of constant aileron deflection.

b. Thermal Behavior

In flight, the EMAS motors operated at approximately one to two deg Centigrade (C), regardless of the ambient temperature (always less than one deg C at the test altitudes), with no sudden gradients visible at any time. Steady-heading sideslips also failed to produce any gradient. The left aileron bay temperature was generally warmer than ambient by one to three deg C while the right bay was approximately three to seven deg warmer than ambient. This same bay temperature difference was observed during ground operations. These observations may indicate that the EMAS operated at a lower temperature than the PCU. All tests were performed at lower than standard temperature conditions because of the winter environment. The EMAS flight test in no way represented a certification type trial with temperature extremes in conjunction with different levels of actuator excitation.

c. Fully Powered Roll Performance

(1) The three pilots that flew the modified aircraft were unable to detect any difference in the feel of the control system or aircraft response with the EMAS fully functional. A comparison of slopes for the deflection versus time curves of the two ailerons during opposite rolls indicates identical deflection rates (see Figure A-2). Within the accuracy of the instrumentation, the normal aileron deflection schedule is verified for EMAS. Deflection rate and schedule were not duplicated near the end-of-travel, trailing edge up or down. The deflection rate reduction feature of EMAS is evident in the roll performance plots by the lower slopes of the deflection versus time curves of the left aileron as compared with the right near the travel limits. The hydraulic PCU produced no such pronounced round-off in deflection rate. The EMAS response is clearly a departure from normal aileron travel behavior and results in the left aileron requiring an average of 0.3 sec longer to reach the maximum deflection angle for an identical maneuver and test condition than the right aileron (see Table 6-1). Also, because of the experimental nature of EMAS, this feature was duplicated for travel off of the limits. The reduced rate off of the limits is a characteristic that serves no useful function for such a control surface and, while not objectionable to the pilots, should be avoided.

(2) Table 6-1 and examination of the maximum command roll time histories (see Figure A-2), reveal that the test aircraft rolled at a faster rate to the right than to the left. It is also apparent that the left aileron usually failed to reach the same aileron deflection as the right aileron for similar maneuvers and test conditions. This is attributable to the inability of the EMA to reach the normal aileron travel limits. Most maneuvers succeeded in driving the right aileron to the travel limits, so the left aileron would also be expected to reach the limit in the opposite direction. With the lower end travel deflection rate of the left aileron and the greater end travel deflection angle of the right aileron, the ailerons were consistently off the normal travel schedule near the end of travel. This resulted in a positive yaw rate in rolls in both directions, or proverse yaw in rolls to the right and adverse yaw in rolls to the left. Thus, right rolls had a higher roll rate than left rolls. When EMAS test data is compared with limited baseline data (see Figure 6-1, B-1, and B-2), right rolls are generally higher and left rolls lower than predicted. The difference in the time to maximum deflection angle from the baseline is attributed to the lower EMAS test aircraft gross weight. The average gross weight for all test points was 180,000 lbs, not varying by more than 15,000 lb during the course of a test, with an average mean aerodynamic chord center of gravity location of 31.0%. The inabilities of the EMAS to produce similar performance to the hydromechanical system was the result of the more pronounced slow down feature and the inability to reach the maximum travel limits. These departures from the baseline roll performance were not detectable by the pilots.

d. Deflection Sensitivity

(1) EMAS was more sensitive to control inputs than the hydromechanical PCU. This is evident in the greater magnitude left aileron excursions during autopilot trim shots (see Figure A-9). Near the aileron travel limits or during an abrupt return to neutral, the left aileron occasionally showed abrupt movements not consistent with right aileron travel (see Figure A-2 and C-1). This aberration was traced to a control system nonlinearity, evident as uncommanded deflections of the three aileron control quadrants (see Figure A-3). These deflections are too rapid and the direction change too abrupt to be pilot-induced. The EMA reproduced these rapid deflections more frequently and with sharper resolution than the PCU, supporting the conclusion of higher sensitivity for the electrical system. This sensitivity will be useful in applications such as automatic flight controls, but may require control system modifications in retrofit applications to remove aberrations such as those experienced on the C-141.

(2) During aileron pulses, surface response was discovered to have a sudden bounce or reversal as it returned to neutral (see Figure A-9). This is attributed to a nonlinearity in the yoke centering bungee evident during ground control cycles where the yoke was released at full deflection (see Figure A-1). The response was only apparent on the surfaces as they returned from a trailing edge down deflection. The left aileron displayed a deflection reversal due to this effect whereas the right only showed a short pause. The difference in response between the two surfaces to this aberration is attributed to the higher sensitivity of EMAS.

e. Degraded System Performance

The three pilots that flew the modified aircraft were unable to detect any difference in the feel of the control system or aircraft response in single channel operation. Aileron deflection and roll performance during degraded system rolls with first one and then the other channel of both actuators depowered were comparable. Roll performance was comparable to the fully powered configuration in low dynamic pressure conditions but degraded with an increase in dynamic pressure as expected (see Table 6-2 and Figure A-5). No uncommanded aileron movement was seen as a consequence of deactivating or activating a single channel.

f. Tab Operable Roll Performance

(1) With tab operable selected on the left wing at 200 kts and 9,000 ft, approximately 30 lbs of force and 45 deg of yoke deflection was required to maintain level flight. Right aileron tab required 45 lbs and 55 deg. At 200 kts and 20,000 ft these values were 30 lbs of force and 40 deg with left tab, and 65 lbs and 50 deg with right tab. This difference is caused by the higher forces required to backdrive the aileron against inert motors, resulting in less aileron deflection at the neutral condition (trailing edge up zero hinge moment state, see Table 6-3 and Figure A-5 and A-6). Comparison of roll performance with tab selected on the left aileron and tab on the right for rolls in the same direction at identical test conditions but in the two tab scenarios (see Table 6-3 and Figure A-6) indicates that no degradation in performance has occurred. Baseline tab data is provided in Table 6-3 and Figure B-2, B-3, and B-4 (the baseline tab roll data corresponds to tab selected on both ailerons).

(2) Roll-off resulting from the selection and de-selection of tab was found to be similar in the two scenarios but for the final aileron position becasue of the higher backdrive force requirements of the EMA (see Figure A-6 and A-7). This roll-off is a consequence of the unpowered aileron floating to a zero hinge moment condition that is approximately eight deg trailing edge up.

g. Aileron Damping

(1) Aileron damping was found to be high or deadbeat at all test conditions (see Figure A-9). Wing accelerometer response to the aileron pulses and maximum command inputs showed evidence of a structural mode superimposed on the input response (see Figure A-9 and C-1). This was not considered critical. The intial part of the wing response show a high frequency oscillation superimposed on the wing trace. This is believed to be caused by vibration of the cantilever accelerometer mount. Wing response was considerably greater during maximum command rolls, however inertia overshoots are also included in these oscillations (see Appendix C).

(2) Aileron pulses produced increasingly greater bank angles, from three to 13 deg, as altitude increased. In general, right pulses produced less bank angle than left pulses by approximately two deg. The aircraft slowly rolled to level flight after the pulse. At 35,000 ft PA and 0.81M, and at all 41,000 ft airspeeds, left pulses produced bank angles that required pilot action to recover. These differences are not attributed to an EMAS effect.

h. Autopilot Interaction

(1) The test team had originaly intended to use the autopilot turn knob control for a roll input, but the A/P would not function reliably in this mode. Therefore, the rolls were made in the NAV SELECT mode by moving the heading bug on the horizontal situation indicator (HSI). This limited roll attitude to 30 deg of bank whereas the normal mode was expected to produce 38 deg. Regardless of test conditions, maximum aileron deflection was approximately five deg and maximum roll rate was approximately three to four deg/sec (see Figure A-4). The autopilot malfunction precluded the accomplishment of autocoupled approaches.

(2) The left aileron made greater deflections than the right in response to the small autopilot inputs. See paragraph 6d for EMAS sensitivity to autopilot trim shot commands.

i. Takeoff and Landing

Takeoff and landings were normal and no irregularities in the data were found. No data is provided.

j. System Failure

(1) 85% of the flight test was successfully completed. Testing was terminated on the sixth sortie because of an EMAS malfunction that could not be corrected in time to complete the program prior to scheduled aircraft demodification. Up to that time, the system had performed flawlessly during the flight portion of the test. The inflight malfunction was characterized by three channel A drop outs. The first drop out occurred during a maximum command left roll at 174 knots airspeed (Ve) at 35,000 ft PA. The next drop outs occurred during maximum command left and right rolls at 190 kts (Ve) and 41,000 ft PA. The indicated faults for the last two drop outs were position in limit and 200 V for greater than two seconds. The faults for the initial drop out could not be recorded prior to channel recycle. The channel recycled without difficulty and the aircraft returned to base without incident.

(2) During ground testing to isolate the fault experienced in flight, the following events occurred:

(a) Event 1: Initially less than maximum rate and then maximum rate aileron inputs were made until channel A dropped out. No data was recorded during these cycles.

(b) Event 2: During a maximum command left yoke input, the aileron deflected at a higher than normal maximum rate to the trailing edge up electrical travel limits, without a slow down, and shut down automatically. Motor speeds were abnormally high.

(c) Event 3: Without yoke input, channel B was powered up from the pilot overhead switches. The aileron began to move up at lower than maximum rate. During this excursion, channel A was turned on at which time deflection rate increased to higher than normal maximum rate. The slow down feature worked, however the aileron went to the electrical travel limits and both channels automatically shut down. Channel A voltage command displayed an abnormal oscillatory response before the shut down. Motor speed was normal.

(d) Event 4: Without yoke input, channel A was powered up, as in Event 3, and the aileron traveled at higher than normal maximum rate to the full down electrical limit and automatically shut down. Channel B was not powered during this excursion. No slow down was evident and motor speed was normal.

(e) Event 5: Both channels were powered up and the aileron moved at lower than maximum command in both directions without incident. Maximum command trailing edge up was executed without incident. Upon maximum command trailing edge down input, the aileron traveled at higher than normal maximum rate without a slow down to the retract mechanical stops. The stops were engaged with such force that a material failure occurred. Both channels automatically shut down.

(3) The retract mechanical stop engagement resulted in the screw end fitting being sheared. The tab on the actuator housing bottomed out on the end fitting and bent outward (see Figure 6-2). The end fitting tab had been designed to fail at a lower load than the housing tab. This demonstrated that the fitting served as a backup mechanical stop. The last forceful stop created enough moment to allow the nut holding the B otor resolver against an orientation pin to back off from a restraining lock (safety or tine) washer. The resolver moved forward and the pin fell out of a tight press fit through the motor shaft. This allowed the resolver to change orientation and created multiple problems. After this damage was repaired, the basic fault that appeared to have produced the inflight drop outs remained. This fault was a current imbalance, evidently caused by a phase error in motor A. This fault was not evident in data from previous flights and only became critical during high deflection rate situations with low air loads such as in a maximum command condition at low dynamic pressures; just the conditions at which the inflight drop outs occurred. This however, does not explain the motor runaways without inputs experienced in the ground tests. The results of contractor laboratory evaluation of the EMAS failure were not available prior to publication of this report.

k. Structural Design

The mechanical stops on the actuator were not required to be designed to withstand the loads imparted by a stop from normal EMAS maximum travel rate, programmed to be identical to the normal PCU system maximum rate. The EMAS was capable of producing higher deflection rates when unrestrained. The inability to withstand the normal or unrestrained maximum deflection rate is not a logical design criteria and defeats the purpose of the stops.

Table 6-1 Maximum Deflection Rolls

Yaw Rate prior to rudder coordination

+ = positive
(R) = rudder coordination
(N) = no rudder data

Airspeed kts	Altitude ft	Roll Rate deg/sec	Time to Max Rate sec	Max LH Ail Def deg	Max RH Ail Def deg	Time to Max LH sec	Time to Max RH sec	Yaw Rate deg/sec
204	9,200	-17.5	1.9	23.0	-13.0	1.3	0.5	-0.3 (N)
208	9,200	22.0	2.0	-13.0	24.5	0.9	0.8	-0.5 (N)
255	9,500	-19.8	2.0	22.5	-12.0	1.3	0.6	0.3 (N)
263	9,500	23.2	1.9	-13.0	24.0	0.9	1.0	0.7 (N)
335	9,000	-13.6	2.6	21.0	-11.5	1.5	0.6	(R) +
335	9,000	17.0	3.2	-11.0	21.5	1.0	0.8	(R) +
195	20,600	-19.6	2.7	23.0	-13.0	1.3	0.7	0.3
187	20,300	21.0	2.0	-13.0	25.0	0.9	0.8	0.7
250	20,500	-21.1	2.0	22.0	-13.0	1.4	0.7	0.2
252	20,200	24.6	2.3	-12.5	25.0	1.0	1.0	0.6
333	20,500	-13.7	3.2	20.0	-13.0	1.2	0.7	0.5
332	20,300	16.5	1.6	-11.0	21.5	1.0	0.8	0.2
160	36,500	-20.6	3.1	22.5	-14.0	0.9	0.7	(R)
160	36,500	20.5	3.3	-13.0	25.0	0.9	0.8	0.8
265	35,700	-20.3	2.7	21.5	-13.5	1.3	0.6	0.7
265	35,900	23.1	2.6	-13.5	24.5	1.1	1.0	0.5
280	35,000	-18.1	2.7	23.0	-13.5	1.3	0.6	0.8
279	36,000	21.8	2.7	-13.0	24.0	0.9	1.0	0.3
186	40,500	-18.6	2.0	23.9	-13.5	1.0	0.7	(R)
186	40,500	21.0	2.0	-13.5	24.0	0.7	0.6	0.5

Figure 6-1 Baeeline/EMAS Roll Comparison

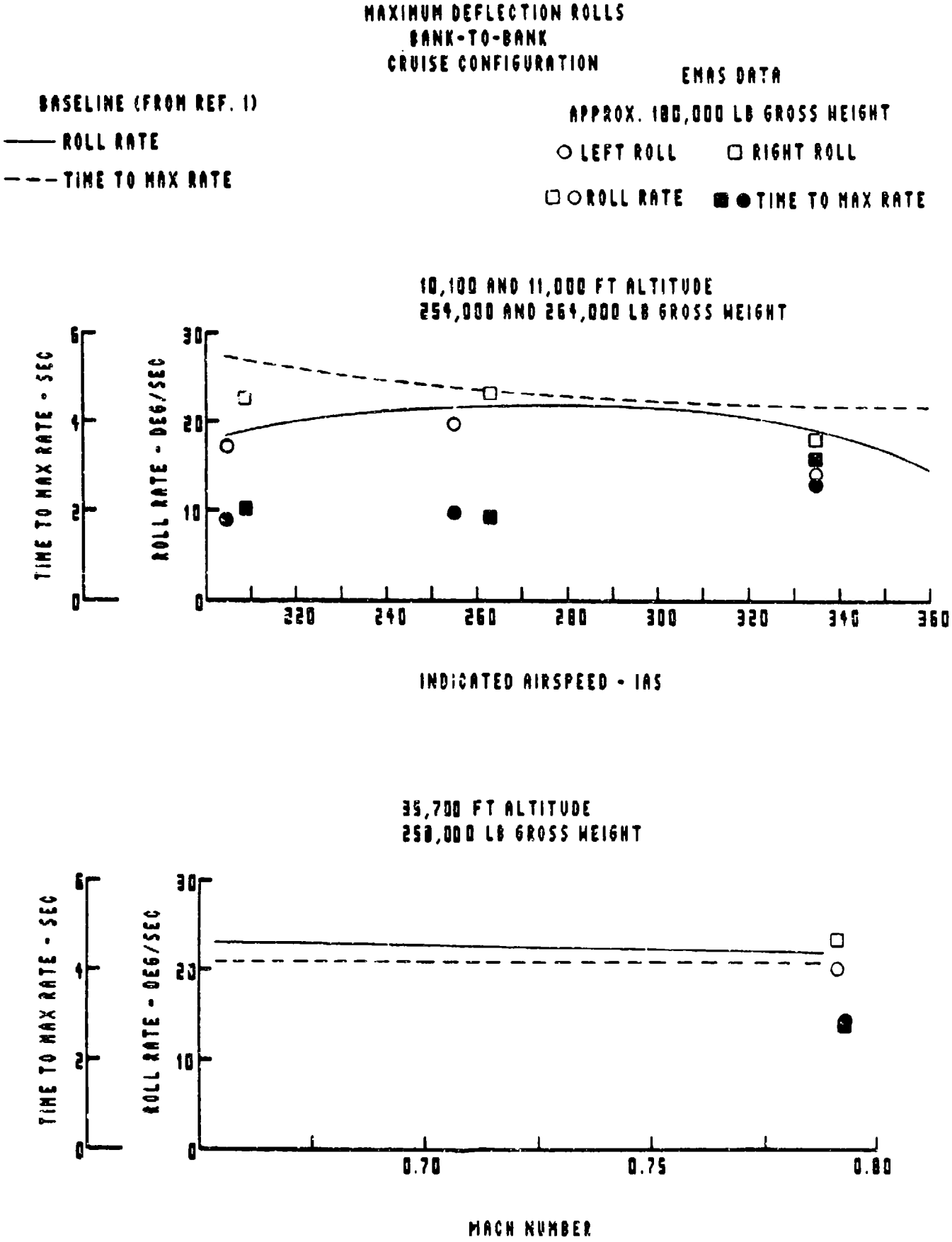


Table 6-2 Degraded System Roll Performance

Yaw Rate prior to rudder coordination

(R) = rudder coordination
(N) = no rudder data

x/y Off = EMAS Channel x & PCU System y Off

Airspeed kts	Altitude ft	Roll Rate deg/sec	Time to Max Rate sec	Max LH Ail Def deg	Max RH Ail Def deg	Time To Max LH sec	Time To Max RH sec	Yaw Rate deg/sec	Condition
190	9,100	-17.5	2.3	22.9	-13.5	1.3	0.6	-0.2 (N)	A/1 Off
190	9,100	20.0	2.3	-13.5	25.0	0.9	1.1	0.7 (N)	"
191	9,100	-17.7	2.3	22.7	-13.7	1.3	0.7	-1.2 (N)	B/2 Off
19[illegible]	9,000	19.8	2.3	-14.0	25.0	0.8	1.0	0.5 (N)	"
336	9,000	-11.5	1.3	18.3	-10.5	0.6	0.5	-0.4	A/1 Off
33[illegible]	8,900	14.9	1.6	-16.8	18.5	0.6	0.6	(R)	"
335	9,000	-12.0	1.6	18.4	-10.5	0.7	0.6	(R)	B/2 Off
334	9,000	14.6	2.0	-11.2	18.5	0.5	0.5	(R)	"
196	20,200	-17.4	2.0	21.8	-13.3	1.1	0.6	-2.0	A/1 Off
199	20,300	21.3	2.6	-12.9	25.0	0.8	1.3	0.9	"
197	20,400	-18.6	2.1	22.3	-13.5	1.1	0.7	-0.3	B/2 Off
197	20,500	20.3	1.5	-13.3	24.9	0.8	1.0	-0.2	"
334	20,500	-11.1	1.7	17.0	-9.5	0.8	0.5	0.2	A/1 Off
334	20,400	13.2	1.5	-10.2	17.3	0.6	0.6	(R)	"
334	20,500	-10.3	1.2	16.8	-9.3	0.4	0.4	(R)	B/2 Off
334	20,500	11.6	2.0	-10.3	17.2	0.4	0.4	(R)	"
160	36,600	-15.1	1.3	22.1	-13.6	0.5	0.6	1.0	A/1 Off
160	36,600	17.5	2.0	-13.0	25.0	1.2	0.9	(R)	"
162	36,600	-14.8	1.6	21.5	-13.5	1.1	0.9	-0.5	B/2 Off
160	35,500	18.7	2.3	-14.0	25.0	0.8	1.0	-0.3	"
25[illegible]	35,700	-10.5	2.3	12.0	-8.2	1.7	1.5	0.0	A/1 Off
285	35,700	15.9	2.0	11.8	18.5	0.9	0.8	0.0	"
285	35,600	-15.4	1.6	20.0	13.0	1.4	1.1	-0.2	B/2 Off
285	35,600	15.5	2.6	-11.7	18.3	0.6	0.8	-0.2	"
182	40,700	-14.1	2.0	17.0	-12.1	1.1	0.8	0.0	A/1 Off
183	40,700	16.5	2.3	-11.2	16.5	1.0	1.3	0.3	"
18[illegible]	40,700	-16.8	2.0	19.0	-13.5	1.1	1.0	-0.3	B/2 Off
17[illegible]	40,500	19.5	2.3	-12.5	20.8	0.9	1.4	0.6	"

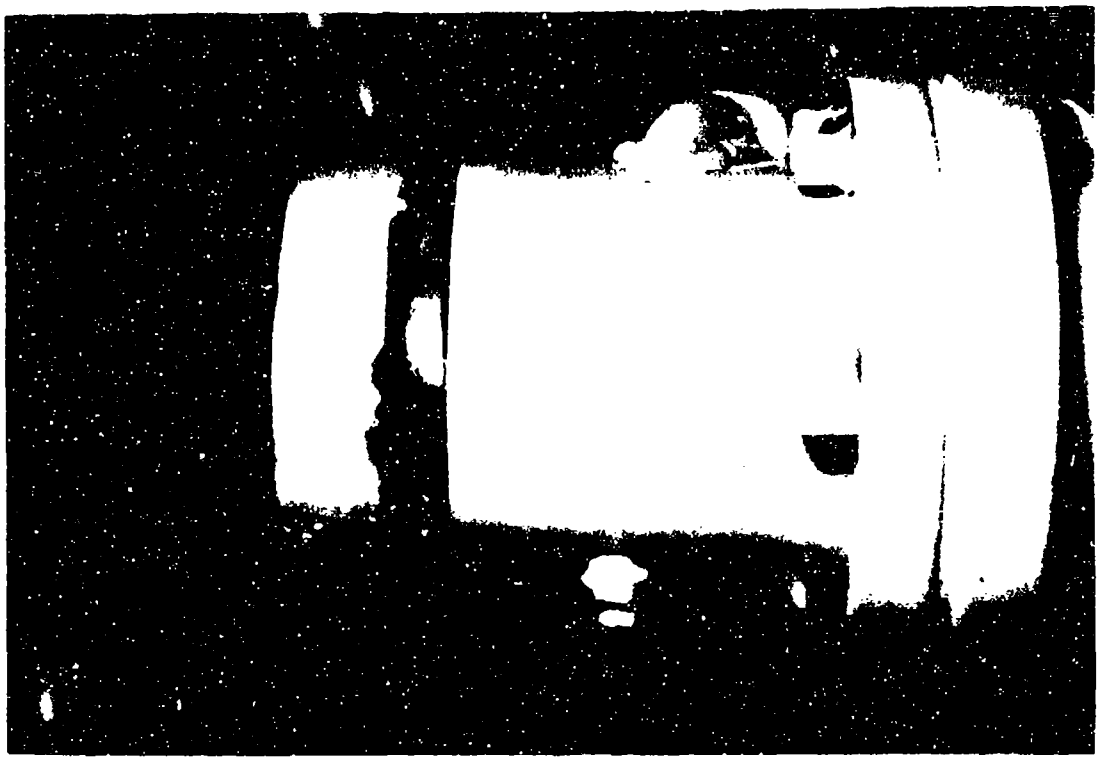
Table 6-3 Tab Operable Roll Performance

Yaw Rate prior to rudder coordination

(N) = no rudder data

Airspeed kts	Altitude ft	Roll Rate deg/sec	Time to Max Rate sec	Initial LH Ail Def deg	Initial RH Ail Def deg	Max LH Ail Def deg	Max RH Ail Def deg	Time to Max LH sec	Time to Max RH sec	Yaw Rate deg/sec	Condition
200	9,100	-14.4	1.6	6.8	8.0	18.6	-13.7	1.0	0.6	0.3 (N)	LH Tab
201	9,400	18.2	2.0	"	"	-7.8	25.0	1.1	0.7	0.8 (N)	"
202	9,300	-14.0	2.3	7.5	8.2	22.6	-3.0	1.4	1.3	-1.0 (N)	RH Tab
201	9,500	16.5	2.3	"	"	-13.8	20.0	2.0	1.9	0.4 (N)	"
192	20,400	-15.5	1.9	7.0	9.1	18.9	-13.7	0.9	0.8	(R)	LH Tab
188	20,300	18.0	2.4	"	"	-6.1	25.0	1.2	0.7	0.9	"
192	20,500	-14.4	2.3	7.5	8.5	22.4	-1.1	1.1	1.1	0.7	RH Tab
194	20,600	17.6	2.3	"	"	-13.3	19.5	1.1	0.4	(R)	"
195	12,000	-	-	8.5	-6.0	-	-	-	-	-	LH Tab
* 205	10,000	-	-	5.6	-4.2	-	-	-	-	-	Full Tab
* 222	34,650	-	-	7.8	-5.0	-	-	-	-	-	"
* 246	36,590	6.5	2.2	-	-	-7.3	18.0	1.0	2.0	-	Full Tab

* from Reference 1



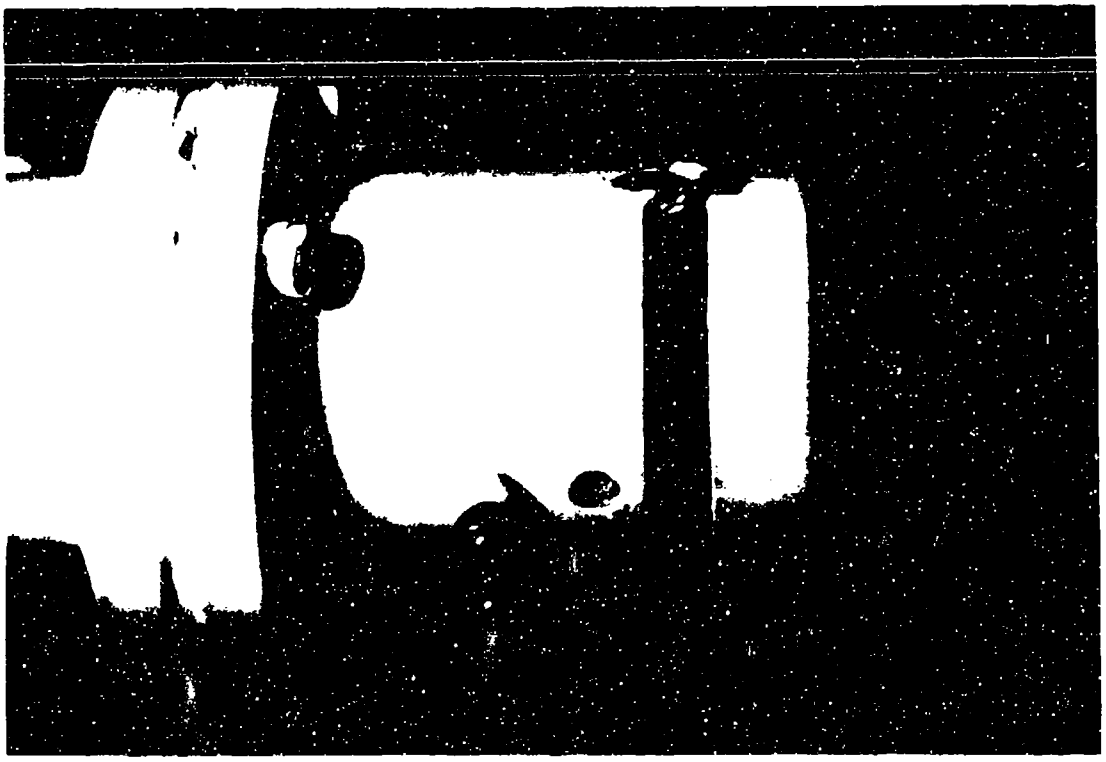


Figure 6-2 EMA Mechanical Stop Damage

CONCLUSIONS AND RECOMMENDATIONS

The EMAS flight test successfully demonstrated the feasibility of powering a primary flight control surface with an electromechanical actuator and tailoring it to specific performance requirements. It has shown that the installation of such a system into a pre-existing airframe can be effected without structural modification and with only minor electrical changes. Specific conclusions and recommendations are given in the order presented in section six with the appropriate page number in parenthesis.

1. A single aileron electromechanical actuator (EMA) can be installed in a C-141A without structural modification to the airframe and without changes to the capacities of existing electrical and hydraulic systems.

2. A dramatic failure of an EMAS system has the potential for producing unusual flight attitudes and motion rates making it difficult for the flight crew to reach system circuit breakers.

CIRCUIT BREAKERS FOR FUTURE INSTALLATIONS MUST BE PLACED IN THE COCKPIT WHERE THEY ARE ACCESSIBLE TO THE FLIGHT CREW DURING EMERGENCIES. (see page 6-1)

3. A single electromechanical actuation system (EMAS) channel displayed sensitivity to the momentary loss of power during power source switching operations, such as switching from aircraft to ground power. It could not be determined if this fault lay with EMAS or the aircraft.

4. The EMAS travel limit and electrical stop settings drifted. This created the potential for inadvertent system shutdown until the difference between the two settings was increased.

FUTURE SYSTEMS SHOULD ELIMINATE THE TENDENCY FOR ELECTRICALLY-SET DEFLECTION LIMITS TO DRIFT. (see page 6-1)

5. Sudden movement of the aileron during initial power up with the surface deflection not in the proper position relative to the yoke may present a hazard to ground personnel. This is also a characteristic of the standard hydromechanical actuator, but the nature of the electromechanical actuator may allow a remedy to this hazard.

FUTURE SYSTEMS SHOULD INCORPORATE A MEANS OF ELIMINATING THE INITIAL POWER-UP MOVEMENT HAZARD WHEN AILERON AND YOKE DEFLECTIONS ARE NOT COINCIDENT. (see page 6-2)

6. With the EMAS unpowered, higher forces were required to move the left aileron than the right.

FUTURE SYSTEMS SHOULD HAVE UNPOWERED BACKDRIVE FORCES OF LESS THAN TEN POUNDS TO EASE MAINTENANCE TASKS. (see page 6-2)

7. Manual positioning of the control surface by rotating the actuator ball nut was not possible in temperatures approximately below freezing. The loss of this capability may hamper maintenance actions.

FUTURE SYSTEMS SHOULD CONSIDER MANUAL COLD-WEATHER POSITIONING OF THE ACTUATOR AS A DESIRABLE FEATURE FOR EASE OF MAINTENANCE EFFORTS. (see page 6-2)

8. Control surface EMAs have the potential for disturbing flux gate compass transmitters normally placed in wing tips and tails.

FUTURE INSTALLATIONS MAY REQUIRE THE RELOCATION OR IMPROVED SHIELDING OF FLUX GATE COMPASSES TO ELIMINATE POTENTIAL INTERFERENCE FROM EMA ELECTRICAL FIELDS. (see page 6-2)

9. The EMAS installation and operation did not alter the damping of the aileron or wing structure.

10. The maximum EMAS current draw of 12.5 amps during the flight test was compatible with available excesses on the aircraft buses; however, future retrofits to existing airframes may require electrical system changes to handle the increased requirements for emergency supply.

11. In flight and on the ground, the EMAS aileron actuator bay temperature was two to seven degrees Centigrade cooler than the right hydraulic actuator bay. No thermal gradients were observed during any maneuvers.

12. During fully functional EMAS operation, the roll control system felt normal to the pilots and the data indicated that the system duplicated the hydromechanical actuator performance with both pilot and autopilot inputs, but for the exceptions noted in this section.

13. The EMAS slow-down feature for aileron deflection rate approaching travel limit is more pronounced than that for the hydraulic PCU and resulted in the left aileron requiring an average of 0.3 seconds longer to reach the same deflection angle as the right aileron for similar test conditions. This contributed to a departure from normal aircraft roll rate but was not evident to the pilot.

14. The EMAS reduced aileron travel rate coming off the surface travel limit. This is not a useful feature.

FUTURE SYSTEMS SHOULD NOT INCORPORATE A REDUCED TRAVEL RATE FOR TRAVEL OFF OF A DEFLECTION LIMIT STOP. (see page 6-4)

15. Right rolls occurred at a higher rate than left rolls and also higher than baseline roll rates for maximum command roll inputs. Left rolls were at a lower rate than baseline. This is attributable to the lower deflection rate of the left aileron near the travel limit and the failure of the EMAS to produce the maximum aileron travel.

16. The EMAS-driven aileron made more small position corrections than the normal aileron system during autopilot trim shots. The EMAS-driven aileron also made small deflection excursions in response to control system aberrations that the hydromechanical system did not repond to. These excursions were not felt by the pilot and did not affect roll performance.

FUTURE SYSTEMS MATED TO A MECHANICAL CONTROL SYSTEM SHOULD CONSIDER THE INCREASED SENSITIVITY OF AN EMAS TO SMALL CONTROL SYSTEM ABERRATIONS THAT MAY PRODUCE SHARP SURFACE DEFLECTIONS INCONSISTENT WITH CONTROL INPUT. (see page 6-4)

17. Deactivation or reactivation of one of the two redundant channels produced no aileron motion. Single channel performance was comparable to dual channel operation.

18. Aileron tab operable tests indicated that the EMAS installation resulted in no roll performance degradation for single aileron powered configurations. The increased forces required to backdrive the unpowered EMA as compared with the hydromechanical system resulted in a less severe asymmetric control yoke deflection and less control force with tab operable on the left aileron only.

19. EMAS interfaced with the autopilot without difficulty but displayed increased sensitivity to small autopilot inputs.

20. Takeoffs and landings showed no abnormal aileron behavior.

21. The EMAS suffered an inflight fault and, during a subsequent ground incident, experienced motor runaways with and without control inputs.

22. The emergency mechanical stops on the actuator were not designed to withstand the impact forces of the programmed normal maximum travel rate or the unrestrained full motor capacity travel rate and failed during a ground incident.

FUTURE SYSTEMS SHOULD INCORPORATE A PRIMARY AND BACKUP MECHANICAL TRAVEL LIMIT STOP THAT WILL WITHSTAND FORCES IMPARTED BY A FULL MOTOR SPEED IMPACT. (see page 6-7)

LIST OF ABBREVIATIONS

Item	Definition	Units
AEA	all-electric airplane	---
AFWAL	Air Force Wright Aeronautical Laboratories	---
A/P	autopilot	---
C	Centigrade	C
CE	Controller Electronics	---
CWS	control wheel steering	---
DC	direct current	Amps
deg	degrees	deg
EMA	Electromechanical Actuator	---
EMAS	Electromechanical Actuation System	---
EMC/EMI	electromagnetic compatibility/electromagnetic interference	---
F	Fahrenheit	F
GeLac	Lockheed-Georgia Co.	---
GVT	ground vibration test	---
Hz	Hertz	Hz
ILS	Instrument Landing System	---
KCAS	Knots Calibrated Airspeed	knots
M	Mach	M
MILSPEC	Military Specification	---
PA	pressure altitude	feet
PCU	power control unit	---
q	dynamic pressure	lb/sq in
RVDT	Rotary Variable Differential Transformer	---

LIST OF ABBREVIATIONS (continued)

Item	Definition	Units
T.E.	trailing edge	---
VAC	volts alternating current	volts
VDC	volts direct current	volts
Ve	best endurance airspeed	knots

REFERENCES

1. McCabe, J. M., Capt, USAF and Benefield, T. D., Maj, UASF, C-141A Category II Stability and Control Test, FTC-TR-66-5, Air Force Flight Test Center, Edwards AFB, CA, June 1966.

