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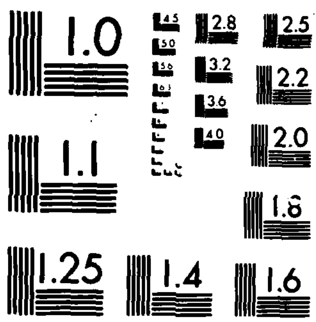

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USE OF STATE ESTIMATION TO CALCULATE ANGLE-OF-ATTACK POSITION ERROR FROM FLIGHT TEST DATA

THESIS

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Captain, USAF

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THESIS

Presented to the Faculty of the School of Engineering

of the Air Force Institute of Technology

Air University

In Partial Fulfillment of the

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Master of Science in Aeronautical Engineering

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Preface

The purpose of this project was to determine the position errors of the angle-of-attack (AOA) sensors on aircraft using state estimation with flight test data. Aircraft from the USAF Test Pilot School (TPS) were used to obtain flight test data, and Kalman filtering was used to process the data. The results of this project are significant to future flight test projects where an accurate AOA measurement is required.

Aircraft AOA position errors are caused by aerodynamic factors such as local flow and upwash. The first step in finding those errors was to determine the equations for calculating the true AOA from other available flight test parameters. Since the inputs to those equations were from instrumentation on flight test aircraft, they were noise corrupted and had to be filtered. I used state estimation in a Kalman filter program to calculate an "optimal" true AOA. The data were obtained from flights in a T-38A Talon, a two-seat supersonic trainer modified with an instrumented Vought yaw and pitch system noseboom. The position errors calculated in this report are only good for that aircraft and nose boom configuration. However, the methods used are applicable to all properly instrumented aircraft.

I would like to thank my thesis advisors, Major (Dr.) James T. Silverthorn of the USAF TPS and Dr. Robert A. Calico of AFIT, for their help in this project. I would also like to thank the test pilots I flew with on the data flights, Major Philip B. Arnold and Captain David J. Eichhorn of the USAF TPS. But most of all I would like to thank my wife, Diana, for her help and understanding over the last two years of AFIT and TPS.

Table of Contents

	Page

List of Figures

Figure		Page

List of Tables

List of Symbols and Abbreviations

Symbol	Title	Units
AOA	angle-of-attack	degrees
a_z	normal acceleration	feet/sec^2
cg	center of gravity	---
DAS	data acquisition system	---
deg	degrees	degrees
$\overline{F}$	applied force	pounds
F_z	component of force in z direction	pounds
g	acceleration due to gravity	feet/sec^2
$\dot{h}$	vertical velocity	feet/sec
H_c	pressure altitude	feet
hz	hertz	cycles/sec
K_α	AOA position error correction factor	---
lbs	pounds	pounds
M	Mach number	---
m	mass	slugs
MAC	mean aerodynamic chord	feet
NIU	noseboom instrumentation unit	---
n_z	normal load factor	g
P, P_o, p	roll rate	radians/sec
Q, Q_o, q	pitch rate	radians/sec
R	yaw rate	radians/sec
rms	root mean squared	---
sec	seconds	seconds
S/N	serial number	---

Symbol	Title	Units
~~U, U_o~~	~~true airspeed~~	~~feet/sec~~
U, U_o, u	component of vehicle velocity along x-axis	feet/sec
$\underline{u}(t)$, $\underline{u}(t_j)$	control input vector	---
V, V_o, v	component of vehicle velocity along y-axis	feet/sec
$\overline{V}_T$	vehicle velocity vector	feet/sec
$\underline{v}(t)$	measurement noise vector	---
W, W_o, w	component of vehicle velocity along z-axis	feet/sec
$\underline{w}(t)$	input noise vector	---
x_α	longitudinal distance from fixed point to AOA sensor	feet
$\underline{x}(t)$	state vector	---
$\hat{\underline{x}}(t)$, $\hat{\underline{x}}(t_j)$	estimated state vector	---
$\underline{x}(t_o)$	initial condition of state vector	---
XYZ	fixed earth axis	---
xyz	vehicle body axis	---
y_α	lateral distance from centerline to AOA sensor	feet
YAPS	yaw and pitch system	---
$\underline{z}(t)$, $\underline{z}(t_j)$	measurement history vector	---
α	angle-of-attack	degrees
α_c	true AOA	degrees
α_m	measured AOA at aircraft sensor	degrees
α_o	initial value of AOA	degrees
α_{o_T}	bias between true and measured AOA	degrees
$\Delta\alpha$	rate of change of AOA	deg/sec

Symbol	Title	Units
Δcg	longitudinal distance from cg to fixed point	feet
Δt	time interval	seconds
γ	flight path angle	degrees
$\bar{\omega}$	vehicle rotation vector	radians/sec
$\sigma^2_{a_z}$	variance of normal acceleration	$feet^2/sec^4$
σ^2_q	variance of pitch rate	deg^2/sec^2
σ^2_θ	variance of pitch angle	$degrees^2$
θ	pitch angle	degrees
θ_m	measured pitch angle	degrees
$(\dot{\ })$	time rate of change	---
$\tilde{q}$	strength of system noise	---
$\tilde{r}$	strength of measurement noise	---

Abstract

This project determined the position errors of an aircraft's angle-of-attack (AOA) sensor using state estimation with flight test data. The position errors were caused by local flow and upwash and were found to be a function of AOA and Mach number. The test aircraft used in this project was a T-38A Talon supersonic trainer from the USAF Test Pilot School configured with a Vought yaw and pitch system noseboom and an internal Aydin-Vector data acquisition system (DAS).

The position errors were found by calculating the true AOA using equations of motion and DAS parameters. The data from the DAS were noise corrupted and had to be filtered. This was accomplished using state estimation in a Kalman filter. The estimated AOA was compared to the measured AOA from the noseboom sensor to obtain the position error. Accurate position errors were obtained, even in dynamic maneuvers. The method was accurate enough to identify a hysteresis error in the T-38A's AOA sensor of +/- 0.5 degrees, which was confirmed by ground calibration. This method should be considered in future AOA error testing.

USE OF STATE ESTIMATION TO CALCULATE ANGLE-OF-ATTACK POSITION ERROR FROM FLIGHT TEST DATA

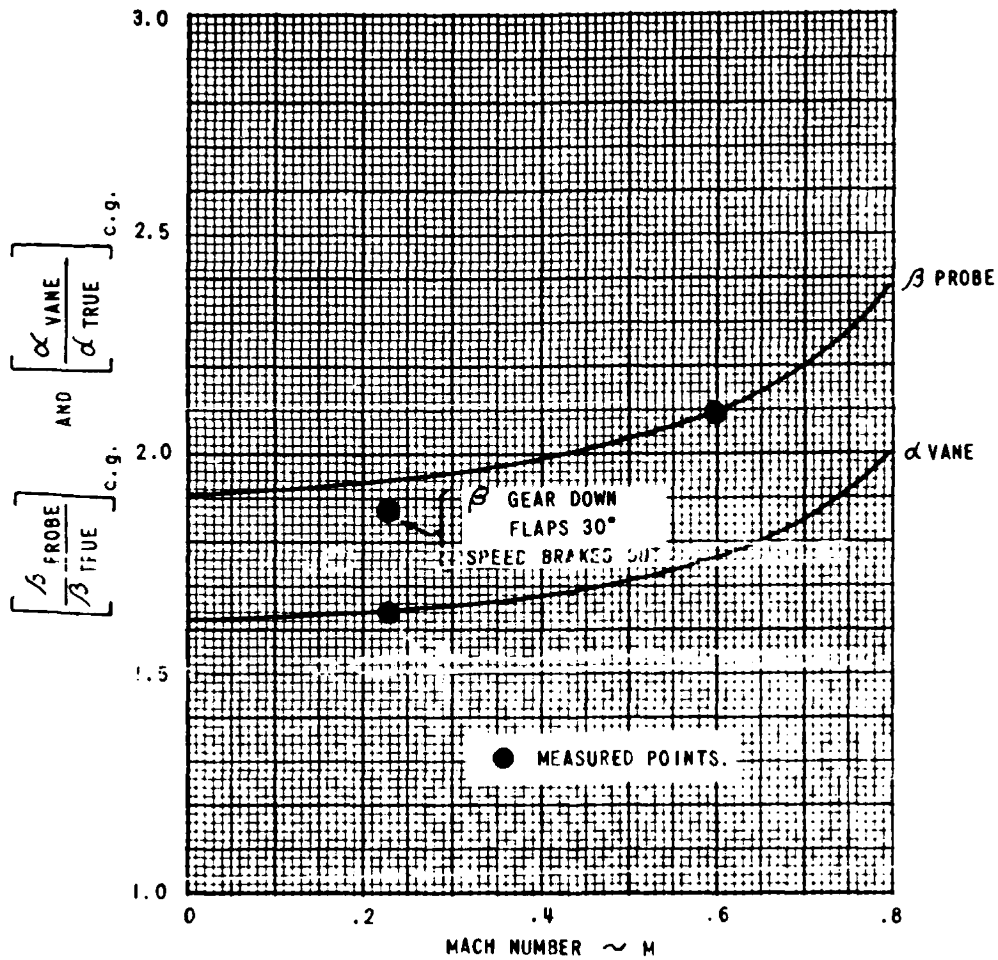
I. Introduction

Problem

Angle-of-attack (AOA) is a primary parameter of performance and stability-and-control in flight test. Unfortunately, the AOA measured by the aircraft sensors has a position error caused by the aerodynamic influence of the aircraft body. The first source of this position error is local flow about the AOA sensor caused by aerodynamic interference, boundary layer effects, and shock interaction. The second source of the position error is upwash from aircraft components such as the fuselage and wing. The accurate determination of AOA position error is a significant problem in flight test (1:7).

Background

The magnitude of the AOA position error is evident with the USAF/CAL variable stability NT-33 airplane, a jet trainer used in the USAF Test Pilot School (TPS) curriculum. The NT-33 has a fuselage-mounted AOA vane which is subject to large flow and upwash effects. Figure 1 shows the NT-33 AOA (and sideslip, which has similar errors) position error correction factors (2:163). At Mach 0.6, the NT-33 has an AOA position error correction factor of 1.75, which means the measured AOA is 1.75 times the true AOA. The AOA position errors were determined by combining wind tunnel and flight test data. One data point was determined, and a line was extrapolated over a range of Mach numbers. AOA position error was assumed to be a function of Mach number alone (2:162-163).



ESTIMATED MACH NUMBER VARIATION CORRESPONDS TO FLOW AROUND AN ELLIPSOID OF REVOLUTION REF: SHAPIRO, COMPRESSIBLE FLOW, VOL I. PG 399

Figure 1. USAF/CAL NT-33 Angle-of-Attack and Sideslip Position Error Correction Factors (2:163)

Wind tunnel calibration is a commonly used method of determining AOA position error at the Air Force Flight Test Center (AFFTC). An entire noseboom instrumentation unit (NBIU) can be installed in a wind tunnel and tested over a range of conditions. One of the AFFTC standard NBIUs, a Conrac adapter with a Rosemount Model 852G pitot-static probe, was tested in the NASA/Ames Research Center wind tunnels in 1973 (3). The AOA position error was found to be small, less than 7.5% (3:38). The wind tunnel test showed AOA position error to be a function of Mach number and sideslip angle. Reynold's number effects were not discovered.