2. USAF Technical Order 1C-141A-2-1GA-2, Flight Controls.

APPENDIX A

TEST DATA

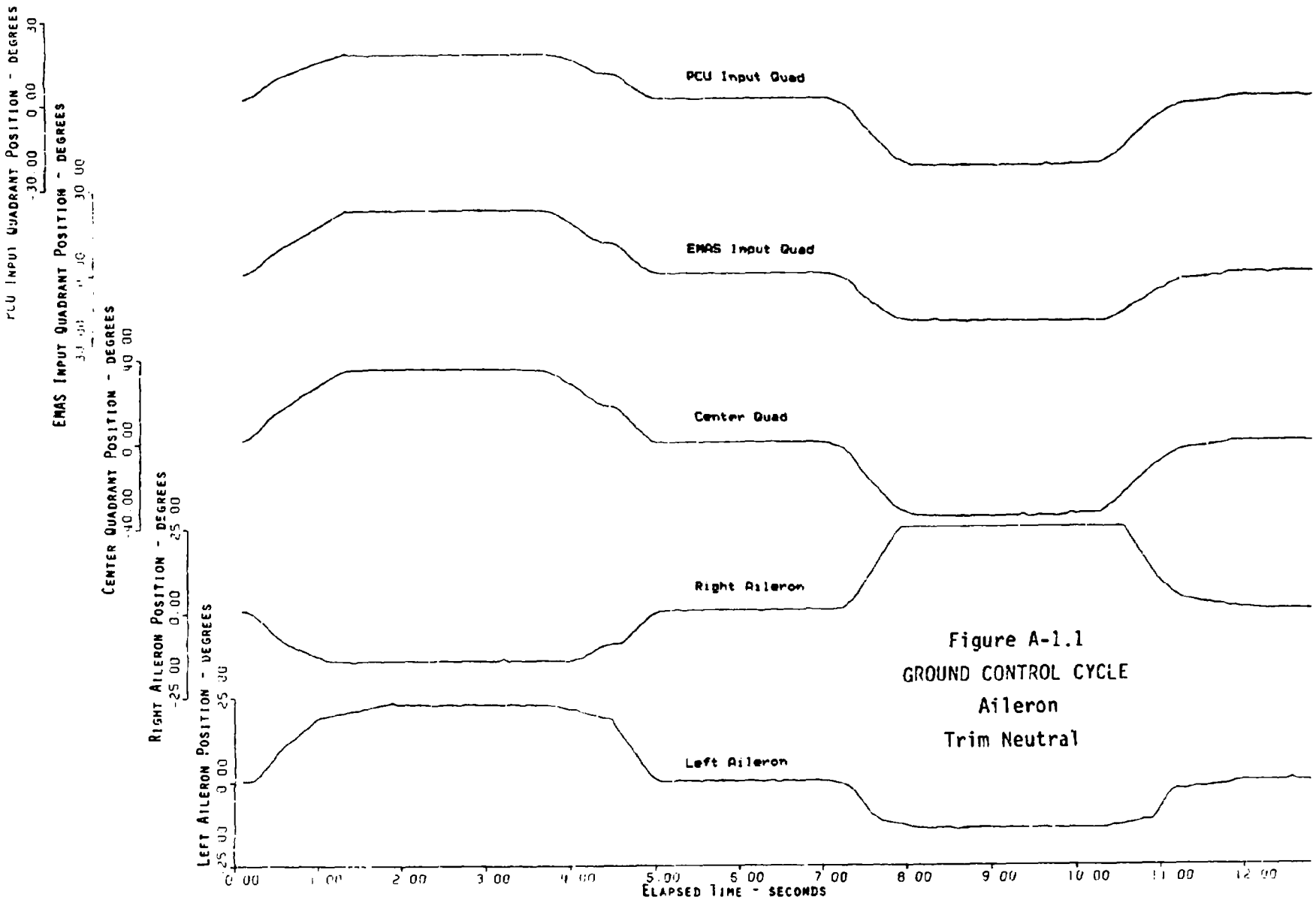


Figure A-1.1
GROUND CONTROL CYCLE
Aileron
Trim Neutral

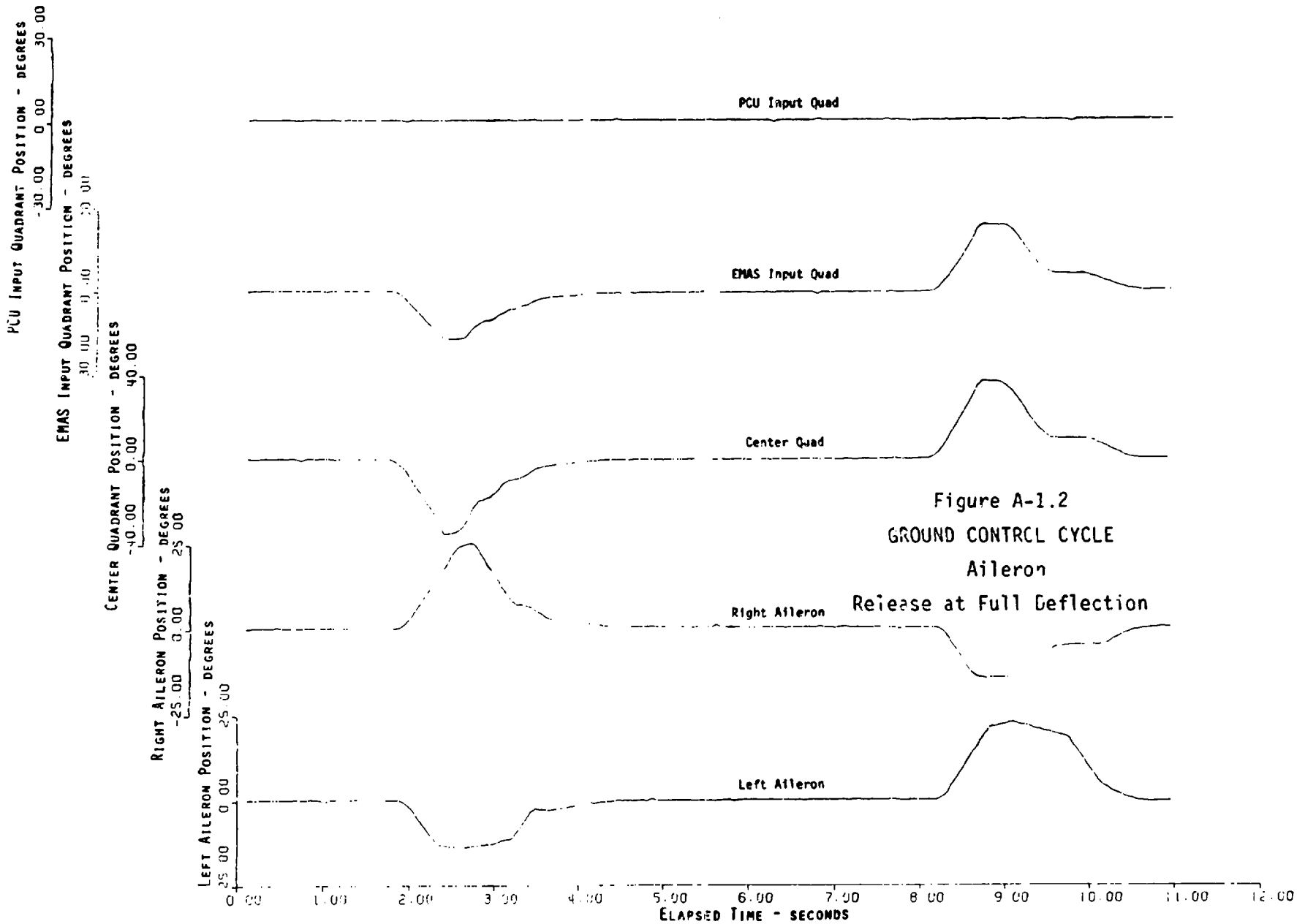


Figure A-1.2
GROUND CONTROL CYCLE
Aileron
Release at Full Deflection

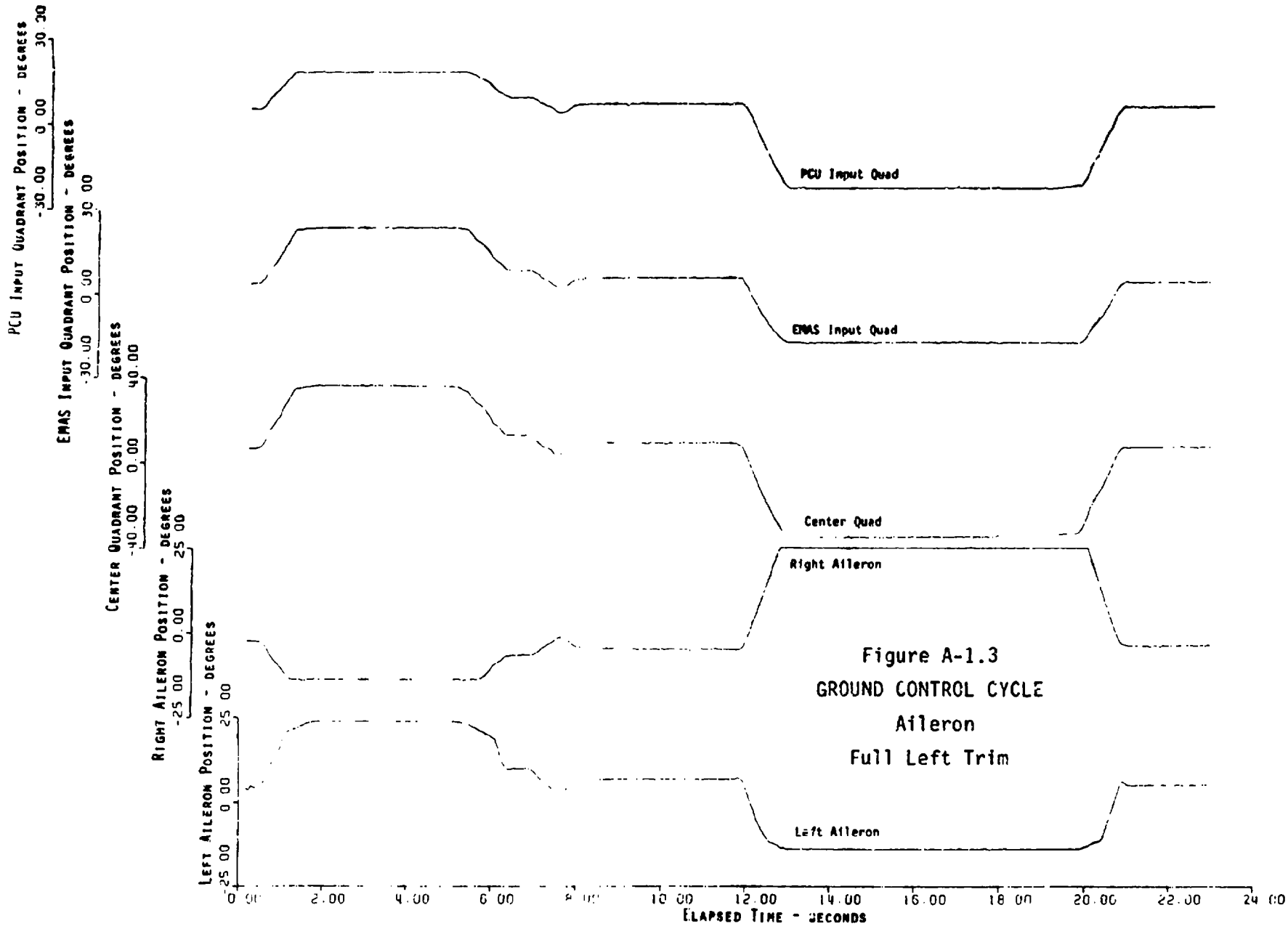


Figure A-1.3
GROUND CONTROL CYCLE
Aileron
Full Left Trim

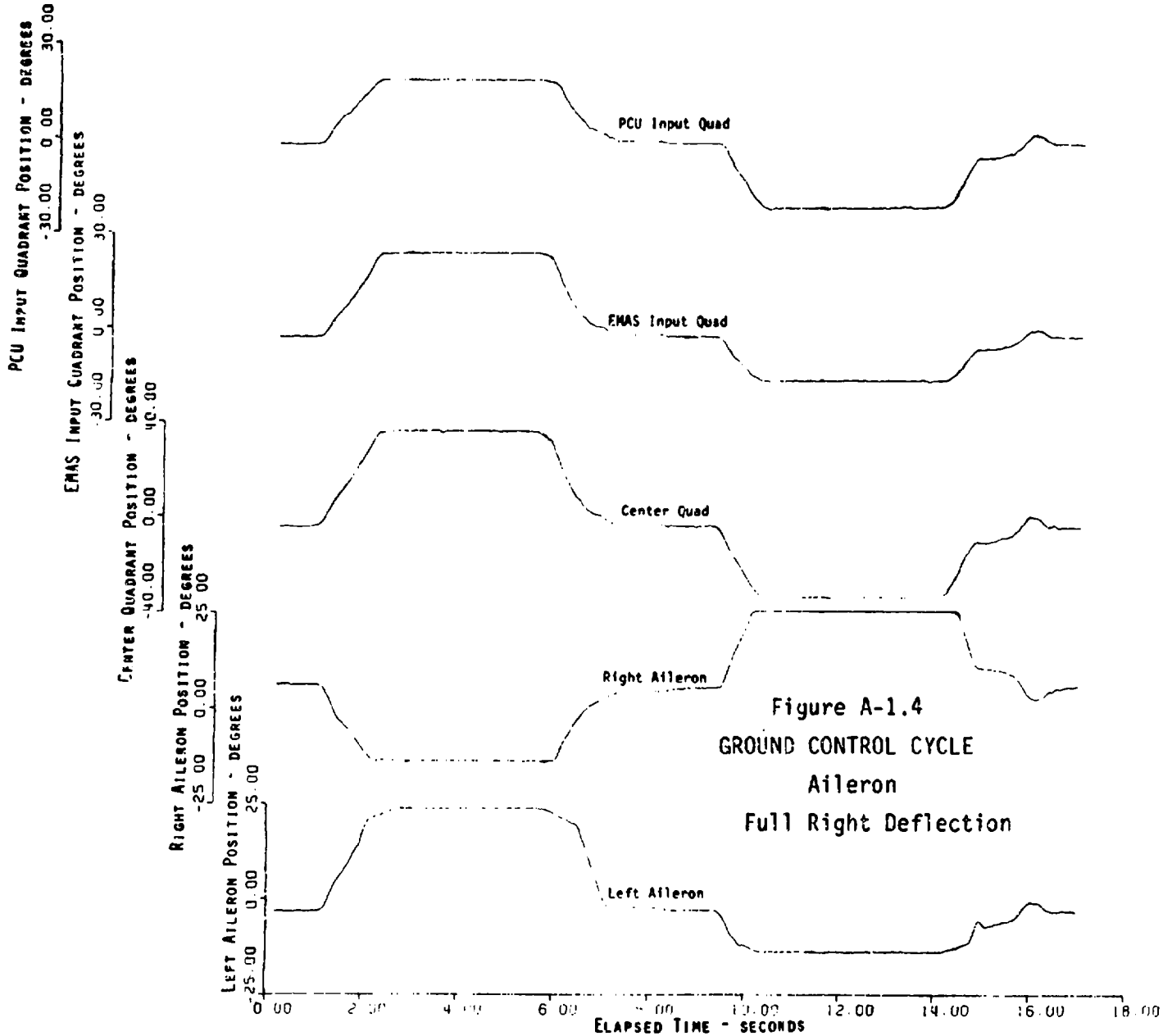
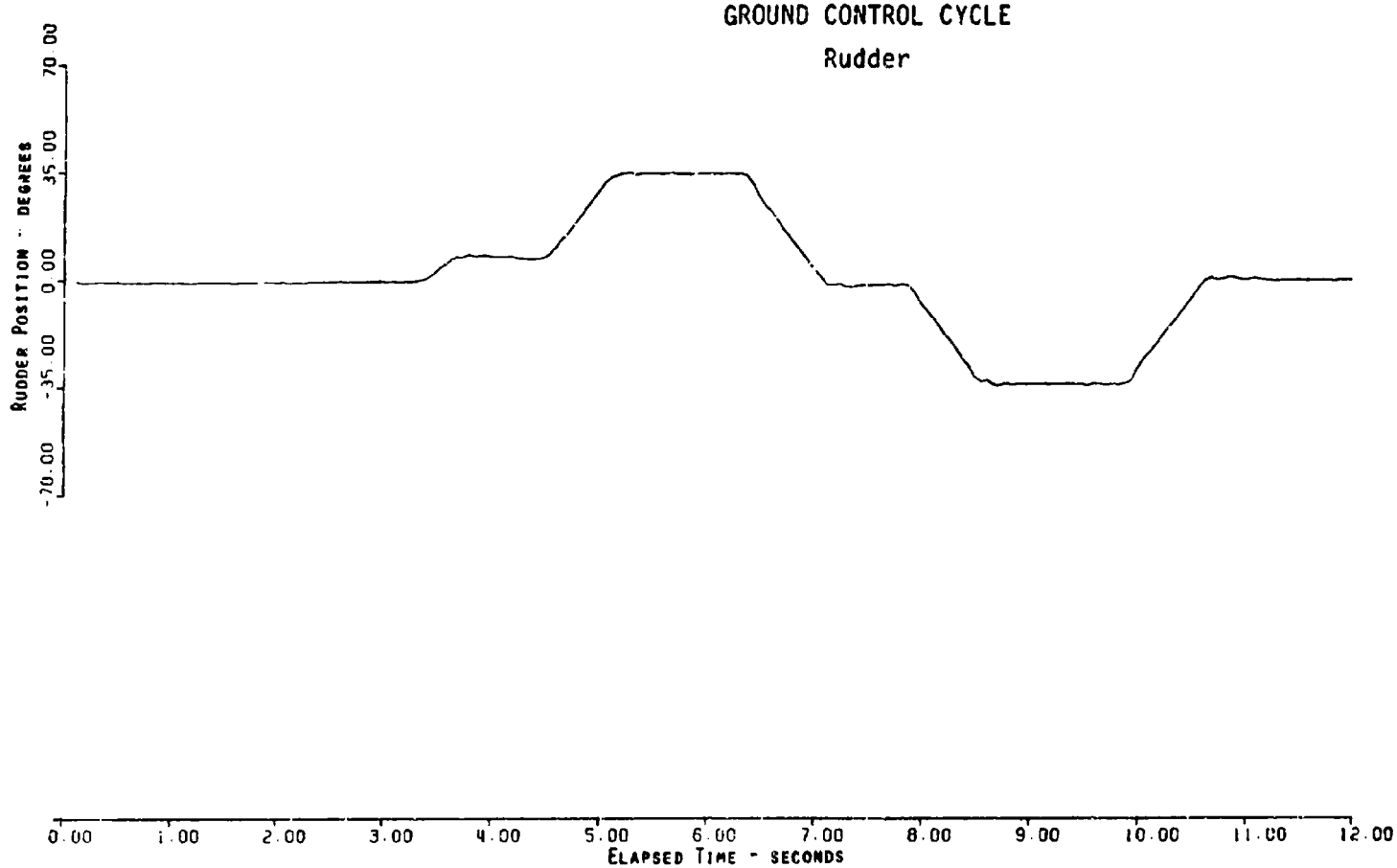


Figure A-1.4
GROUND CONTROL CYCLE
Aileron
Full Right Deflection

Figure A-1.5

GROUND CONTROL CYCLE

Rudder



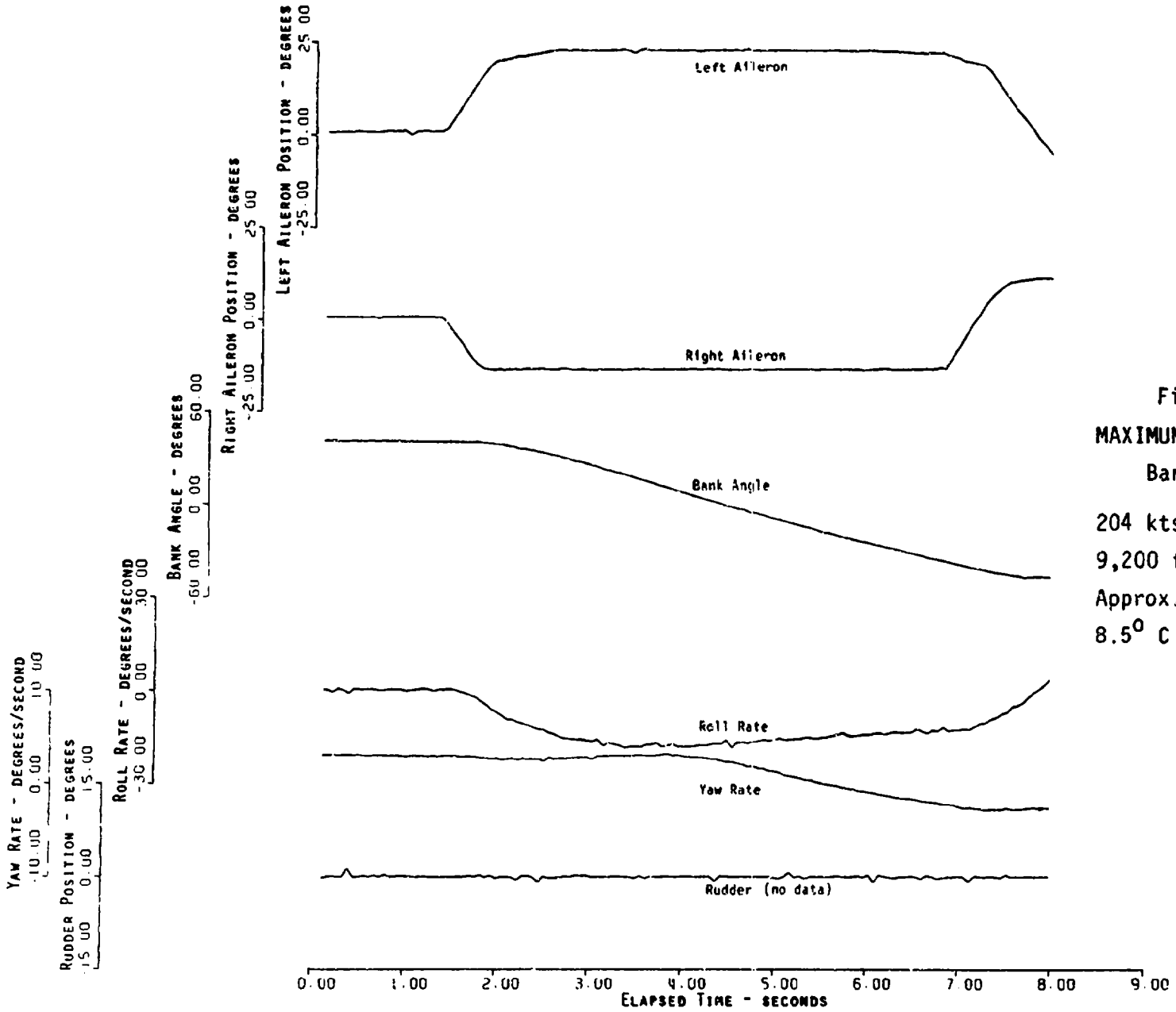


Figure A-2.1a

MAXIMUM DEFLECTION ROLL

Bank-to-Bank

204 kts Airspeed

9,200 ft Altitude

Approx. 180,000 lb G.W.

8.5° C OAT

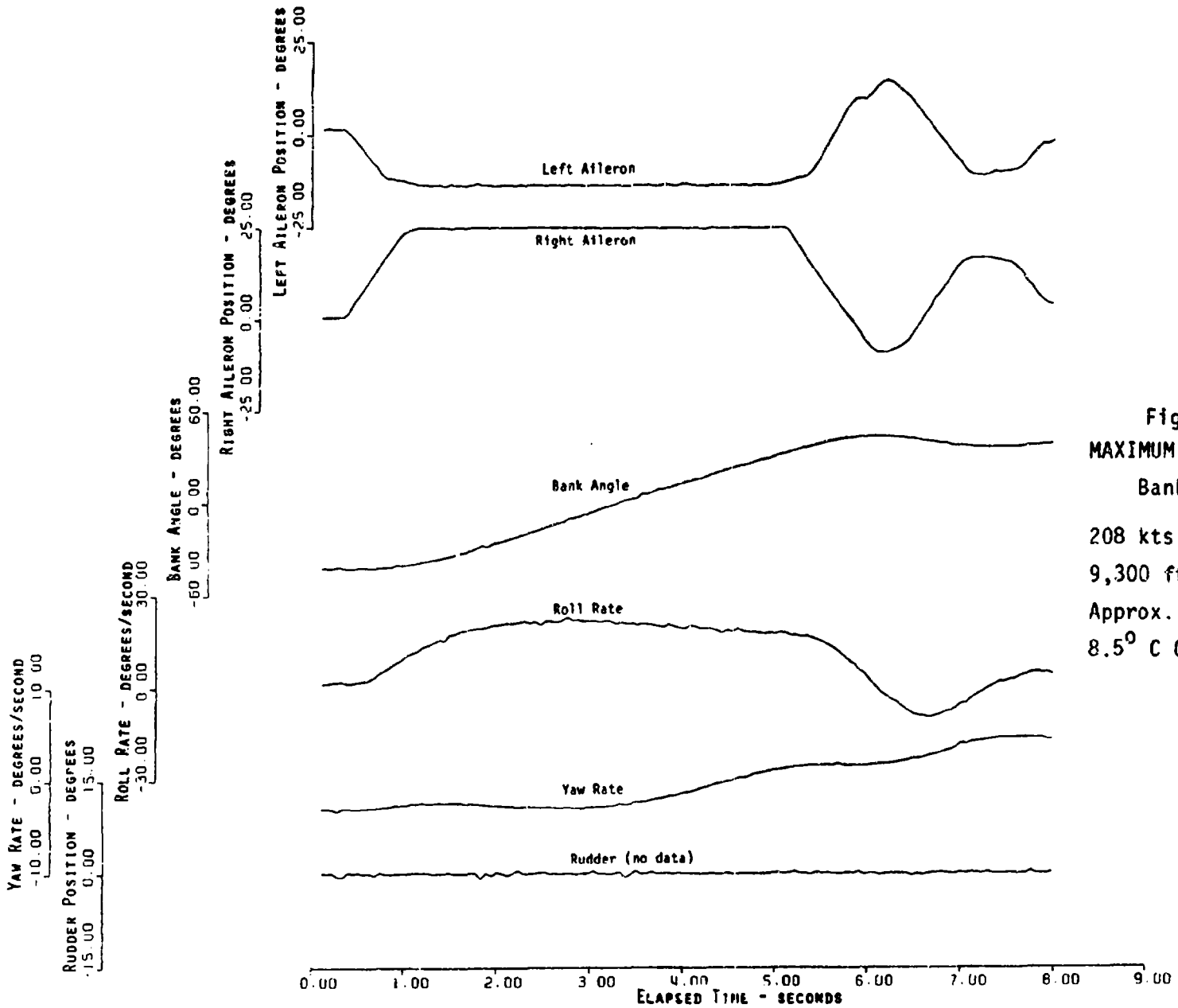


Figure A-2.1b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

208 kts Airspeed
9,300 ft Altitude
Approx. 180,000 lb G.W.
8.5° C OAT

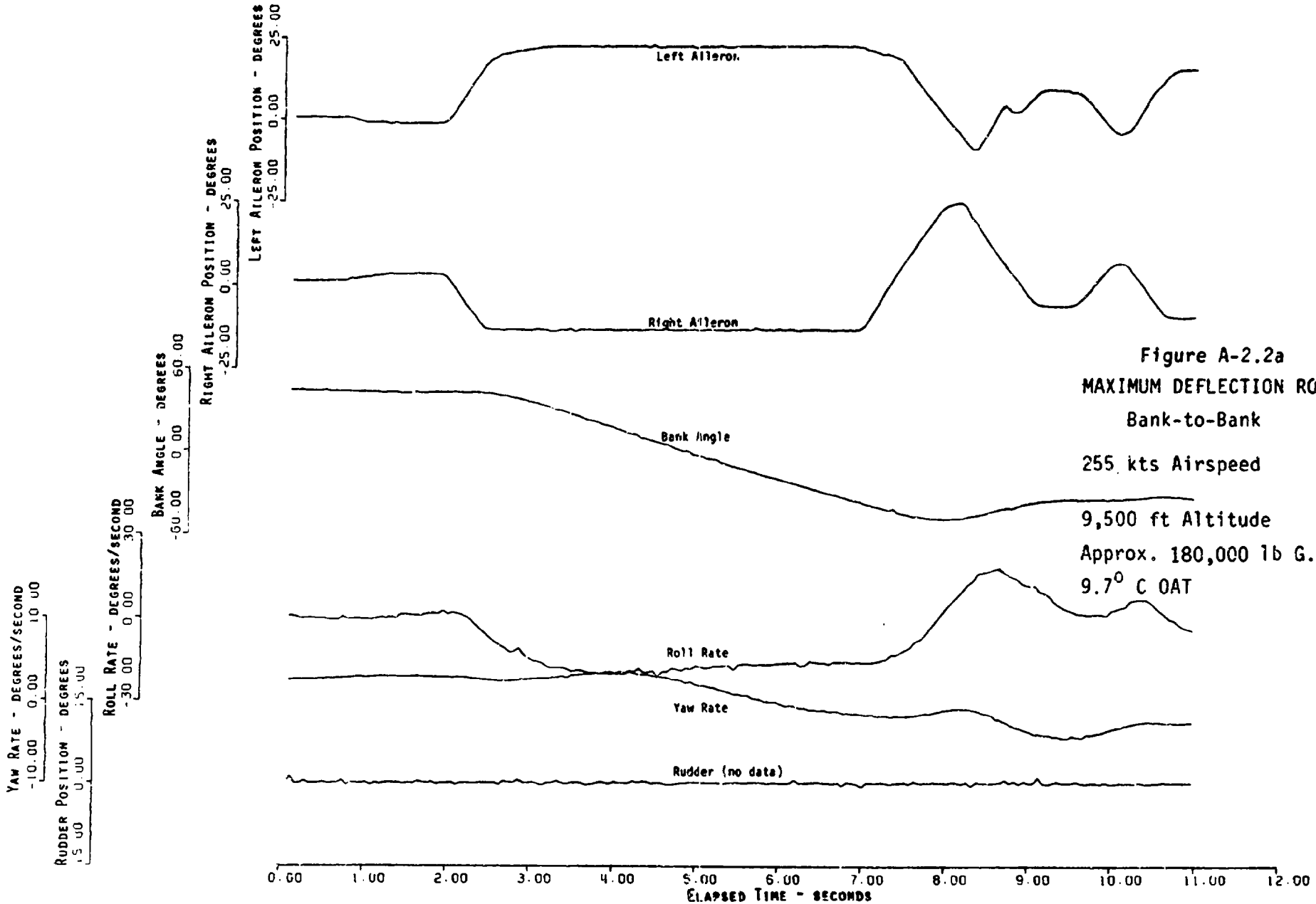


Figure A-2.2a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

255 kts Airspeed

9,500 ft Altitude
Approx. 180,000 lb G.W.
9.7° C OAT

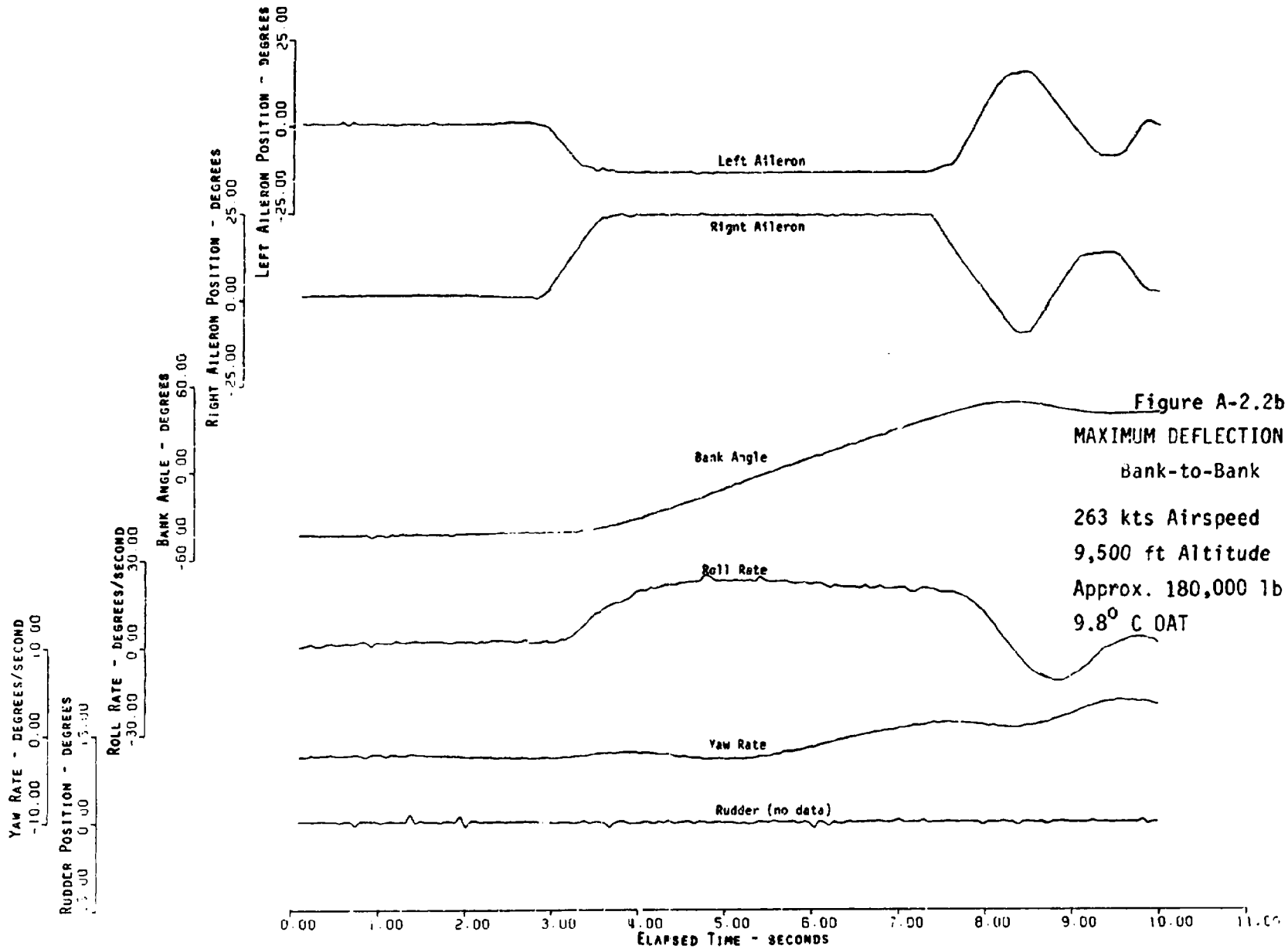


Figure A-2.2b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

263 kts Airspeed
9,500 ft Altitude
Approx. 180,000 lb G.W.
9.8° C OAT

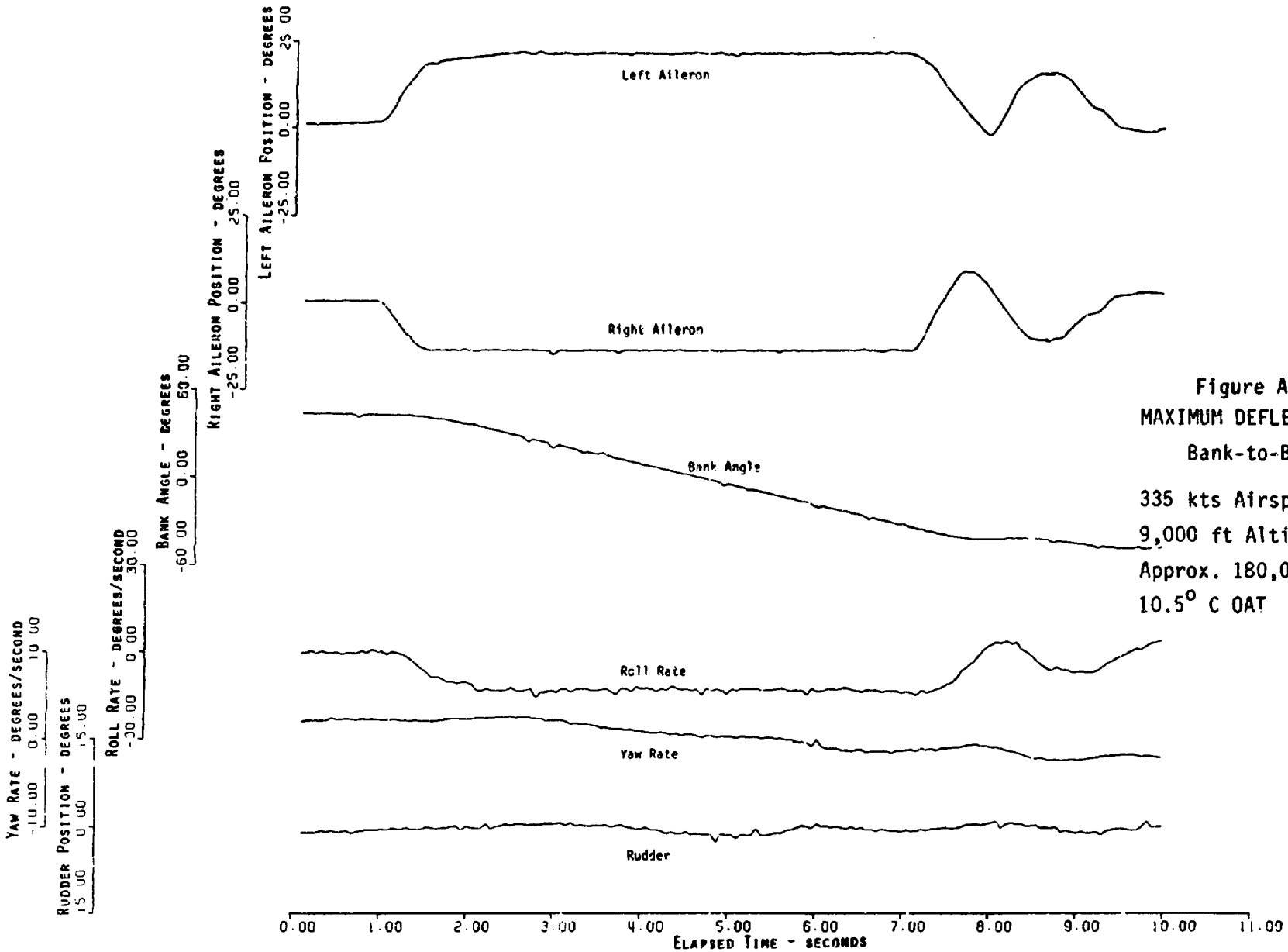


Figure A-2.3a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

335 kts Airspeed
9,000 ft Altitude
Approx. 180,000 lb G.W.
10.5° C OAT

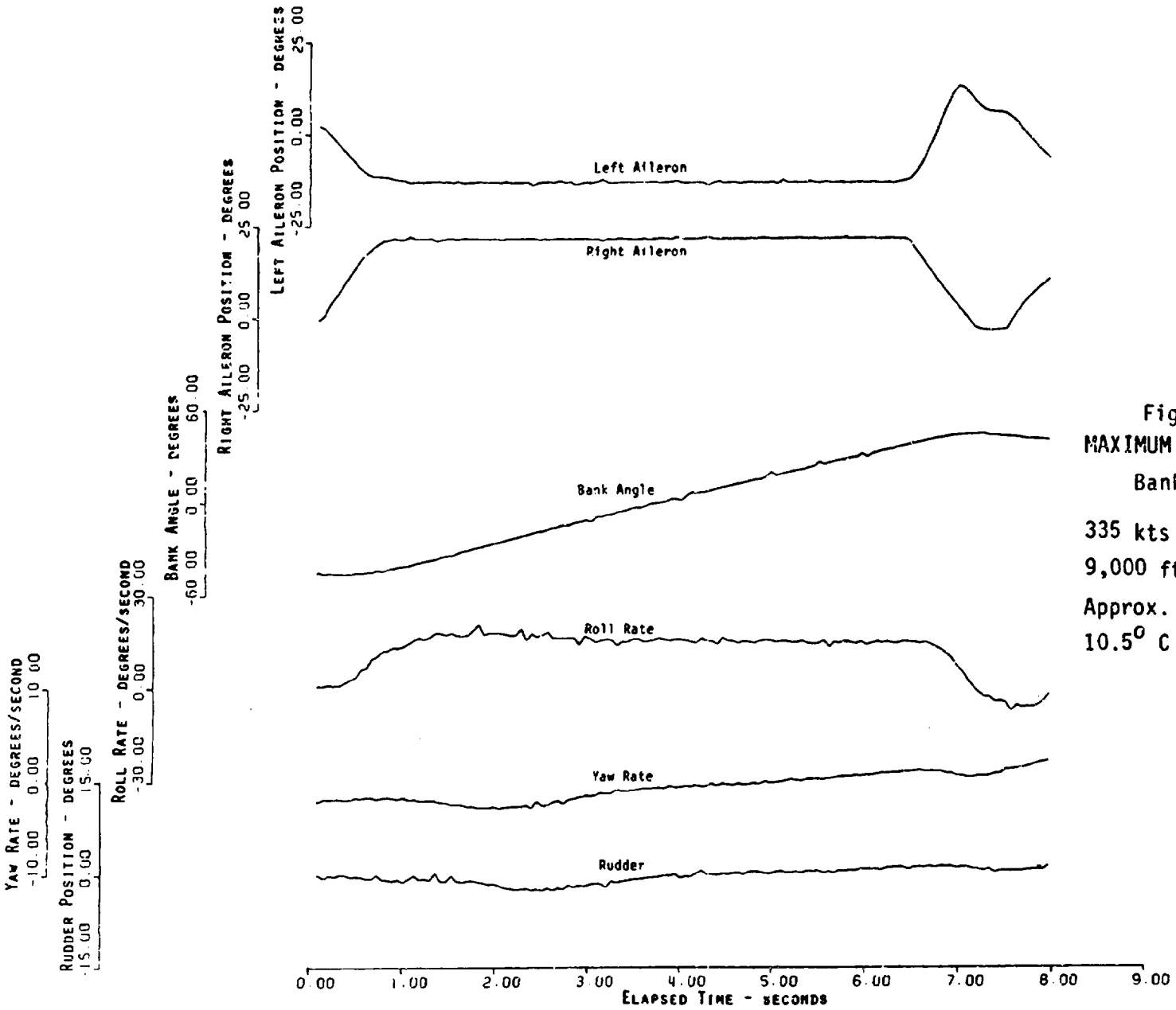
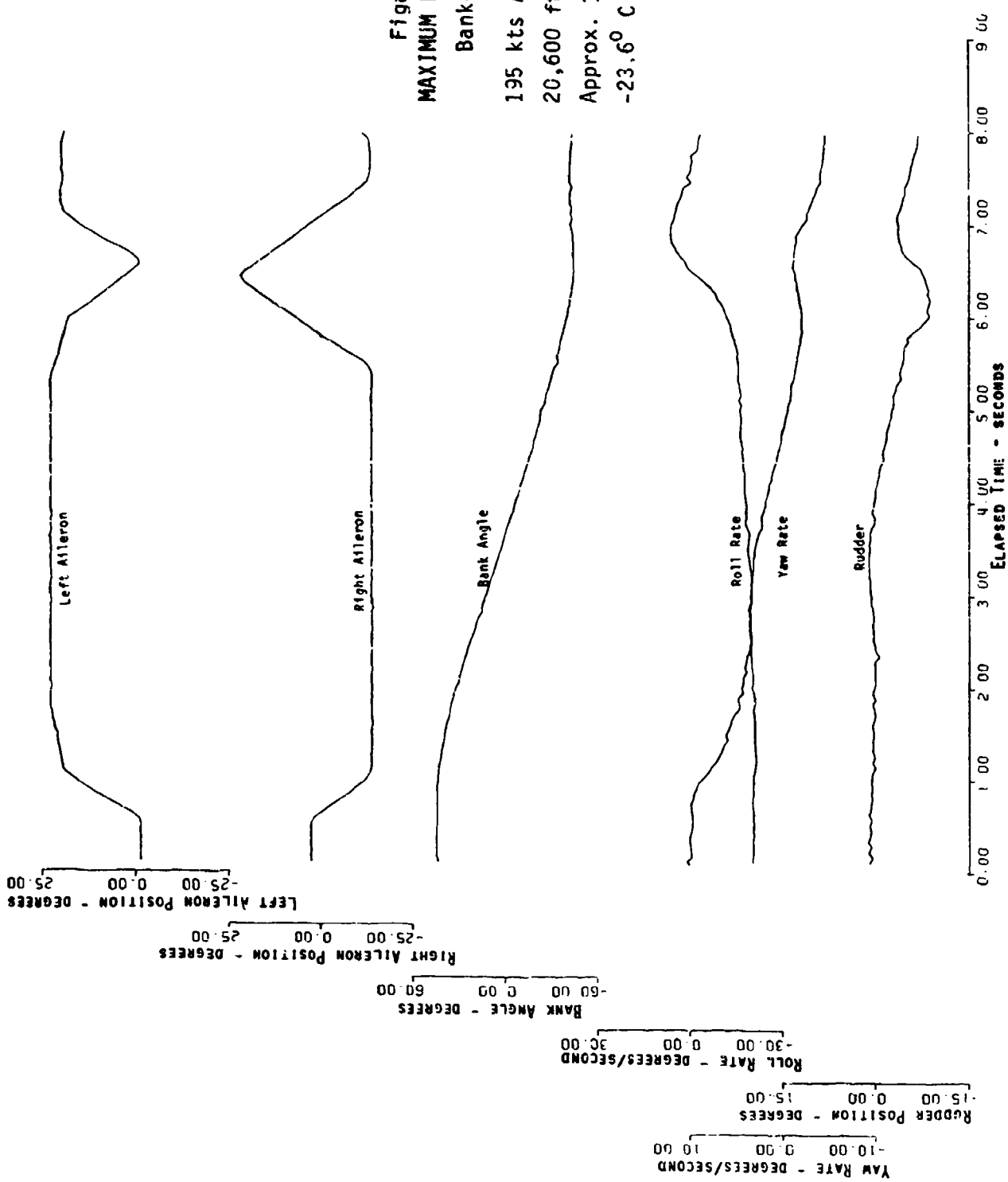


Figure A-2.3b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

335 kts Airspeed
9,000 ft Altitude
Approx. 180,000 lb G.W.
10.5^{o} C OAT

Figure A-2.4a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

195 kts Airspeed
20,600 ft Altitude
Approx. 180,000 lb G.W.
-23.6° C OAT



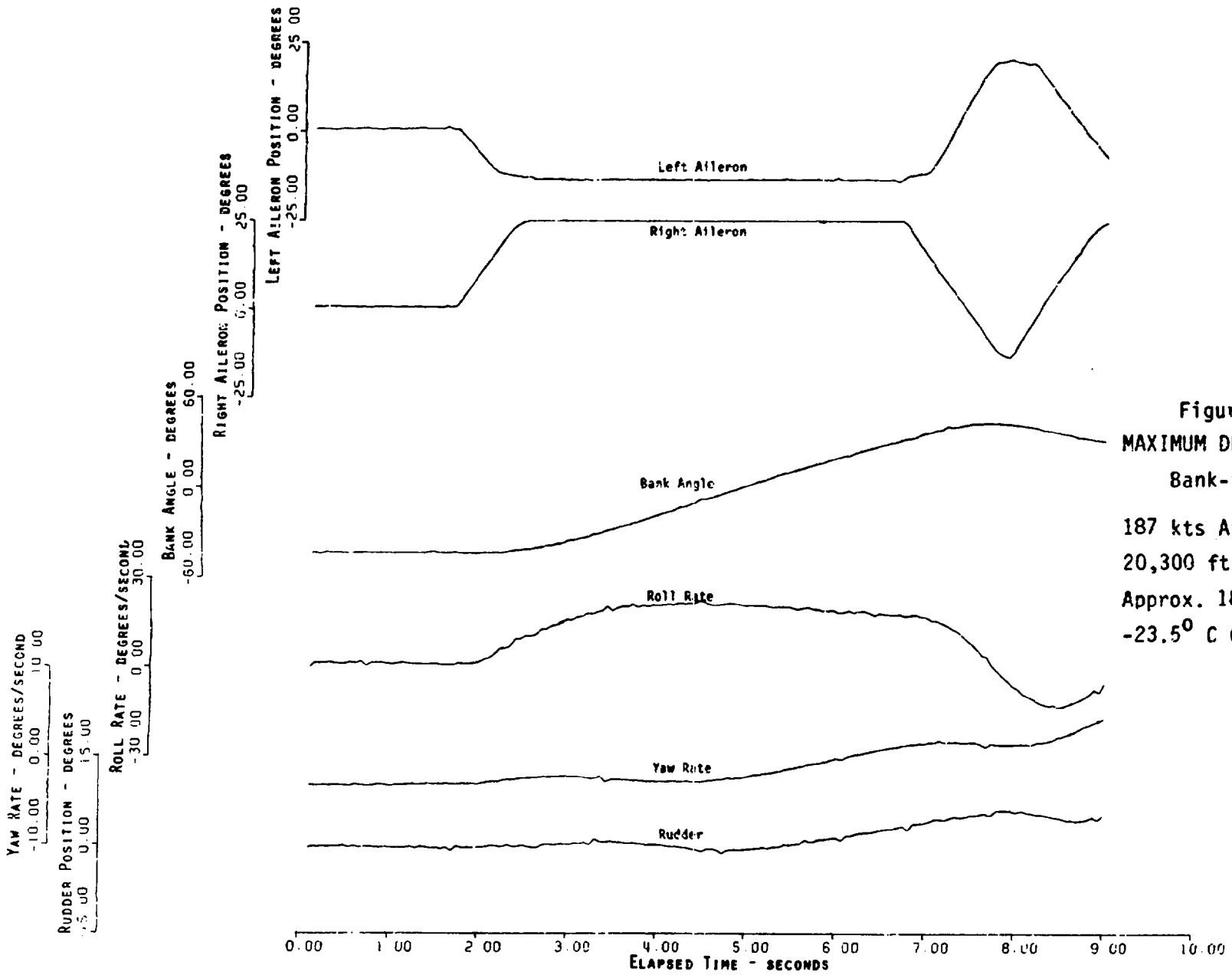


Figure A-2.4b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

187 kts Airspeed
20,300 ft Altitude
Approx. 180,000 lb G.W.
-23.5° C OAT

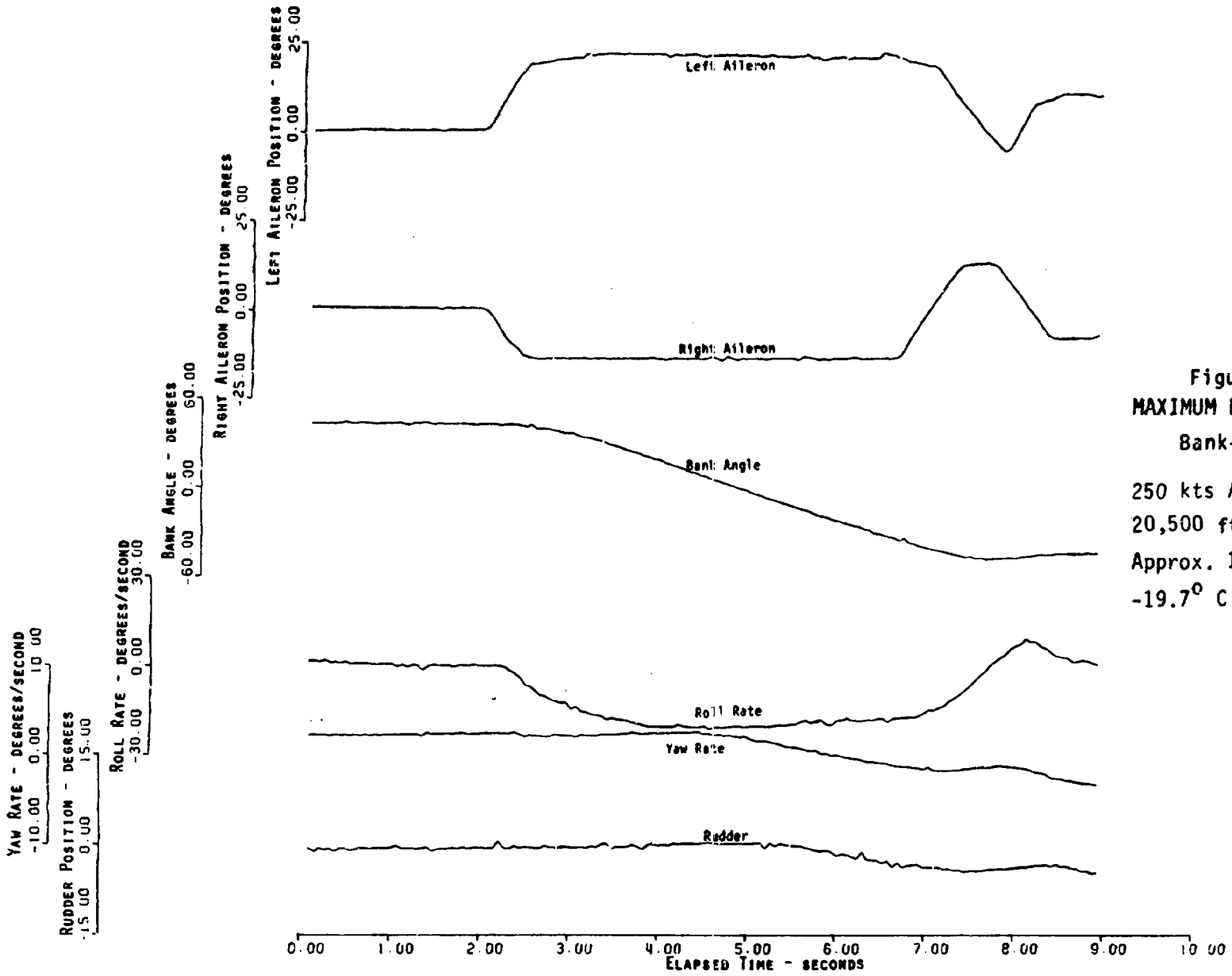


Figure A-2.5a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

250 kts Airspeed
20,500 ft Altitude
Approx. 180,000 lb G.W.
-19.7° C OAT

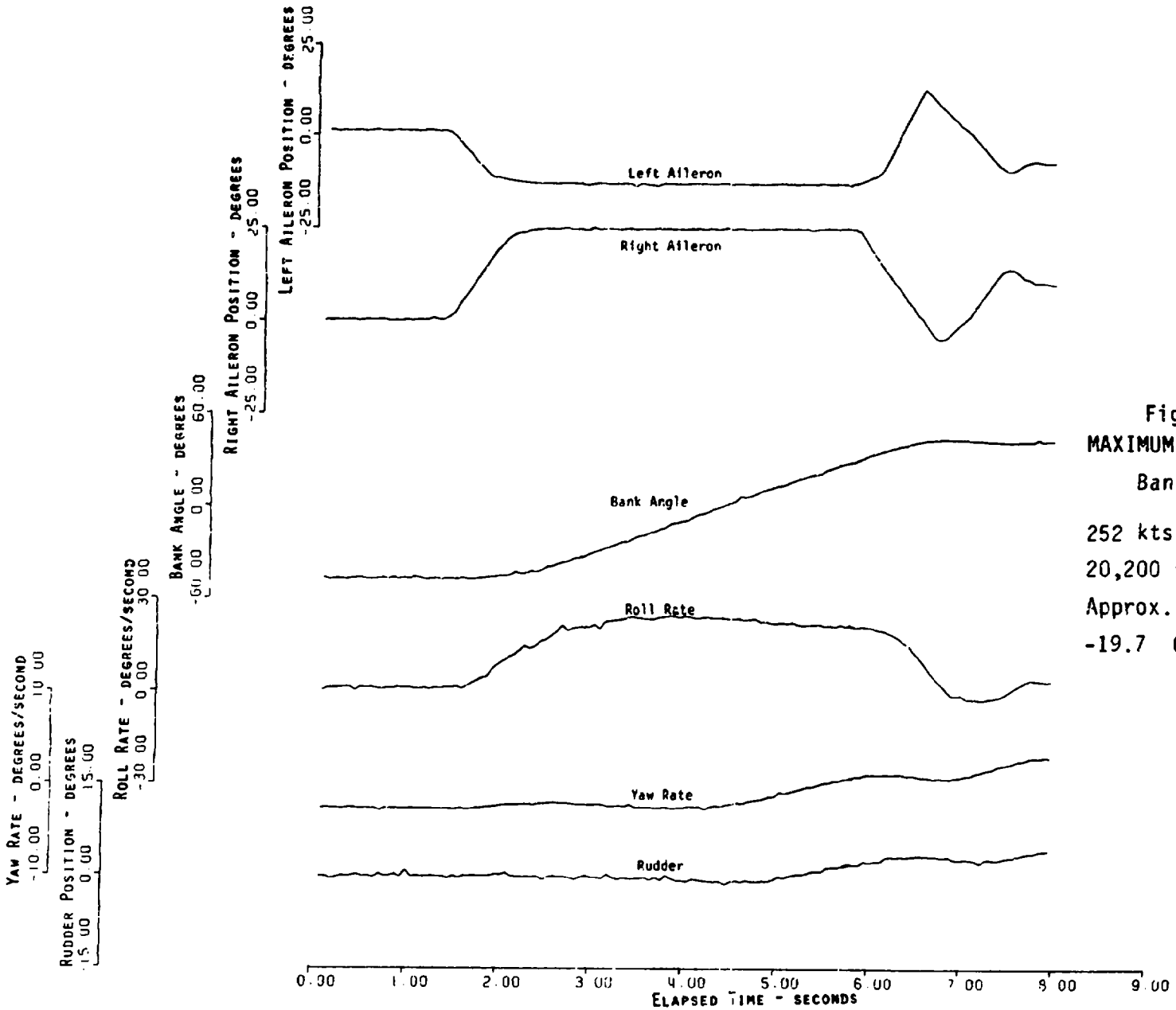


Figure A-2.5b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

252 kts Airspeed
20,200 ft Altitude
Approx. 180,000 lb G.W.
-19.7 C OAT

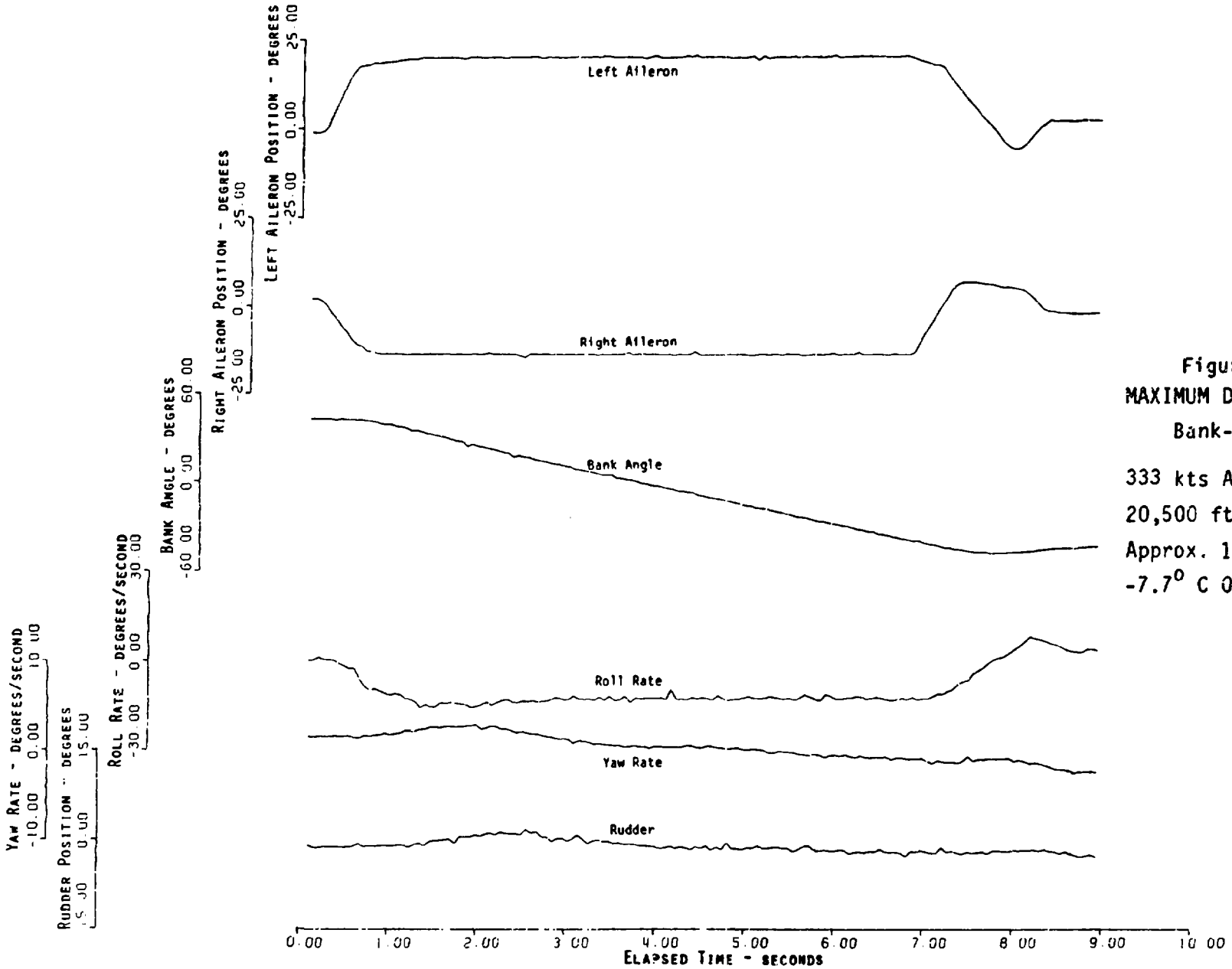


Figure A-2.6a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

333 kts Airspeed
20,500 ft Altitude
Approx. 180,000 lb G.W.
-7.7° C OAT

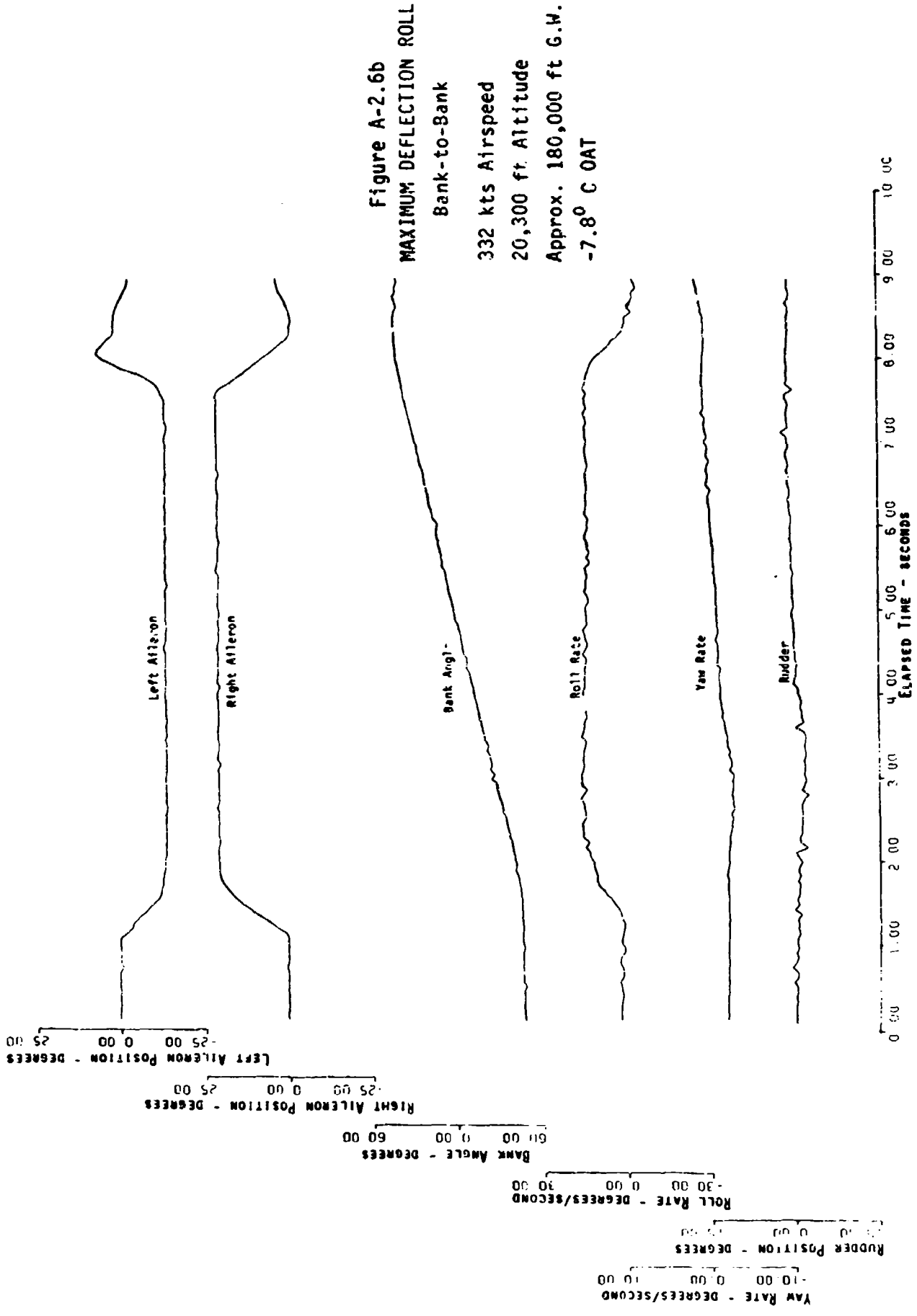


Figure A-2.6b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

332 kts Airspeed
20,300 ft Altitude
Approx. 180,000 ft G.W.
-7.8° C OAT

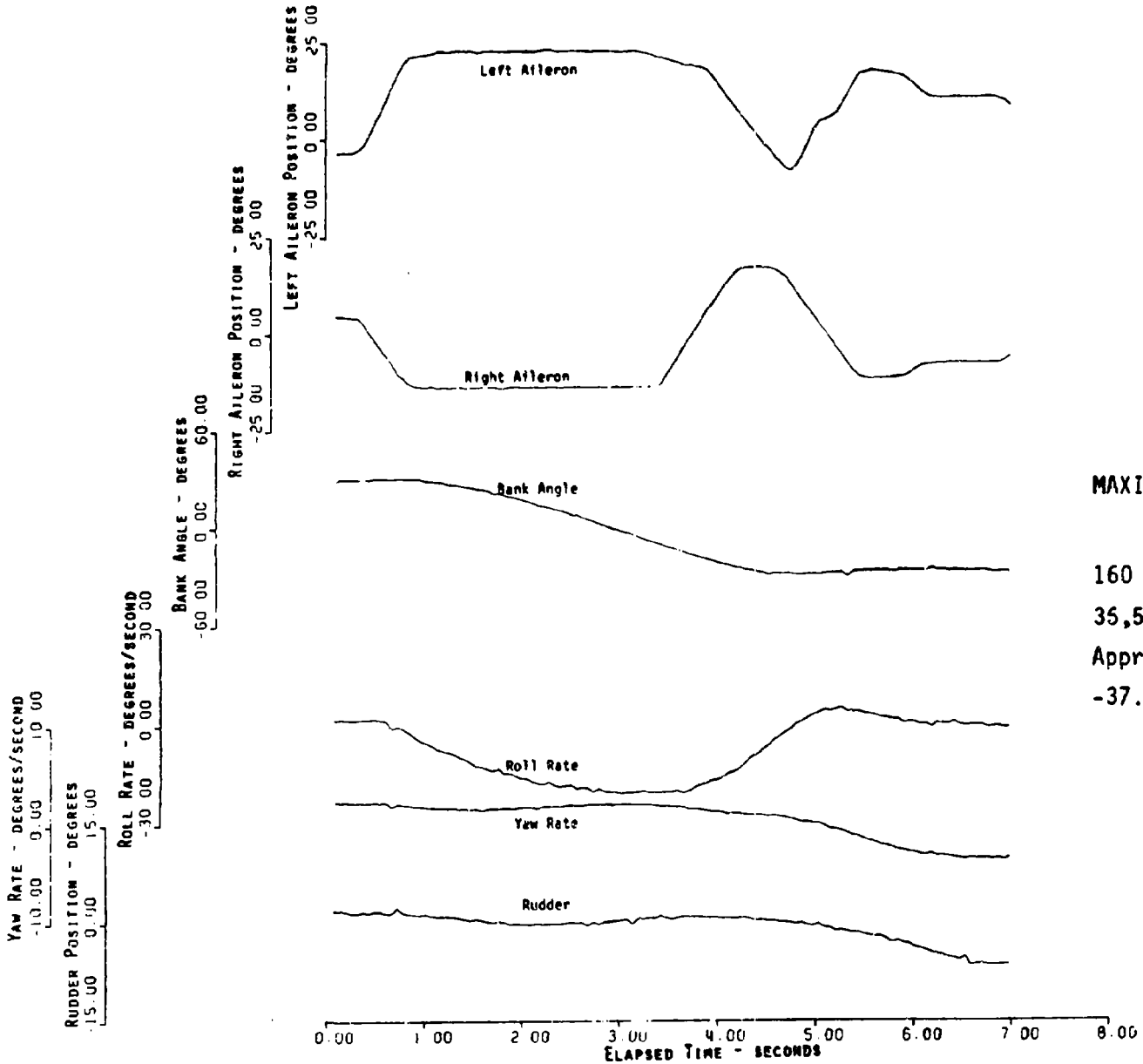


Figure A-2.7a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

160 kts Airspeed
35,500 ft Altitude
Approx. 180,000 lb G.W.
-37.1° C OAT

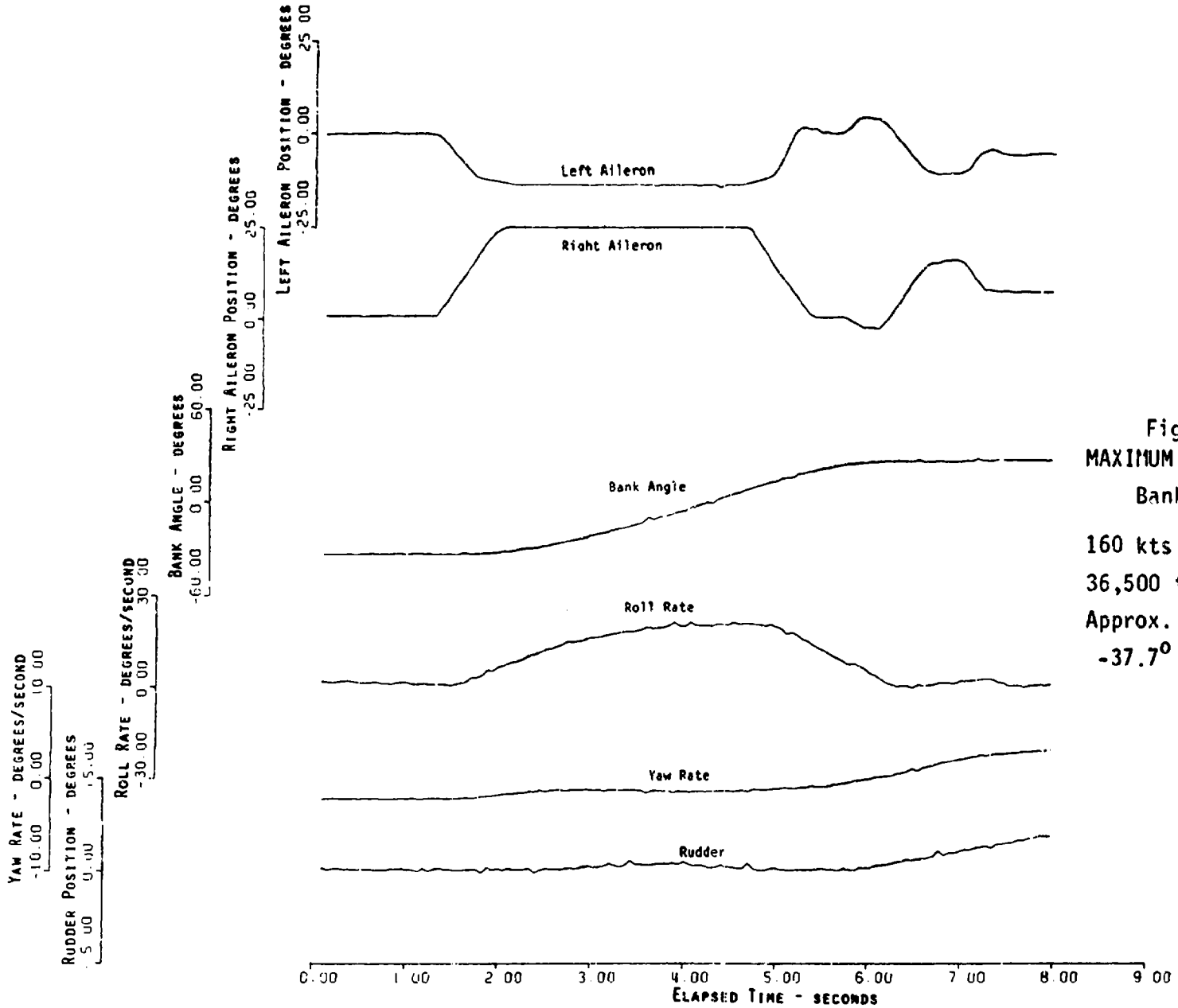


Figure A-2.7b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

160 kts Airspeed
36,500 ft Altitude
Approx. 180,000 lb G.W.
-37.7° C OAT

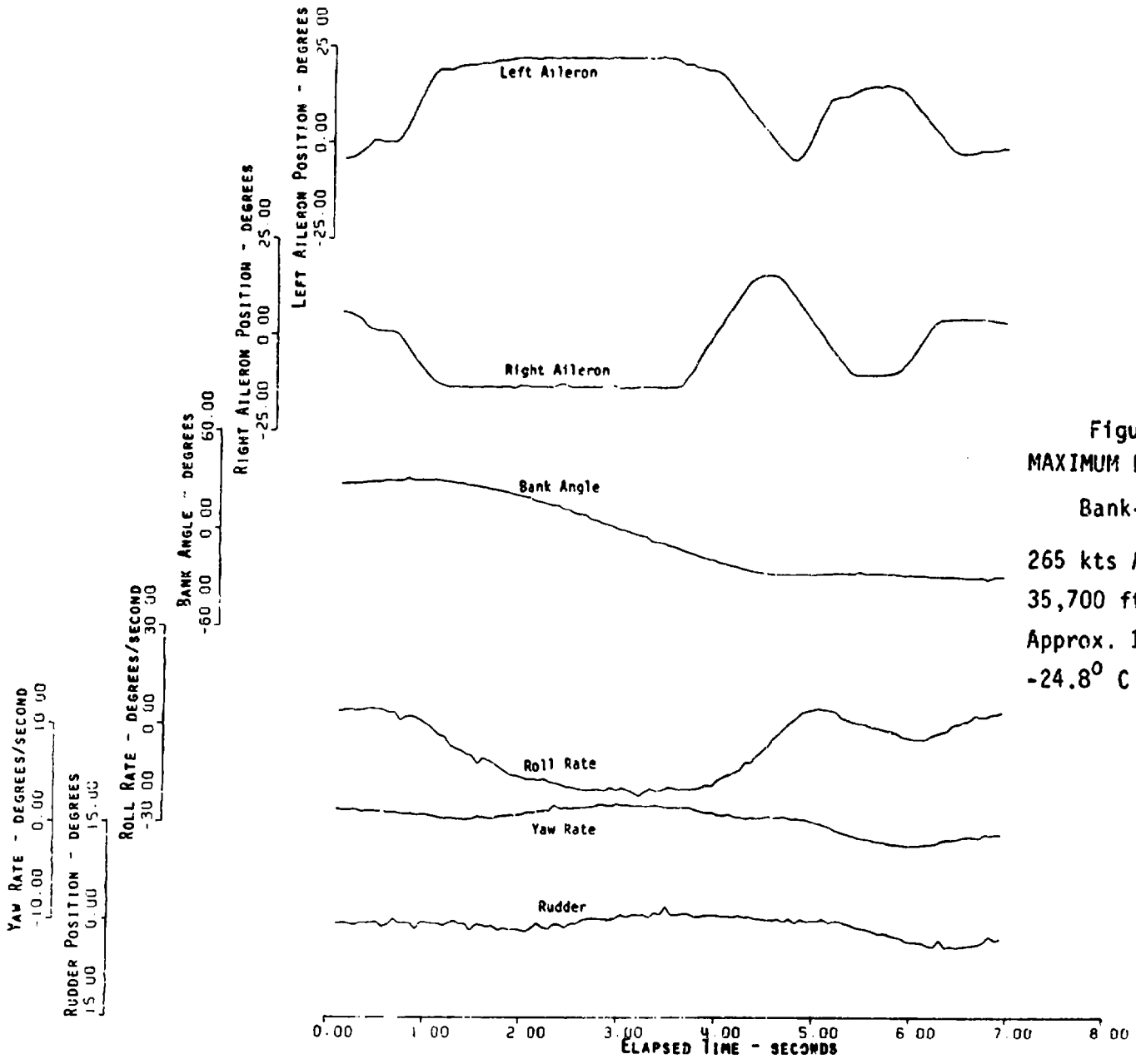


Figure A-2.8a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

265 kts Airspeed
35,700 ft Altitude
Approx. 180,000 lb G.W.
-24.8° C OAT

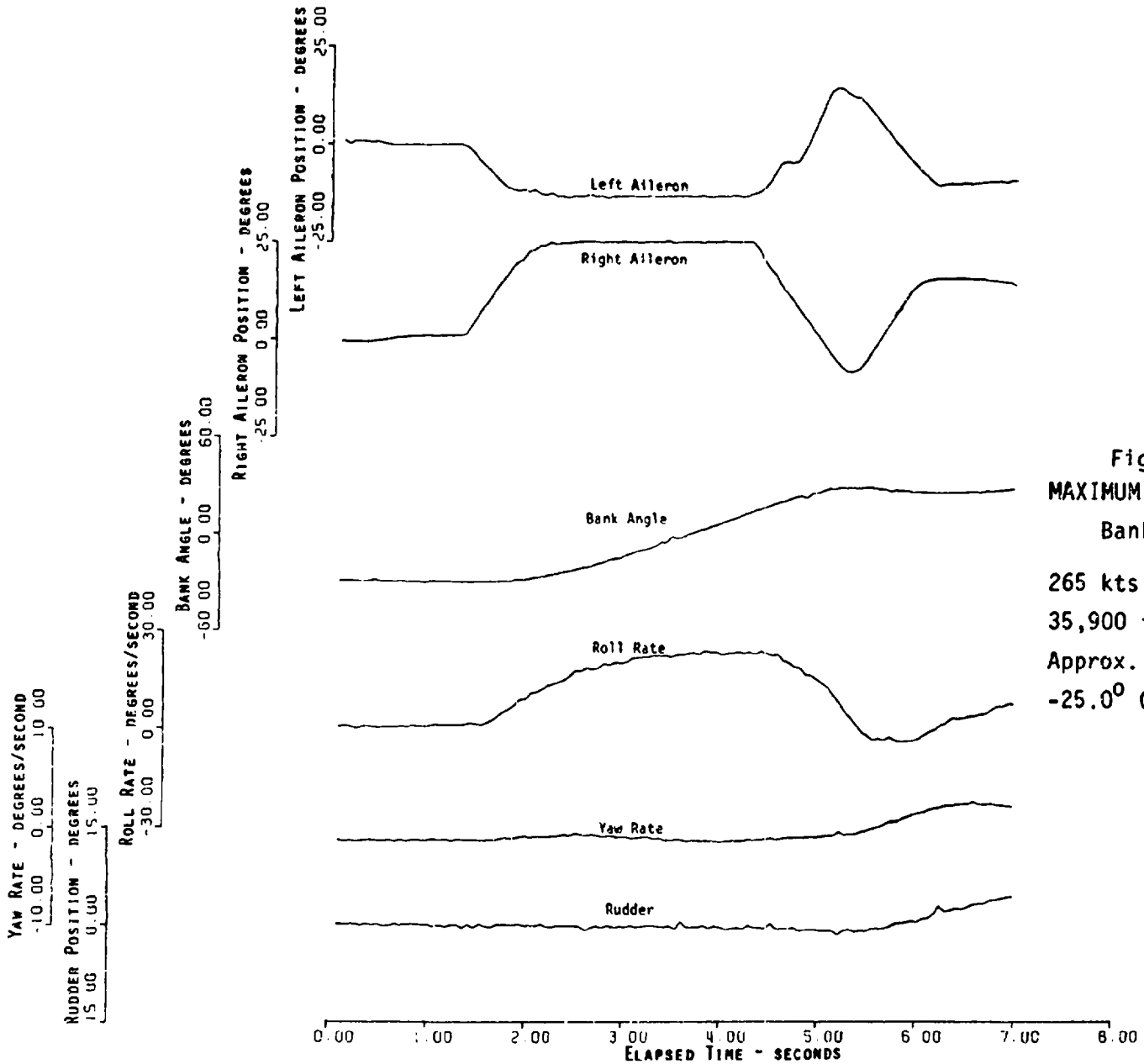


Figure A-2.8b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

265 kts Airspeed
35,900 ft Altitude
Approx. 180,000 lb G.W.
-25.0° C OAT

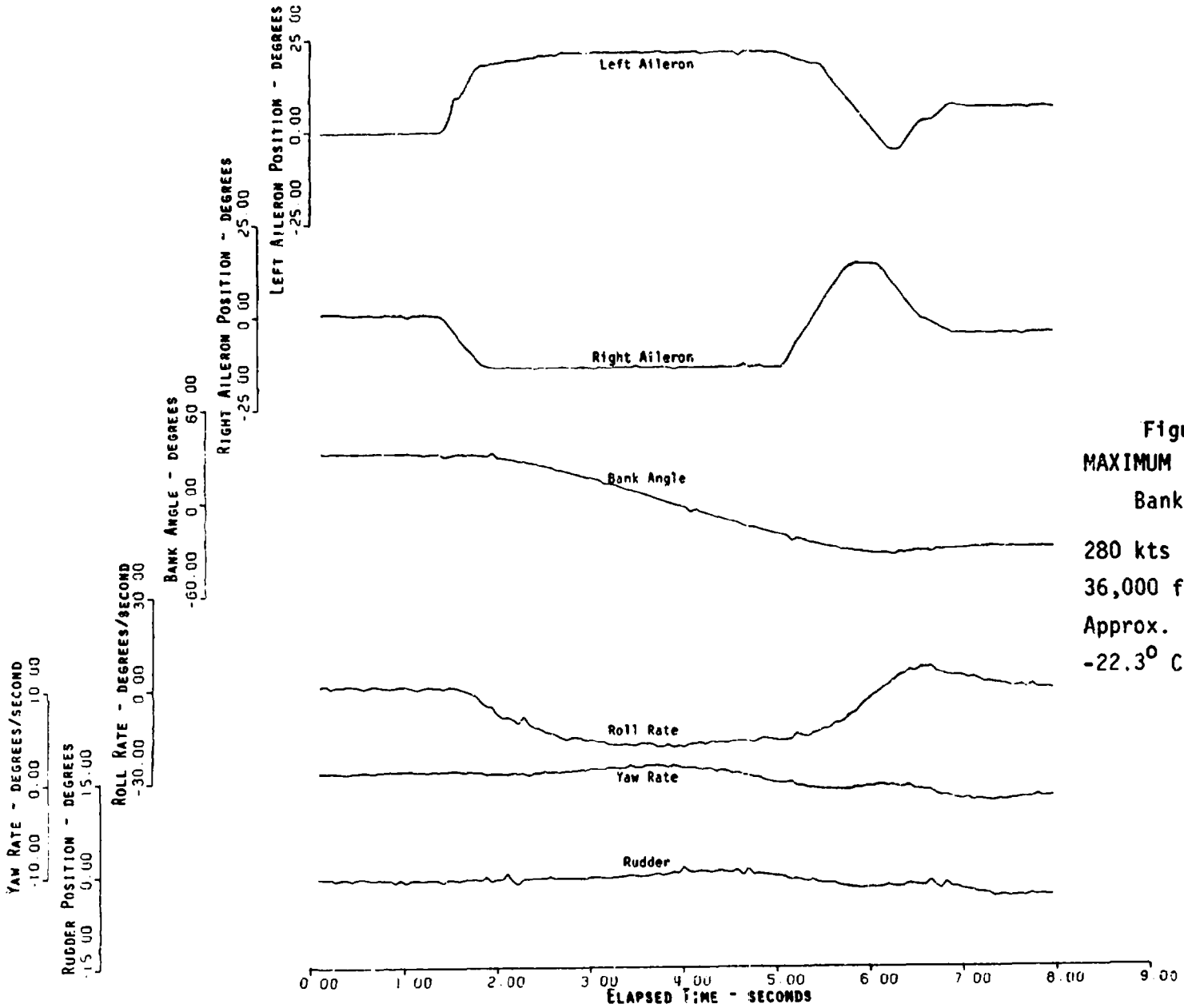


Figure A-2.9a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

280 kts Airspeed
36,000 ft Altitude
Approx. 180,000 lb G.W.
-22.3° C OAT

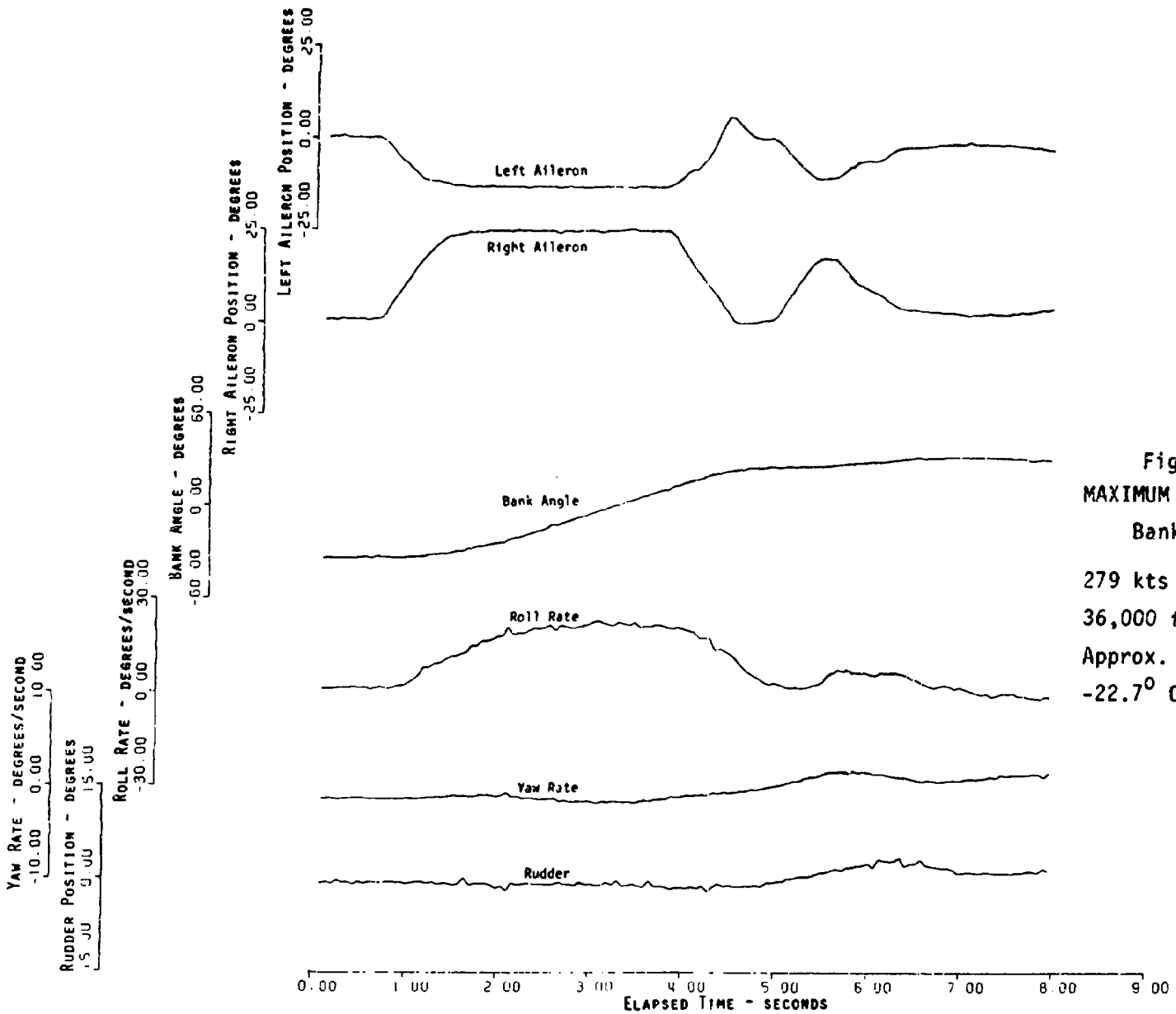


Figure A-2.9b
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

279 kts Airspeed