Very little flight testing has been accomplished to determine AOA position error. One technique that has been used is to mount flight path accelerometers on the test aircraft and fly 1 g, wings level stable points over a range of Mach numbers, sideslips, and aircraft weights. True AOA (α) is determined from the equation:

$$\boxed{\alpha = \theta - \gamma} \tag{1}$$

where θ is pitch angle in wings level flight and γ is the flight path angle. The T-46 jet trainer Combined Test Force is planning to use this technique to calibrate their AOA sensors when flight testing begins in October 1985. They plan to eliminate noise in the data by using a 2 hz low bypass Butterworth filter. Unfortunately, not all aircraft can be equipped with flight path accelerometers due to the size and required cost. Furthermore, the range of AOA that is attainable at a particular Mach number is very limited for straight and level flight, since altitude is the only variable that can be adjusted. As an example, the T-38A, at Mach 0.83, flies at + 2.5 degrees AOA at 25,000 feet and at + 1 degrees AOA at 15,000 feet. A technique to obtain AOA position error during dynamic maneuvers is required.

A new flight test method of determining AOA position error is through the use of MMLE3, a modified maximum likelihood estimation program (4) MMLE3 uses the aircraft mathematical model with estimated stability and control (S&C) derivatives. Flight test maneuvers such as elevator doublets are flown, and MMLE3 tries to match the time history of the maneuver with the time history of the math model by changing the estimated S&C derivatives. MMLE3 also calculates an AOA position error factor for the maneuver (4:3). MMLE3 is not extremely accurate and requires numerous flight test maneuvers to increase its accuracy. An easier and more accurate technique is needed to calibrate AOA sensors.

Scope

The purpose of this project was to determine the position errors on the AOA sensors of aircraft using flight test data available from standard data acquisition systems (DAS). Initially, AOA position error was determined using deterministic equations from straight and level flight.

Problems with this technique suggested a more general approach. State estimation in the form of Kalman filtering was used to filter out noise on the flight test data and calculate an "optimal" true AOA during dynamic maneuvers. This true AOA was compared to the measured AOA to determine the position errors. USAF TPS T-38A aircraft were used to collect data and the AOA position errors are valid for those aircraft. However, the technique will work for any properly instrumented aircraft.

Objectives

The objectives of this project were to:

(1) Determine the equations necessary to calculate the true AOA from flight test data.

(2) Use state estimation (Kalman filtering) to filter noise from the flight test data and calculate an "optimal" true AOA.

(3) Collect the flight test data needed to compute the "optimal" true AOA.

(4) Calculate the AOA position error correction factors for the test aircraft.

II. Angle-of-Attack Equations

AOA Correction Factor

The AOA position error correction factor, K_α, is calculated from the equation (4:3):

Pitch rate Roll Rate

$$\alpha_c = \frac{\alpha_m}{K_\alpha} + \frac{q\,(x_\alpha + \Delta cg)}{U} - \frac{p\,(y_\alpha)}{U} \qquad (2)$$

where α_c is the true AOA of the aircraft and α_m is the AOA measured by the aircraft's sensor.

The term:

$$\frac{q\,(x_\alpha + \Delta cg)}{U} \qquad (3)$$

corrects the measured AOA for pitch rate (q) effects. The terms x_α and Δcg account for the longitudinal distance from the cg to the AOA sensor. U is aircraft true airspeed.

The term:

$$\frac{p\,(y_\alpha)}{U} \qquad (4)$$

corrects the AOA for roll rate (p) effects. The term y_α is the lateral distance from the aircraft centerline to the AOA sensor.

All flight testing for this project was accomplished wings level. Since there was no roll rate, equation (4) drops out of equation (2). Pitch rate, true airspeed, and measured AOA are parameters measured by the aircraft DAS. Longitudinal distance from the cg to the AOA sensor is a function of aircraft fuel weight and is easily calculated. The only remaining unknown is true AOA.

Equations of Motion

In order to calculate the true AOA, an equation was needed that used parameters available from the aircraft DAS. Equation (1) showed the angular relationship between AOA, pitch angle, and flight path angle in wings level flight. Since many test aircraft, including the TPS aircraft used in this project, do not have flight path accelerometers, flight path angle (γ) must be calculated by:

$$\sin \gamma = \frac{\dot{h}}{U} \qquad \dot{h} = VV \quad U = TAS \tag{5}$$

where $\dot{h}$ is the vertical velocity of the aircraft. Vertical velocity can be calculated as the time rate of change of the altitude from the DAS.

Another equation to calculate true AOA comes from the aircraft's equations of motion (5:3.21-3.51). The vector equation for applied force (F) is:

$$\bar{F} = m \left. \frac{d\bar{V}}{dt} \right|_{XYZ} \tag{6}$$

which applies to inertial space. Assuming the forces resulting from the earth's rotation and coriolis effects to be negligible, a fixed earth axis system can be used instead of inertial space. The movement of a vehicle with respect to a fixed earth axis is shown in Figure 2.

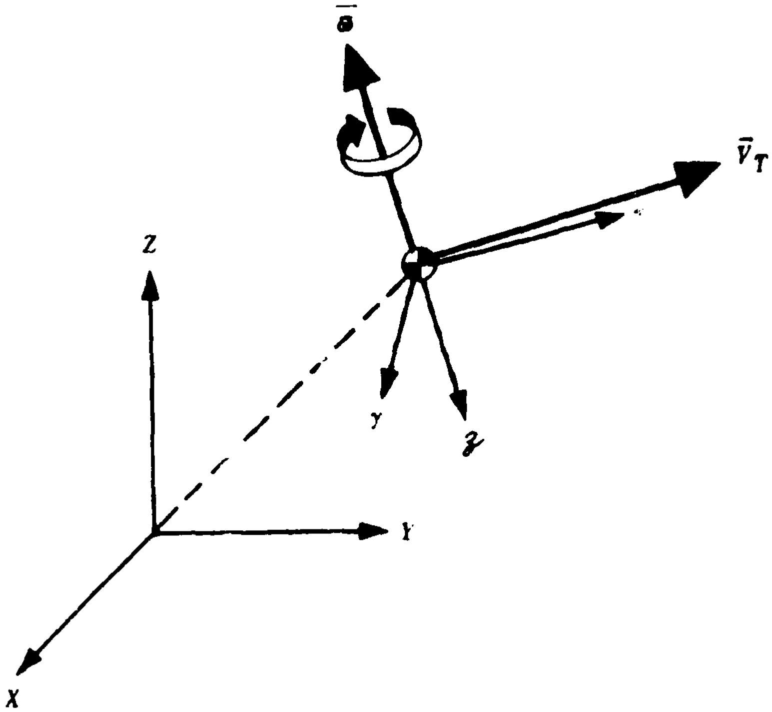


Figure 2. Relationship of Fixed Earth Axis (XYZ) to Vehicle Body Axis (xyz) (5:3.22)

The vector equation for the time rate of change of velocity from one axis system to another is:

$$\left.\frac{d\bar{V}_T}{dt}\right|_{XYZ} = \left.\frac{d\bar{V}_T}{dt}\right|_{xyz} + \bar{\omega} \times \bar{V}_T \qquad (7)$$

where XYZ is the fixed earth axis and xyz is the aircraft body axis.

Equation (6) now becomes:

$$\bar{F} = m \left[\left. \frac{d\bar{V}_T}{dt} \right|_{xyz} + \bar{\omega} \times \bar{V}_T \right] \tag{8}$$

where aircraft velocity ($\bar{V}_T$) can be written as:

$$\bar{V}_T = U\,\bar{i} + V\,\bar{j} + W\,\bar{k}$$

and aircraft rotation ($\bar{\omega}$) can be written:

$$\bar{\omega} = P\,\bar{i} + Q\,\bar{j} + R\,\bar{k}$$

Equation (8) now becomes:

$$\bar{F} = m \left[\dot{U}\bar{i} + \dot{V}\bar{j} + \dot{W}\bar{k} + \begin{vmatrix} \bar{i} & \bar{j} & \bar{k} \\ P & Q & R \\ U & V & W \end{vmatrix} \right] \tag{9}$$

Taking the cross product of the inner term and expanding:

$$\bar{F} = m \; [\dot{U}\bar{i} + \dot{V}\bar{j} + \dot{W}\bar{k} + (QW - RV)\bar{i} - (PW - RU)\bar{j} + (PV - QU)\bar{k}] \tag{10}$$

Looking at only the z component of force gives:

$$F_z = m\,(\dot{W} + PV - QU) = m\,(a_z) \tag{11}$$

where ($\dot{W} + PV - QU$) equals the normal acceleration, a_z.

Assuming that the aircraft motion consists of small deviations from an initial reference condition, the above values can be written as:

$$\dot{W} = \dot{W}_o + \dot{w}$$

$$P = P_o + p$$

$$V = V_o + v$$

$$Q = Q_o + q$$

$$U = U_o + u$$

where the small case values are the small perturbations from the initial values. Assuming the aircraft starts from wings level, steady straight symmetrical flight:

$$\dot{W}_o = P_o = V_o = Q_o = 0$$

Equation (11) now becomes:

$$m\,(\dot{w} + pv - qU) = m\,(a_z) \tag{12}$$

All testing during this project was done wings level, so roll rate (p) is zero. The change in velocity (u) is assumed to be small, so $U = U_o$. Dividing each side by $m(U_o)$ gives:

$$\frac{\dot{w}}{U_o} - q = \frac{a_z}{U_o} \tag{13}$$

Assuming small AOA gives the relationship:

$$\dot{\alpha} = \frac{\dot{w}}{U_o} \tag{14}$$

which can be substituted into equation (13) and rearranged to give:

$$\dot{\alpha} = q + \frac{a_z}{U_o} \tag{15}$$

where both pitch rate and normal acceleration are measured by the aircraft DAS. Assuming a finite time interval Δt, $\dot{\alpha} = \Delta\alpha$. An iterative equation can be formed where:

$$\alpha_{i+1} = \alpha_i + \Delta\alpha\,(\Delta t) \qquad (16)$$

Computer Program AOAOPT

A FORTRAN computer program was designed to calculate true AOA using both the angle relationship (equations (1) and (5)) and the iterative relationship (equations (15) and (16)). This program is called AOAOPT and is shown in Appendix B. The program reads the required flight test parameters from the DAS, makes necessary pitot-static corrections, calculates true AOA using both methods, and calculates the AOA position error correction factor (from equation (2)). The program works with flight test data from either USAF TPS T-38As or RF-4Cs. No data filtering is accomplished.

Results

Sample RF-4C flight test data from 1 g, wings level flight was processed by the program AOAOPT. Test data was sampled at the highest rate possible for the aircraft DAS, 8 times per second (a complete DAS description is in Chapter IV). Figure 3 is a plot of the AOA calculated using the angle relationship. The calculated AOA was very sensitive to noise in the altitude channel and was only accurate by averaging over a time span of 3 to 4 seconds in 1 g, wings level flight.

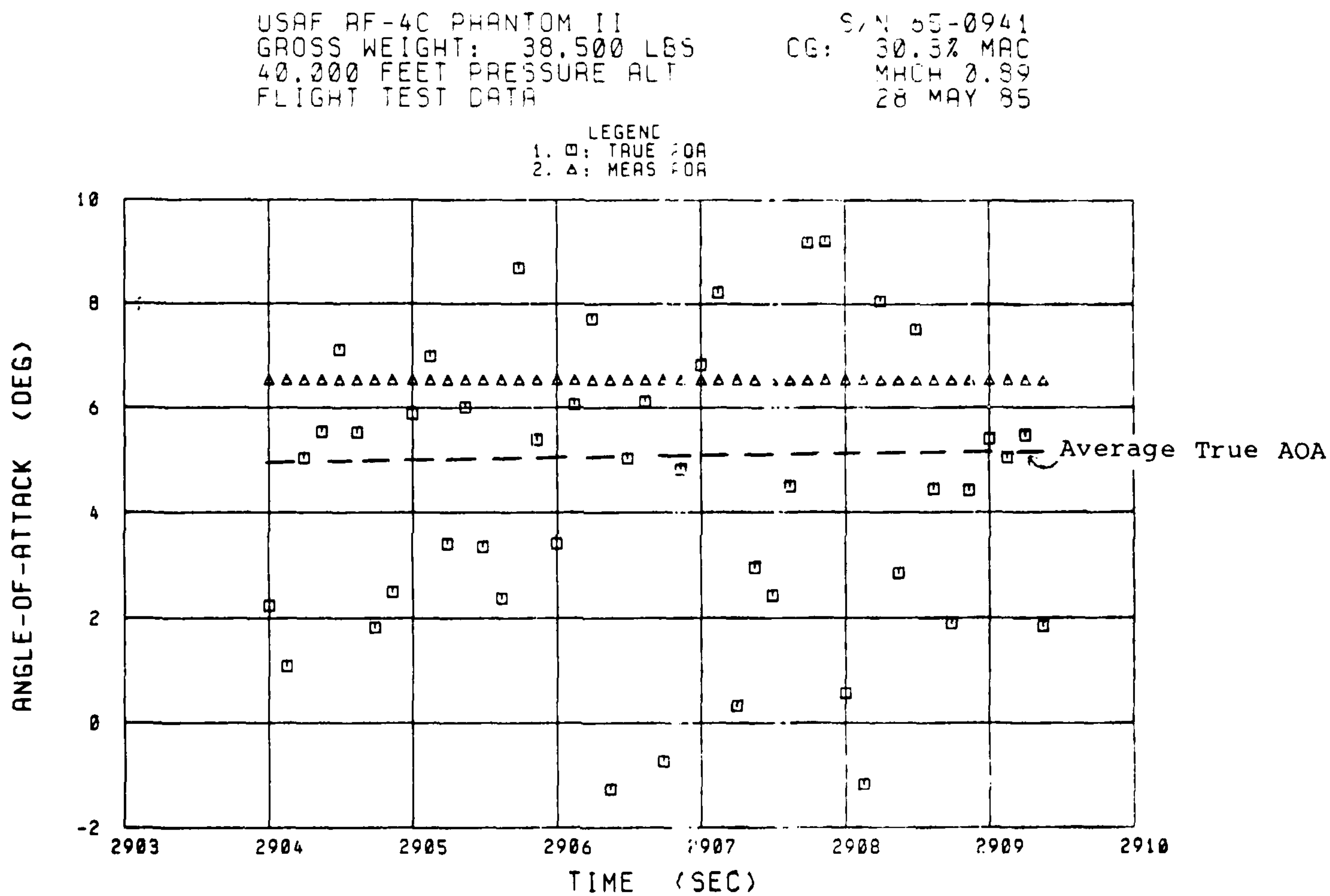


Figure 3. True AOA Calculated Using the Angle Method With Unfiltered Data Compared to Measured AOA From the Aircraft AOA Sensor

Figure 4 is a plot of the AOA calculated using the iterative method. The resulting AOA is less subject to noise, as the pitch rate and normal acceleration channels were fairly noise free. The data pitch rate and normal acceleration values were also corrected for bias measured while on the ground (bias was measured from 0 deg/sec pitch rate and 1 g normal acceleration). The major problem with the iterative method is its initial value, α_o, which must be calculated beforehand. The program AOAOPT uses as α_o the AOA calculated from the angle relationship averaged over a time interval of 3 to 4 seconds of 1 g flight. However, the AOA values are still corrupted by noise and are only as good as the resolution of the DAS. Some type of filtering is needed to optimize the true AOA.

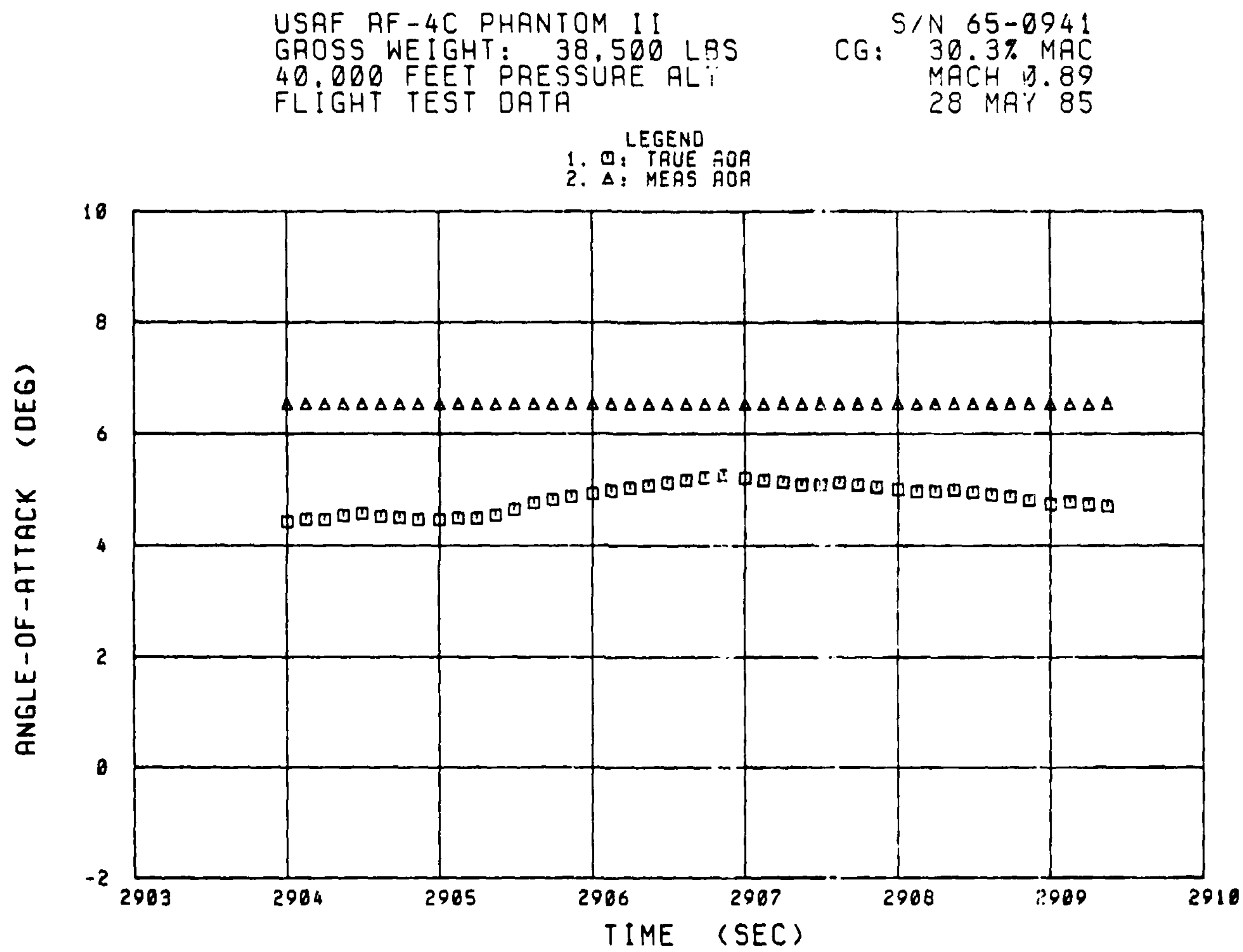


Figure 4. True AOA Calculated Using the Iterative Method With Unfiltered Data Compared to Measured AOA From the Aircraft AOA Sensor

III. State Estimation

State Equations

In order to use a digital computer to filter the flight test data and compute an "optimal" true AOA, the system dynamics need to be modelled (6:174). One way to model the system is with linear differential equations of the form:

$$\dot{\underline{x}}(t) = \underline{F}(t)\,\underline{x}(t) + \underline{B}(t)\,\underline{u}(t) \tag{17}$$

$$\underline{z}(t) = \underline{H}(t)\,\underline{x}(t) \tag{18}$$

where $\underline{x}(t)$ is the state vector, $\underline{u}(t)$ is the control input, and $\underline{z}(t)$ is the measurement history. One differential equation for angle-of-attack comes from equation (15):

$$\dot{\alpha} = q + \frac{a_z}{U_o} \tag{15}$$

Another equation to use in the state equations is the pitch rate equation valid for wings level flight:

$$\dot{\theta} = q \tag{19}$$

Since α and θ are not directly related, they become functions of the inputs q and a_z. The only useful parameter to measure is pitch angle, θ, since the measured AOA has an undetermined position error. Combining equations (15) and (19) together gives the state equations:

$$\begin{bmatrix} \dot{\alpha} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} 0 & 0 \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \alpha \\ \theta \end{bmatrix} + \begin{bmatrix} 1 & \frac{1}{U_o} \\ 1 & 0 \end{bmatrix} \begin{bmatrix} q \\ a_z \end{bmatrix} \tag{20}$$

The measurement equation is:

$$\begin{bmatrix} \theta_m \end{bmatrix} = [\,0 \quad 1\,] \begin{bmatrix} \alpha \\ \theta \end{bmatrix} \tag{21}$$

The above equations define the matrices $\underline{F}(t)$, $\underline{B}(t)$, and $\underline{H}(t)$. This is the math model to compute a true AOA, but nothing in the model filters the noise in the data.

Kalman Filtering

A Kalman filter provides the best method to "optimize" the true AOA from available flight test data. A Kalman filter will combine the pitch angle measurements, plus prior knowledge about the system and measuring devices, to produce an estimate of true AOA in such a manner that the error in true AOA is minimized statistically (7:5). The filter uses the state equations plus a statistical description of the system noises, measurement noises, and uncertainty in the dynamics model (7:4). The Kalman filter assumes that the system can be described by a linear model, and that system and measurement noises are white and Gaussian (7:7).