36,000 ft Altitude
Approx. 180,000 lb G.W.
-22.7° C OAT

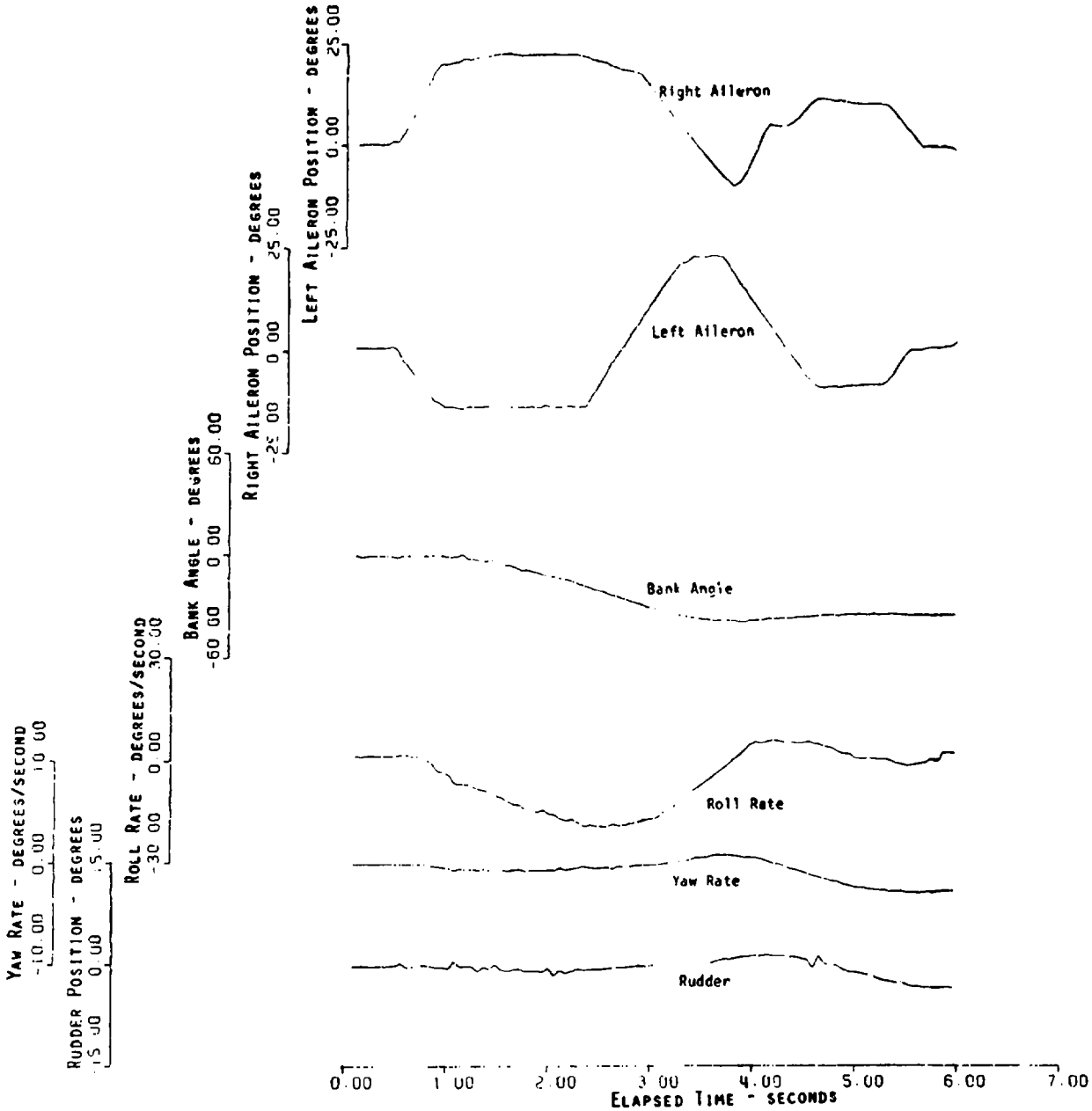


Figure A-2.10a
MAXIMUM DEFLECTION ROLL
From Level Flight

186 kts Airspeed
40,500 ft Altitude
Approx. 180,000 lb G.W.
-30.0° C OAT

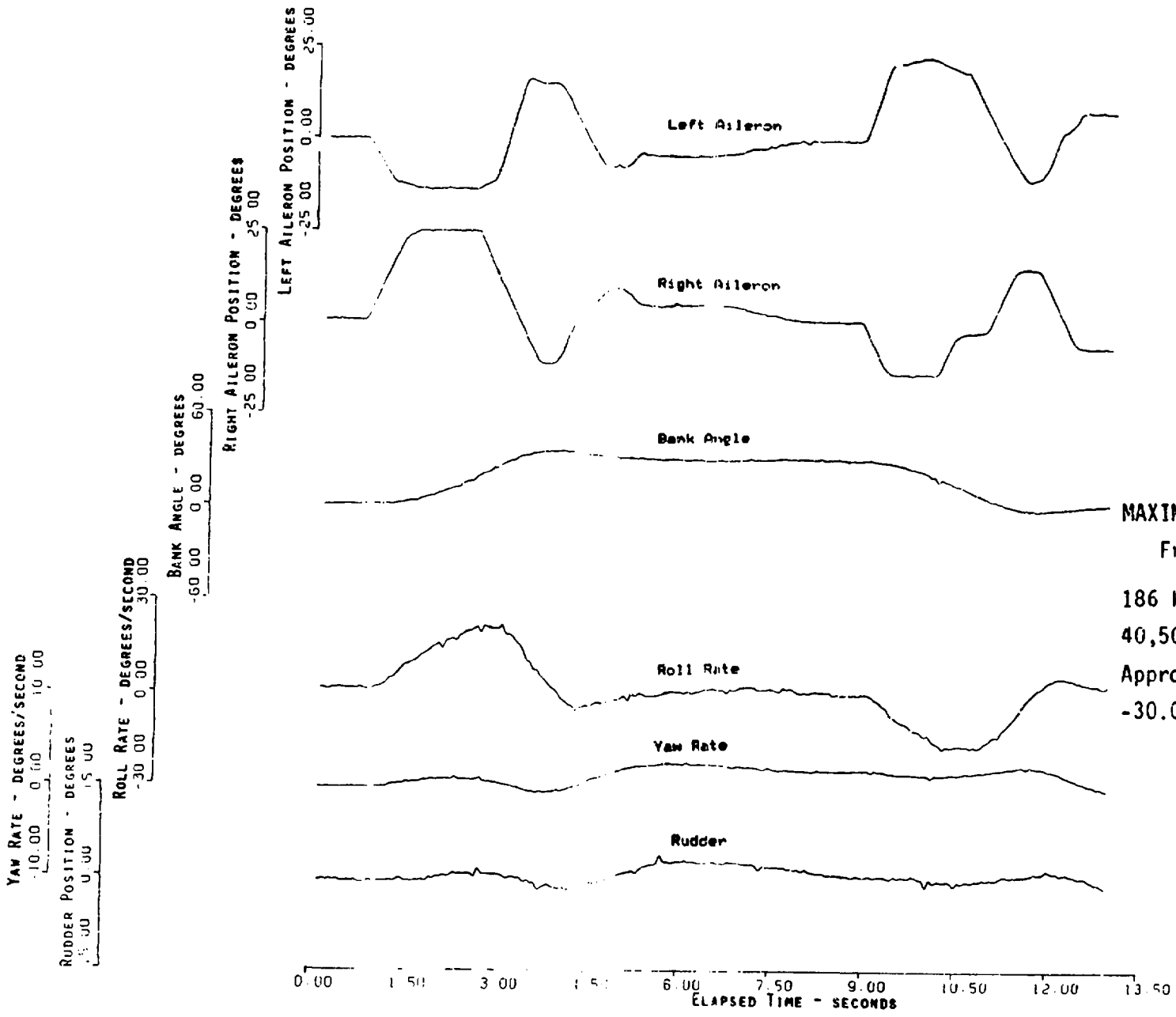


Figure A-2.10b
MAXIMUM DEFLECTION ROLL
From Level Flight

186 kts Airspeed
40,500 ft Altitude
Approx. 180,000 lb G.W.
-30.0° C OAT

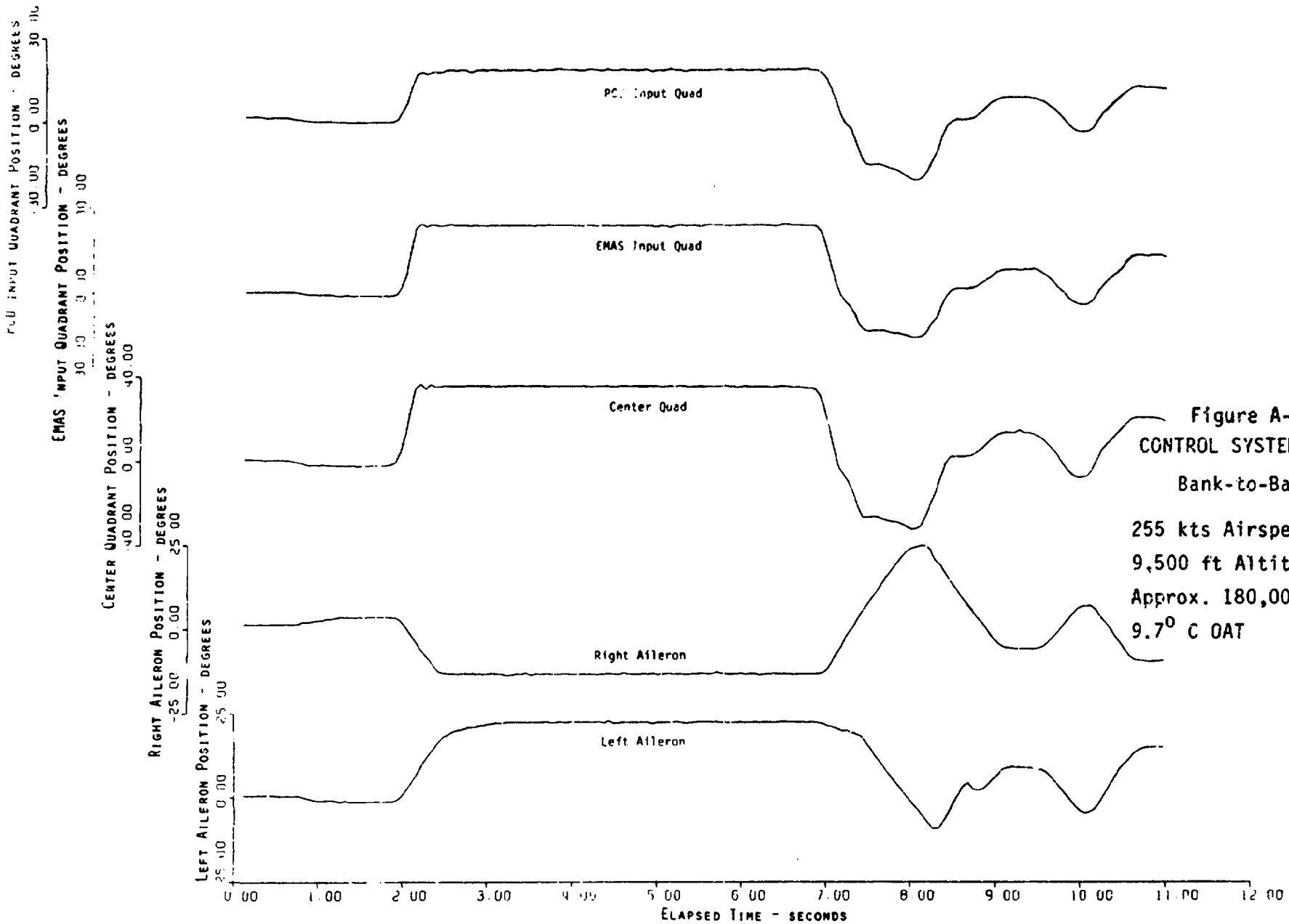


Figure A-3.1a
CONTROL SYSTEM RESPONSE
Bank-to-Bank

255 kts Airspeed
9,500 ft Altitude
Approx. 180,000 lb G.W.
9.7° C OAT

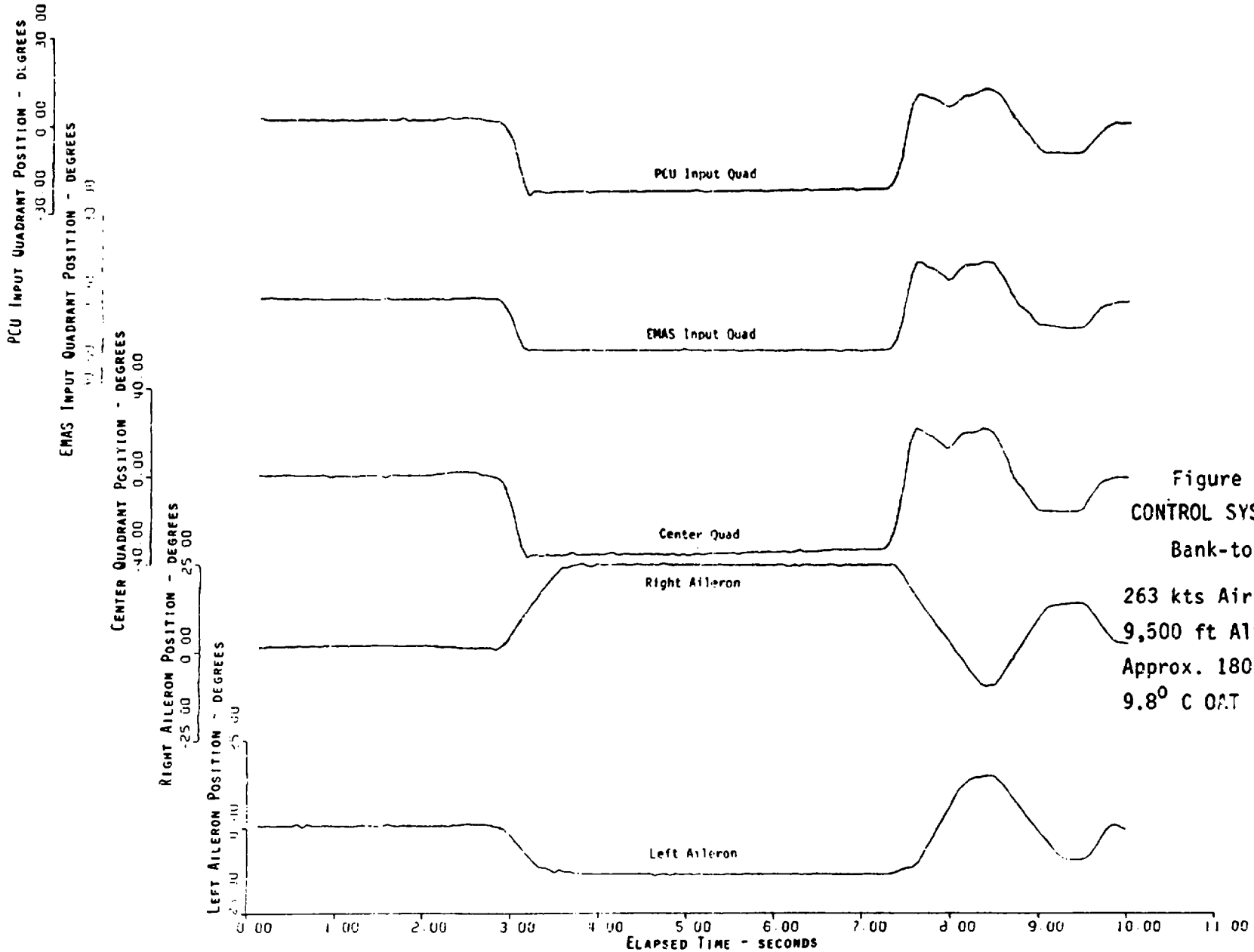


Figure A-3.1b
CONTROL SYSTEM RESPONSE
Bank-to-Bank

263 kts Airspeed
9,500 ft Altitude
Approx. 180,000 lb G.W.
9.8° C OAT

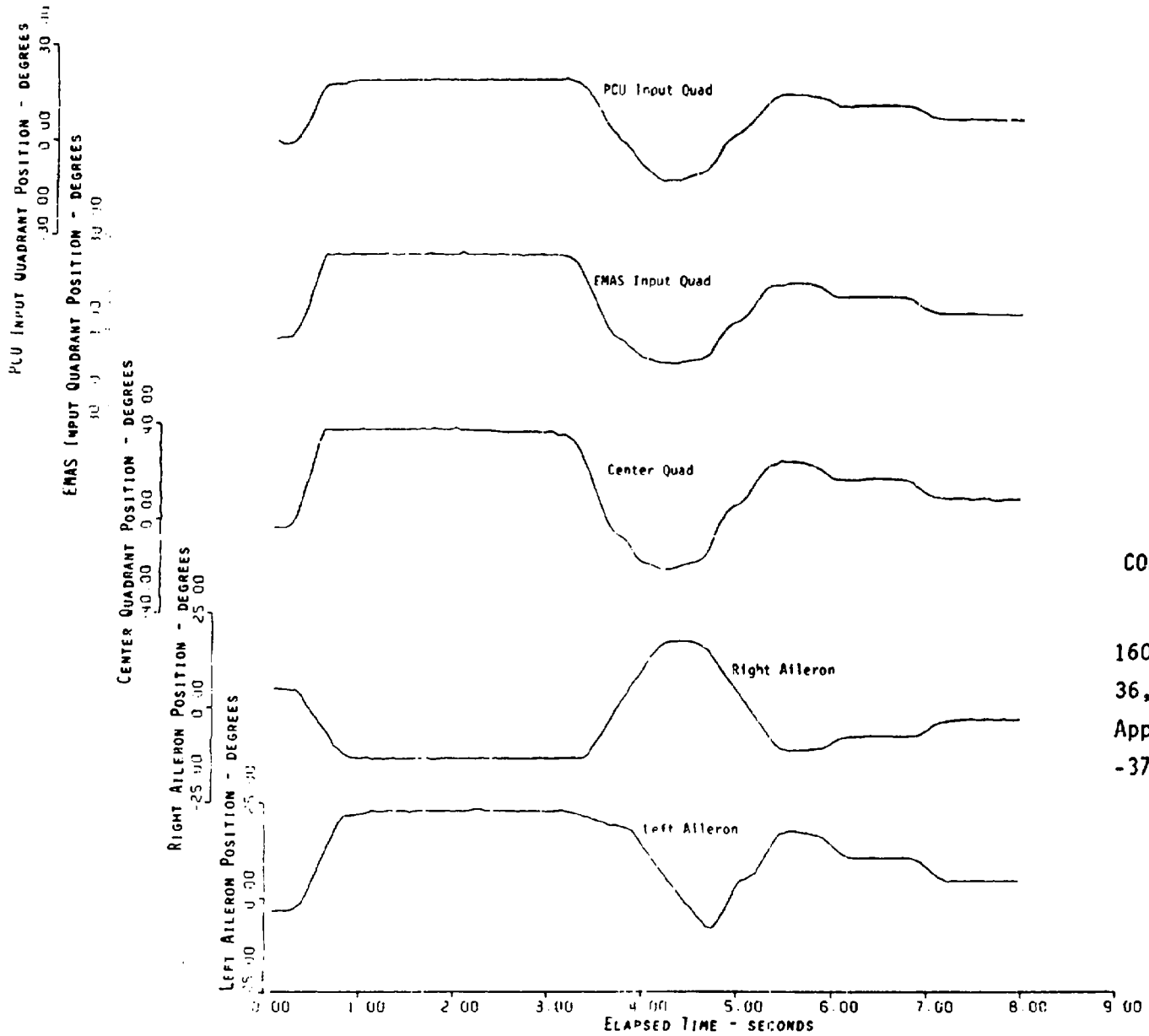


Figure A-3.2a
CONTROL SYSTEM RESPONSE
Bank-to-Bank

160 kts Airspeed
36,500 ft Altitude
Approx. 180,000 lb G.W.
-37.1° C OAT

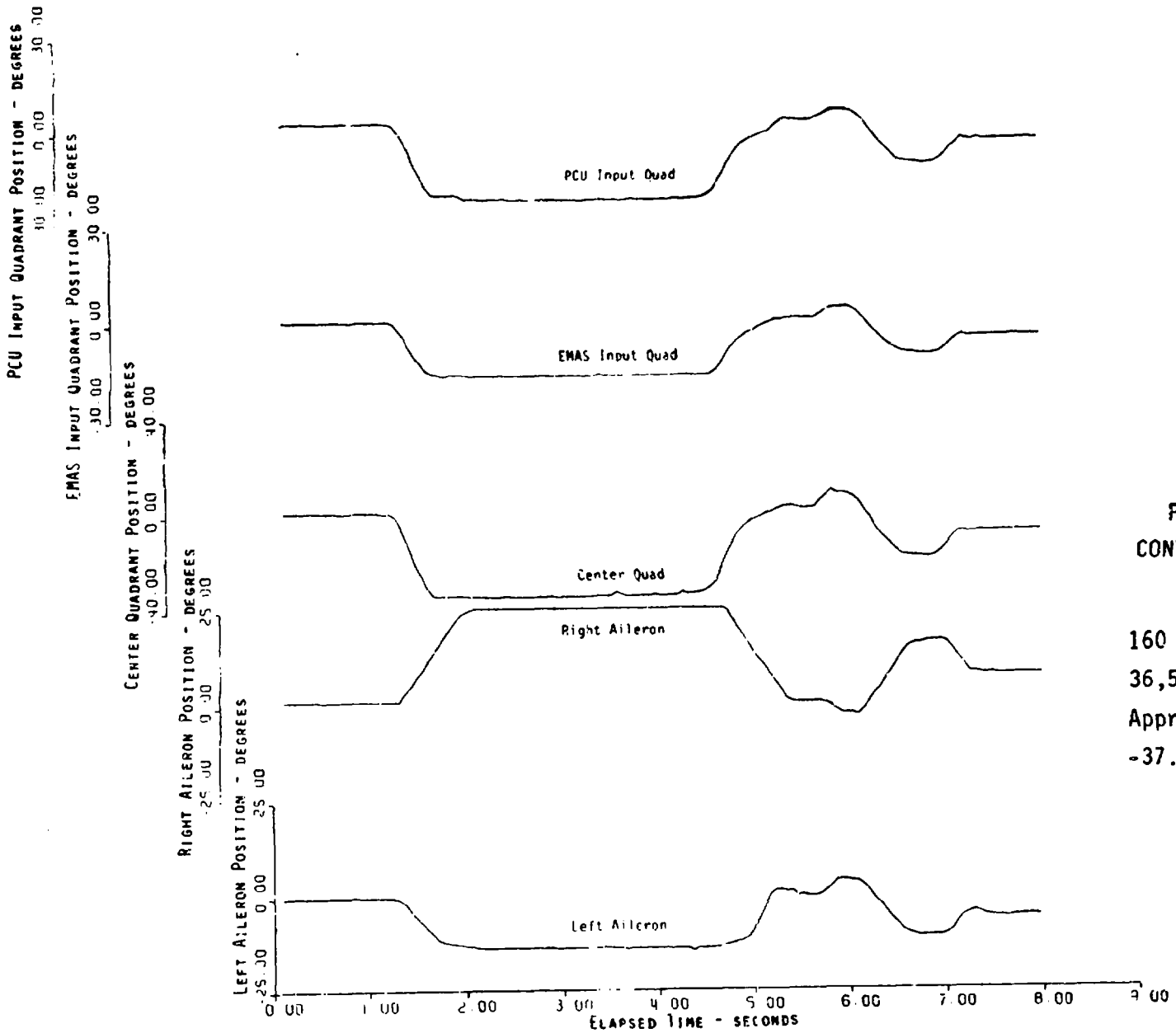


Figure A-3.2b
CONTROL SYSTEM RESPONSE
Bank-to-Bank

160 kts Airspeed
36,500 ft Altitude
Approx. 180,000 lb G.W.
-37.7° C OAT

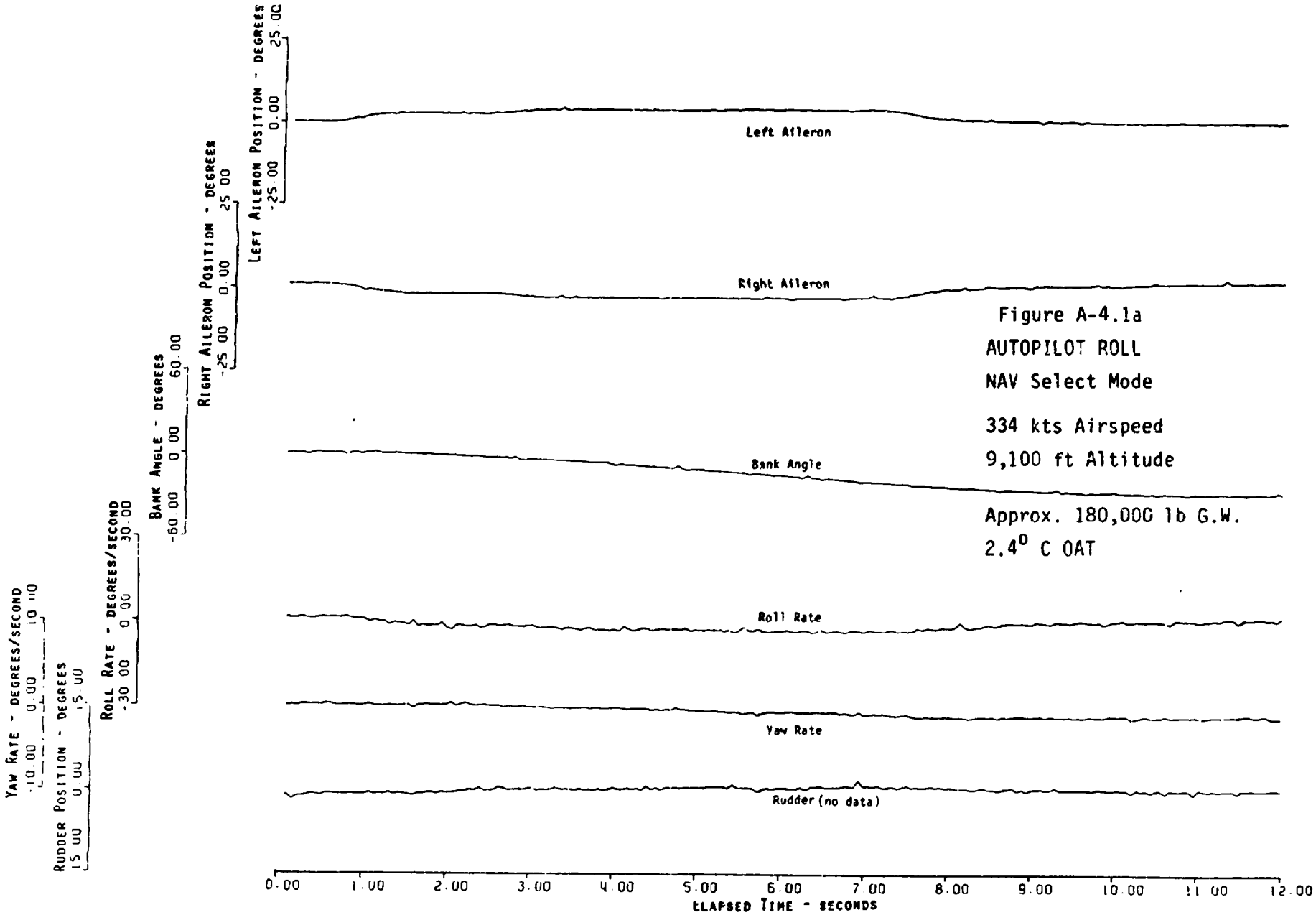


Figure A-4.1a
AUTOPILOT ROLL
NAV Select Mode

334 kts Airspeed
9,100 ft Altitude

Approx. 180,000 lb G.W.
2.4° C OAT

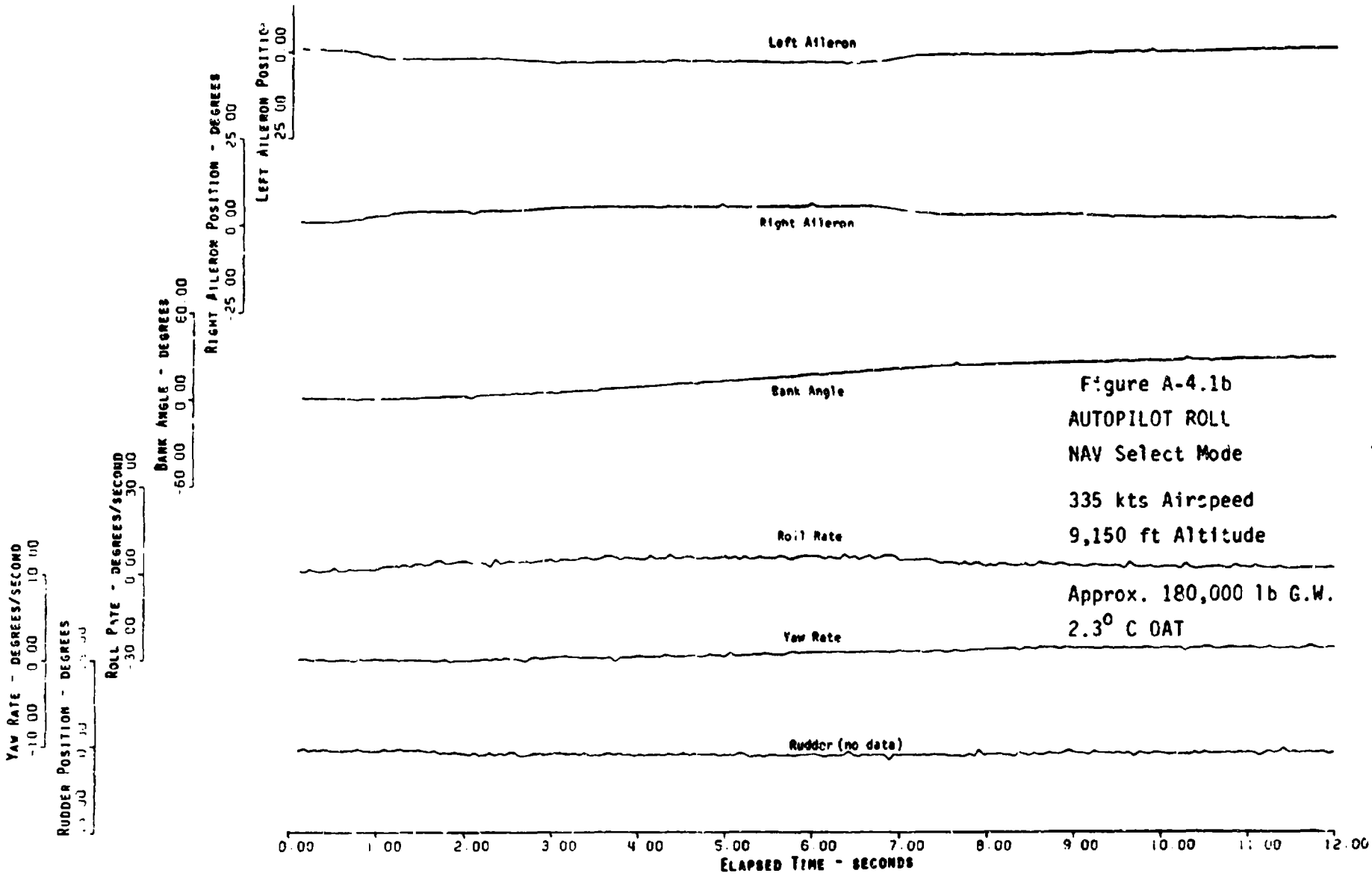


Figure A-4.1b
AUTOPILOT ROLL
NAV Select Mode

335 kts Airspeed
9,150 ft Altitude

Approx. 180,000 lb G.W.
2.3° C OAT

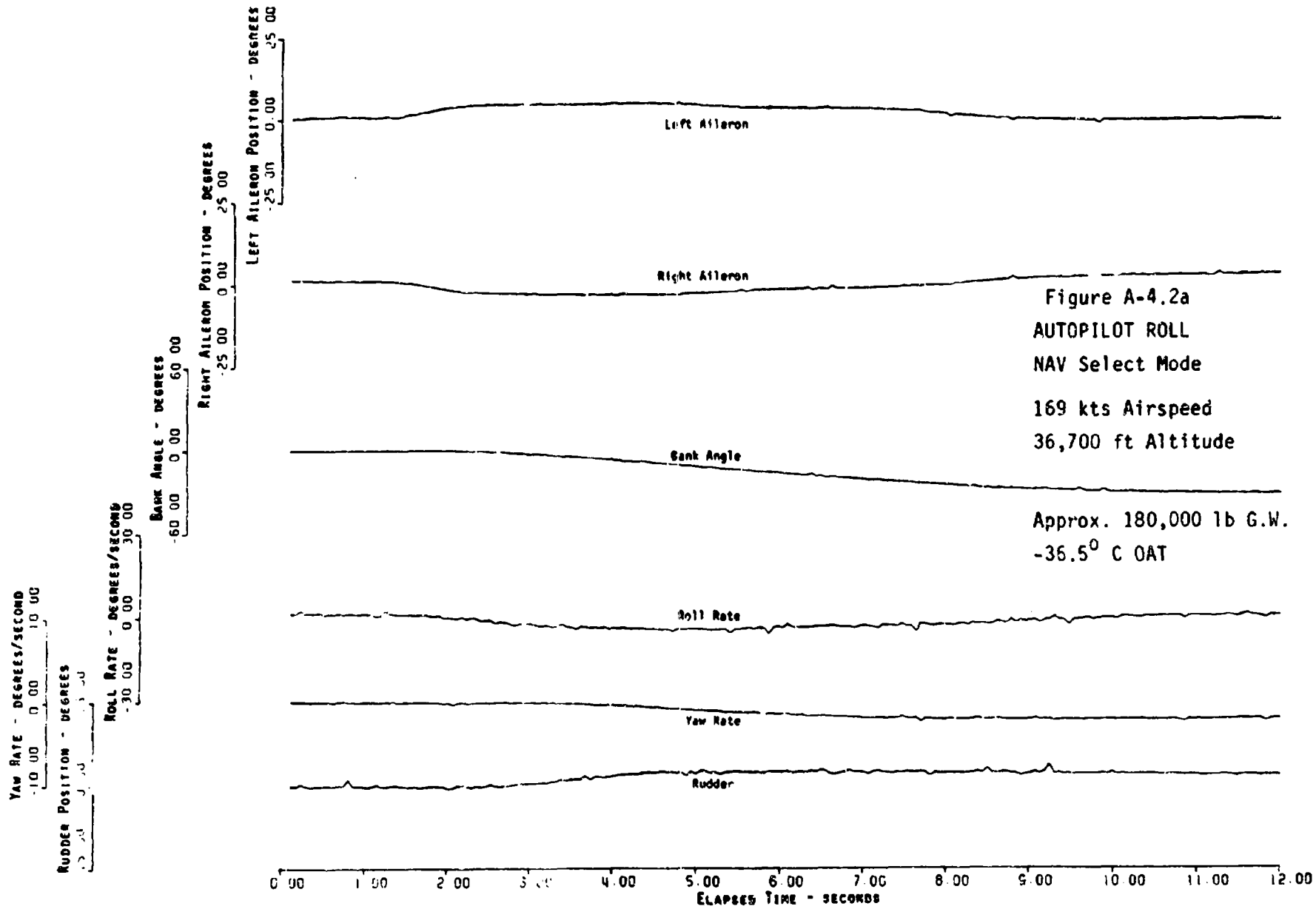


Figure A-4.2a
AUTOPILOT ROLL
NAV Select Mode

169 kts Airspeed
36,700 ft Altitude

Approx. 180,000 lb G.W.
-36.5° C OAT

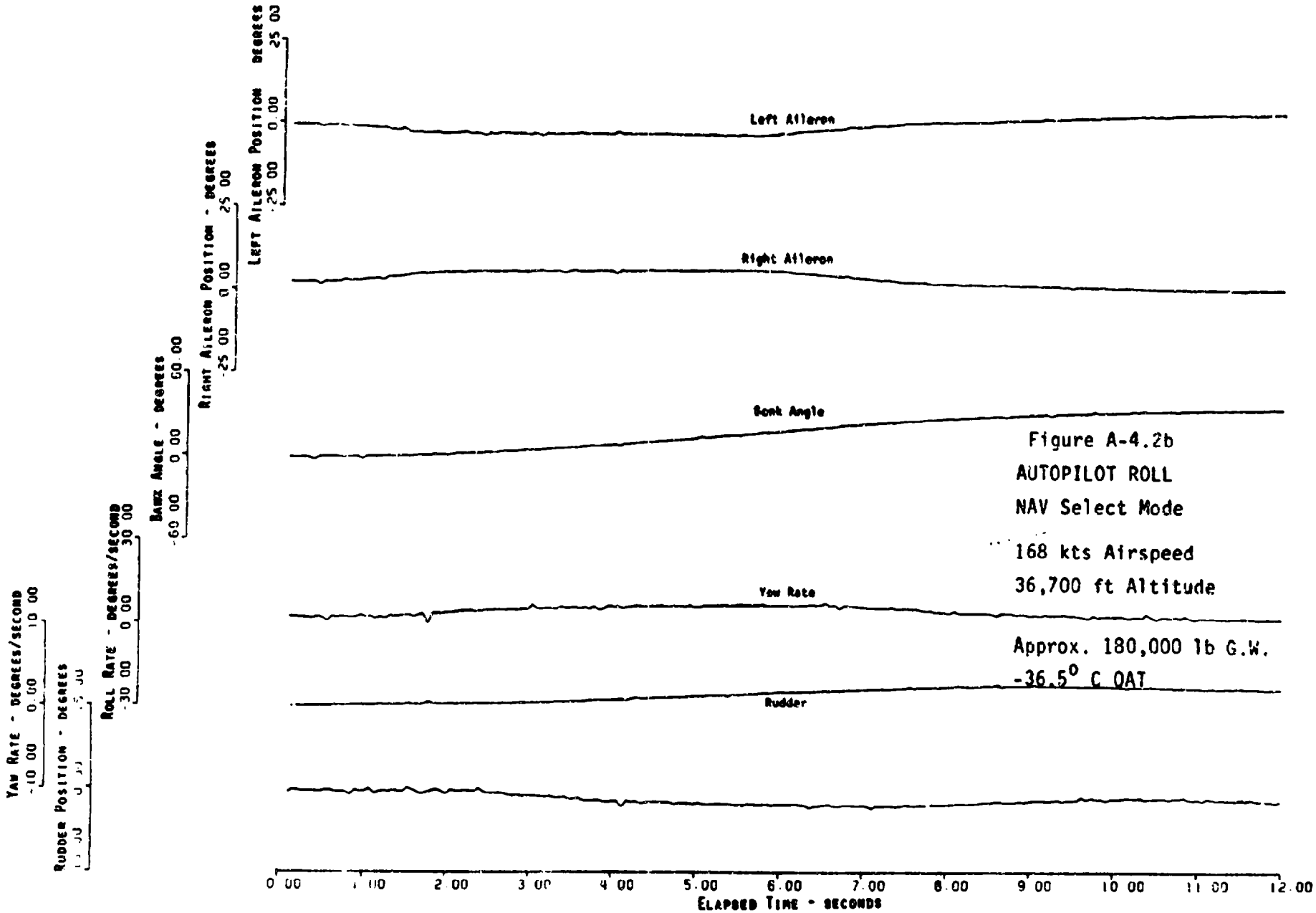


Figure A-4.2b
AUTOPILOT ROLL
NAV Select Mode

168 kts Airspeed
36,700 ft Altitude

Approx. 180,000 lb G.W.
-36.5° C OAT

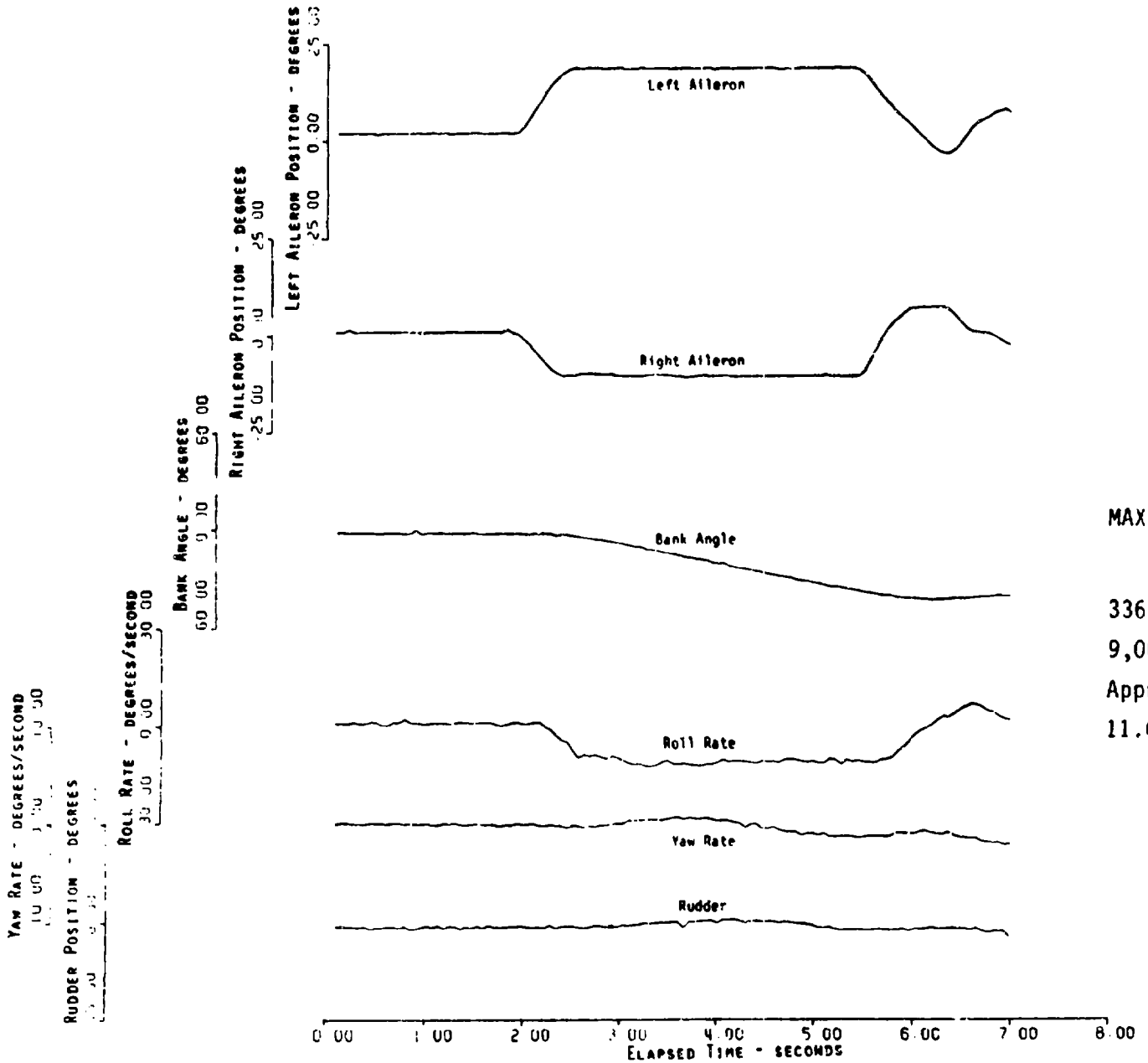


Figure A-5.1a
MAXIMUM DEFLECTION DEGRADED ROLL
Chan A & Sys 1 Off

336 kts Airspeed
9,000 ft Altitude
Approx. 180,000 lb G.W.
11.0° CAT

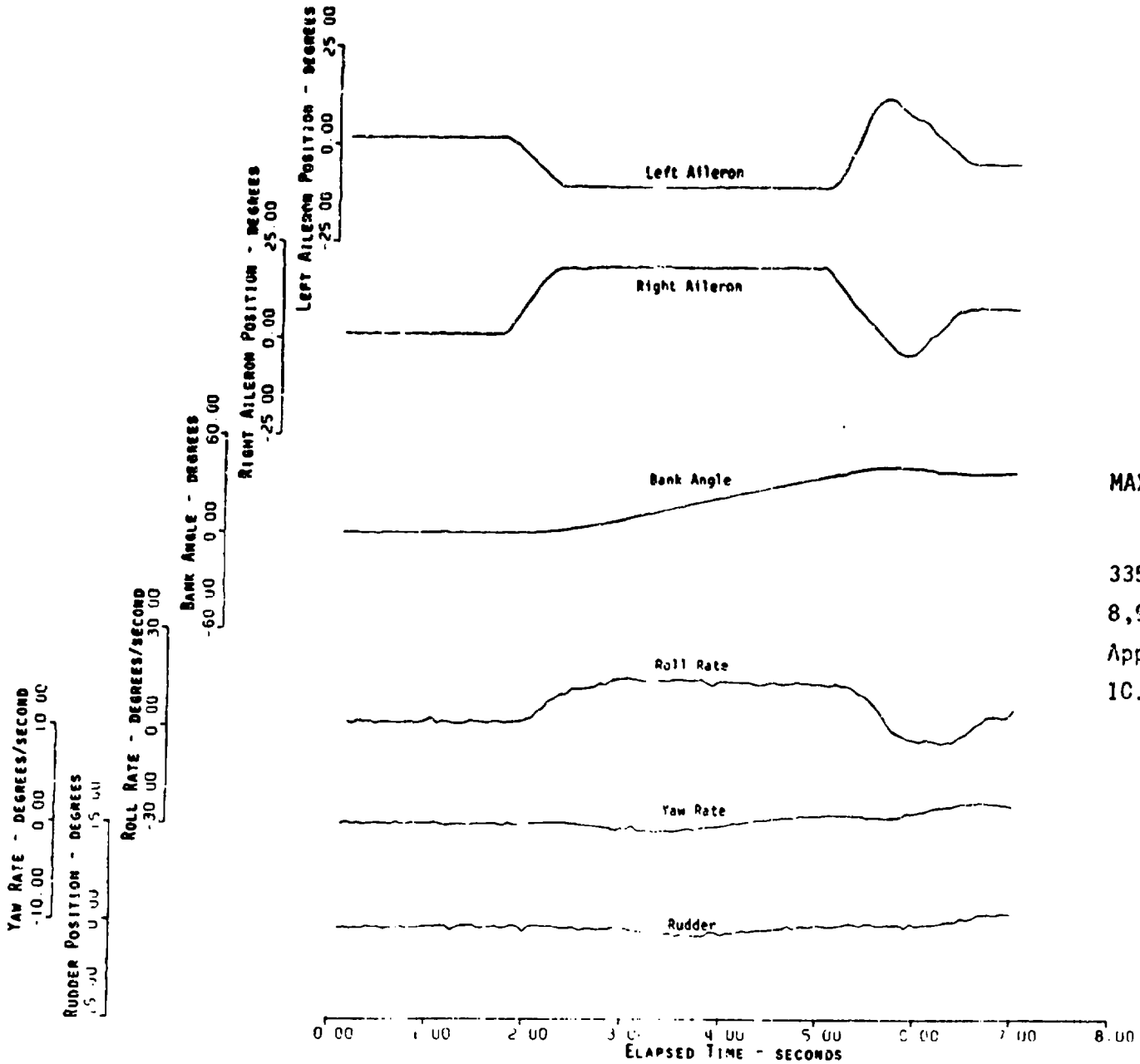


Figure A-5.1b

MAXIMUM DEFLECTION DEGRADED ROLL

Chan A & Sys 1 Off

335 kts Airpseed

8,900 ft Altitude

Approx. 180,000 lb G.W.

10.7° C OAT

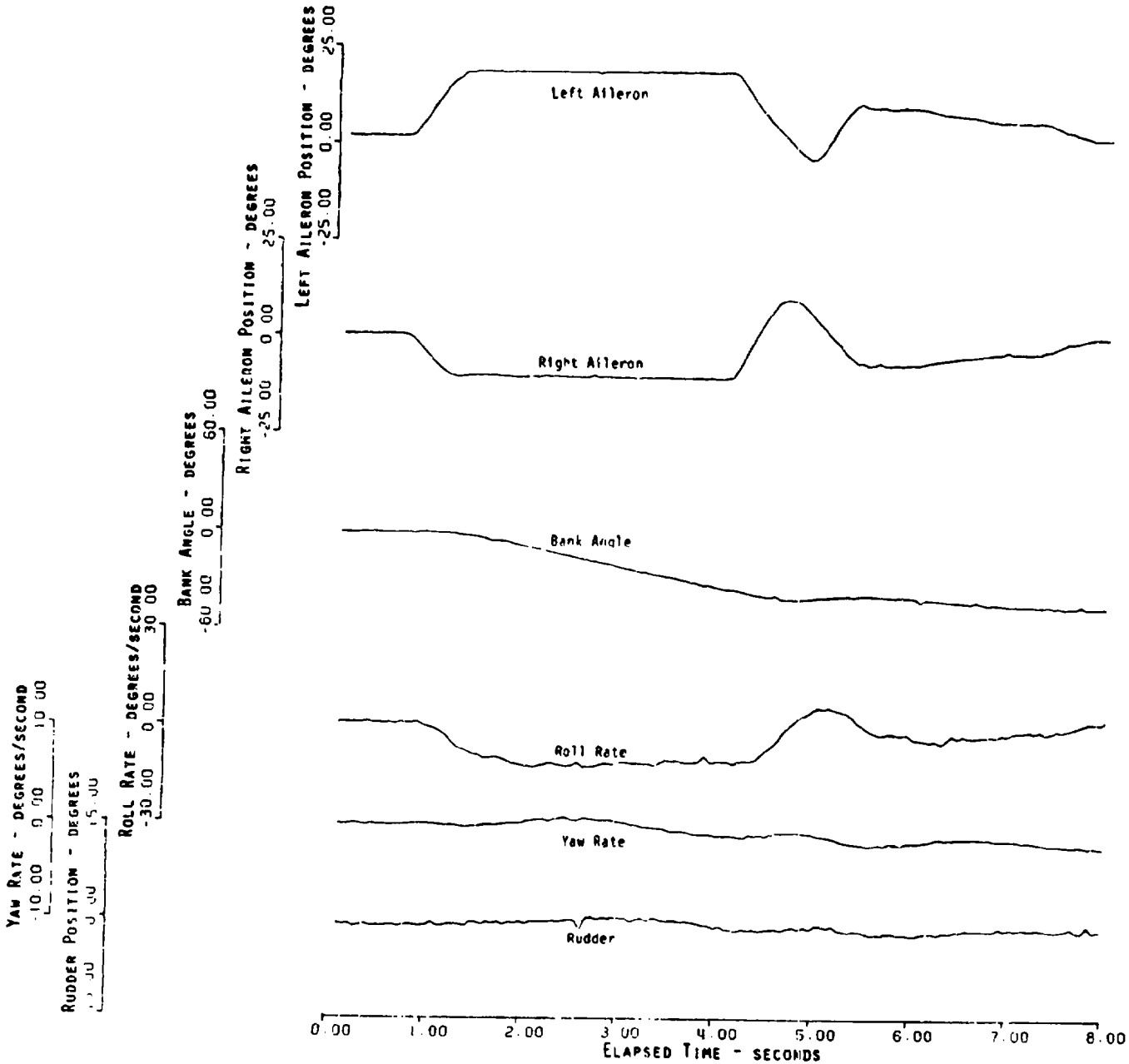


Figure A-5.2a

MAXIMUM DEFLECTION DEGRADED ROLL

Chan B & Sys 2 Off

335 kts Airpseed

9,000 ft Altitude

Approx. 180,000 lb G.W.

10.7^{0} C OAT

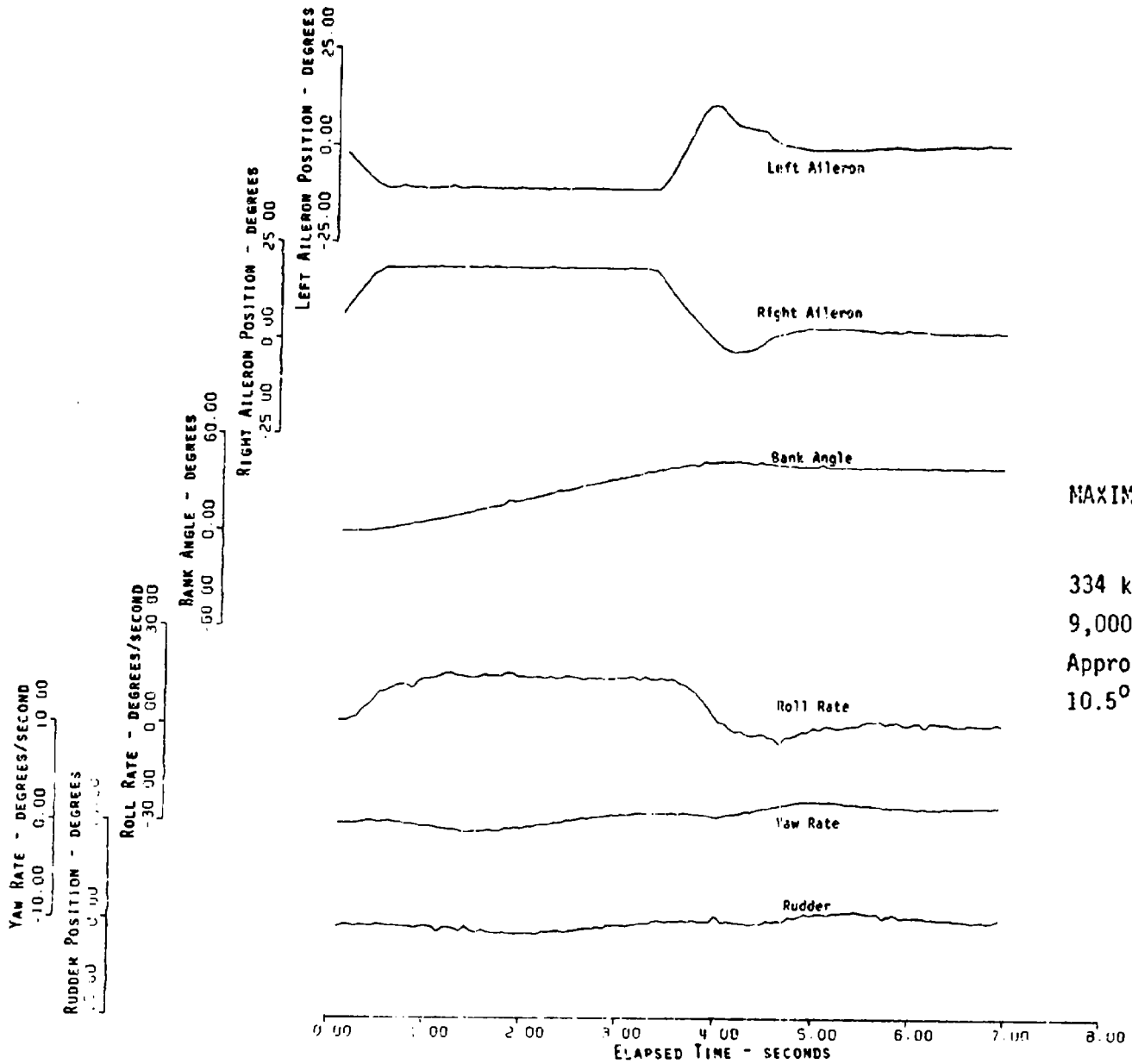


Figure A-5.2b
MAXIMUM DEFLECTION DEGRADED ROLL
Chan A & Sys 2 Off

334 kts Airspeed
9,000 ft Altitude
Approx. 180,000 lb G.W.
10.5° C OAT

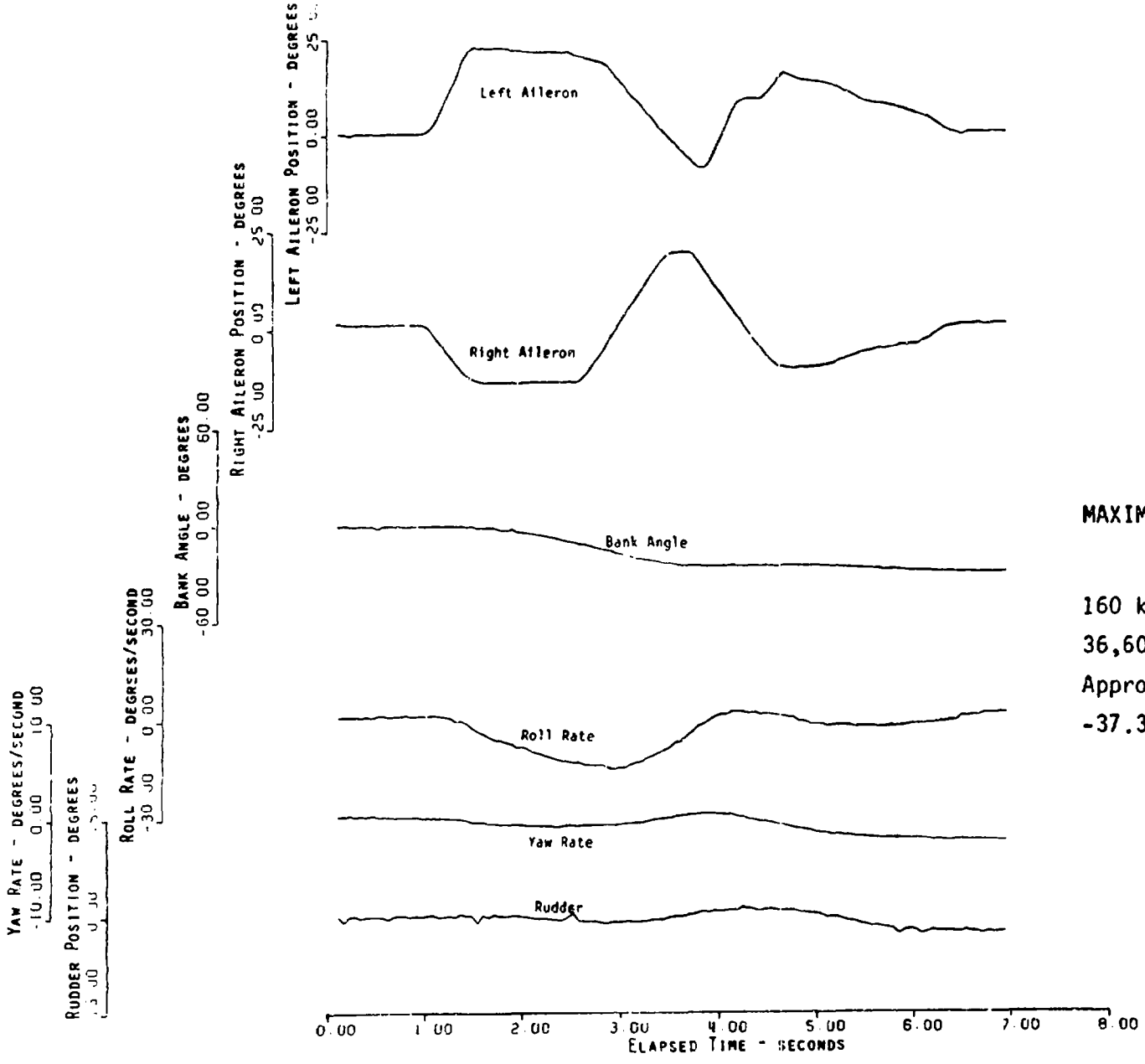


Figure A-5.3a

MAXIMUM DEFLECTION DEGRADED ROLL

Chan A & Sys 1 Off

160 kts Airspeed

36,600 ft Altitude

Approx. 180,000 lb G.W.

-37.3^{o} C OAT

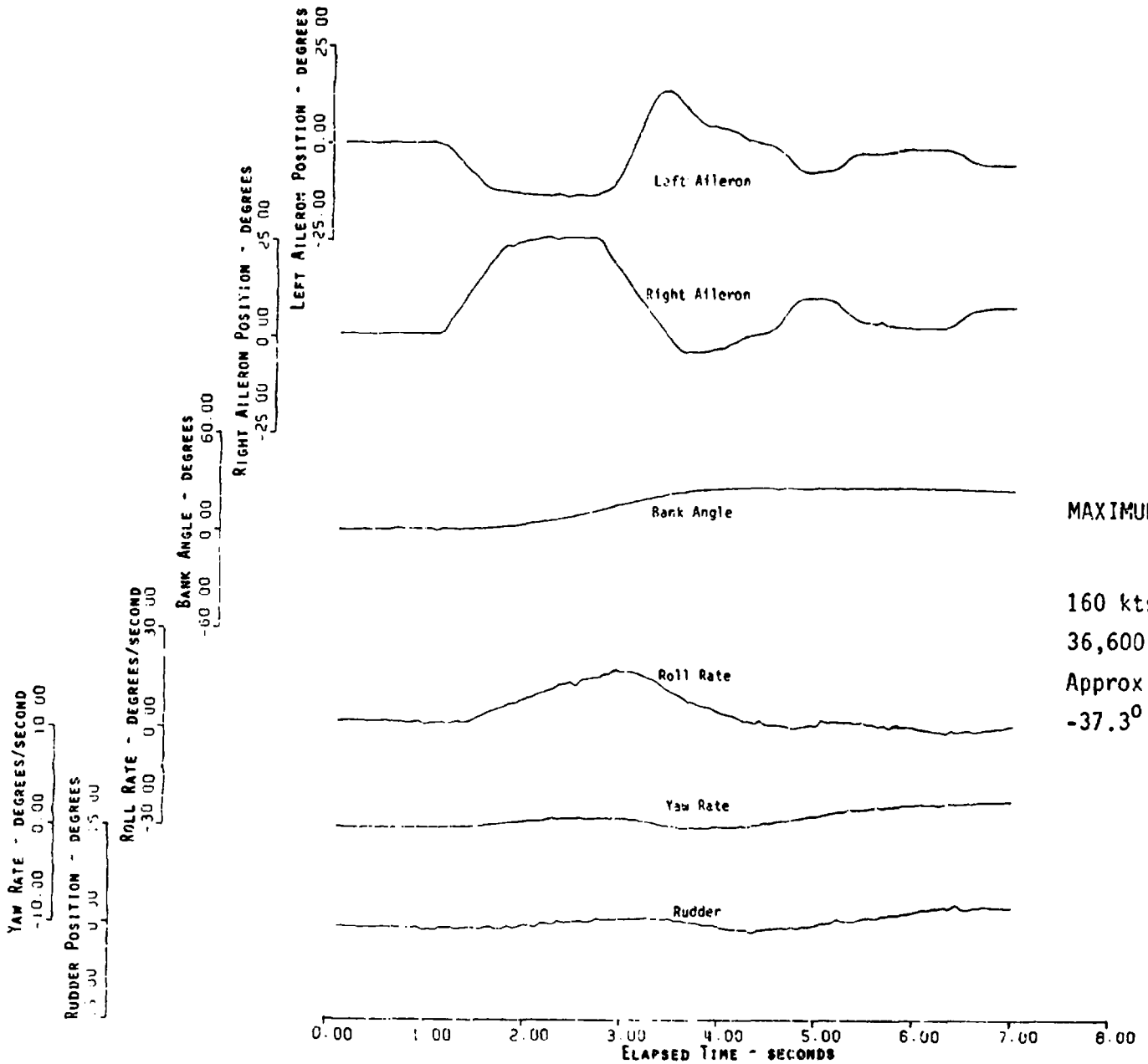


Figure A-5.3b

MAXIMUM DEFLECTION DEGRADED ROLL

Chan A & Sys 1 Off

160 kts Airspeed

36,600 ft Altitude

Approx. 180,000 lb G.W.

-37.3° C OAT

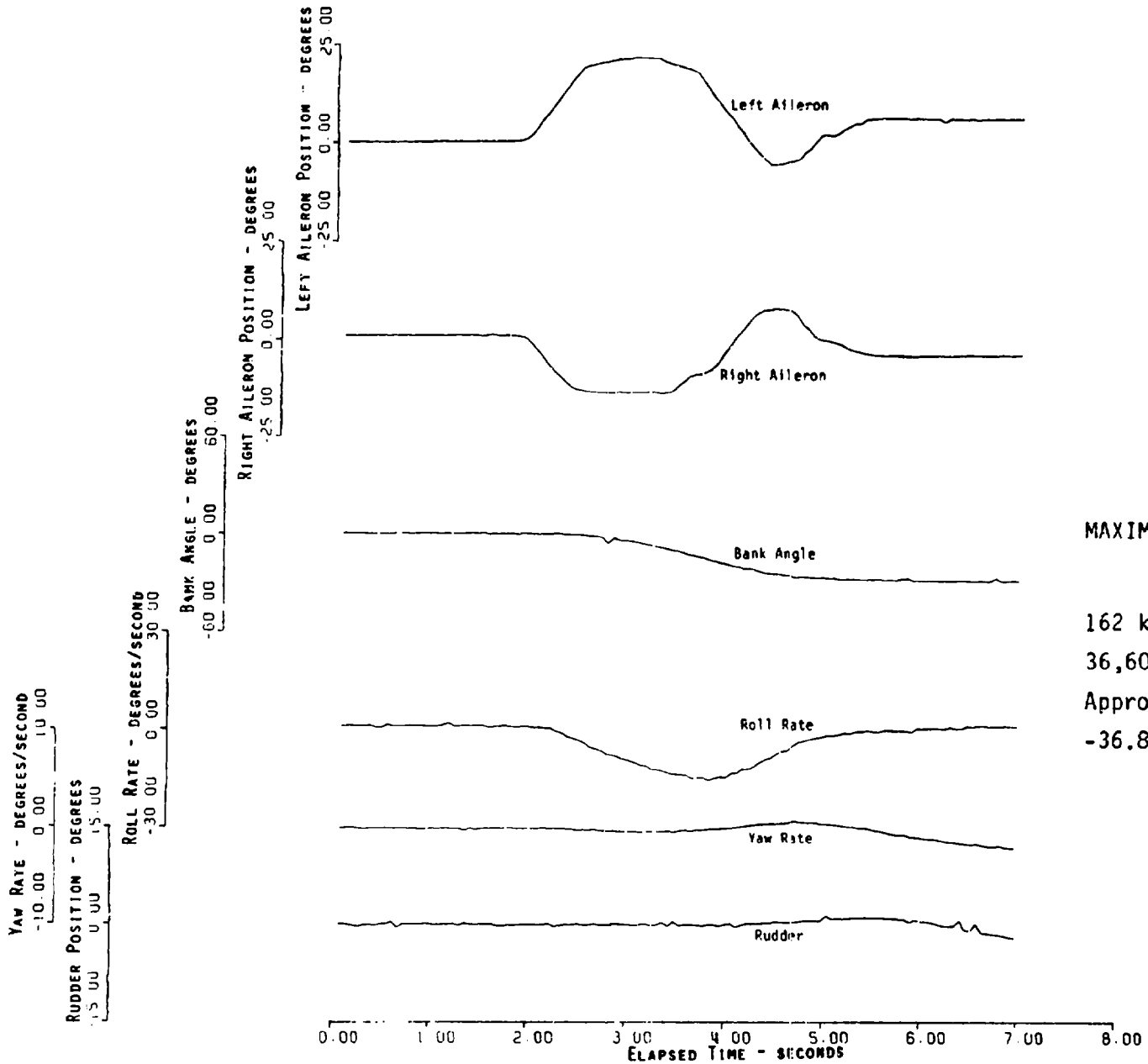


Figure A-5.4a

MAXIMUM DEFLECTION DEGRADED ROLL

Chan B & Sys 2 Off

162 kts Airspeed

36,600 ft Altitude

Approx. 180,000 lb G.W.

-36.8° C OAT

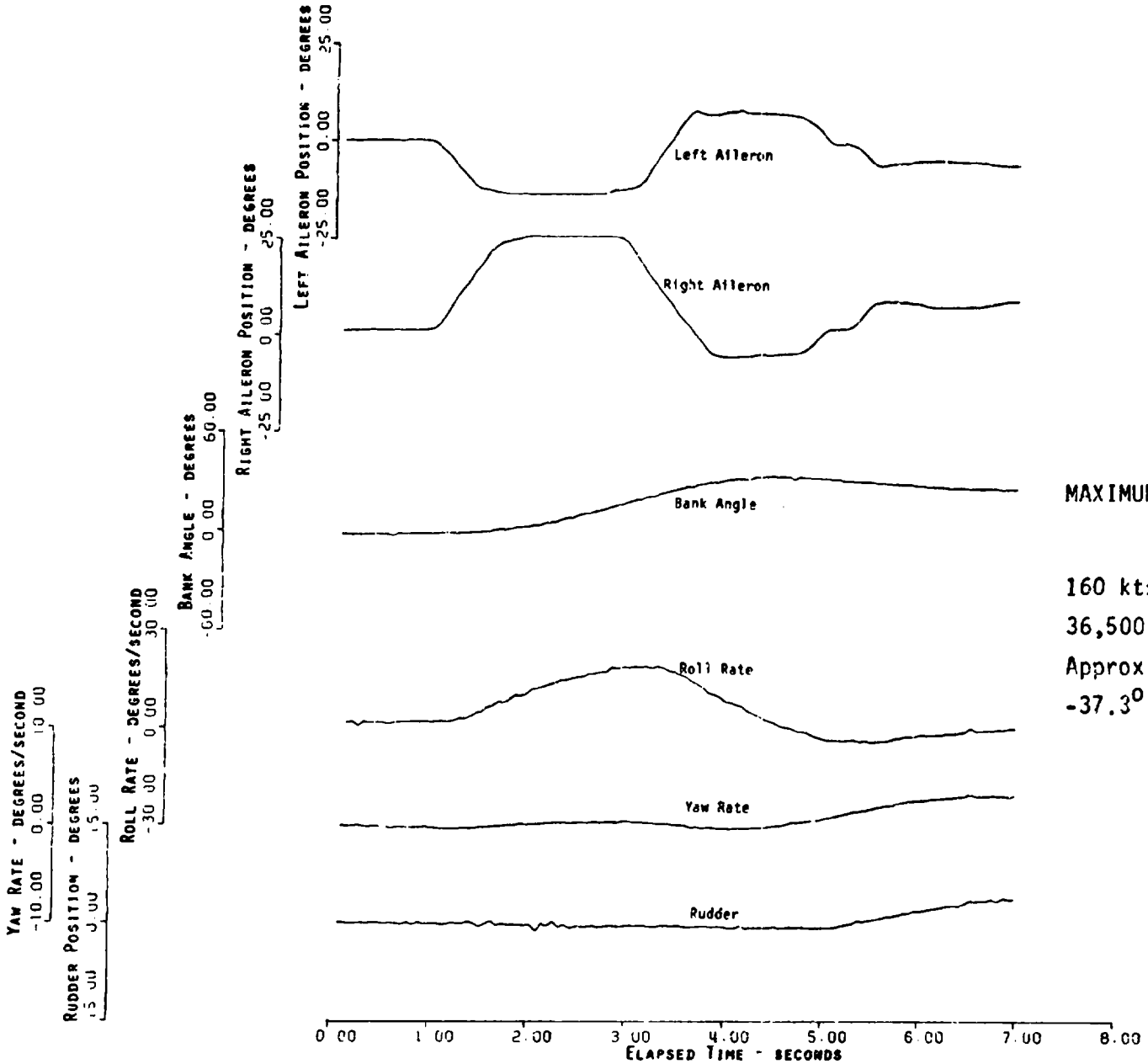


Figure A-5.4b

MAXIMUM DEFLECTION DEGRADED ROLL

Chan B & Sys 2 Off

160 kts Airspeed

36,500 ft Altitude

Approx. 180,000 lb G.W.

-37.3° C OAT

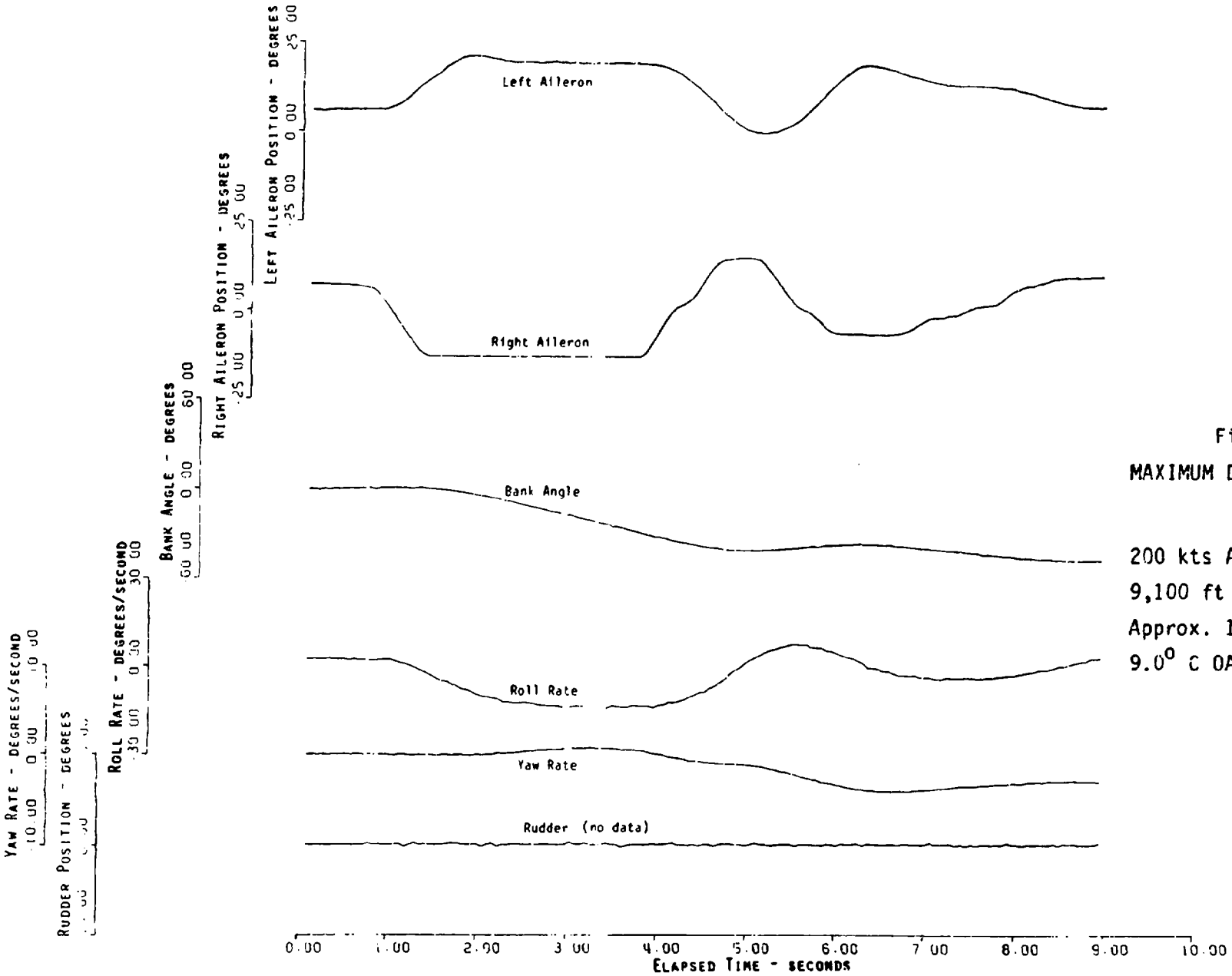


Figure A-6.1a

MAXIMUM DEFLECTION TAB ROLL

Left Tab

200 kts Airspeed

9,100 ft Altitude

Approx. 180,000 lb G.W.

9.0° C OAT

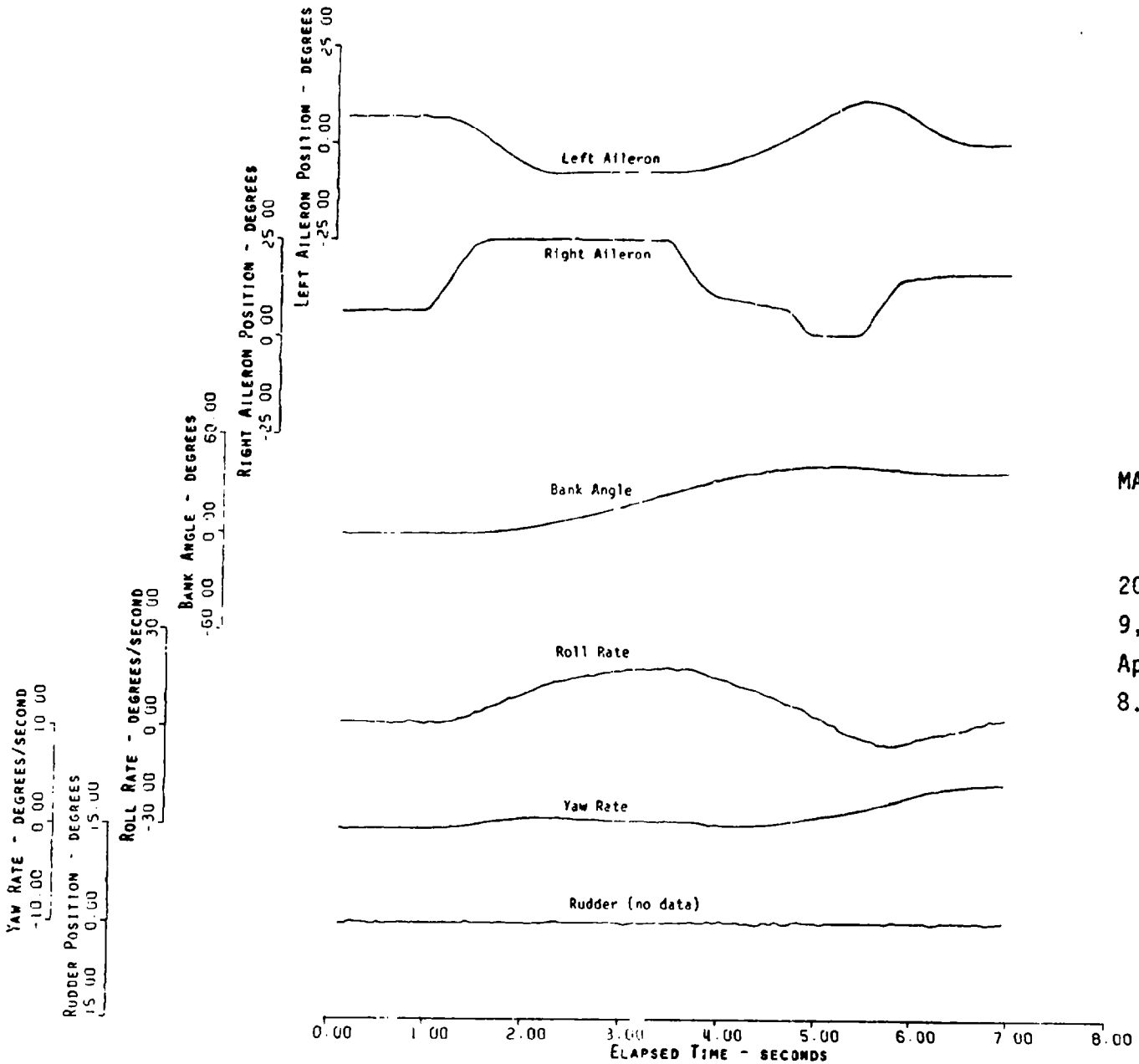


Figure A-6.1b
MAXIMUM DEFLECTION TAB ROLL
Left Tab

201 kts Airspeed
9,400 ft Altittude
Approx. 180,000 lb G.W.
8.5° C OAT

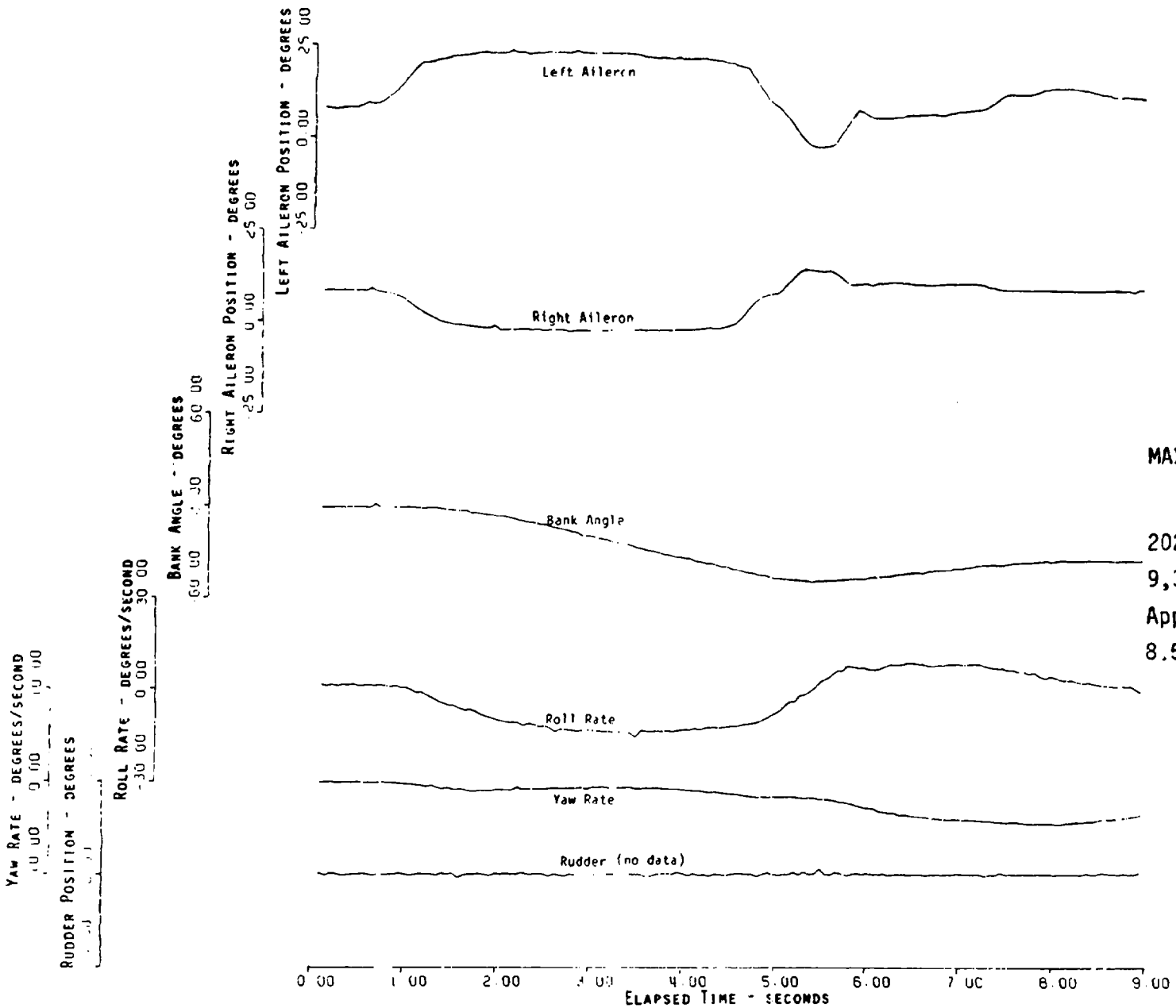


Figure A-6.2a
MAXIMUM DEFLECTION TAB ROLL
Right Tab

202 kts Airspeed
9,300 ft Altitude
Approx. 180,000 lb G.W.
8.5° C OAT

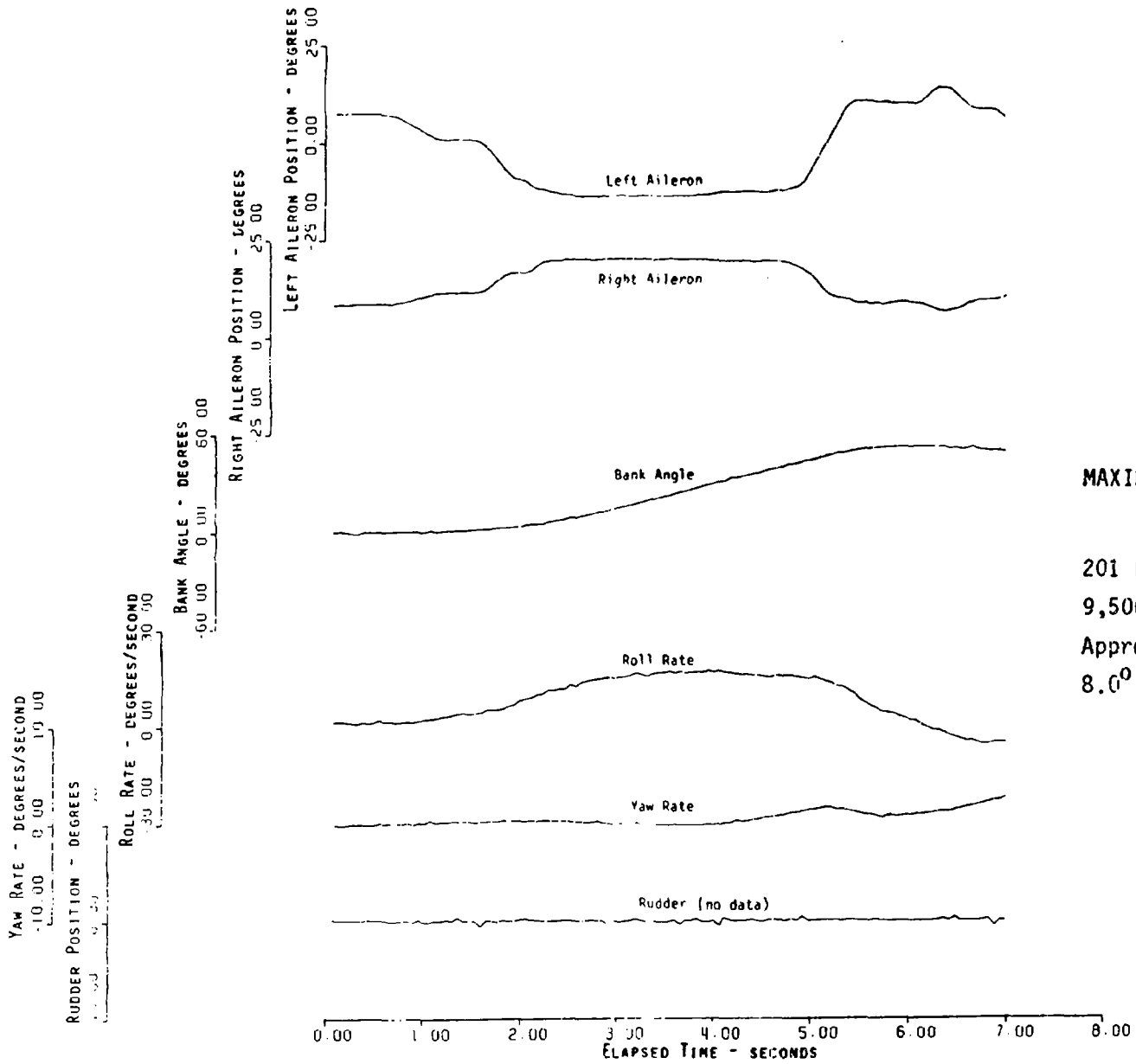


Figure A-6.2b
MAXIMUM DEFLECTION TAB ROLL
Right Tab

201 kts Airspeed
9,500 ft Altitude
Approx. 180,000 lb G.W.
8.0° C OAT

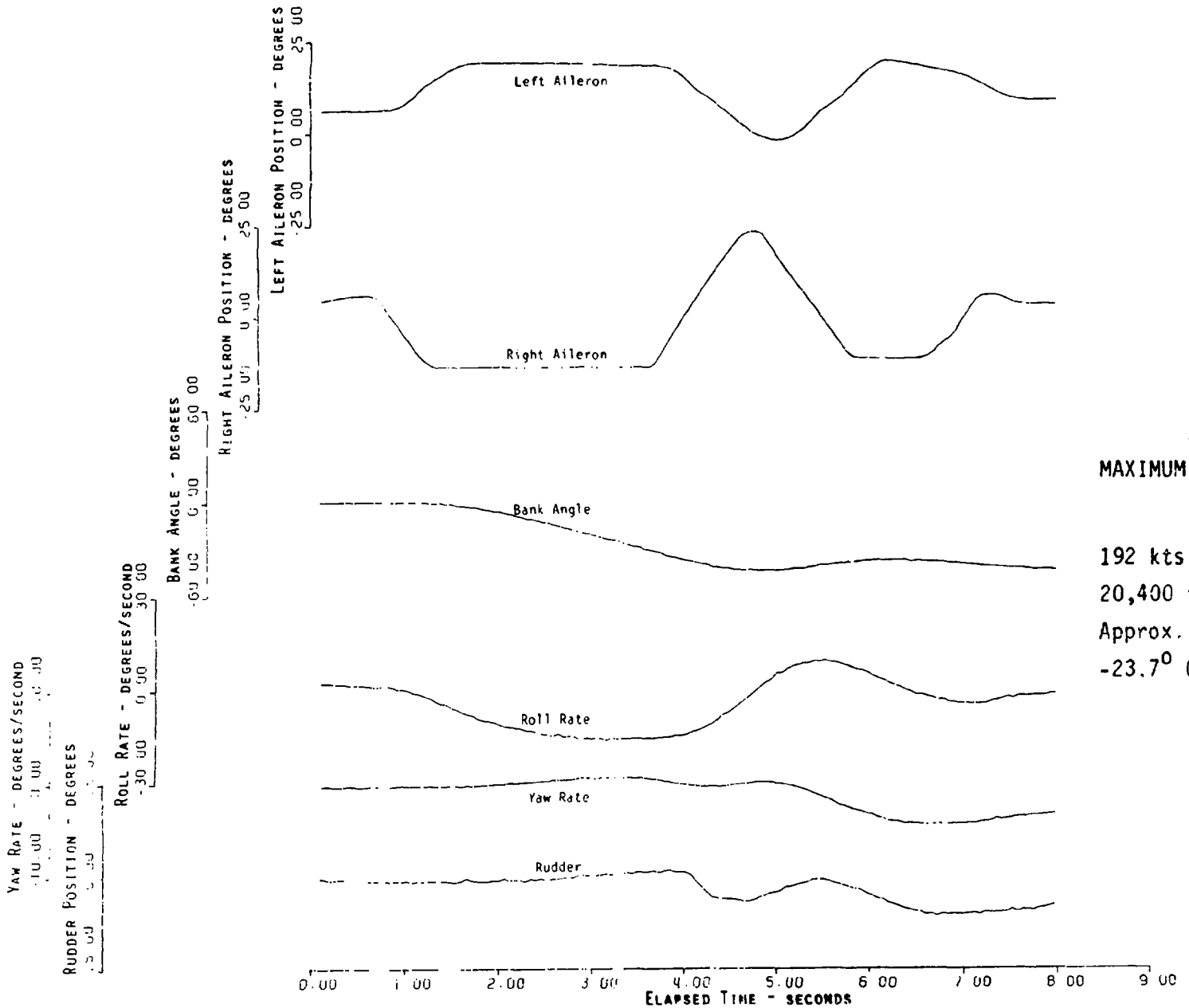


Figure A-6.3a
MAXIMUM DEFLECTION TAB ROLL
Left Tab

192 kts Airspeed
20,400 ft Altitude
Approx. 180,000 lb G.W.
-23.7° C OAT

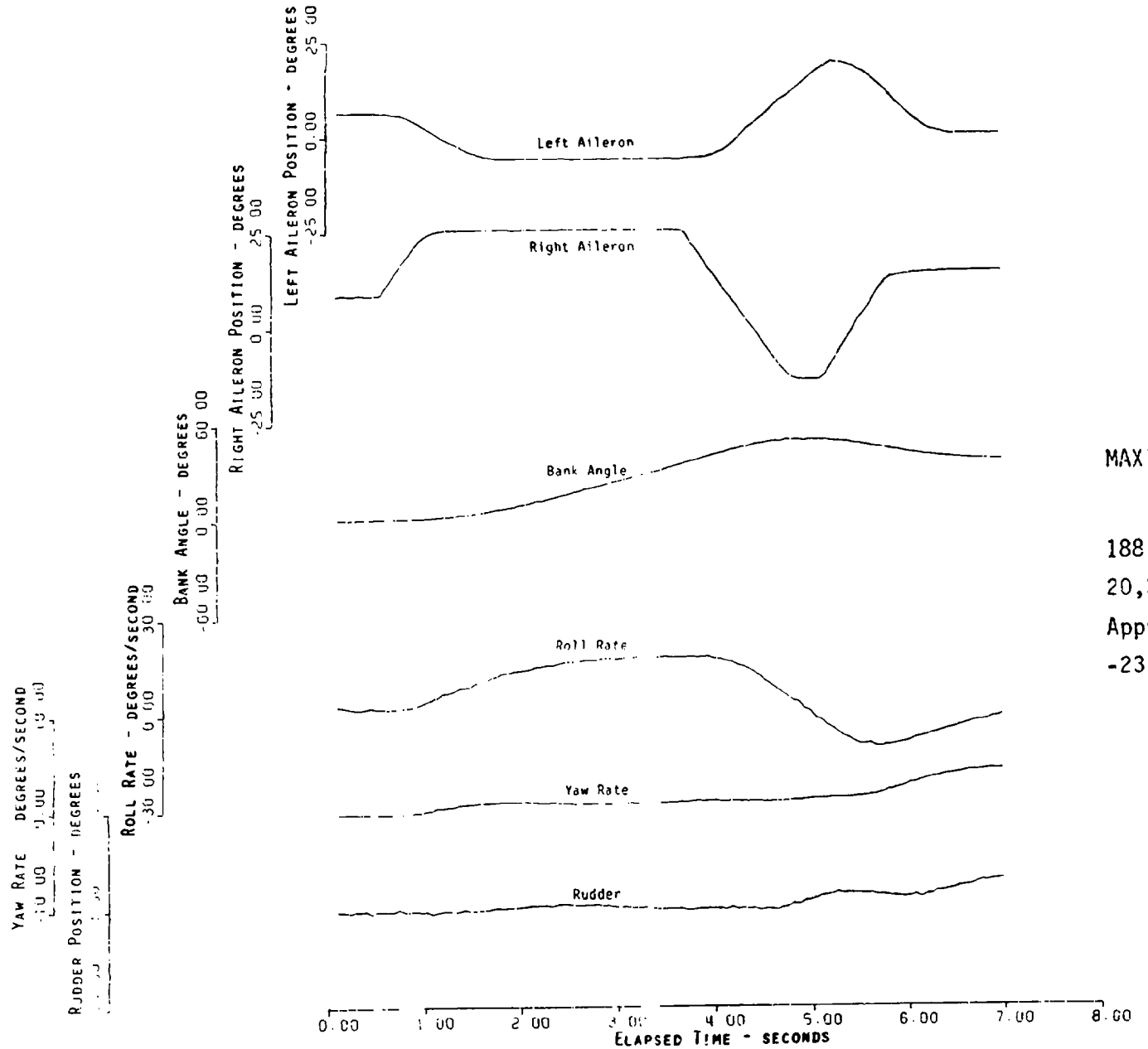


Figure A-6.3b
MAXIMUM DEFLECTION TAB ROLL
Left Tab

188 kts Airspeed
20,300 ft Altitude
Approx. 180,000 lb G.W.
-23.8° C OAT

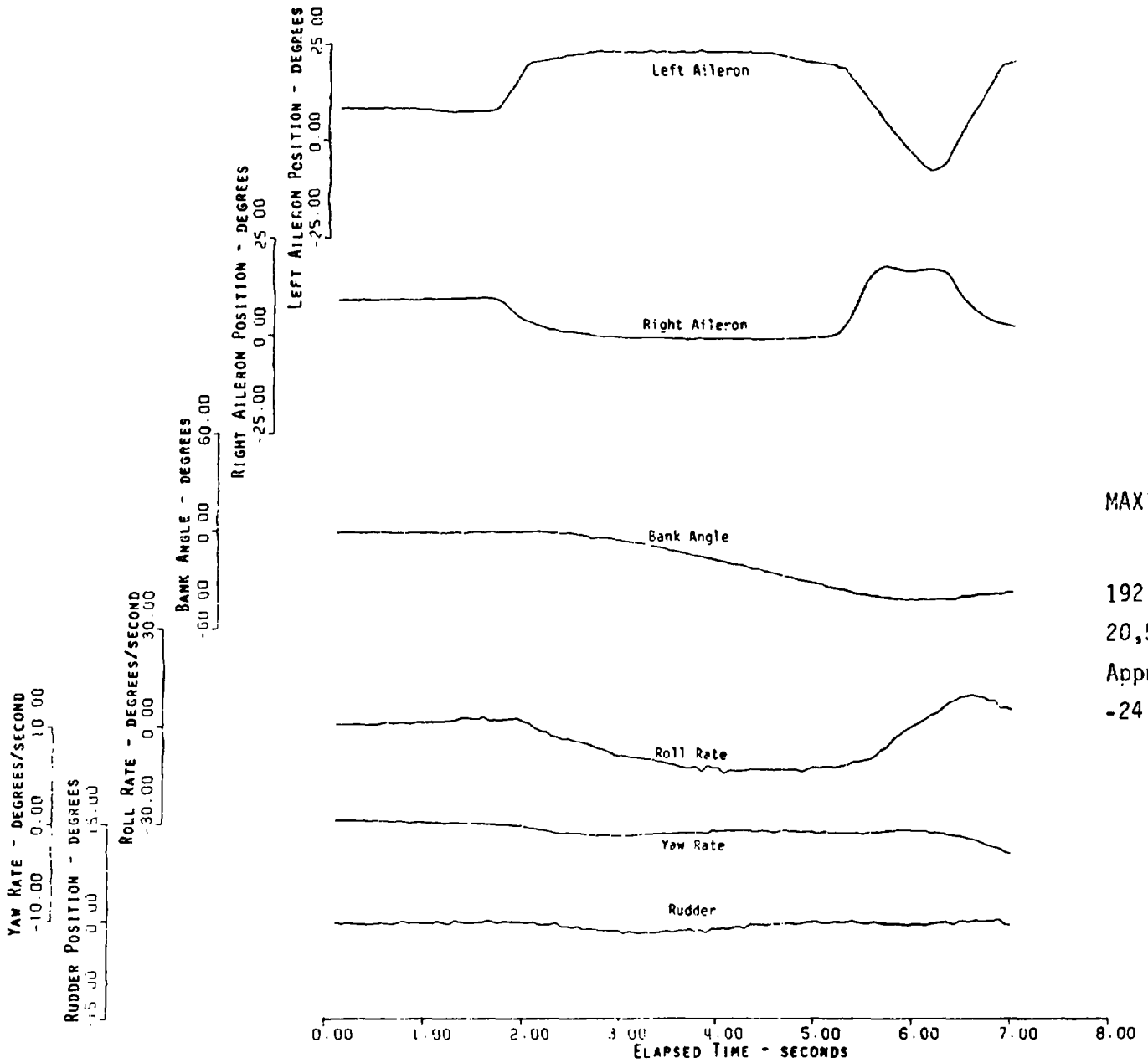


Figure A-6.4a

MAXIMUM DEFLECTION TAB ROLL

Right Tab

192 kts Airspeed

20,500 ft Altitude

Approx. 180,000 lb G.W.