The original system model, equations (17) and (18), is augmented by (7:146):

$$\dot{\underline{x}}(t) = \underline{F}(t)\,\underline{x}(t) + \underline{B}(t)\,\underline{u}(t) + \underline{G}(t)\,\underline{w}(t) \tag{22}$$

$$\underline{z}(t) = \underline{H}(t)\,\underline{x}(t) + \underline{v}(t) \tag{23}$$

where the system is now driven by the input vector $\underline{u}(t)$ and noise vectors $\underline{w}(t)$ and $\underline{v}(t)$.

The vector $\underline{w}(t)$ models the system noise as white and Gaussian with mean of zero and strength $\tilde{q}$, described as (7:154-155):

$$E[\underline{w}(t)] = 0 \tag{24}$$

$$E[\underline{w}(t)\ \underline{w}^T(t+\tau)] = \tilde{q}\ \delta(\tau) \tag{25}$$

where $\tilde{q}$ is a measure of the uncertainty in the input vector $\underline{u}(t)$. The noise in the values from the aircraft DAS is assumed to be white since it is random and uncorrelated.

The vector $\underline{v}(t)$ models the measurement noise as white and Gaussian with mean zero and strength $\tilde{r}$, described as (7:174):

$$E[\underline{v}(t)] = 0 \tag{26}$$

$$E[\underline{v}(t)\ \underline{v}^T(t)] = \underline{R} \tag{27}$$

where $\underline{R}$ is a measure of the uncertainty in the measurement vector $\underline{z}(t)$.

In order for the Kalman filter to propogate the system, the estimated state vector (denoted by $\hat{\ }$) must be given an initial condition, $\underline{x}(t_o)$, where (6:233):

$$E[\underline{x}(t_o)] = \hat{\underline{x}}(t_o) \tag{28}$$

$$E[\ [\underline{x}(t_o) - \hat{\underline{x}}(t_o)][\underline{x}(t_o) - \hat{\underline{x}}(t_o)]^T] = \underline{P}(t_o) \tag{29}$$

The equations to propogate and update the optimal estimate using Kalman filtering are fully derived in <u>Stochastic Estimation and Control Systems</u> (6:210-233). Since the matrix $\underline{F}(t)$ is a zero matrix (equation (20)), the propogation equations from a measurement at time t_{i-1} to time t_i become:

$$\hat{\underline{x}}(t_i^-) = \hat{\underline{x}}(t_{i-1}^+) + \Delta t \; \underline{B}(t_i) \; \underline{u}(t_i) \tag{30}$$

$$\underline{P}(t_i^-) = \underline{P}(t_{i-1}^+) + \Delta t \; \underline{G}(t_i) \; \underline{Q} \; \underline{G}^T(t_i) \tag{31}$$

where $^-$ denotes prior to the update and $^+$ denotes after the update. The matrix $\underline{B}(t_i)$ is defined in equation (20). Matrix $\underline{G}(t_i)$ is set equal to $\underline{B}(t_i)$ so the noise in the input vector, $\underline{u}(t_i)$, is modeled by the values in matrix $\underline{Q}$. $\underline{Q}$ becomes a 2x2 matrix which contains the uncertainties in the inputs q and a_z. These two inputs are assumed independent, therefore $\underline{Q}$ becomes a diagonal matrix of the form:

$$\underline{Q} = \begin{bmatrix} \sigma^2_q & 0 \\ 0 & \sigma^2_{a_z} \end{bmatrix} \tag{32}$$

where the diagonal values are constant with time.

The update equations at measurement time t_i are:

$$\underline{K}(t_i) = \underline{P}(t_i^-) \; \underline{H}^T(t_i) \; [\, \underline{H}(t_i) \; \underline{P}(t_i^-) \; \underline{H}^T(t_i) + \underline{R} \,]^{-1} \tag{33}$$

$$\hat{\underline{x}}(t_i^+) = \hat{\underline{x}}(t_i^-) + \underline{K}(t_i) \; [\, \underline{Z}(t_i) - \underline{H}(t_i) \; \hat{\underline{x}}(t_i^-) \,] \tag{34}$$

$$\underline{P}(t_i^+) = \underline{P}(t_i^-) - \underline{K}(t_i) \; \underline{H}(t_i) \; \underline{P}(t_i^-) \tag{35}$$

The matrix $\underline{H}(t_i)$ is defined in equation (21). $\underline{K}(t_i)$ is the gain matrix which specifies how much the measurement, $\underline{Z}(t_i)$, is weighted in the update. $\underline{R}$ is a 1x1 value modelling the noise in the measurement, $\underline{Z}(t_j)$, which is pitch angle. $\underline{R}$ is in the form:

$$\underline{R} = \begin{bmatrix} \sigma^2_\theta \end{bmatrix} \tag{36}$$

and is also constant in time.

The Kalman filter is ready to be put into a digital computer routine. Equations (30), (31), (33), (34), and (35) will propogate and update the system over time. The matrix $\underline{x}(t_j^+)$ contains the "optimal" values for α and θ, of which α is the "optimal" true AOA desired. Before the routine can be implemented, the Kalman filter must be "tuned" to determine the values for $\underline{P}(t_o)$, $\underline{Q}$, and $\underline{R}$.

Filter Tuning

The objective of filter tuning is to achieve the best possible estimation performance from a filter that is totally specified except for $\underline{P}(t_o)$, $\underline{Q}$, and $\underline{R}$. The covariance values in those matrices account for the actual noises and disturbances in the system and determine how adequately the model represents the real world system. The $\underline{P}(t_o)$ matrix determines the initial performance of the filter, and the $\underline{Q}$ and $\underline{R}$ matrices determine the long term performance (7:337).

The method of filter tuning used here is "covariance analysis" (7:337-339). The filter program is run with some assumed covariance values in the three matrices, $\underline{P}(t_o)$, $\underline{Q}$, and $\underline{R}$. The "true" root mean squared (rms) error, which is the error at each update between the filter's estimate and the actual measurement, is plotted over time. The time history of the computed rms error, or what the filter calculates as its error, is plotted with the true error.

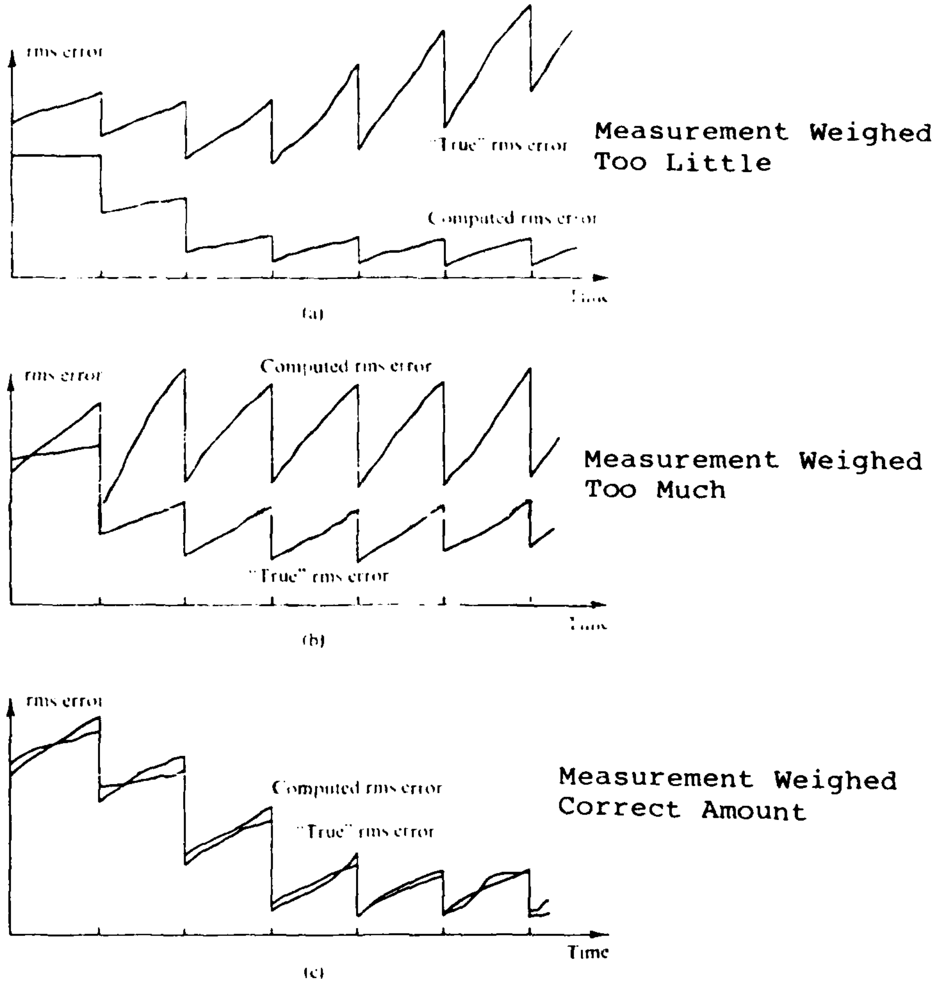


Figure 5. An Example of Kalman Filter Tuning Through Covariance Analysis (7:338)

Figure 5 is an example of what these plots show. Plot 5(a) shows a filter that has a low computed error and weighs the measurement too little. Plot 5(b) shows a filter that has too high a computed error and weighs the measurements too much. Plot 5(c) shows a filter that is just right - its computed error and true error are equal (7:339). For the Kalman filter used in this project, the values of $\underline{P}(t_o)$, $\underline{Q}$, and $\underline{R}$ were varied until the computed rms errors and true rms errors were about equal. The Kalman filter was complete and ready to filter test data.

Computer Program KALOPT

A FORTRAN computer program was designed to calculate an "optimal" true AOA using the Kalman filter equations. This program is called KALOPT and is shown in Appendix C. The program reads the required flight test parameters from the DAS, makes necessary pitot-static corrections, and calculates an "optimal" true AOA each iteration. The program works with flight test data from either USAF TPS T-38As or RF-4Cs.

The Aydin-Vector DAS does not read normal acceleration, a_z, but instead reads normal load factor, n_z. The sign of n_z is opposite from the standard body axis system: positive n_z is through the top of the canopy. Also, n_z includes acceleration due to gravity. The following equation, which assumes small pitch and roll angles, corrects n_z to a_z:

$$a_z = -(n_z - 1)\ 32.2 \qquad (37)$$

The Kalman filter needs to know the initial values for AOA and pitch angle to use as $\underline{x}(t_o)$. KALOPT uses the first

DAS value for pitch angle as θ_o. However, an initial AOA needs to be calculated since the DAS AOA values have the yet-to-be-determined position error. The value of α_o is calculated using the angle method (equations (1) and (5)) used in the program AOAOPT. It averages the unfiltered true AOAs over a 3 to 4 second period in 1 g flight to calculate α_o.