-24.0° C OAT

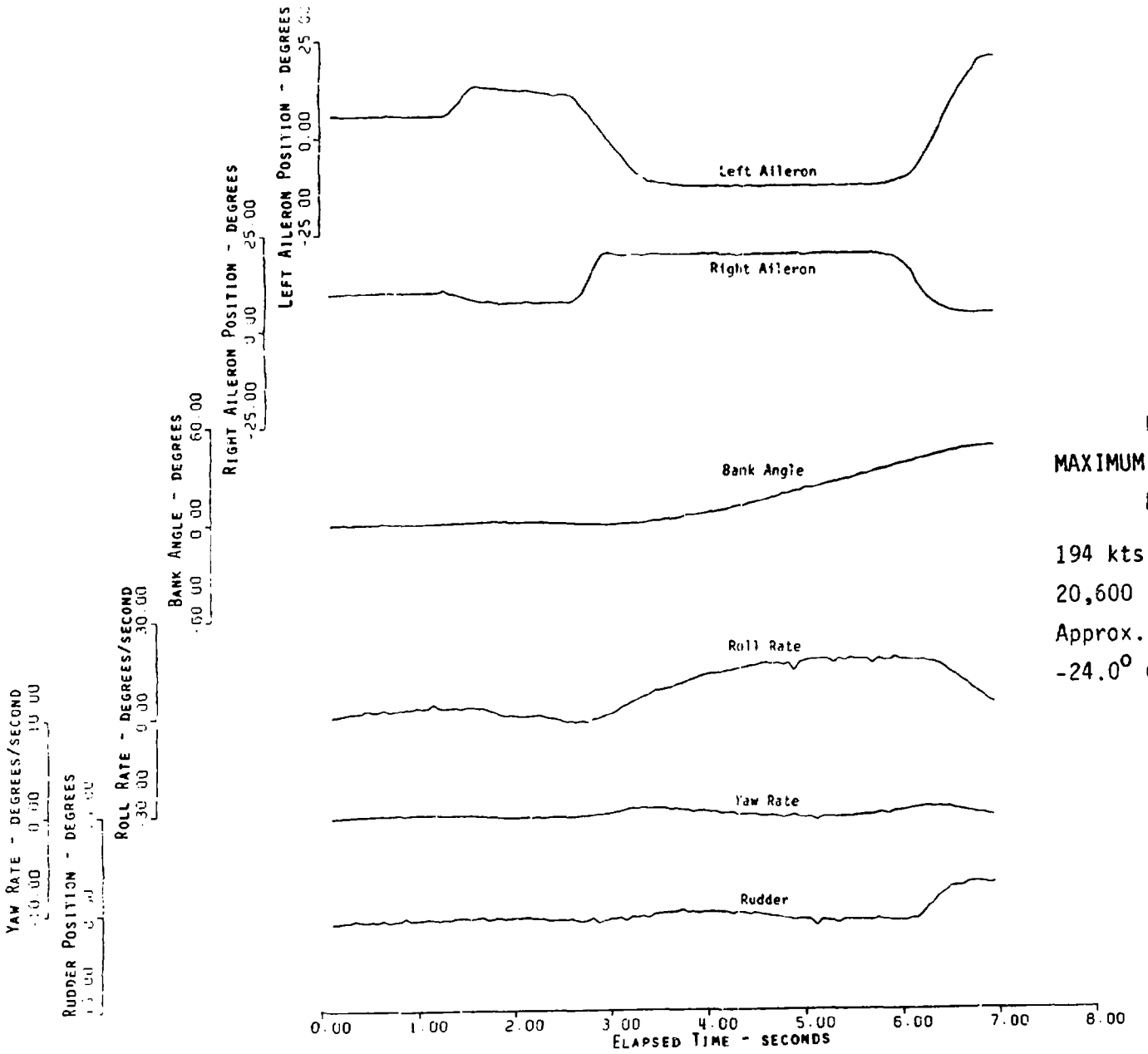


Figure A-6.4b
MAXIMUM DEFLECTION TAB ROLL
Right Tab

194 kts Airspeed
20,600 ft Altitude
Approx. 180,000 lb G.W.
-24.0° C OAT

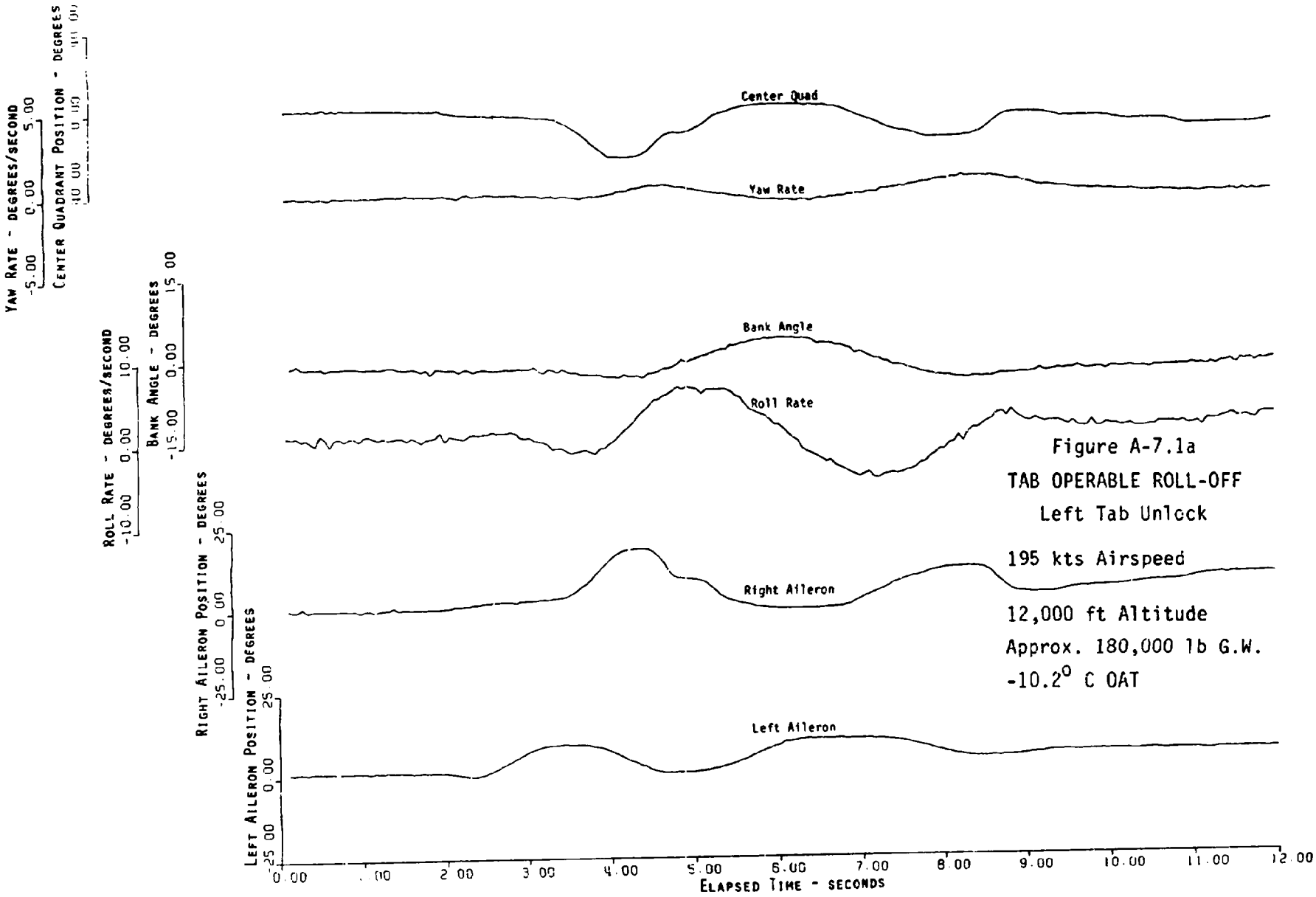


Figure A-7.1a
TAB OPERABLE ROLL-OFF
Left Tab Unlock

195 kts Airspeed

12,000 ft Altitude
Approx. 180,000 lb G.W.
-10.2° C OAT

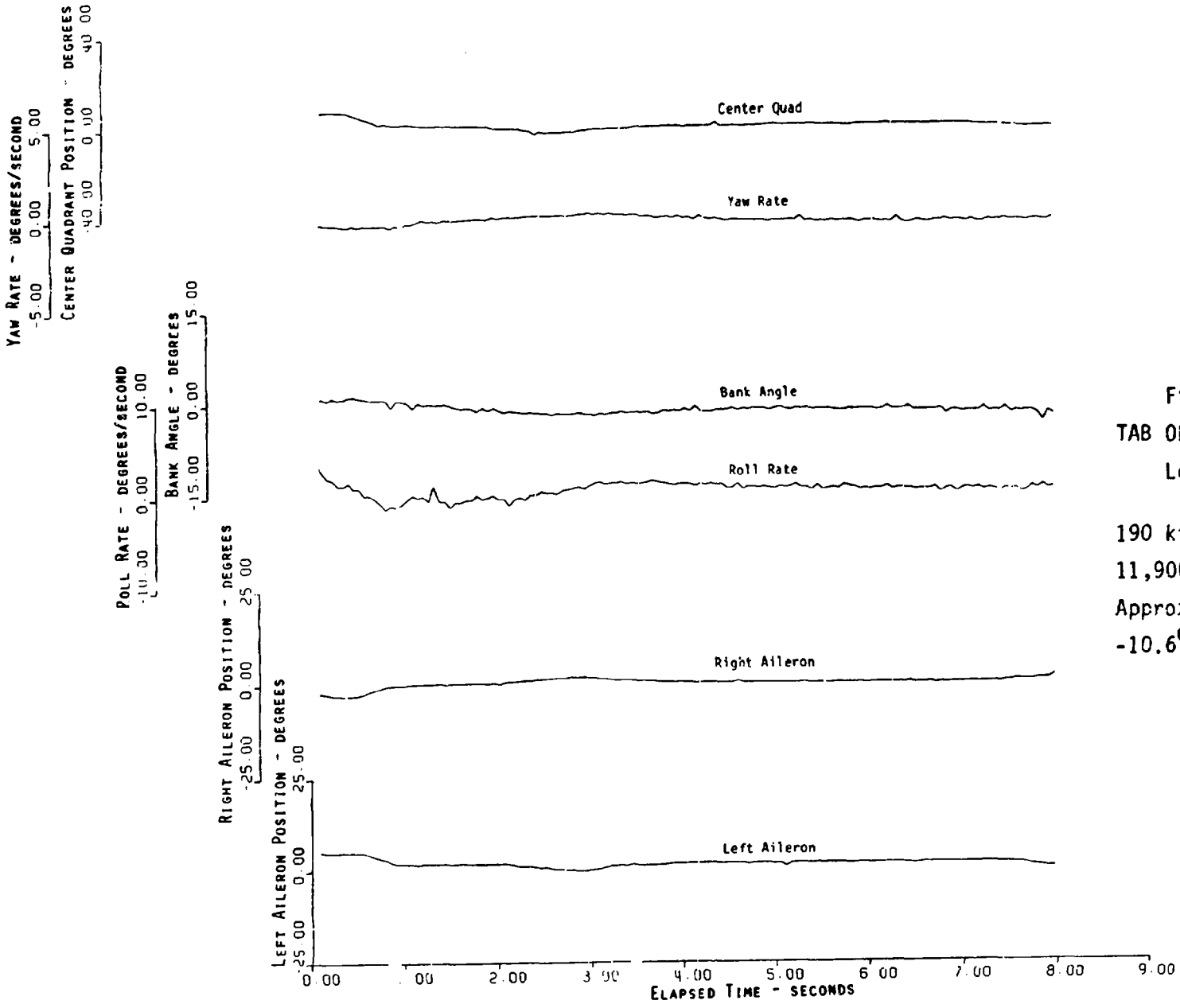


Figure A-7.1b
TAB OPERABLE ROLL-OFF
Left Tab Lock

190 kts Airspeed
11,900 ft Altitude
Approx. 180,000 ft G.W.
-10.6° C OAT

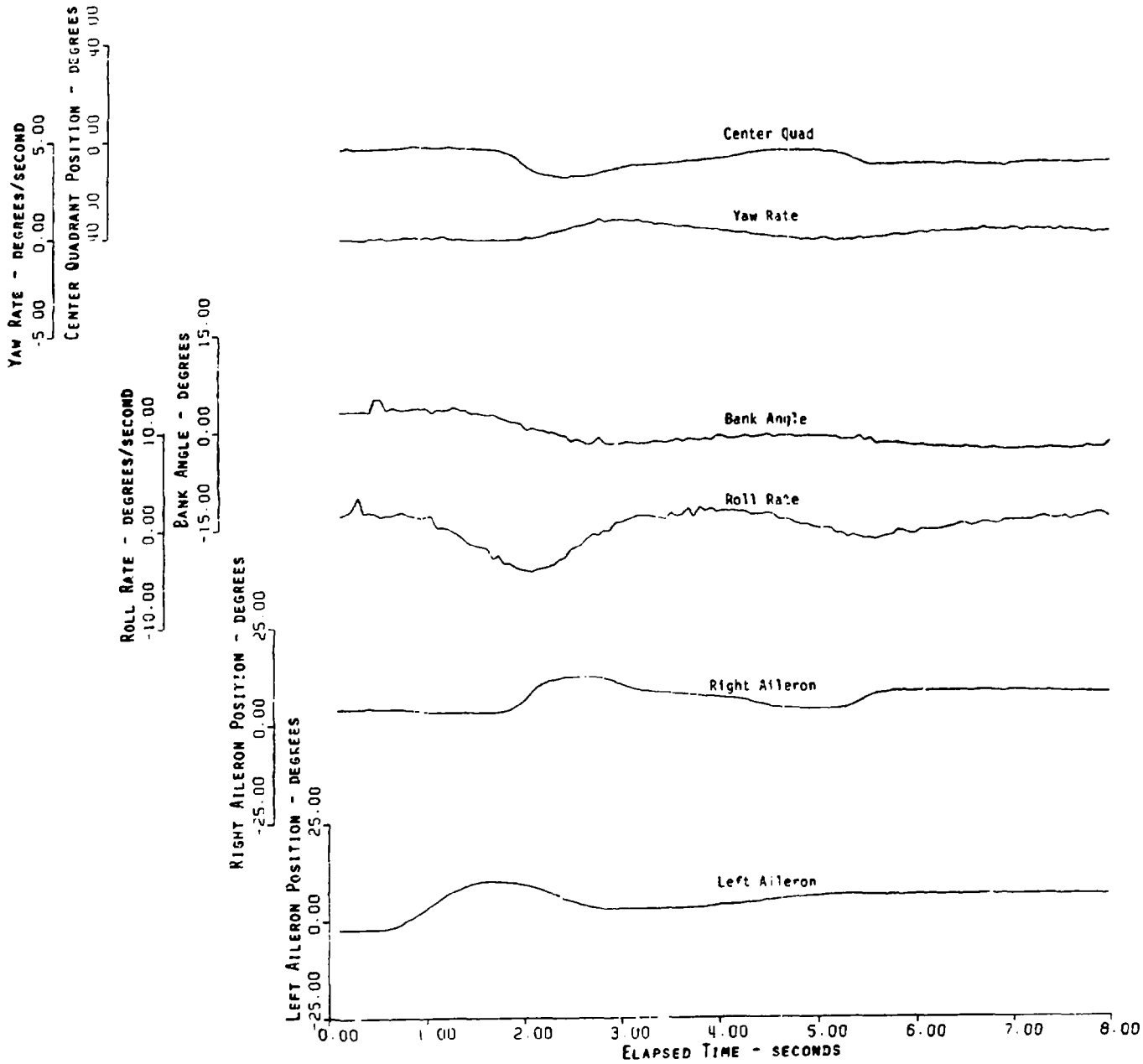


Figure A-7.2a
TAB OPERABLE ROLL-OFF
Left Tab Unlock

202 kts Airspeed
9,200 ft Altitude
Approx. 180,000 lb G.W.
8.5° C OAT

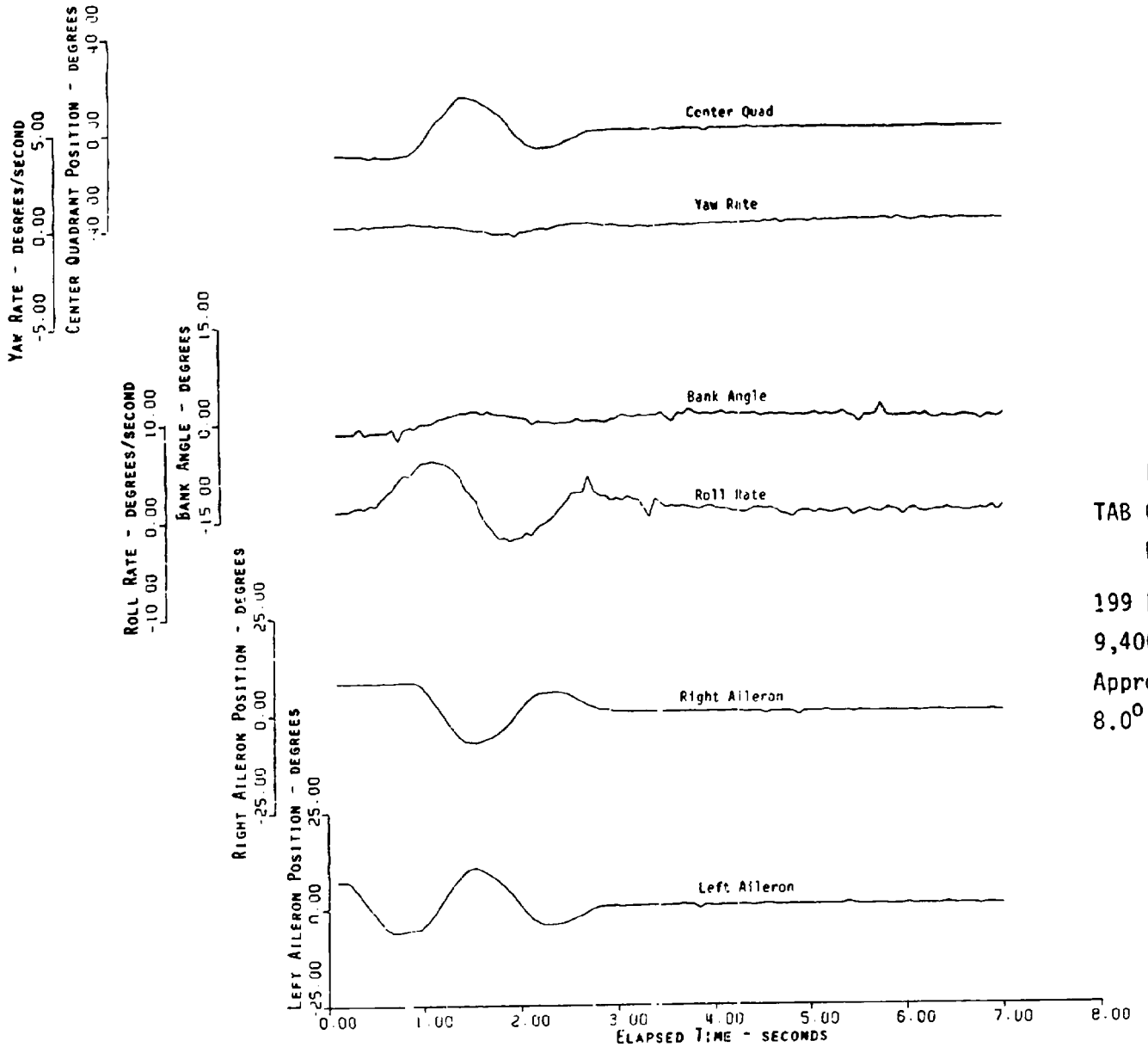


Figure A-7.2b
TAB OPERABLE ROLL-OFF
Left Tab Lock

199 kts Airspeed
9,400 ft Altitude
Approx. 180,000 lb G.W.
8.0° C OAT

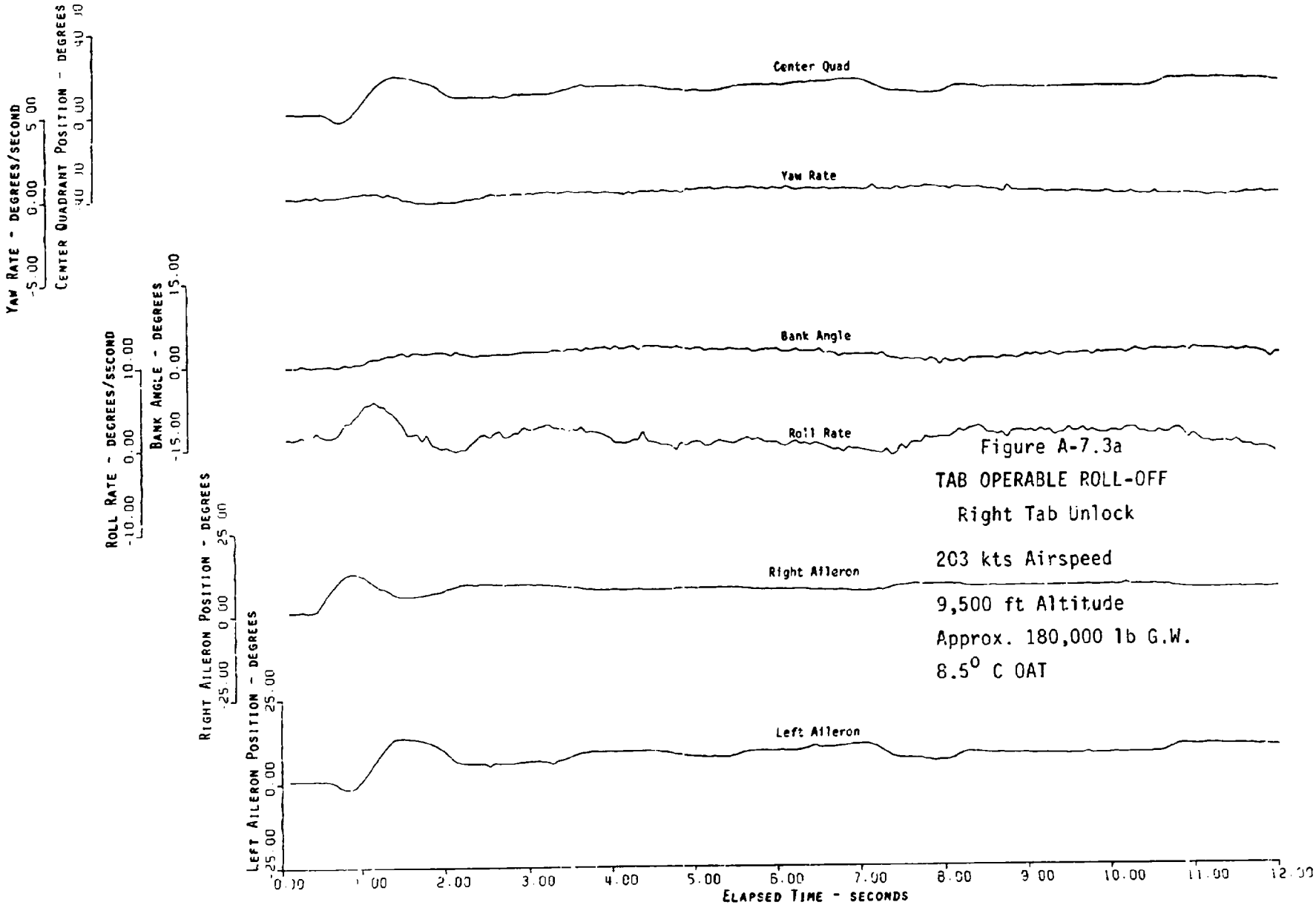


Figure A-7.3a

TAB OPERABLE ROLL-OFF

Right Tab Unlock

203 kts Airspeed

9,500 ft Altitude

Approx. 180,000 lb G.W.

8.5° C OAT

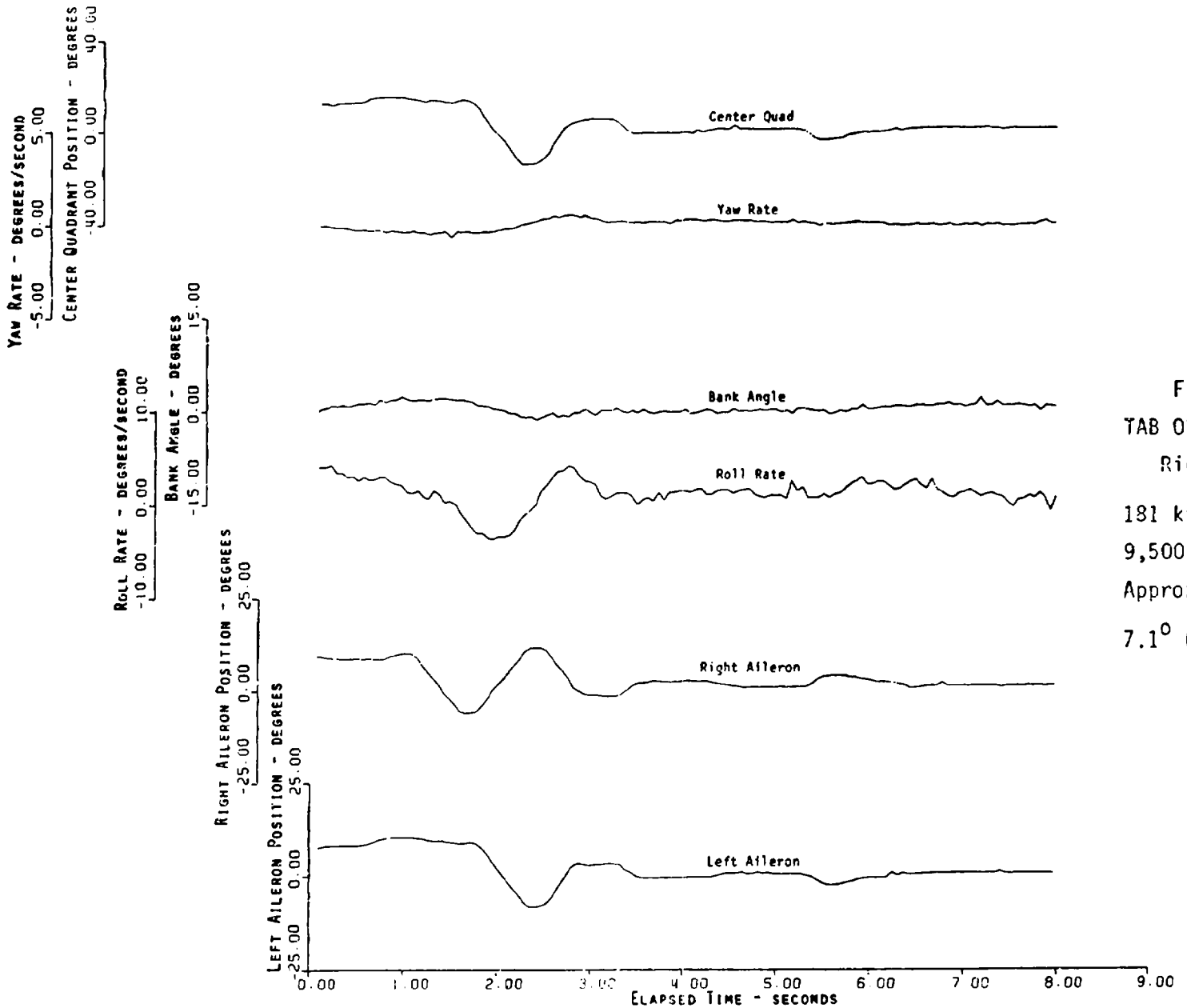
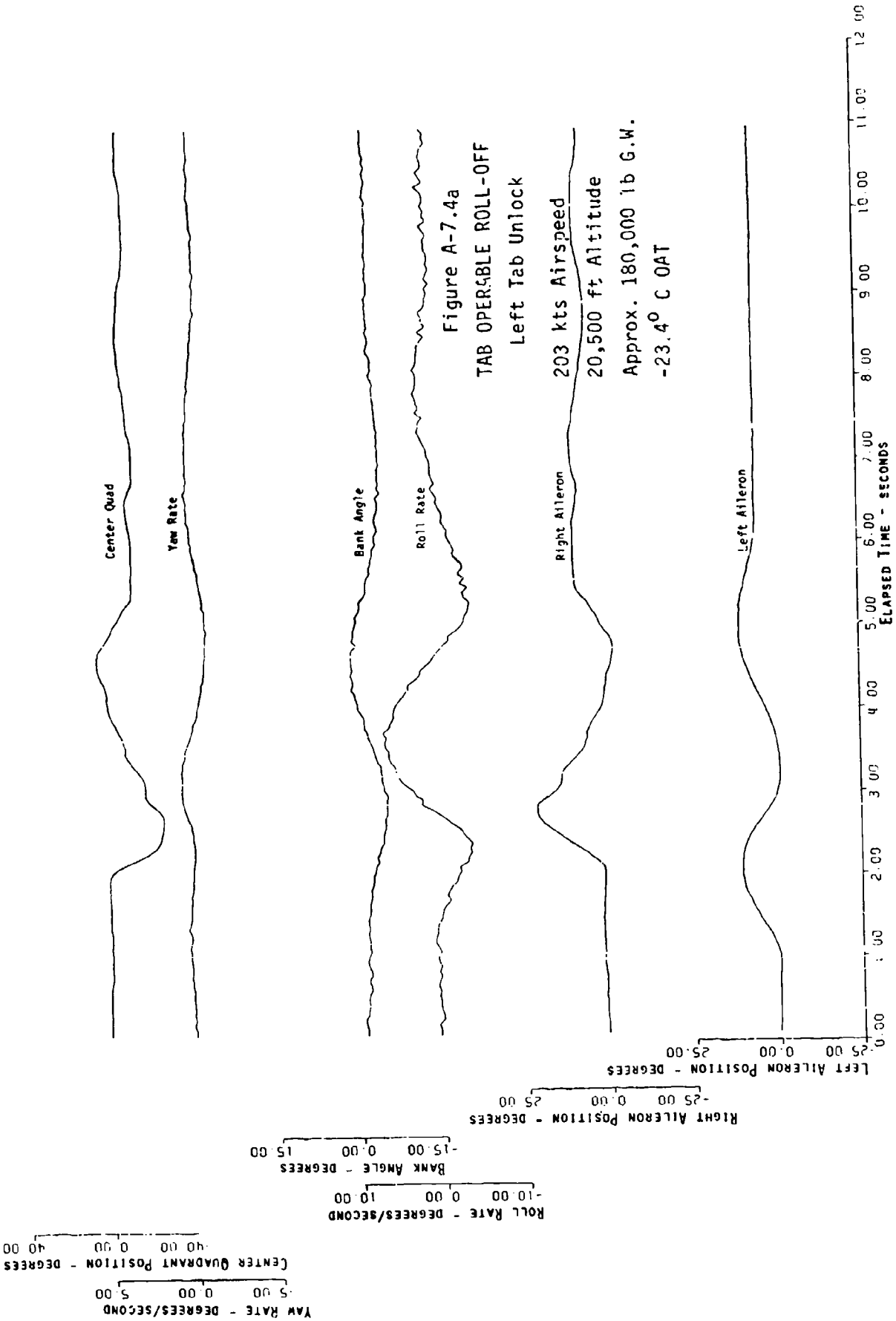


Figure A-7.3b
TAB OPERABLE ROLL-OFF
Right Tab Lock

181 kts Airspeed
9,500 ft Altitude
Approx. 180,000 lb G.W.

7.1° C OAT


Center Quad
Yaw Rate
Bank Angle
Roll Rate
Right Aileron
Left Aileron
Figure A-7.4a
TAB OPERABLE ROLL-OFF
Left Tab Unlock
203 kts Airspeed
20,500 ft Altitude
Approx. 180,000 lb G.W.
-23.4° C OAT
ELAPSED TIME - SECONDS
0.00
1.00
2.00
3.00
4.00
5.00
6.00
7.00
8.00
9.00
10.00
11.00
12.00
LEFT AILERON POSITION - DEGREES
-25.00
0.00
25.00
RIGHT AILERON POSITION - DEGREES
-25.00
0.00
25.00
BANK ANGLE - DEGREES
-15.00
0.00
15.00
ROLL RATE - DEGREES/SECOND
-10.00
0.00
10.00
CENTER QUADRANT POSITION - DEGREES
-40.00
0.00
40.00
YAW RATE - DEGREES/SECOND
-5.00
0.00
5.00