KALOPT calculated the computed and true rms errors after each iteration. The values were varied from 0.001 to 1.0 during the covariance analysis. Changing the values of $\underline{P}(t_o)$ changed the initial rms values, but had little effect on the overall results. When the $\underline{Q}$ values were increased, the measurement was weighted more; the true rms error was less than the computed rms error. Increasing the $\underline{R}$ value caused the measurement to be weighted less; the true rms error was greater than the computed rms error. These results agreed with the theory behind filter tuning. Based on the covariance analysis conducted using T-38A data, the following values caused the true and computed errors to be equal:

$$\underline{P}(t_o) = \begin{bmatrix} 0.100 & 0 \\ 0 & 0.030 \end{bmatrix}$$

$$\underline{Q} = \begin{bmatrix} 0.025 & 0 \\ 0 & 0.002 \end{bmatrix}$$

$$\underline{R} = \begin{bmatrix} 0.300 \end{bmatrix}$$

KALOPT reads in the covariance values from a separate file, so they can be easily changed without changing the program.

Results

The first attempt at Kalman filtering included another state equation formed by combining equations (1) and (5):

$$\dot{h} = (\theta - \alpha)\, U_o$$

The DAS altitude readout was used as a measurement along with pitch angle. Unfortunately, the noise of the altitude transducer in the T-38A DAS was too erratic and could not be modelled as Gaussian. Altitude was not used in KALOPT.

The program KALOPT processed the same RF-4C flight test data that was used in Chapter II. The test data was sampled 8 times per second. Figure 6 is a plot of the "optimal" true AOA calculated by KALOPT from that data. The measured AOA and the unfiltered true AOA calculated by AOAOPT are also shown. The "optimal" true AOA is quicker to return to a steady state value than the unfiltered true AOA. It is impossible to tell which AOA is more accurate as the actual true AOA is unknown.

A better way to see how the Kalman filter is working is to compare the measured pitch angle to the "optimal" pitch angle to see how well it filters over noise and resolution increments. Figure 7 is a plot of measured pitch angle and "optimal" pitch angle. The measured pitch angle only had a resolution of 0.7 degrees, and after two samples it immediately increased by that amount. The "optimal" pitch angle is a fairly smooth curve over the time span, which shows that the Kalman filter is working. The next step is to use flight test data to calculate the AOA position error.

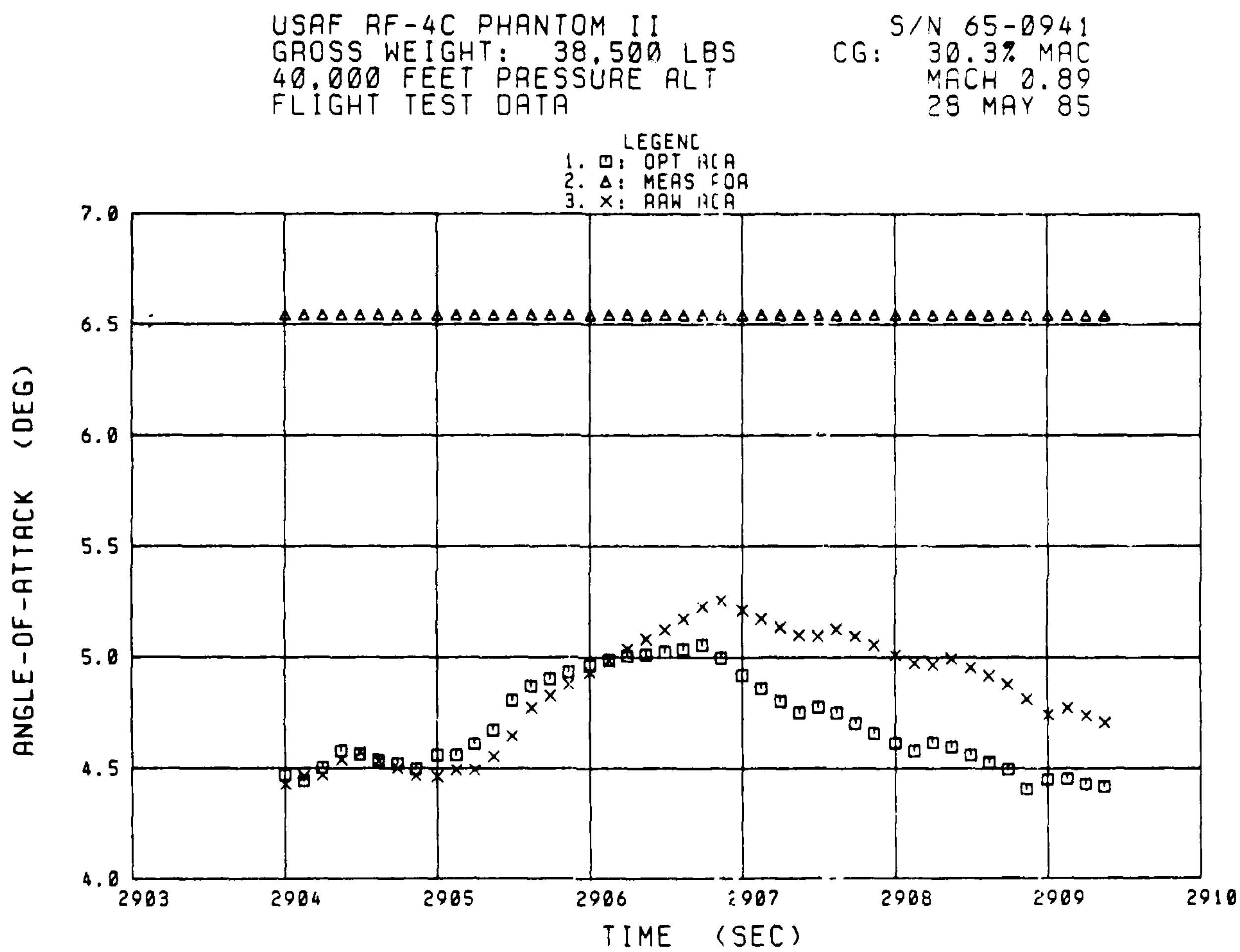


Figure 6. Optimal True AOA Calculated by the Kalman Filter Compared to Measured AOA and Unfiltered True AOA

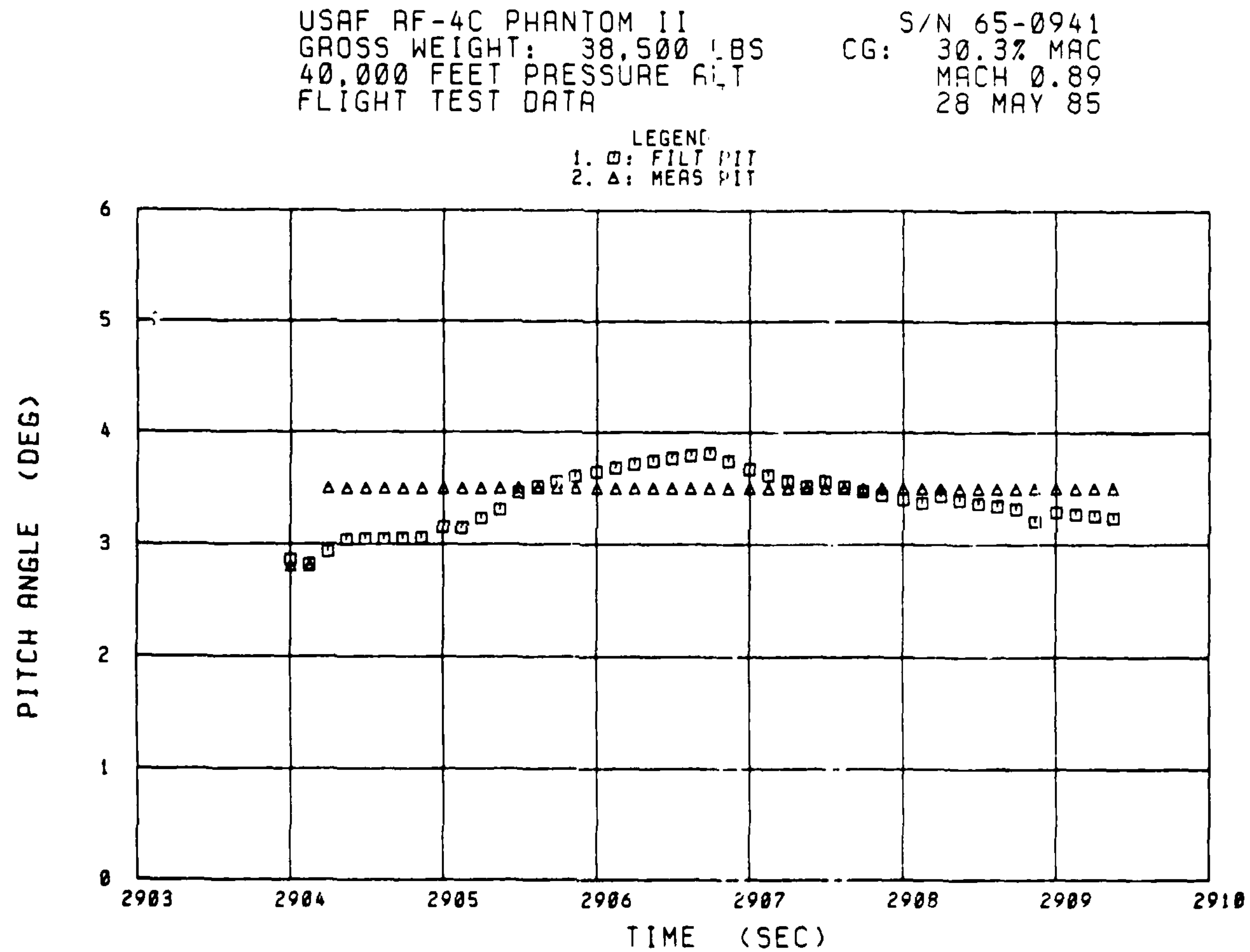


Figure 7. Optimal Pitch Angle Calculated by the Kalman Filter Compared to Measured Pitch Angle

IV. Flight Test

Test Item Description

The test aircraft was a USAF TPS T-38A Talon. The T-38A is a two place (tandem) jet trainer which is used extensively in the TPS curriculum. The aircraft is powered by two J85-GE-5 turbojet engines which give it a maximum capability of Mach 1.2 in level flight (8:6-6). Figure 8 is a photograph of a T-38A used at the TPS. A single T-38A, serial number (S/N) 68-8205, was used for all data flights in this project. Statistics on that airplane are shown in Table I. An important measurement is the distance from the cg of the aircraft to the AOA measuring vane, 25 feet. This length (x_α)