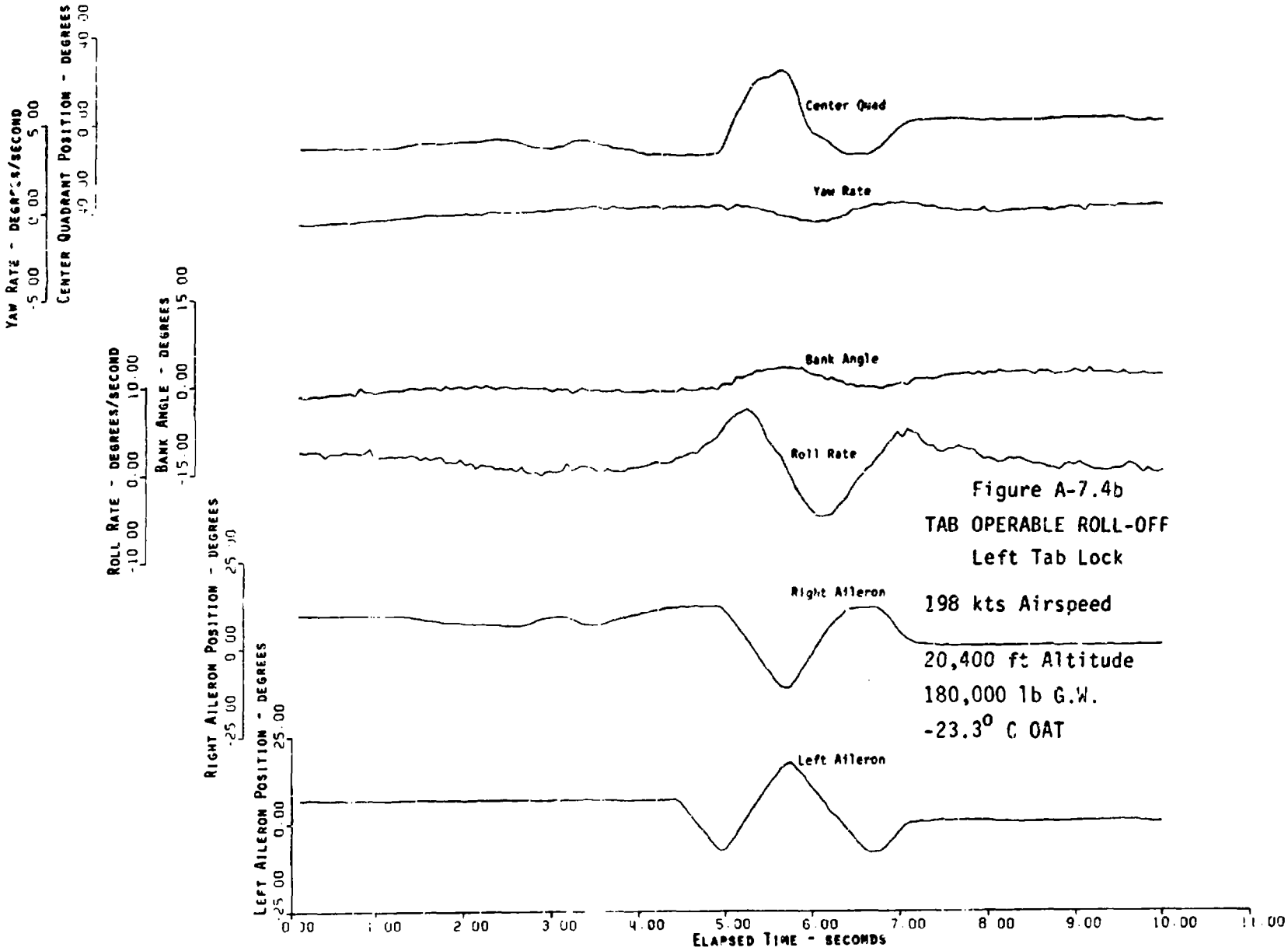


Figure A-7.4b
TAB OPERABLE ROLL-OFF
Left Tab Lock

198 kts Airspeed

20,400 ft Altitude
180,000 lb G.W.
-23.3° C OAT

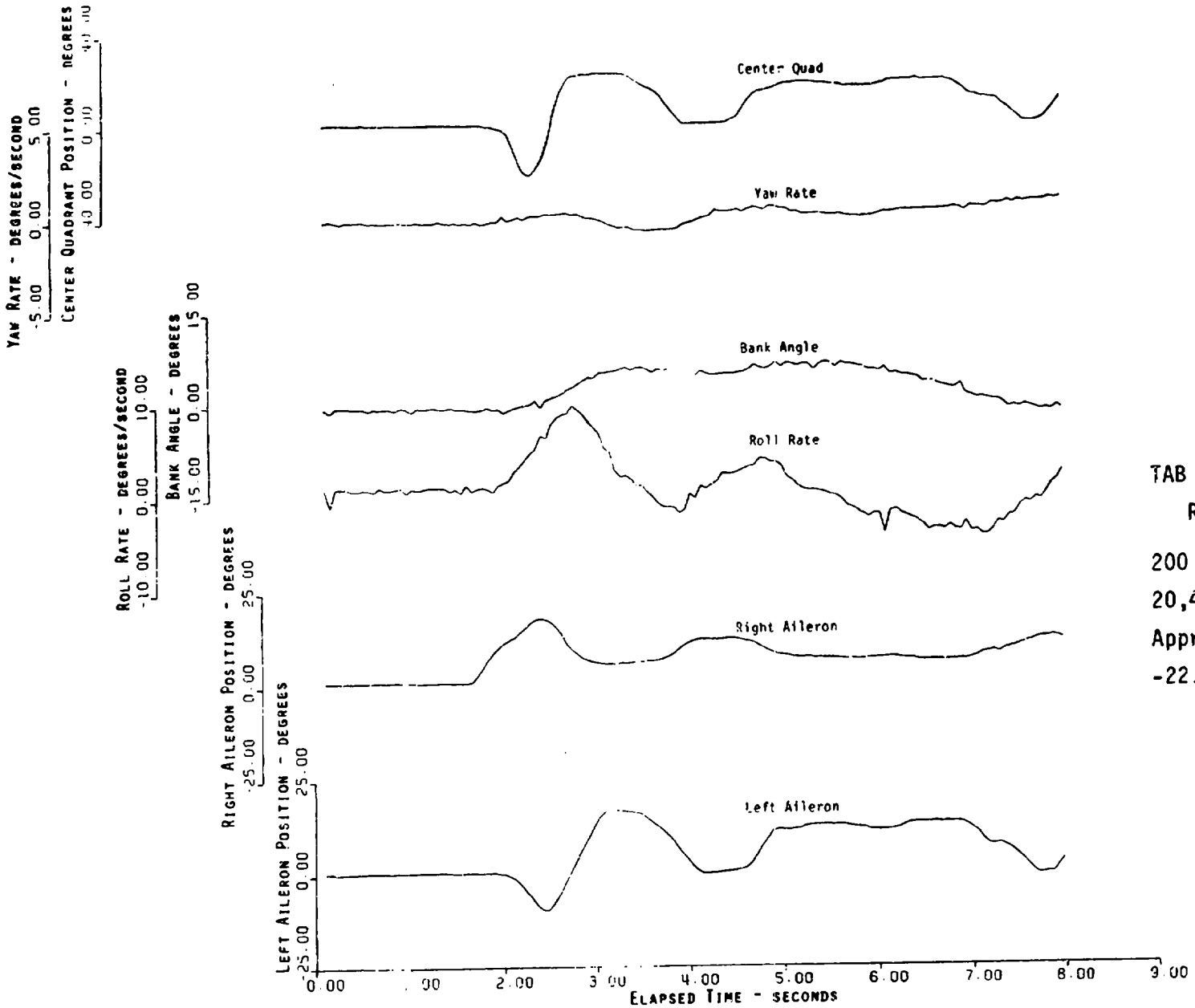


Figure A-7.5a
TAB OPERABLE ROLL-OFF
Right Tab Unlock

200 kts Airspeed
20,400 ft Altitude
Approx. 180,000 lb G.W.
-22.7° C OAT

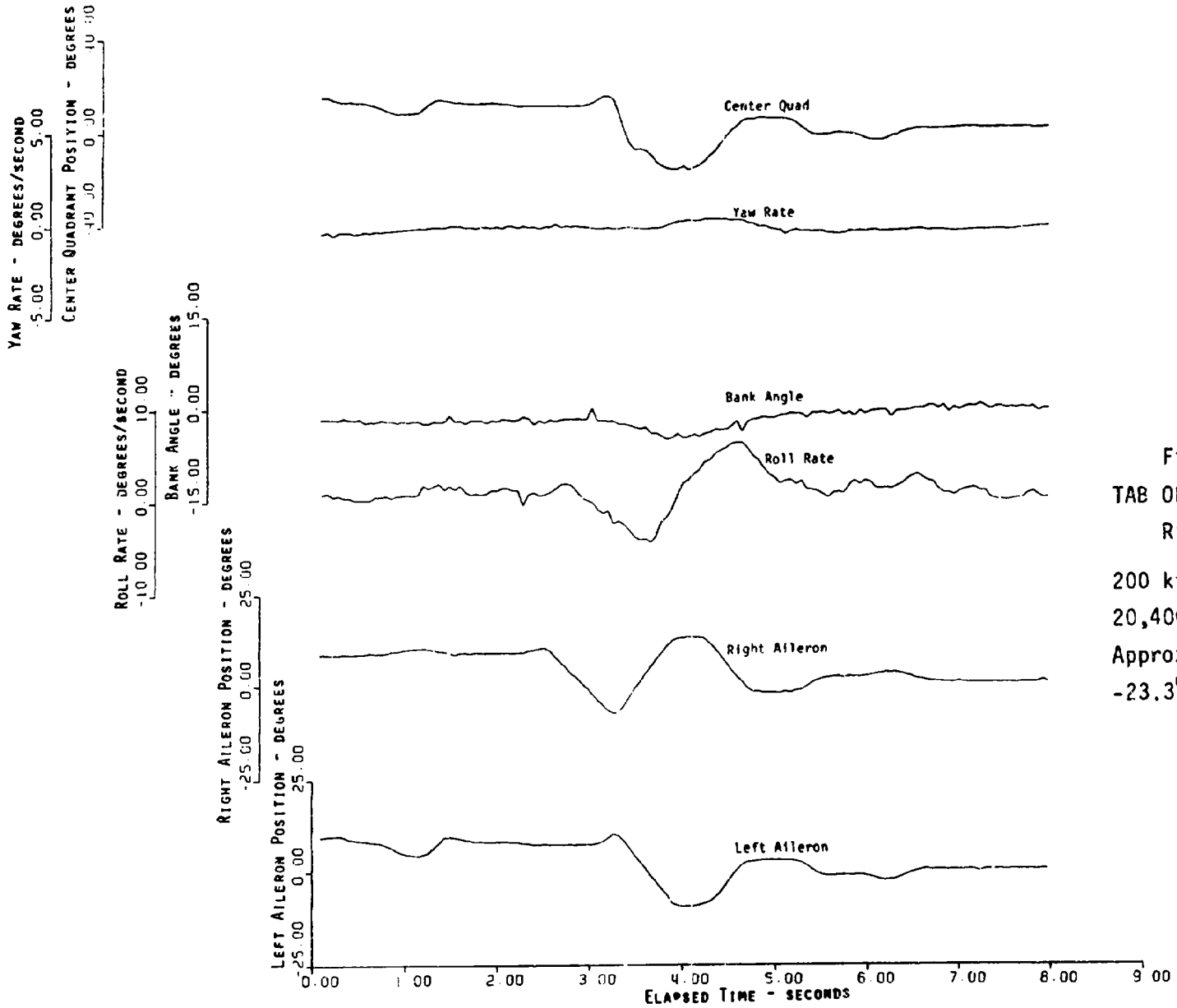


Figure A-7.5b
TAB OPERABLE ROLL-OFF
Right Tab Lock

200 kts Airspeed
20,400 ft Altitude
Approx. 180,000 lb G.W.
-23.3° C OAT

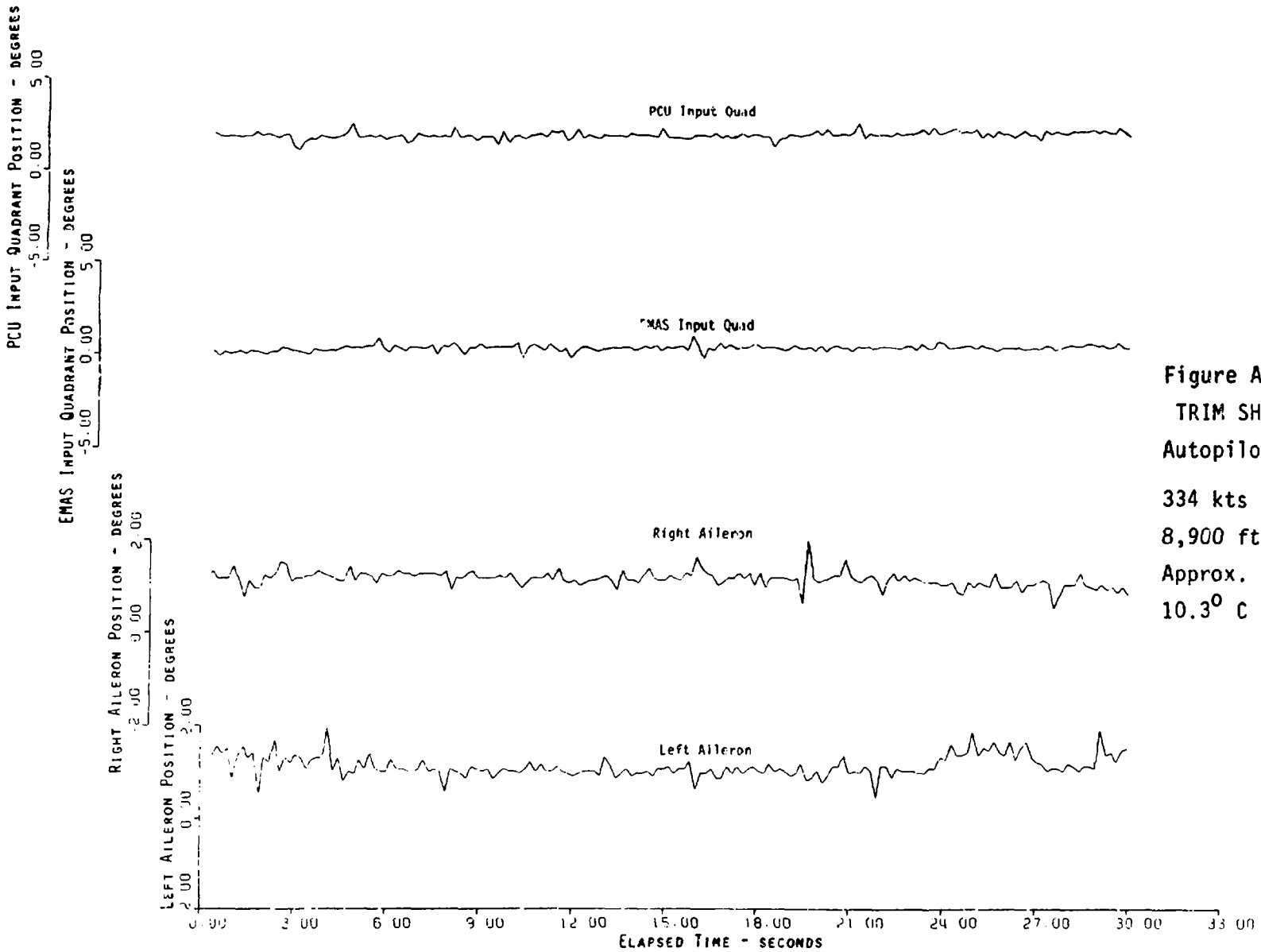


Figure A-8.1
TRIM SHOT
Autopilot Off

334 kts Airspeed
8,900 ft Altitude
Approx. 180,000 lb G,W,
10.3° C OAT

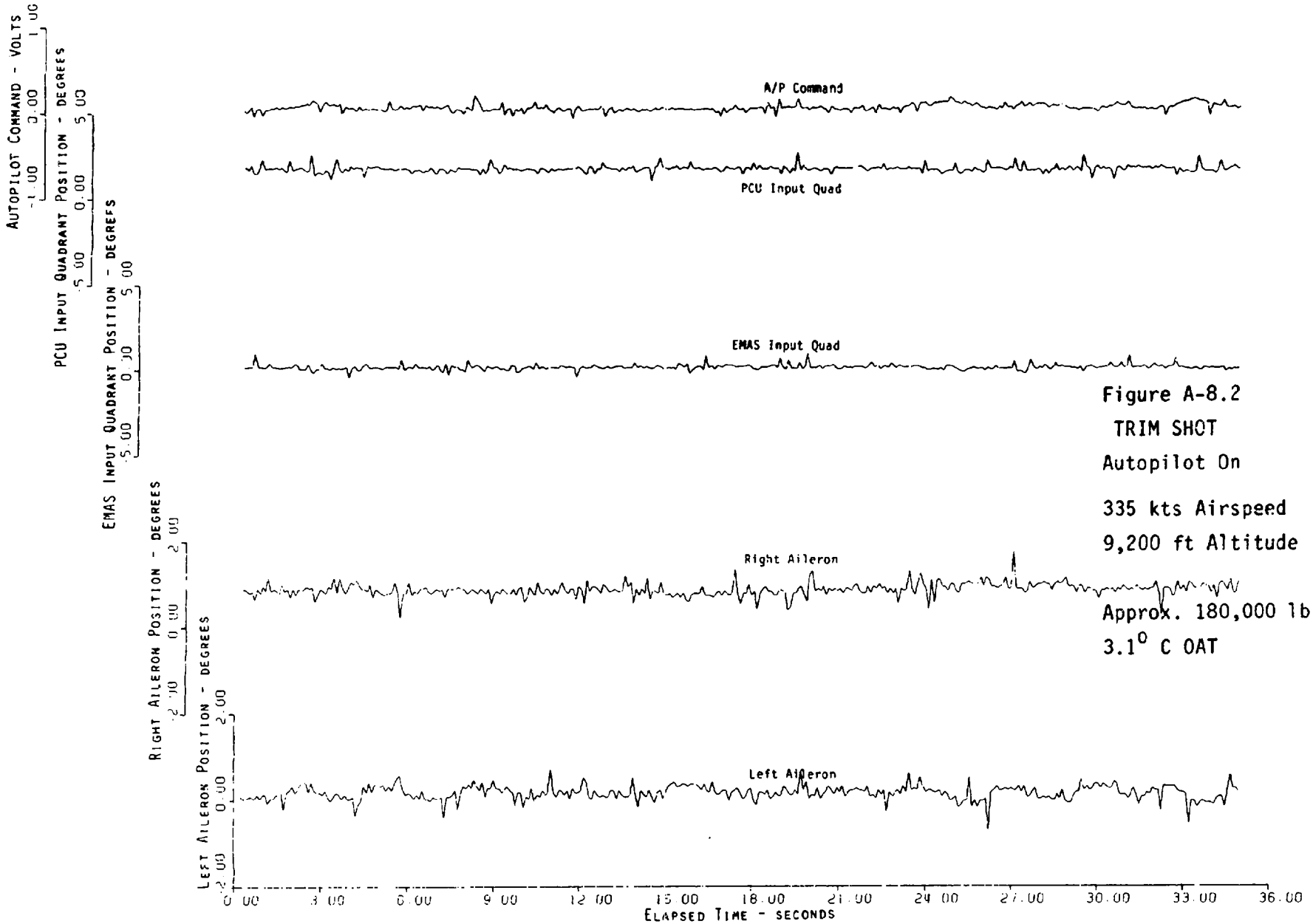


Figure A-8.2
TRIM SHOT
Autopilot On

335 kts Airspeed
9,200 ft Altitude

Approx. 180,000 lb G.W.
3.1° C OAT

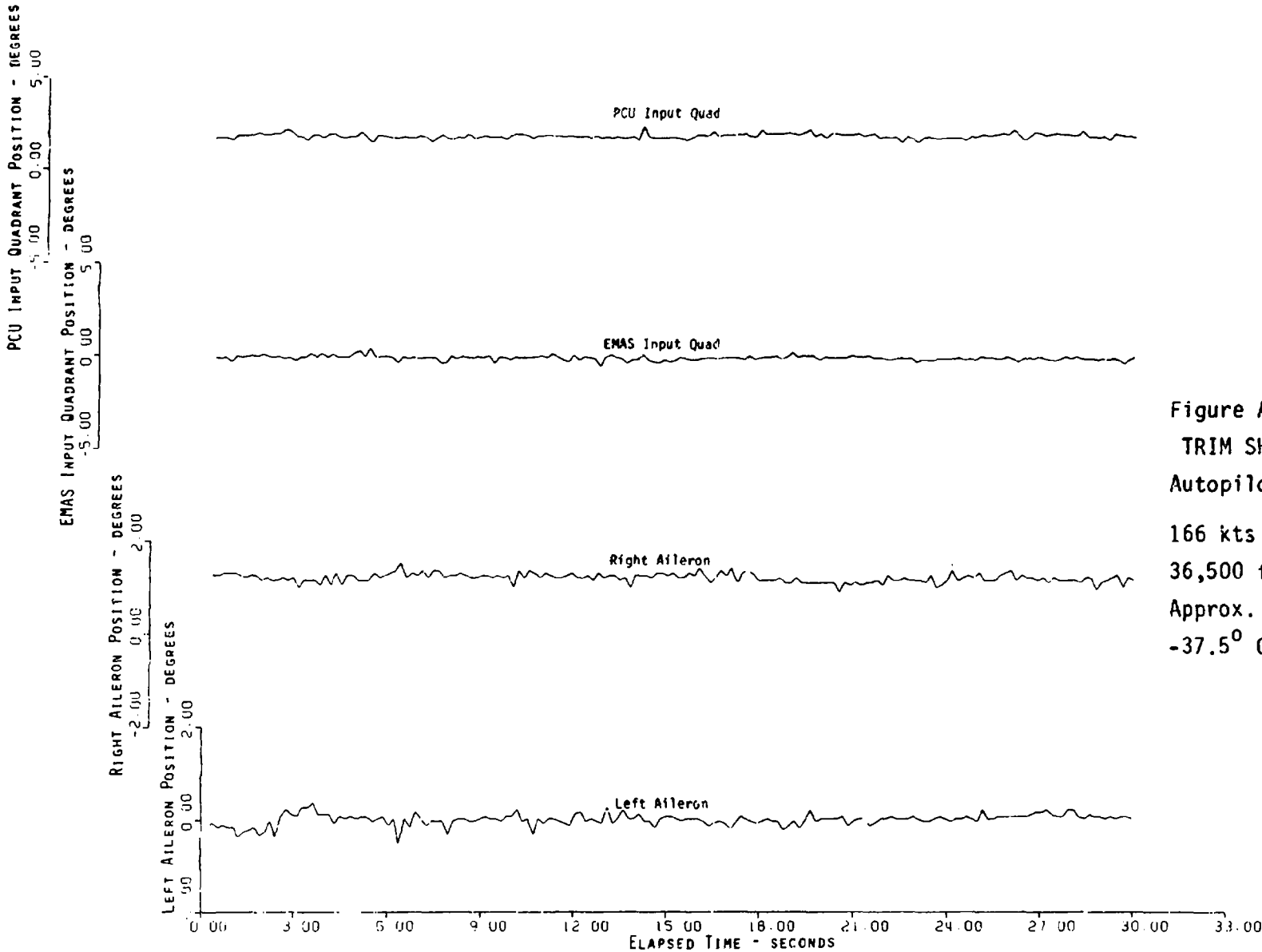


Figure A-8.3
TRIM SHOT
Autopilot Off

166 kts Airspeed
36,500 ft Altitude
Approx. 180,000 lb G.W.
-37.5° C OAT

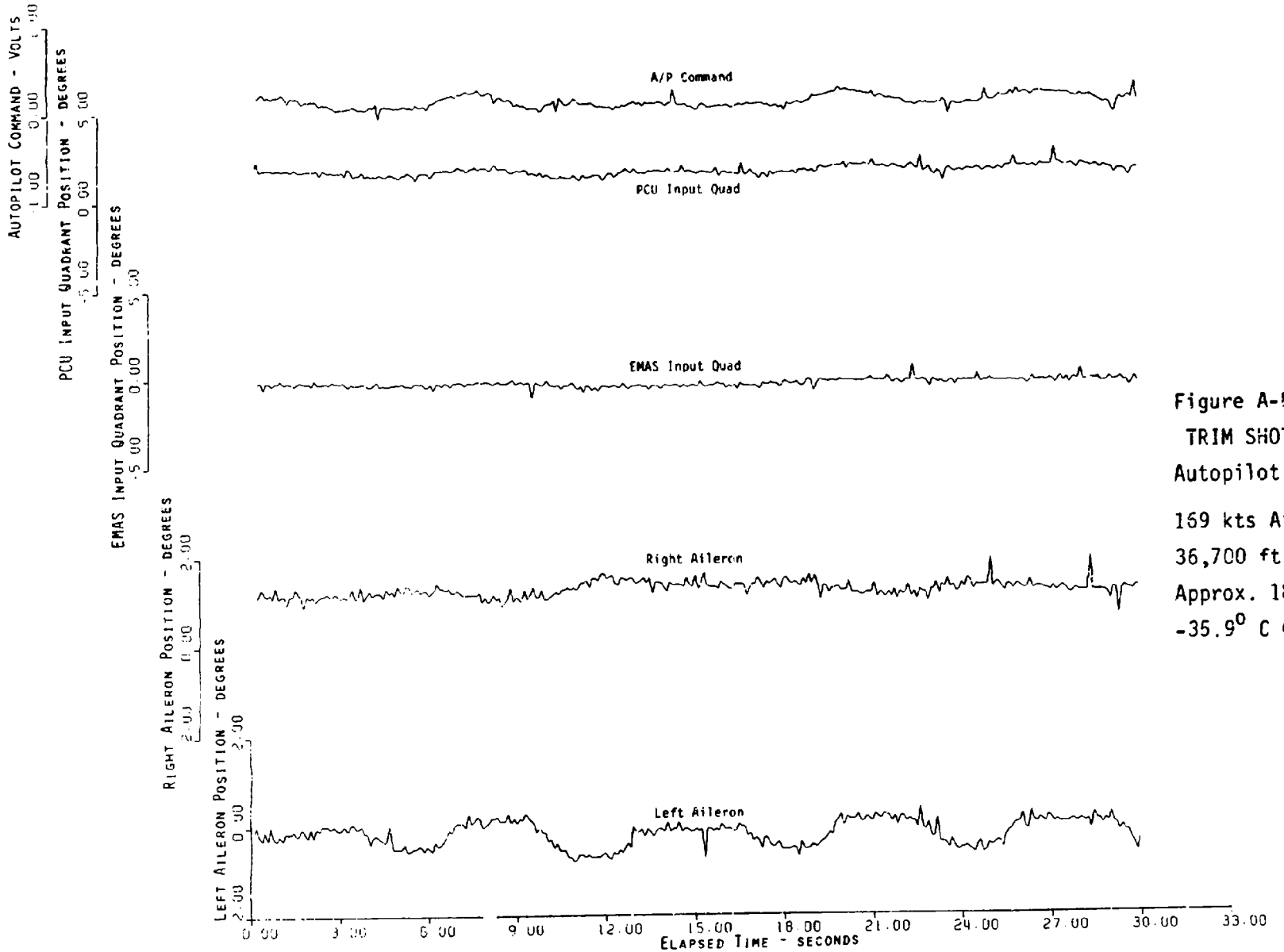


Figure A-8.4
TRIM SHOT
Autopilot On

169 kts Airspeed
36,700 ft Altitude
Approx. 180,000 lb G.W.
-35.9° C OAT

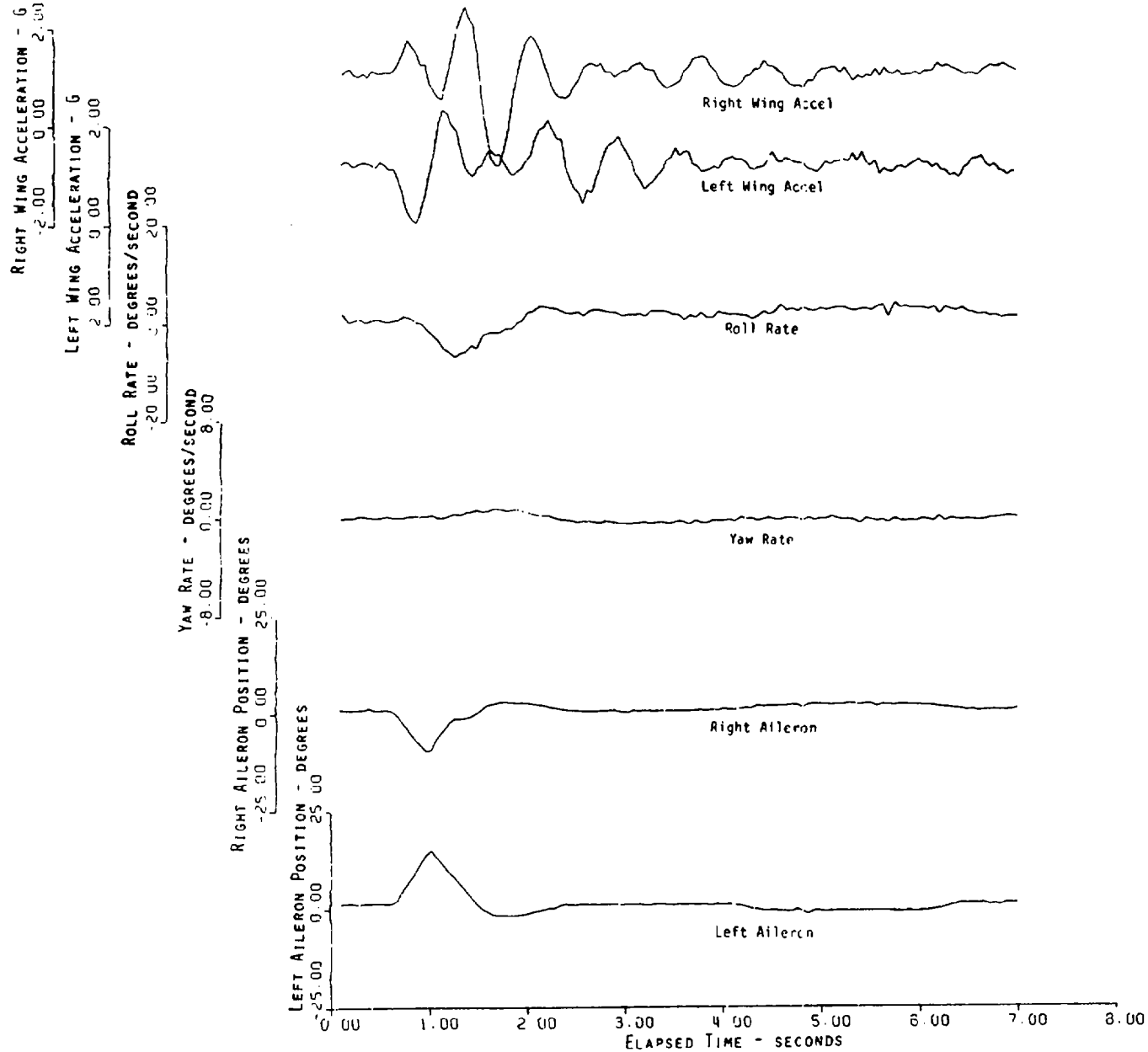


Figure A-9.1a
AILERON PULSE

335 kts Airspeed
20,500 ft Altitude
Approx. 180,000 lb G.W.
-4.3° C OAT

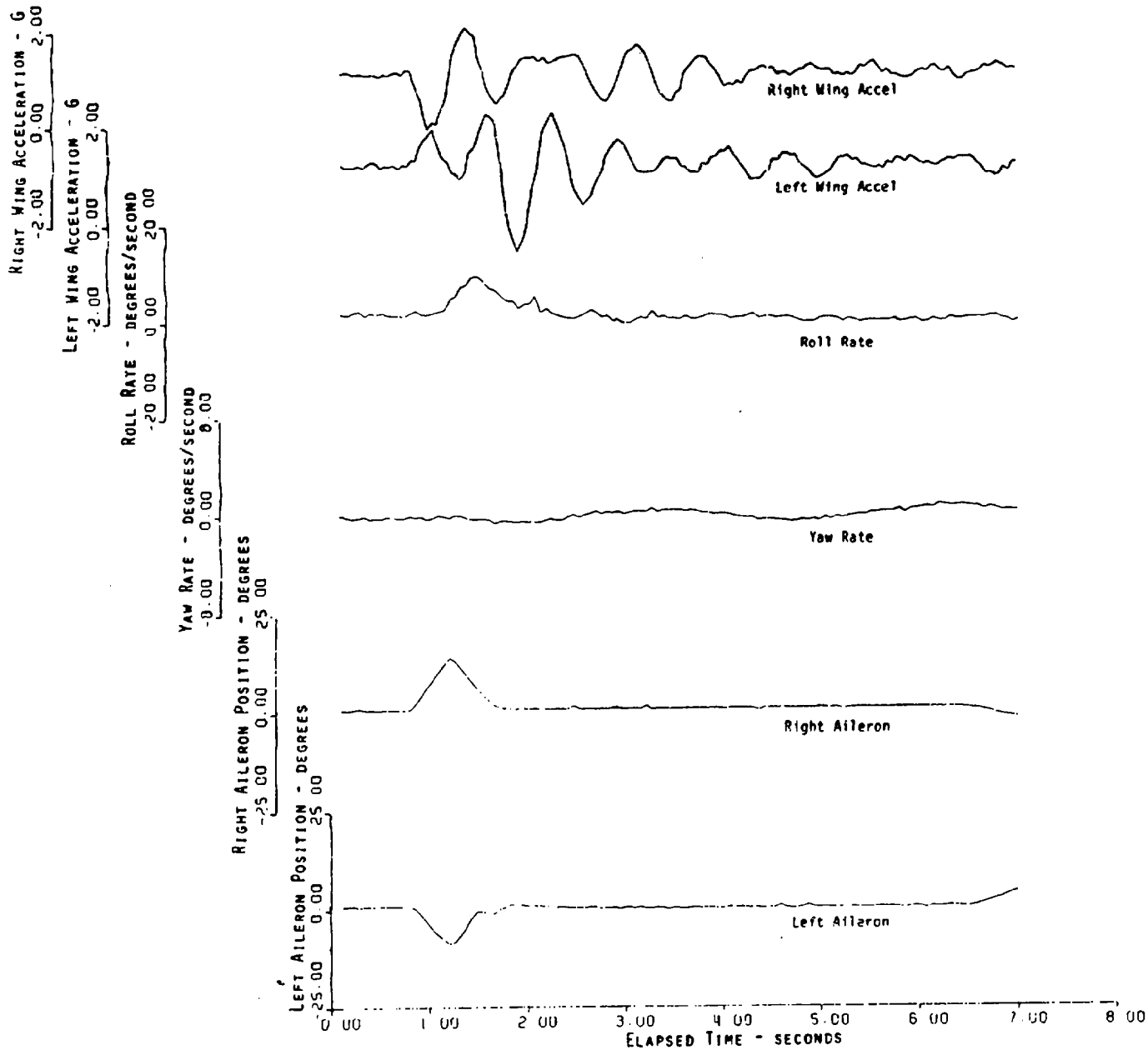


Figure A-9.1b
AILERON PULSE

335 kts Airspeed
20,500 ft Altitude
Approx. 180,000 lb G.W.
-5.0° C OAT

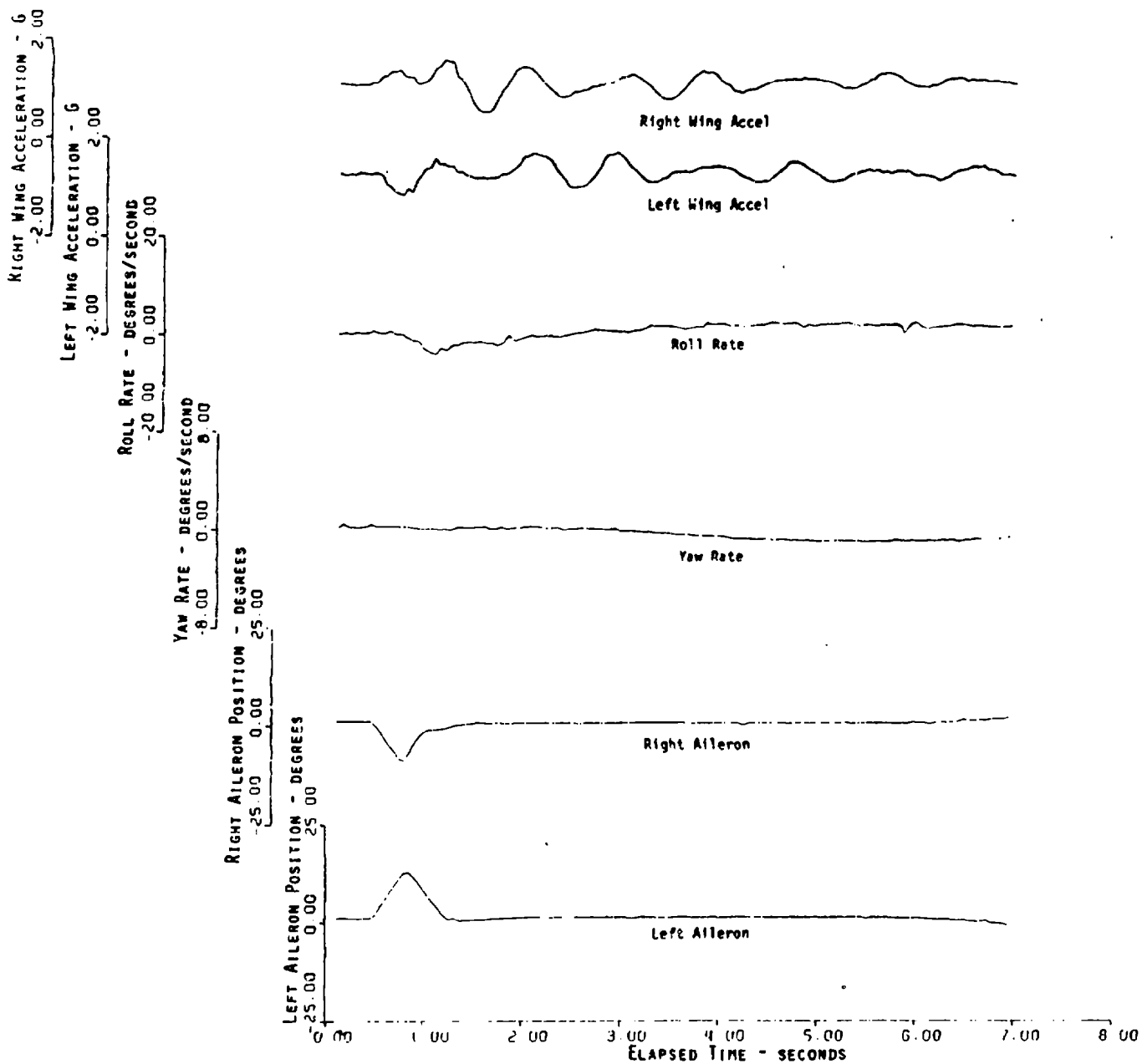


Figure A-9.2a
AILERON PULSE

156 kts Airspeed
35,600 ft Altitude
Approx. 180,000 lb G.W.
-36.6° C OAT

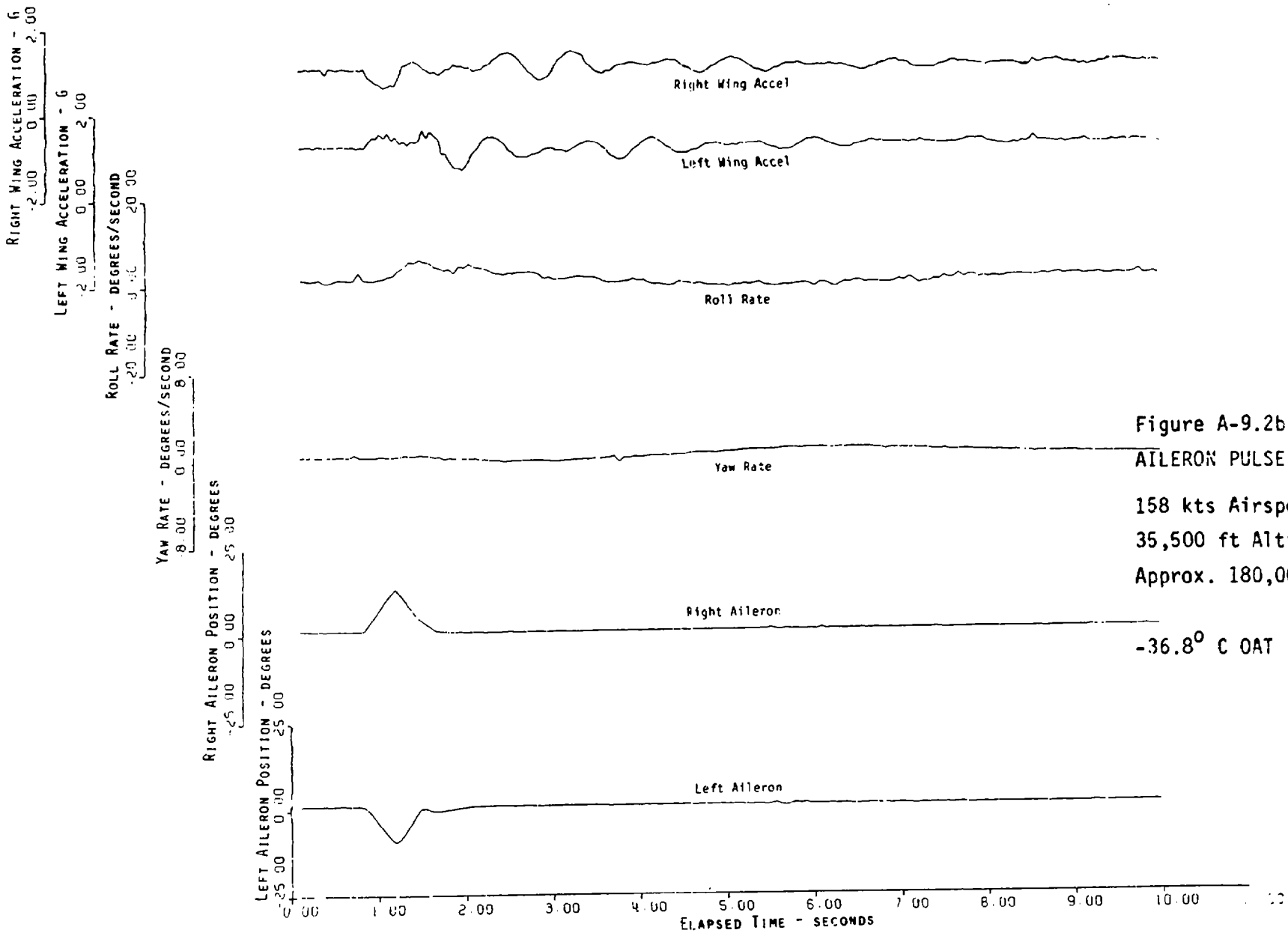


Figure A-9.2b
AILERON PULSE

158 kts Airspeed
35,500 ft Altitude
Approx. 180,000 lb G.W.

-36.8° C OAT

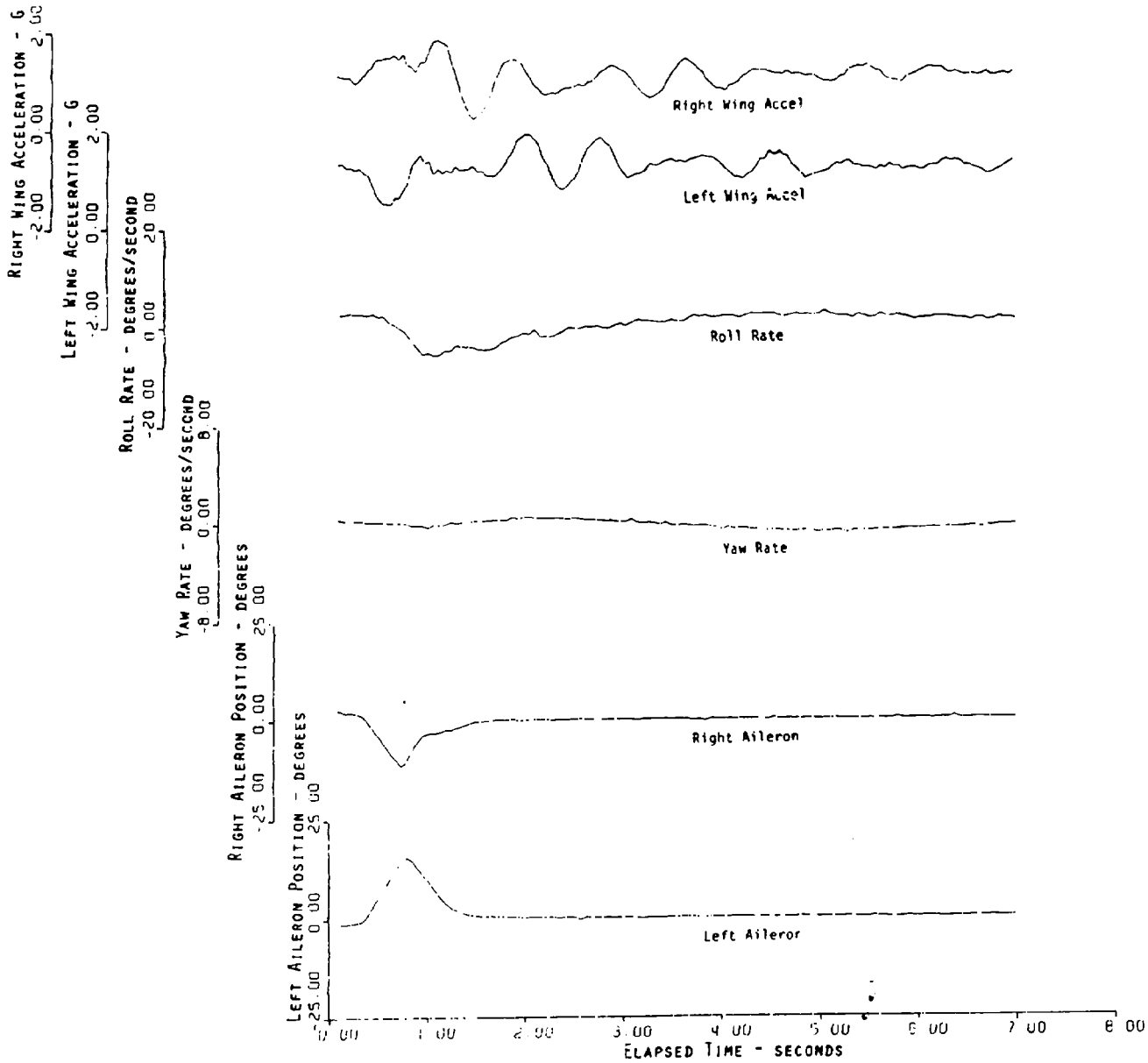


Figure A-9.3a
AILERON PULSE

245 kts Airspeed
40,200 ft Altitude
Approx. 180,000 lb G.W.
-23.8° C OAT

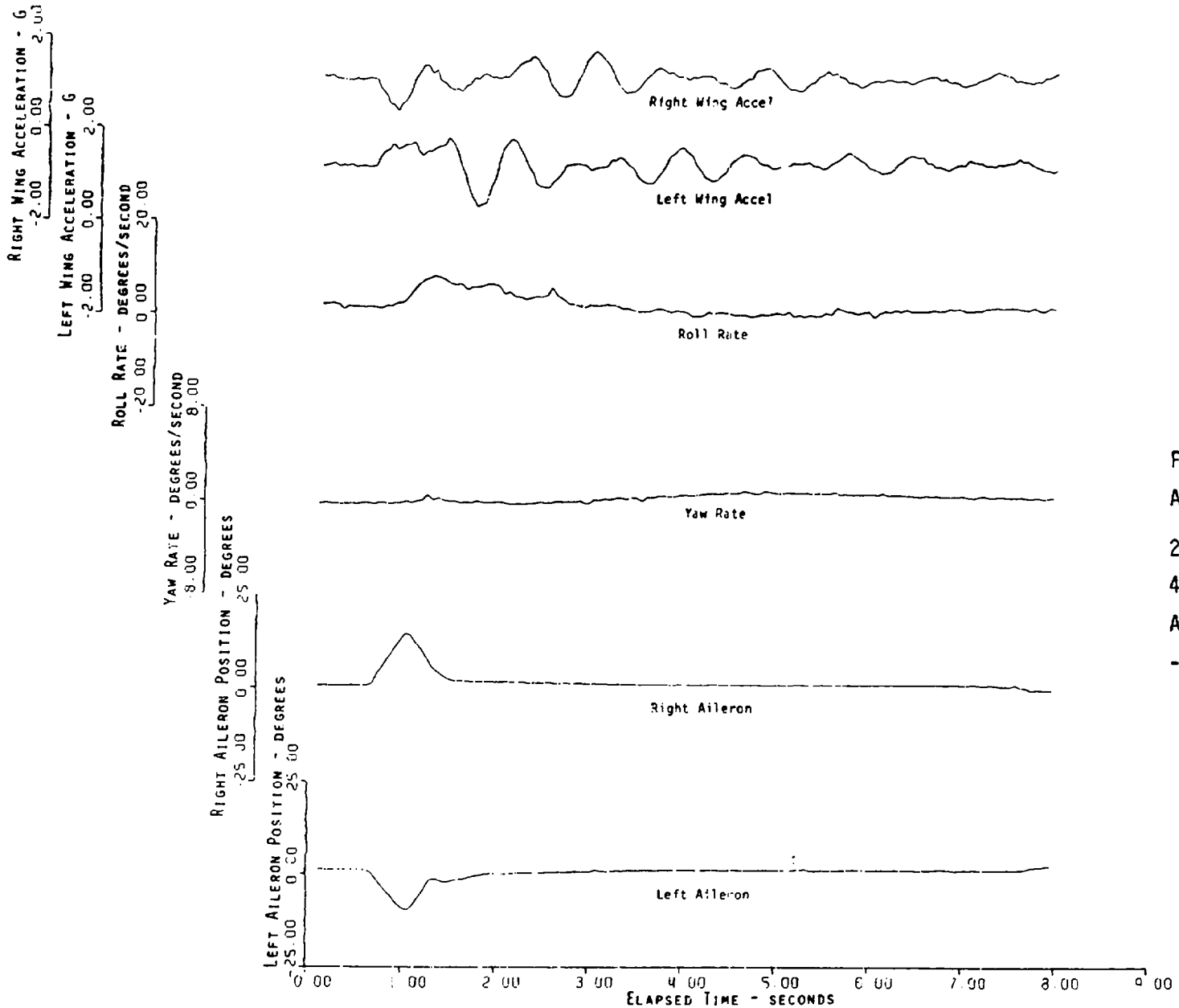


Figure A-9.3b
AILERON PULSE

245 kts Airspeed
40,200 ft Altitude
Approx. 180,000 lb G.W.
-23.8° C OAT

APPENDIX B

BASELINE DATA

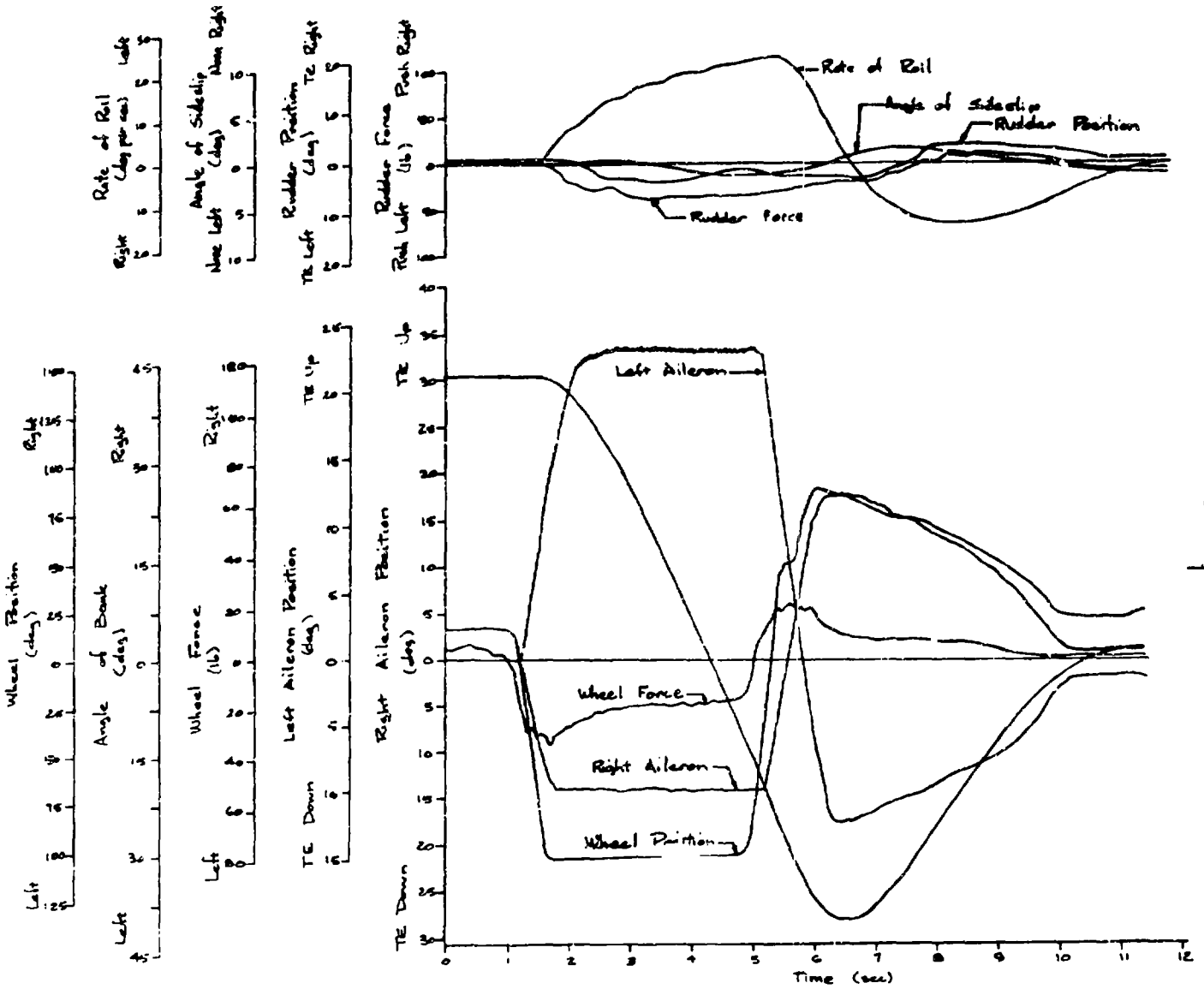


Figure B-1.1

MAXIMUM DEFLECTION ROLL
C-141A AFGI-2775 TF33-P-7 Engines
Cruise Configuration
Rudder Coordinated