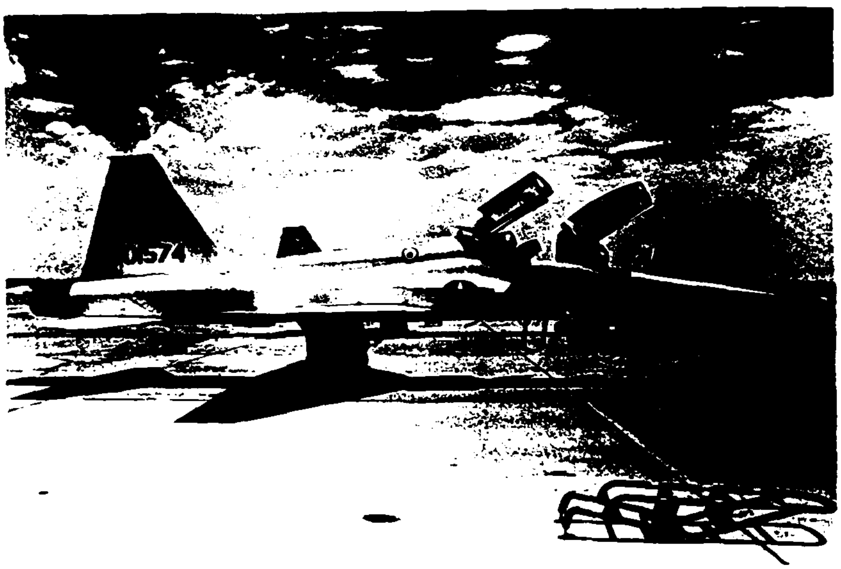


Figure 8. USAF T-38A Talon

is used to correct the true AOA for pitch rate (see equation (3)). The cg of the T-38A only shifts 0.3% mean aerodynamic chord (MAC) while consuming fuel, which is only 0.25 inches. Therefore, the Δcg term from equation (3) can be neglected.

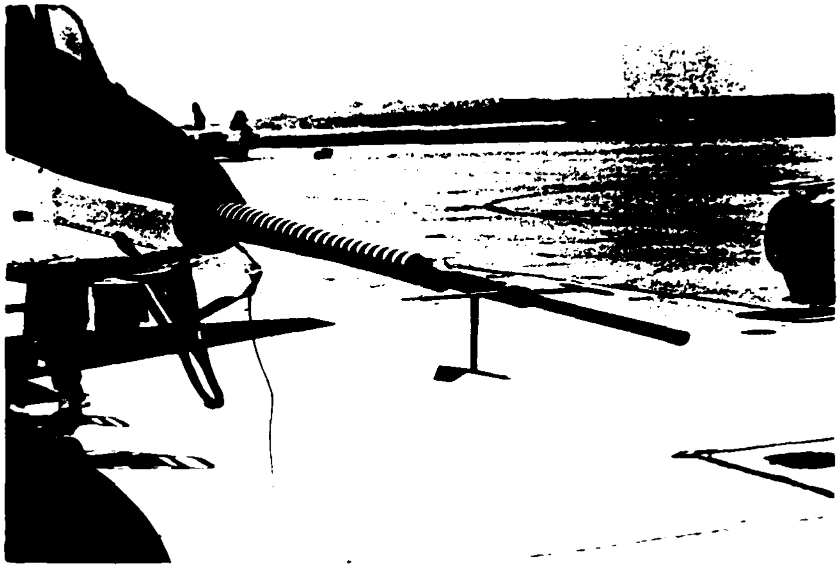
TABLE I

USAF T-38A Talon Statistics (S/N 68-8205)[1]

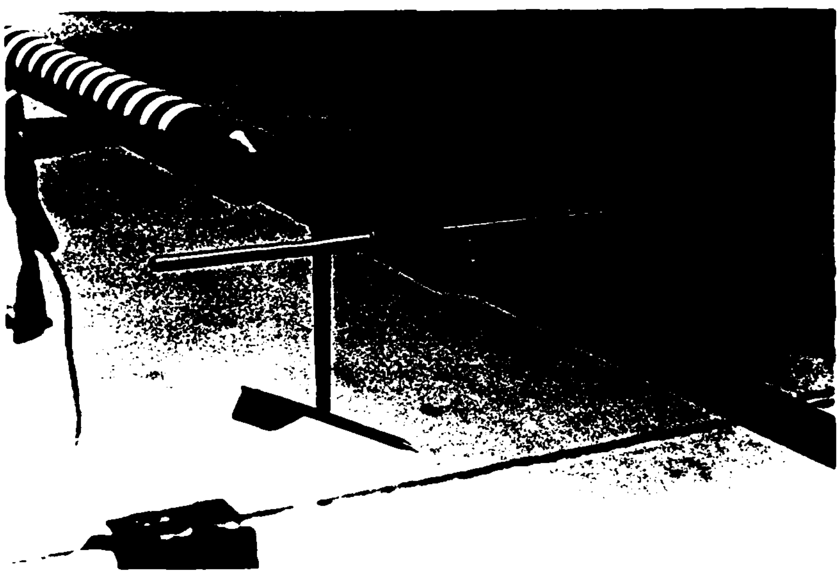
Engines:	Two J85-GE-5 Turbojets	
Dimensions:		
Length:	46 ft	4 in
CG to AOA Vane:	25 ft	0 in
Wingspan:	25 ft	3 in
MAC:	7 ft	0 in
Height:	12 ft	11 in
Weights:		
Operating Weight:	8,533 lbs	
Fuel (JP-4) Weight:	3,790 lbs	
Takeoff Gross Weight:	12,323 lbs	
Center of Gravity Movement:		
CG w/ 3,790 lbs fuel:	18.4% MAC	
CG w/ 400 lbs fuel:	18.1% MAC	

[1](8:1-1)

The test T-38A, S/N 68-8205, was modified for flight testing. The most important modification concerning this project is a fully instrumented yaw and pitch system (YAPS) noseboom (9:A.1), shown in Figures 9(a) and 9(b). A complete diagram of the YAPS noseboom is in Appendix E. The YAPS noseboom has two vane-type sensors, one for AOA and one for sideslip angle. These vanes are in front of the fuselage,



(a)



(b)

Figure 9. T-38A Yaw and Pitch System Noseboom

away from the aerodynamic influence of the aircraft, so the AOA (and sideslip) position errors should be less than for fuselage mounted sensors.

The YAPS noseboom on the T-38A is made out of aluminum alloy. It has been structurally tested up to 8.3 g and only minimal bending resulted (9:D.43-D.53). As a result, bending was ignored during this evaluation. The YAPS noseboom is canted 4 degrees down from the aircraft centerline.

Instrumentation

An internal Aydin-Vector SAU-537 DAS was installed in the aircraft to measure flight test parameters. The following components of the DAS were used in this project: a vertical gyro installed in the nose section to measure pitch and roll angles; a three axis rate gyro installed in the nose section to measure pitch, roll, and yaw rates; a three axis accelerometer installed in the center fuselage (at the nominal cg location) to measure acceleration in the x, y, and z axes (10:1.1-1.8). Other instruments were installed in the aircraft for flight test, but they were not used in this project.

The test aircraft was equipped with an internal Conrac ATR-580T70 magnetic tape recorder in the aft cockpit to record the data parameters (9:A.1). Forty-eight data channels were recorded. Indicated airspeed and altitude were recorded with 16 bit precision, the other parameters for this project had 8 bit precision. Table II is a summary of the parameters used in this project with their maximum/minimum values, precision, and accuracy. The parameters were recorded 8 times per second.

TABLE II

Summary of Flight Test Parameters[1]

USAF T-38A S/N 68-8205
Aydin-Vector SAU-537 DAS

Parameter	Units	Min Value	Max Value	Resolution	Accuracy
Altitude	feet	0	65000	1.030	0.103
Airspeed	knots	0	1250	0.019	0.0019
AOA	degrees	-22	28	0.202	*0.101
Sideslip	degrees	-20	20	0.164	0.082
Pitch	degrees	-80	80	0.704	0.704
Roll	degrees	-180	180	1.408	2.816
Pitch Rate	deg/sec	-20	20	0.163	0.163
N_z	g	-3	6	0.037	0.0037

[1] (9:A.4)

* Actual accuracy +/- 0.5 degrees due to hysteresis

Before any flight test was performed, all of the DAS instruments were ground calibrated and their calibration files updated. All important instruments were found to be working correctly except for the AOA transducer. It had a large hysteresis problem due to wear on its internal gearing. This hysteresis is shown in Figure 10. There is a +/- 0.5 degree error in true AOA depending on whether the vane is moving up or down. The AOA transducer was designed in the 1960s, and no replacement parts are available. The worn gears could not be fixed or replaced. This hysteresis will have a large effect on the flight test data.

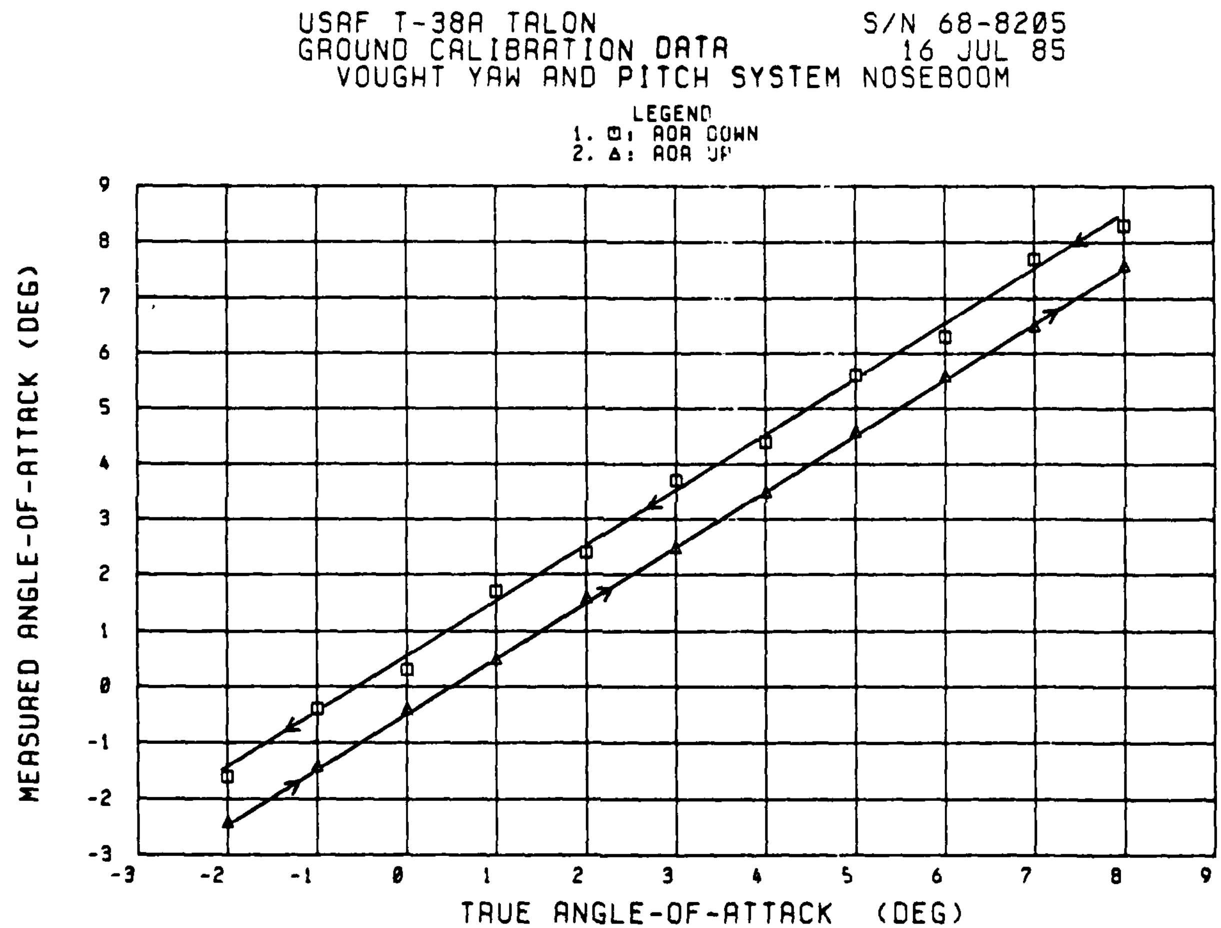


Figure 10. Hysteresis Error in T-38A Angle-of-Attack Transducer

Flight Test Method

The purpose of the flight testing was to gather data to calculate true AOA and to see what factors affected the AOA position error. AOA position error is primarily a function of AOA, therefore the testing covered large AOA changes. Other possible factors that were considered in designing the maneuvers were Mach number, Reynold's number, and sideslip angle. All testing was conducted wings level due to the assumptions used in the AOA equations (see Chapter II).

Since Reynold's number was a possible factor, testing was performed at different altitudes. Due to the altitude restrictions of available supersonic airspace, 25,000 feet and 15,000 feet were chosen for the testing. In order to see the effects of sideslip on AOA, the first maneuver to be performed was a wings level, slowly varying sideslip using maximum rudder deflection in both directions. Thrust was varied to maintain Mach number constant. This maneuver was performed at different Mach numbers. Actual data points are shown in Appendix D (11:3).

After the sideslip maneuver, a roller coaster maneuver was performed to vary AOA as much as possible. From a 1 g trim condition, the aircraft nose was pulled up slightly, then pushed forward to the minimum load factor specified for that data point. An onset rate of 3 seconds per g minimum was desired throughout the maneuver. At the minimum load factor, the aircraft nose was pulled back to the maximum load factor specified for that data point. The aircraft nose was then pushed forward to regain 1 g level flight. All data points and load factor limits are in Appendix D (11:3). Thrust was varied to maintain constant Mach number during the maneuver.

All testing was performed in the cruise configuration (gear and flaps up) with no external stores. All T-38A Flight Manual (8) limitations were complied with. Additional restrictions in the T-38A AOA Position Error Test Plan (11) were followed.

Test Results

Three T-38A test flights were flown at the USAF Flight Test Center, Edwards AFB, California. A summary of these flights is shown in Appendix D. No data were gathered on one flight due to bad weather. The same aircraft, 68-8205, was used on all three flights due to scheduling availability. Future test programs using this method should fly different tail numbers to prevent bias from one aircraft's own peculiarities.

The wings level sideslip maneuver was performed at all data points. The maximum sideslip angle generated was +/- 3.8 degrees at 25,000 feet pressure altitude (H_c), Mach 0.45. No change in AOA was found at this point or any of the others. In wind tunnel testing performed on a Conrac NBIU, a noseboom similar to the T-38A YAPS noseboom, no AOA position error change was discovered until five degrees of sideslip (3:38-39).

The roller coaster maneuver was performed at all data points, and repeated at the 25,000 feet H_c points. The data was reduced using the FORTRAN program KALOPT (see Chapter III). Measured AOA was plotted against the "optimal" true AOA (henceforth referred to as true AOA) at each point tested. These plots are Figures 12 - 29, Appendix A. The data points plot out fairly linear, which shows that the equations and Kalman filtering worked. Most of the test points flown up to 6 g were terminated at that point due to the Mach number decreasing outside tolerances (+/- 0.02 Mach desired). Also, at many of the high g points the data trace becomes erratic. This was due to aerodynamic buffet. Future test maneuvers for this method do not have to go to such high g limits, as the data collected at lower g limits is satisfactory.

As expected, the hysteresis error due to mechanical lag in the AOA gears was evident in the results. All eighteen plots show two lines of data, depending on whether the AOA vane was moving down or up. The error between the two lines ranges from +/- 0.5 to +/- 0.8 degrees, similar to the hysteresis error in Figure 10. In order to average the error, a straight line was drawn down the middle of the two lines. The slope of this line is the AOA position error correction factor, K_α (from equation (2)). The x-axis intercept, α_{o_T}, was also determined from these plots. These values are summarized in Table III. The Reynold's numbers were calculated using MAC (7 feet) as the constant length.

TABLE III

Summary of Flight Test Results

Date of Flight	Altitude (feet)	Mach	Reynolds # ($\times 10^7$)	K_α	α_{O_T} (deg)
23 Jul 85:	25,000	0.84	1.99	1.26	1.25
	25,000	0.94	2.23	1.38	0.60
	25,000	0.98	2.30	1.32	2.00
	25,000	1.07	2.51	1.48	2.30
	25,000	0.62	1.46	1.16	1.20
	25,000	0.44	1.03	1.16	1.20
23 Jul 85:	15,000	0.83	2.69	1.10	1.60
	15,000	0.94	3.03	1.14	3.20
	15,000	0.96	3.10	1.33	1.20
	15,000	1.07	3.44	1.45	0.90
	15,000	0.64	2.08	1.30	0.80
	15,000	0.42	1.35	1.20	1.90
26 Jul 85:	25,000	0.81	1.90	1.10	1.10
	25,000	0.92	2.16	1.20	2.00
	25,000	0.96	2.24	1.30	*-1.9
	25,000	1.06	2.47	1.36	2.20
	25,000	0.65	1.53	1.16	0.70
	25,000	0.44	1.03	1.20	0.50

* Exceeds 2 standard deviations from mean

The values for K_α are plotted versus Mach number in Figure 11. A curve was drawn through the points and shows a large increase in K_α as Mach number increases above 0.8. No apparent Reynold's number effects in K_α are evident in comparing the 15,000 feet points to the 25,000 feet points. The K_α values from the Conrac NBIU wind tunnel testing are also plotted in Figure 11 (3:38). Although the shapes of the curves are similar, the values for K_α are different. Since

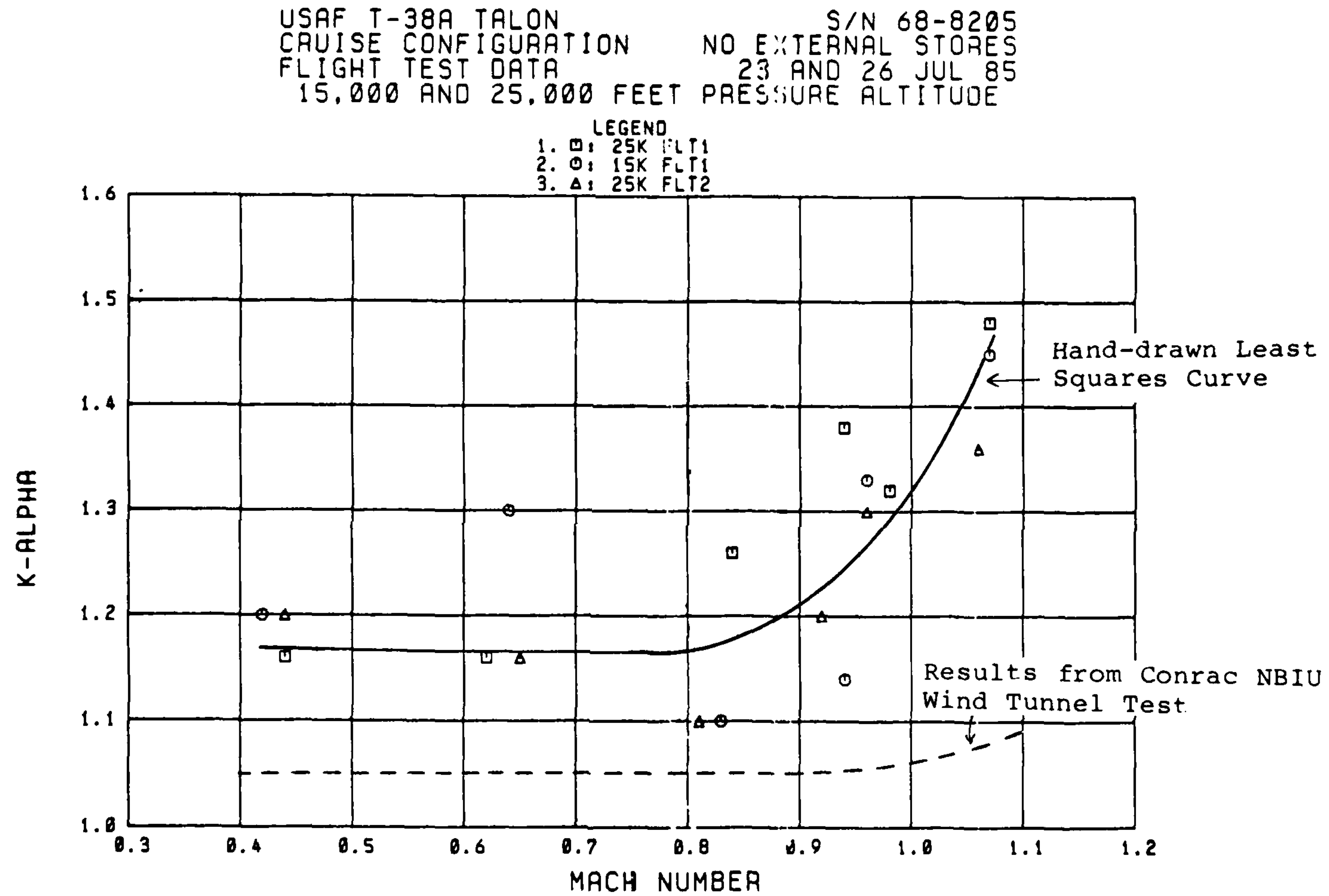


Figure 11. T-38A Angle-of-Attack Position Error Correction Factor for Vought YAPS Noseboom

both nosebooms are so similar in size and shape, the difference is mainly due to the lack of fuselage and wing effects on the wind tunnel results.