Alt (ft)	Vc (kt)	Gr Wt (lb)	OAT (°C)
10800	283	260000	2

Yaw Damper On

from Ref. 1

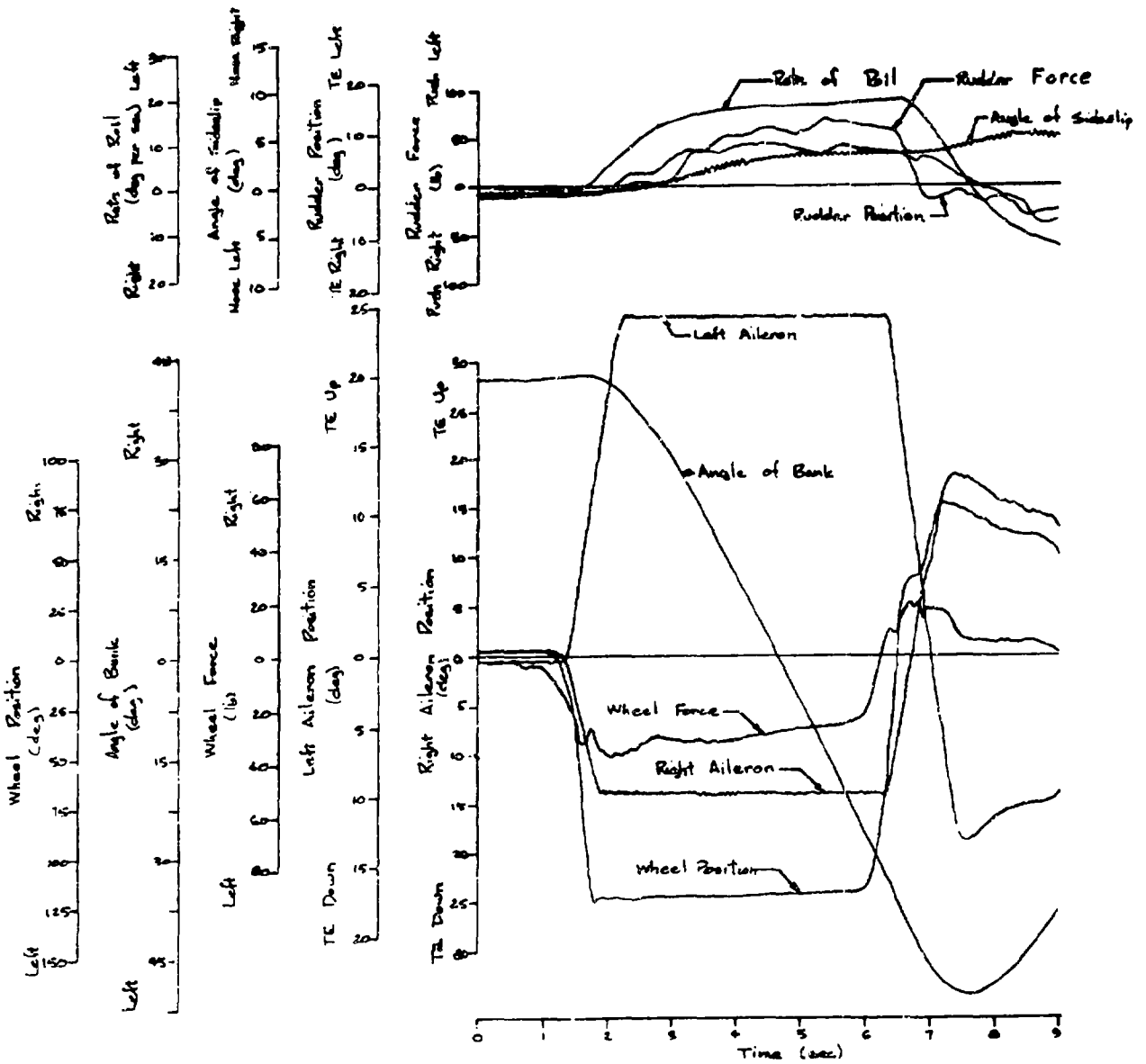


Figure B-1.2

MAXIMUM DEFLECTION ROLLS
C-141A AF61-2775 TF33-P-7 ENGINES

Alt (ft)	Cruise Configuration V_c (mt)	Mach No.	Gr Wt (lb)	OAT (°C)
36600	264	740	250000	-56

Rudder Coordinated
Yaw Damper On

from Ref. 1

Figure B-2

TAB SELECT TRIM CHANGE

C-141A AFSC-ETTS TF33-P-7 Engines

Cruise Configuration

Alt (ft)	V_t (kt)	Gr Wt (lb)
24600	222	139200

from Ref. 1

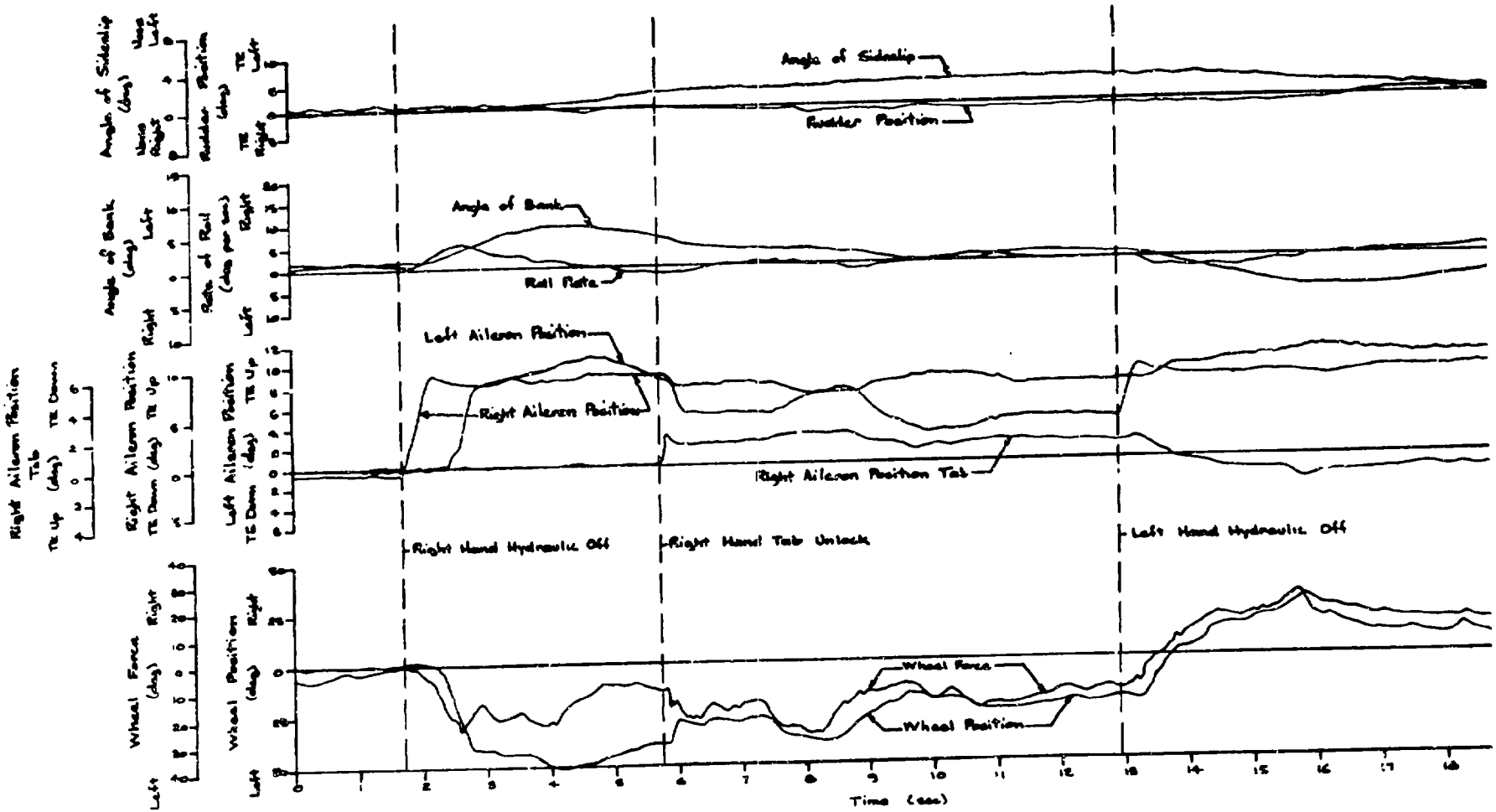


Figure B-3

TAB DE-SELECT TRIM CHANGE

C-141A AFGL-2378 TF33-P-7 Engines

Cruise Configuration

Alt (ft)	V_c (kts)	Gr Wt (lb)
10,000	205	220,000

from Ref. 1

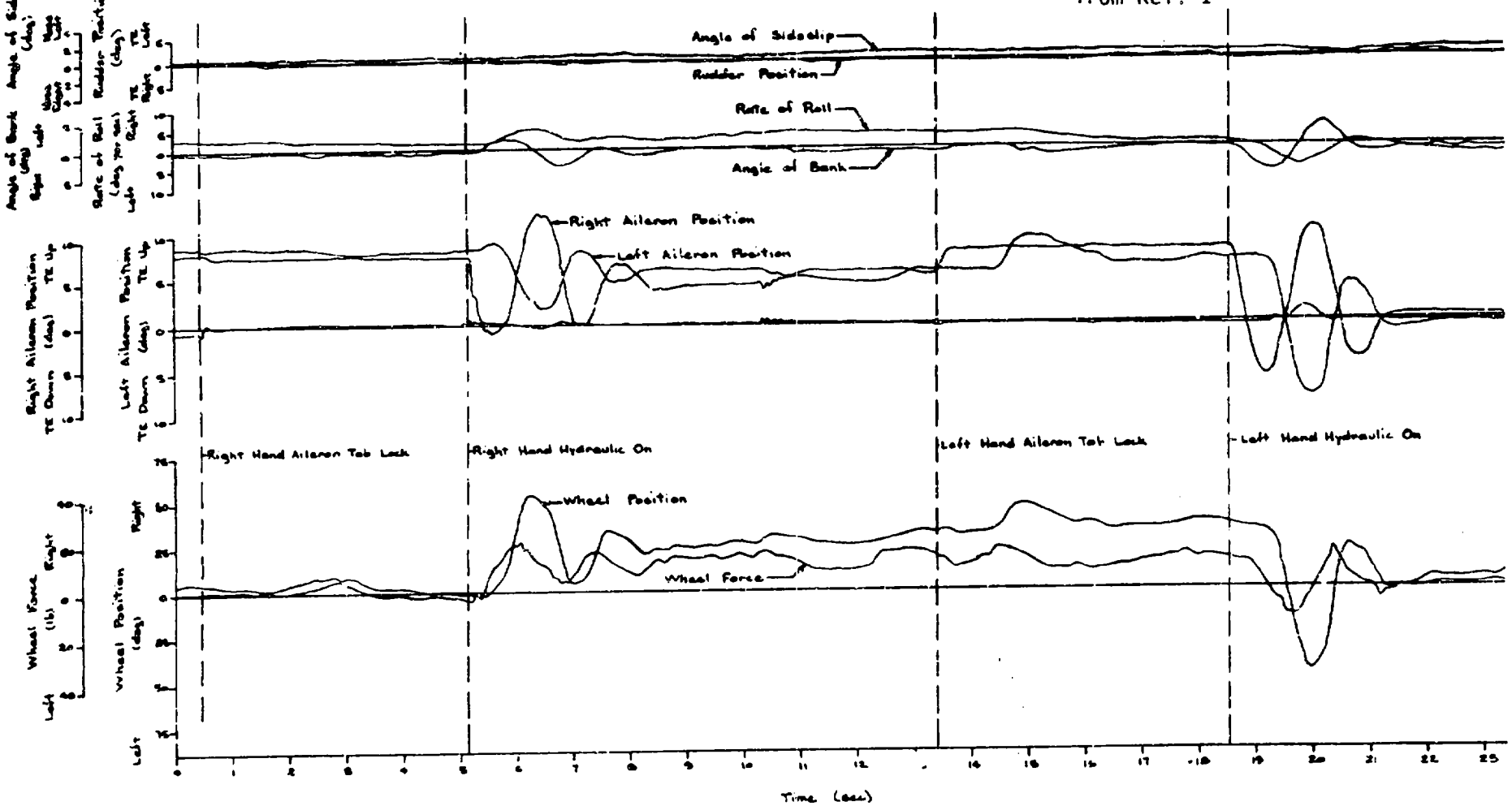


Figure B-4

BANK-TO-BANK TAB ROLL

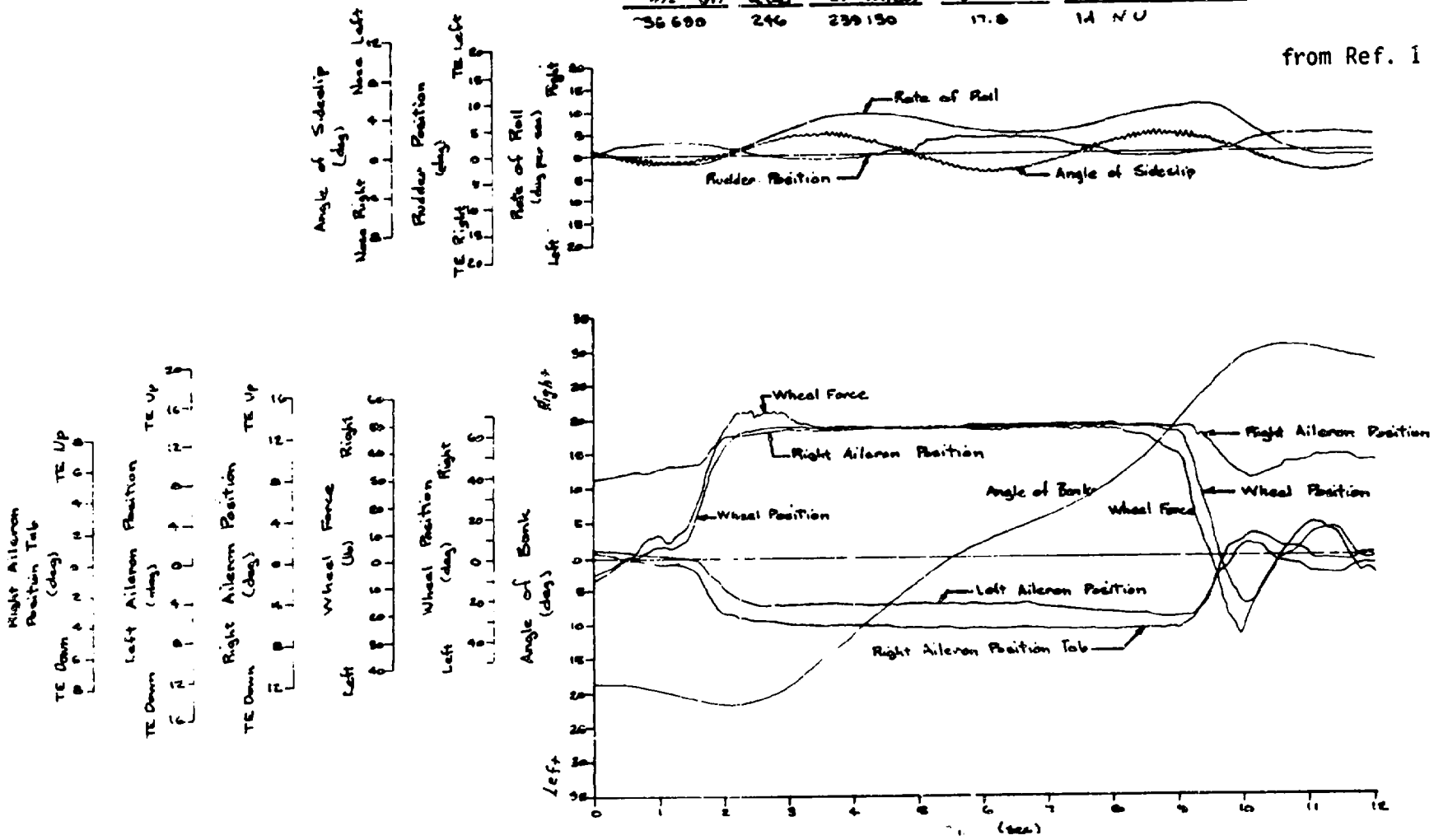
EMERGENCY BACK UP FLIGHT CONTROL SYSTEM

C-141A AFGL-2775 TF33-P-7 Engines

Cruise Configuration

Alt (ft)	V_c (kt)	Gr. Wt. (lb)	c.g. - % MAC	Stab Pos (deg)
36 690	246	239 150	17.8	1d NU

from Ref. 1



APPENDIX C

ADDITIONAL TEST DATA

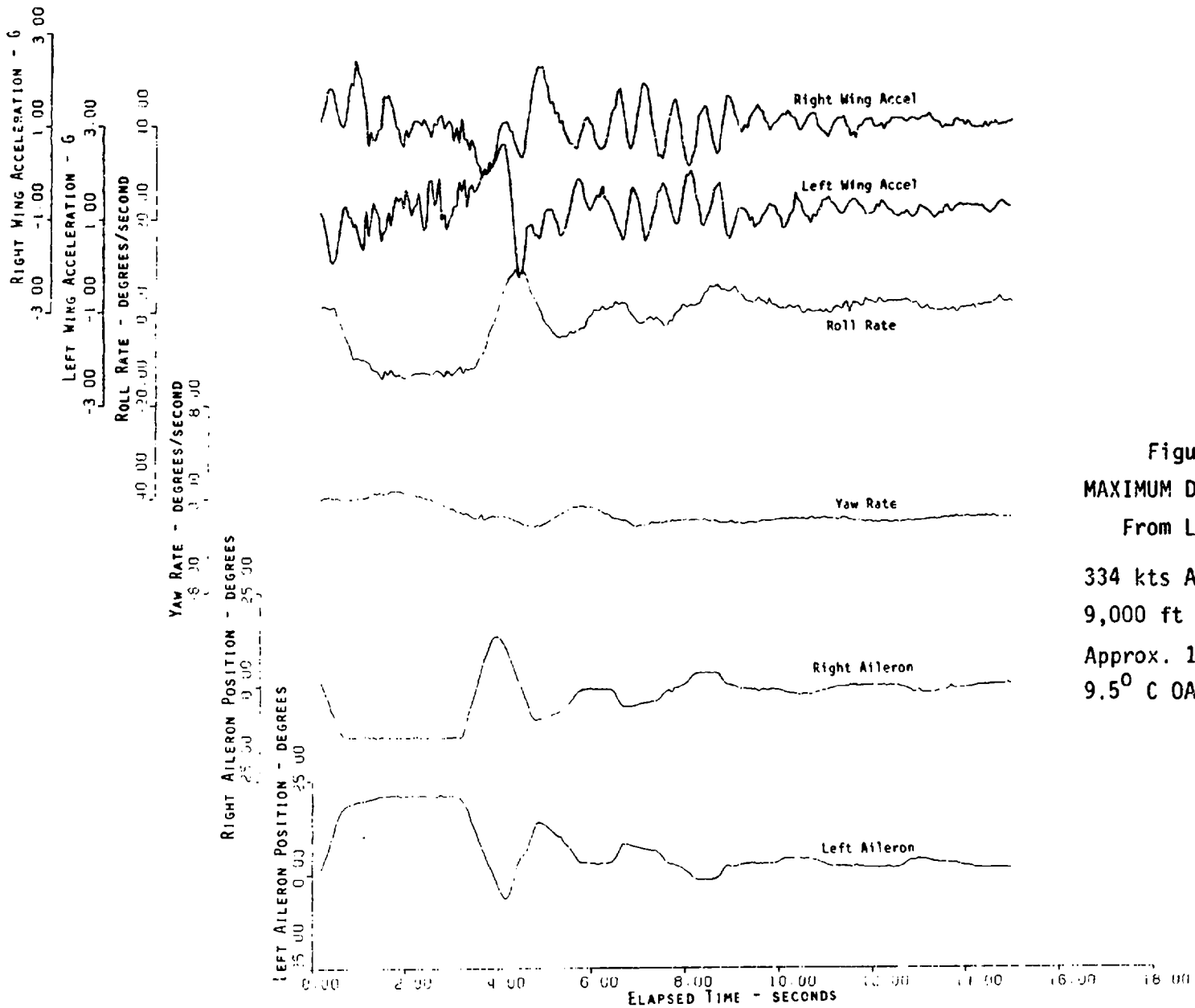


Figure C-1a
MAXIMUM DEFLECTION ROLL
From Level Flight

334 kts Airspeed
9,000 ft Altitude
Approx. 180,000 lb G.W.
9.5^{0} C OAT

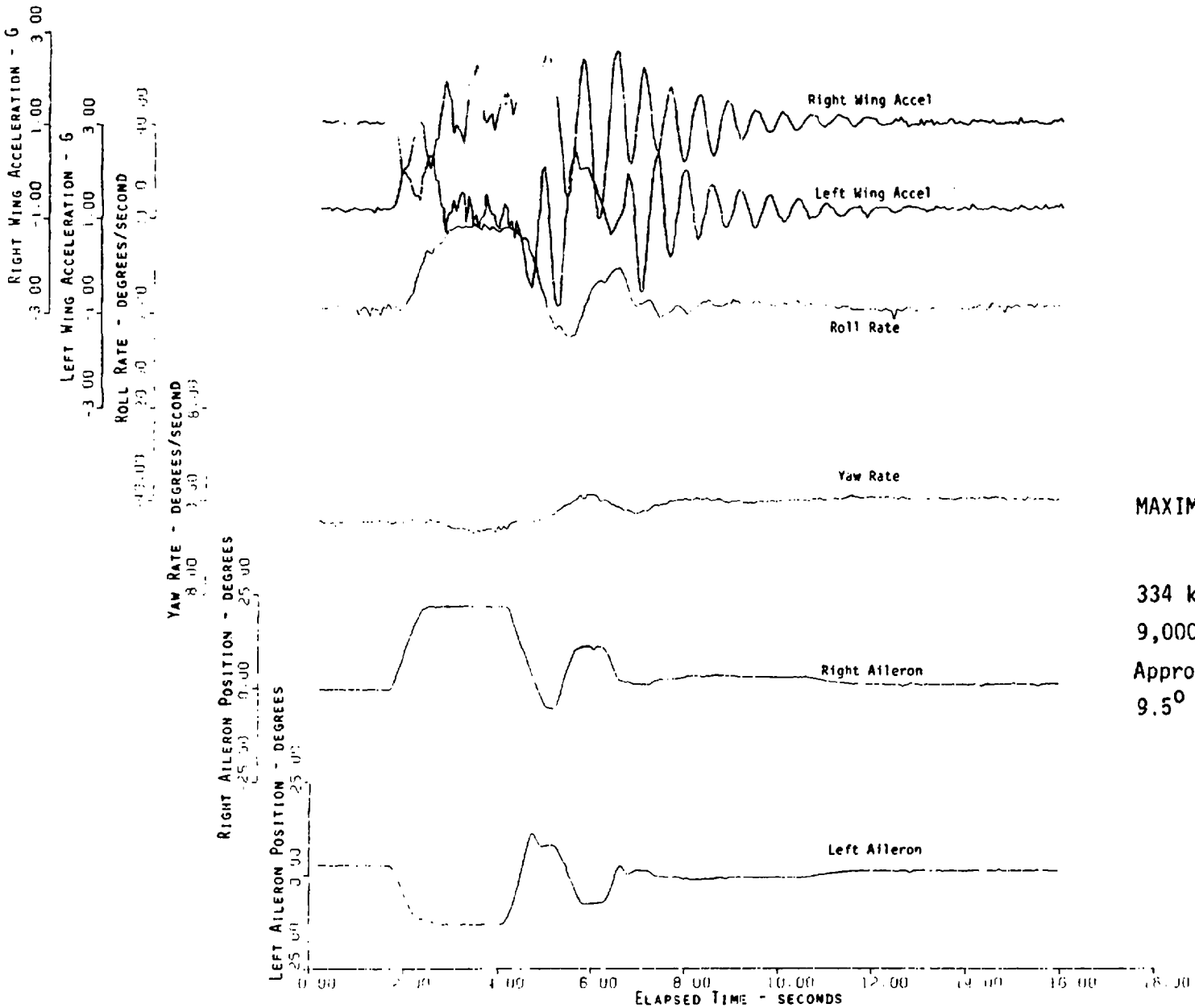


Figure C-1b
MAXIMUM DEFLECTION ROLL
From 45° Bank

334 kts Airspeed
9,000 ft Altitude
Approx. 180,000 lb G.W.
9.5° C OAT

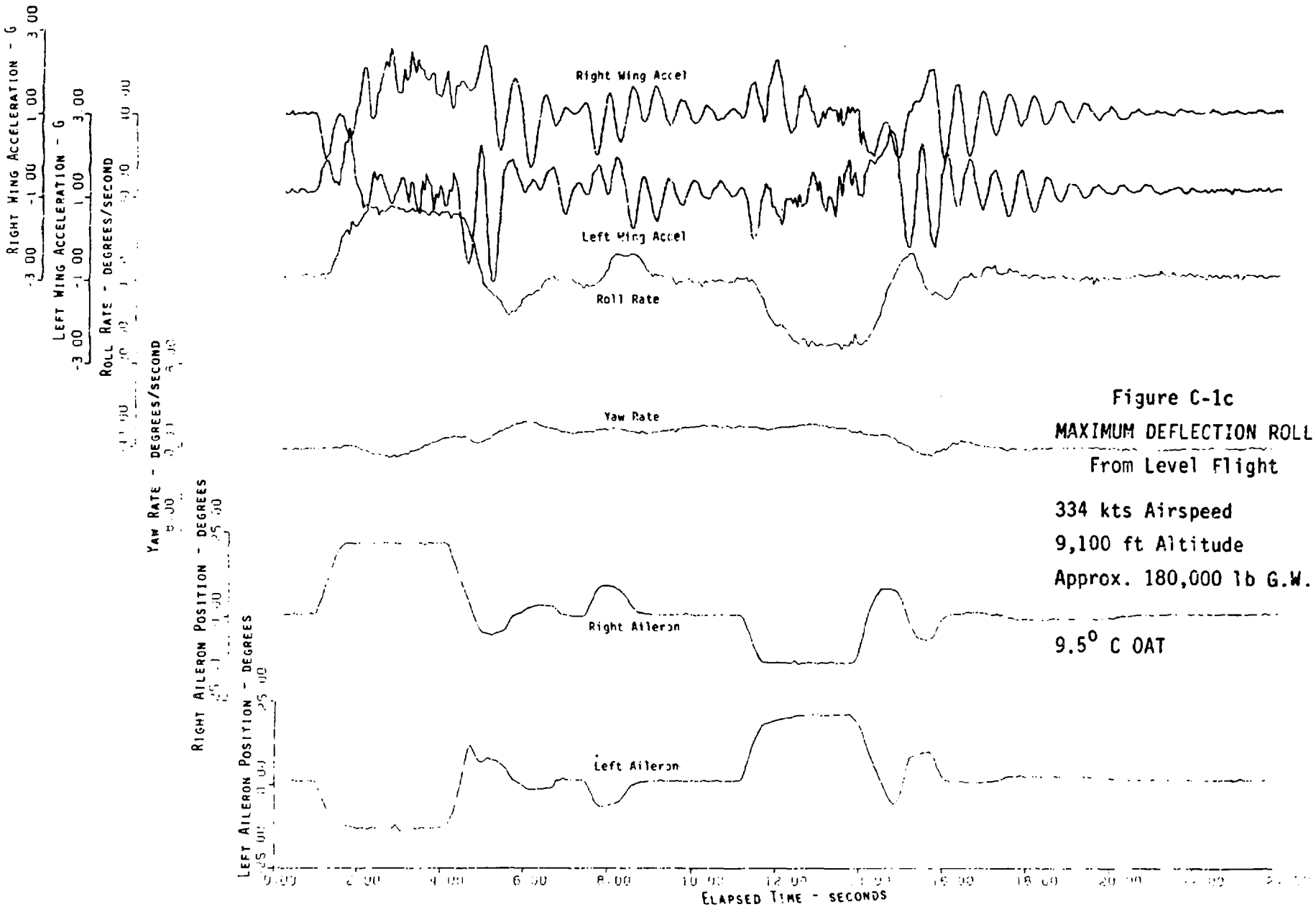


Figure C-1c
MAXIMUM DEFLECTION ROLL
From Level Flight

334 kts Airspeed
9,100 ft Altitude
Approx. 180,000 lb G.W.

9.5° C OAT

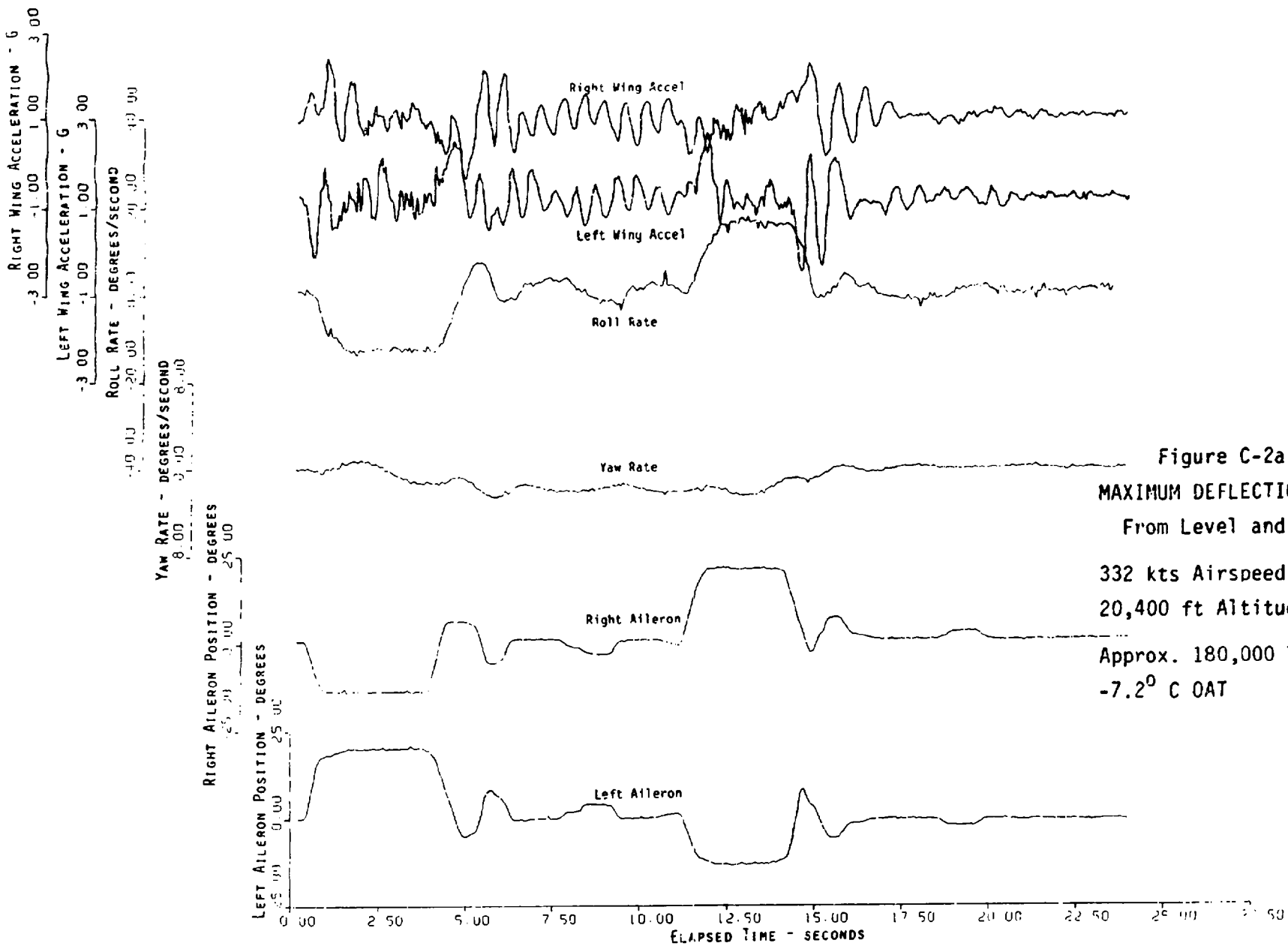


Figure C-2a
MAXIMUM DEFLECTION ROLL
From Level and Back

332 kts Airspeed
20,400 ft Altitude

Approx. 180,000 lb G.W.
-7.2° C OAT

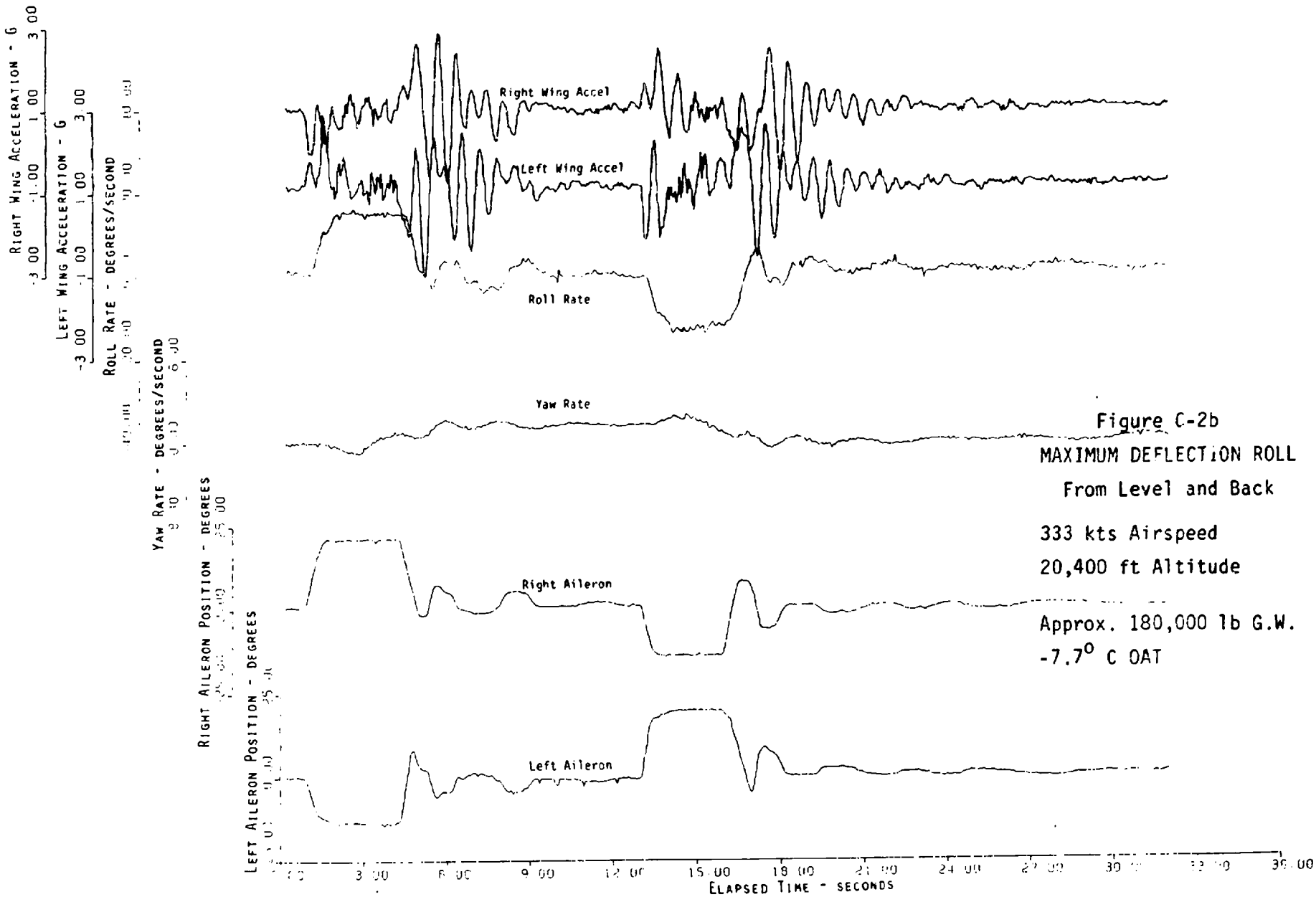


Figure C-2b
MAXIMUM DEFLECTION ROLL
From Level and Back

333 kts Airspeed
20,400 ft Altitude

Approx. 180,000 lb G.W.
-7.7° C OAT

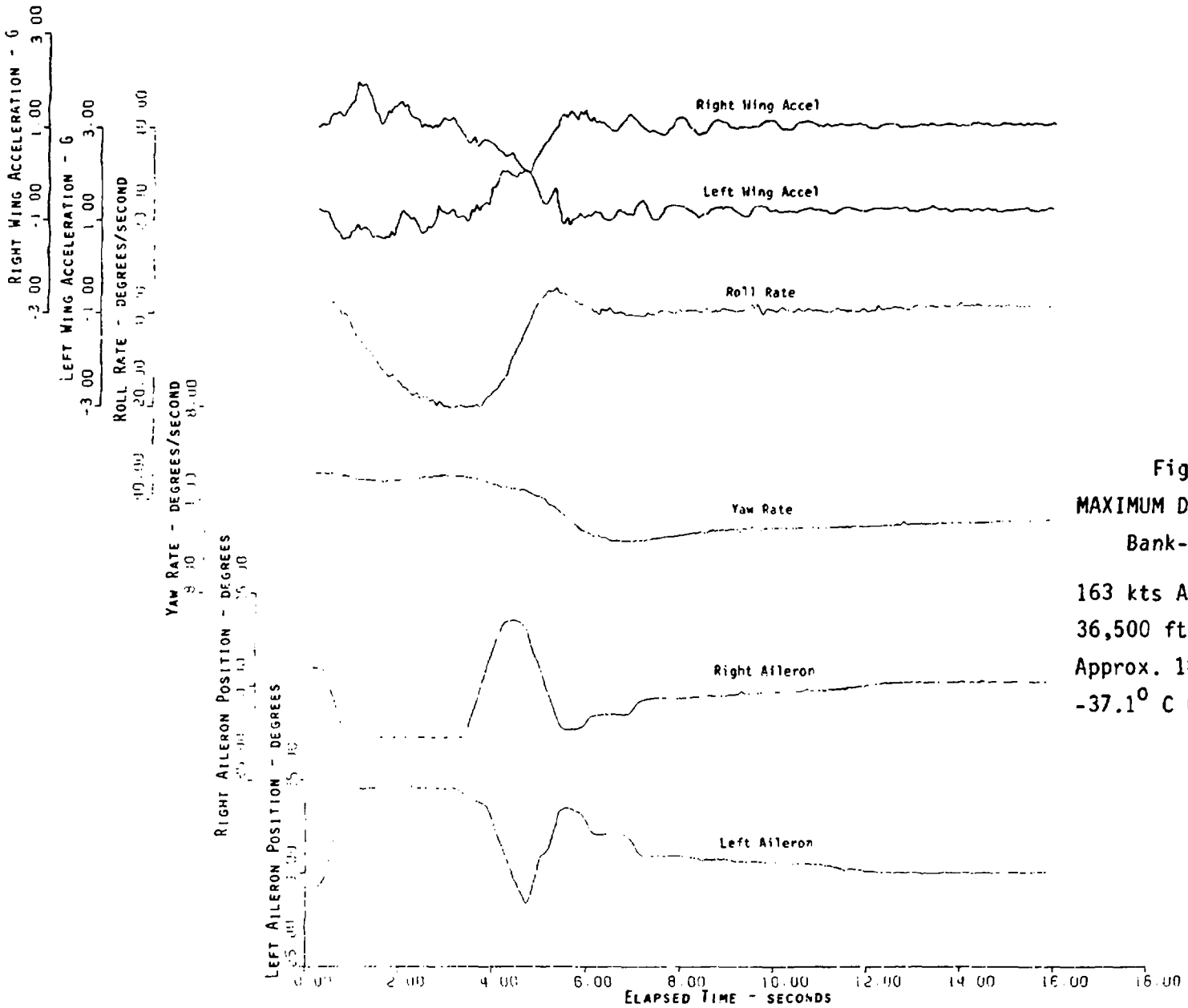


Figure C-3a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

163 kts Airspeed
36,500 ft Altitude
Approx. 180,000 lb G.W.
-37.1° C OAT

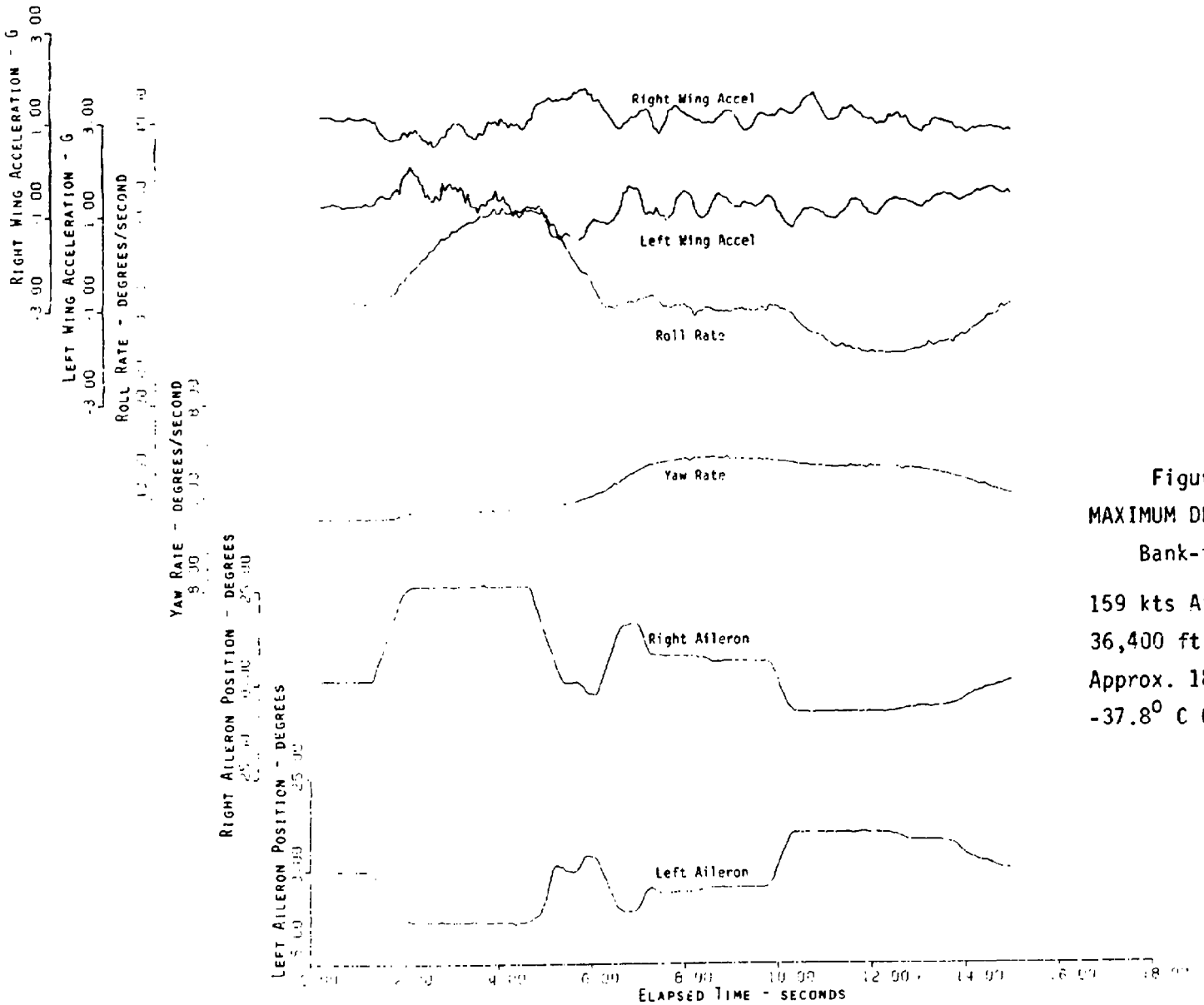


Figure C-4a
MAXIMUM DEFLECTION ROLL
Bank-to-Bank

159 kts Airspeed
36,400 ft Altitude
Approx. 180,000 lb G.W.
-37.8° C OAT

APPENDIX D

ACTUATOR TECHNICAL DATA

COMPARISON OF EMAS AND HYDRAULIC PCU

	EMAS	PCU
Maximum Output Force	19,050 lbs	19,050 lbs
No Load Rate	4.65 in/sec	4.65 in/sec
Stroke *	+3.35 in, -2.08 in	+3.35 in, -2.08 in
Frequency Response	4Hz (1st order)	4Hz (1st order)
Maximum Freeplay	0.018 in (approx. 0.13 deg)	0.095 in
Weight	65 lbs	58 lbs
Reliability	448,632 hrs	115,004 hrs
Stiffness	6.0×10^5 lb/in	5.6×10^5 lb/in

* full mechanical stroke, actual stroke for maximum aileron deflection less

Additional EMAS Information

Motor No Load Speed	-	9,600 rpm
Maximum Power Required	-	8,000 Watts at 115 VAC
Reliability	-	4,000 cycles to single channel failure 5,000,000 cycles to dual channel failure
Limit Load	-	23,300 lbs
Ultimate Load	-	35,000 lbs

power-off backdrive at 0.12 in/sec with less than 360 lbs

(all information from contractor estimated or laboratory test data)

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