Equation (2), the equation to calculate true AOA, is:

$$\alpha_c = \frac{\alpha_m}{K_\alpha} + \frac{q\ (x_\alpha + \Delta cg)}{U} - \frac{p\ (y_\alpha)}{U} \qquad (2)$$

For T-38A 68-8205, $(x_\alpha + \Delta cg)$ is assumed a constant 25 feet. y_α is 6 inches (see Appendix E). The values for K_α are shown in Figure 11 as a function of Mach number. However, equation (2) assumes that no AOA position error exists at zero degrees AOA. According to Figures 12 to 29, this is not true for the T-38A. Some bias exists, which is the x-axis intercept, α_{o_T}. Adding this bias to equation (2), and neglecting pitch and roll rate, gives:

$$\alpha_c = \alpha_{o_T} + \frac{\alpha_m}{K_\alpha} \qquad (38)$$

The values for α_{o_T} from Figures 12 to 29 are shown in Table III. The values vary randomly, and do not seem to be functions of Mach number. The average of all 18 values is 1.26 degrees, with a standard deviation of 1.06 degrees. One value, for 25,000 feet H_c and Mach 0.96, is -1.9 degrees, which exceeds two standard deviations from the mean. Neglecting that point as erroneous, the average of the remaining values is 1.45 degrees, with a standard deviation of 0.73 degrees. The equation to solve for the true AOA of the T-38A with a Vought YAPS noseboom is:

$$\alpha_c = 1.45 + \frac{\alpha_m}{K_\alpha} \tag{39}$$

The pitch and roll rate terms from equation (2) should be included when applicable.

V. Conclusions and Recommendations

All project objectives were met. Conclusions and recommendations follow in order of importance:

The Kalman filter program KALOPT calculated "optimal" true angle-of-attack (AOA) values for a T-38A Talon using equations of motion for wings level flight and pitch angle measurements. From these true AOA values, the AOA position error correction factors were determined and were found to be functions of Mach number. Only standard T-38A flight test instrumentation was used - no flight path accelerometers were needed. This method proved to be accurate in gathering data with minimal instrumentation over a large range of AOAs.

1. **THIS STATE ESTIMATION/KALMAN FILTERING METHOD OF CALCULATING AOA POSITION ERROR SHOULD BE CONSIDERED IN FUTURE AOA ERROR TESTING.**

The Kalman filter needed an initial AOA to start propogating an "optimal" true AOA. This initial AOA was calculated from 1 g wings level flight prior to the roller coaster flight test maneuver. Also, aerodynamic buffet at high load factors caused some data scatter. The data gathered prior to the buffet were enough to calculate the AOA position error.

2. **FUTURE MANEUVERS TO GATHER DATA FOR THIS METHOD SHOULD START FROM A 1 G TRIM SHOT FOR 3 TO 4 SECONDS. THE MANEUVERS SHOULD TERMINATE PRIOR TO AERODYNAMIC BUFFET.**

The T-38A flight test data were not completely accurate due to hysteresis errors in the AOA transducer. The accuracy of the method would be better determined using an aircraft with no AOA hysteresis error. Also, testing with an aircraft with a fuselage-mounted AOA sensor would show larger AOA position errors and would further validate the method.

3. FURTHER TESTING SHOULD BE CONDUCTED USING AN RF-4C OR OTHER SUITABLE AIRCRAFT THAT HAS NO AOA HYSTERESIS ERROR AND HAS A FUSELAGE-MOUNTED AOA SENSOR.

4. THE USAF TEST PILOT SCHOOL NEEDS TO INSTALL NEW AOA TRANSDUCERS IN THEIR T-38A AIRCRAFT TO ELIMINATE THE AOA HYSTERESIS ERRORS.

Appendix A

Flight Test Results

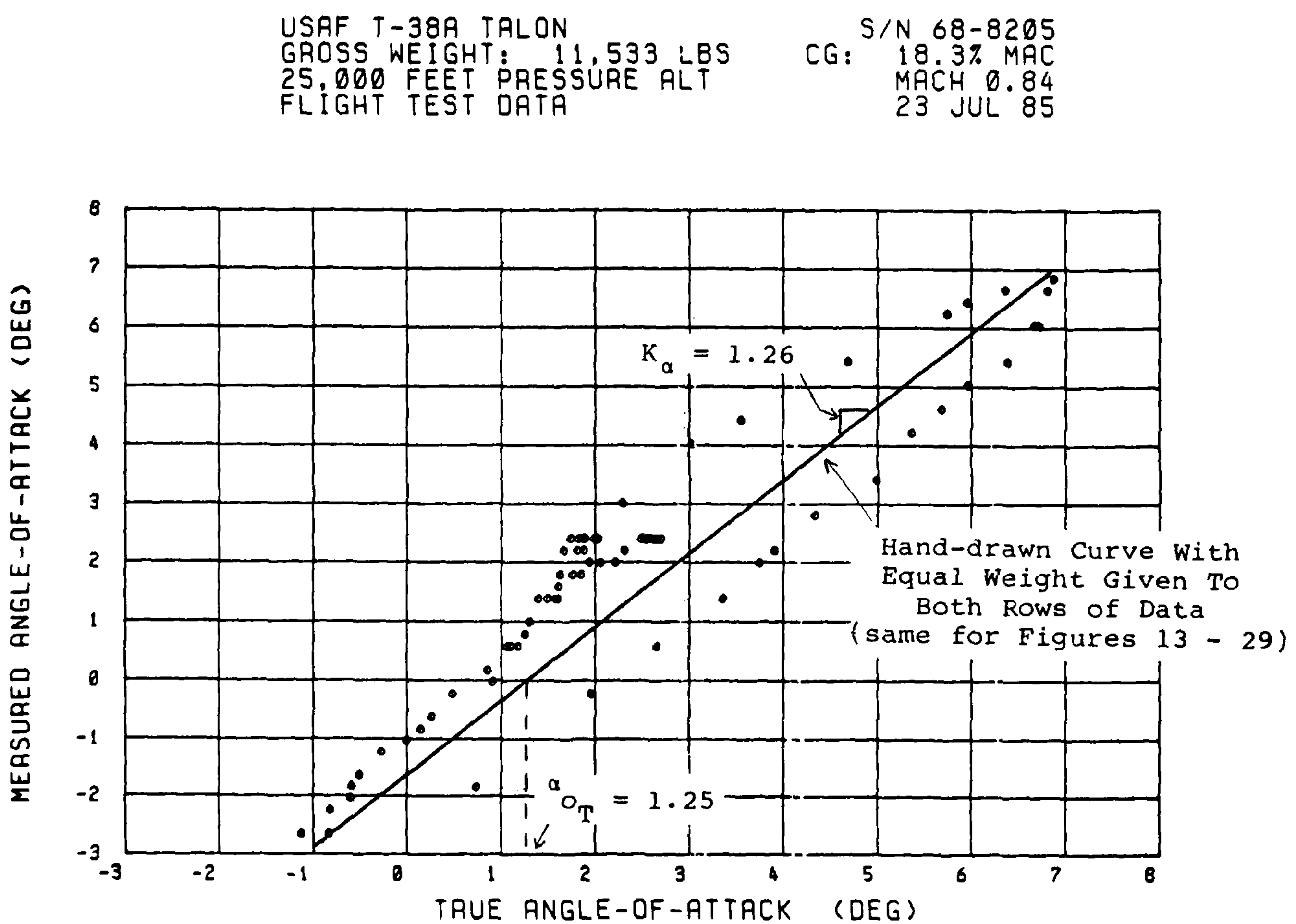


Figure 12. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.84

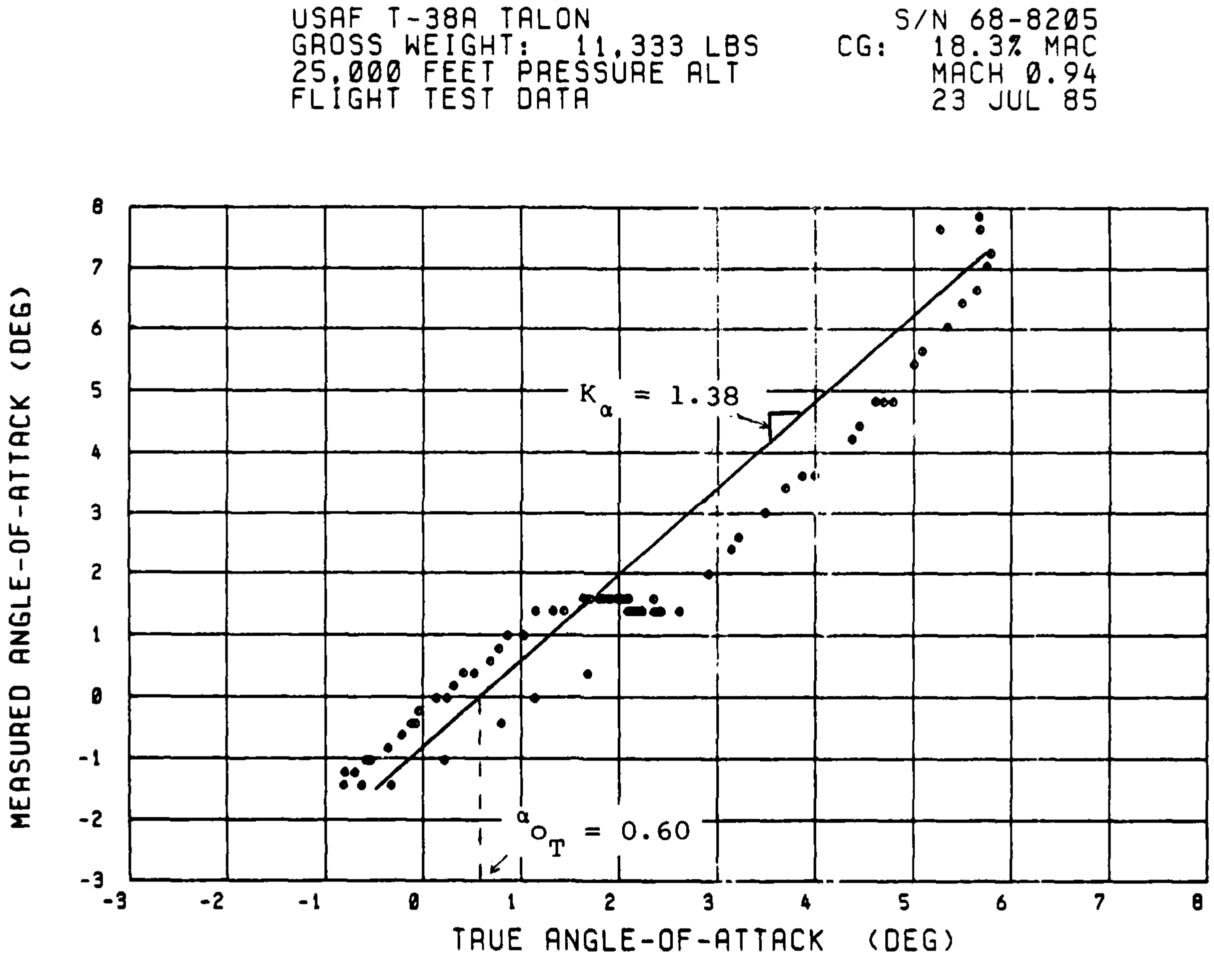


Figure 13. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.94

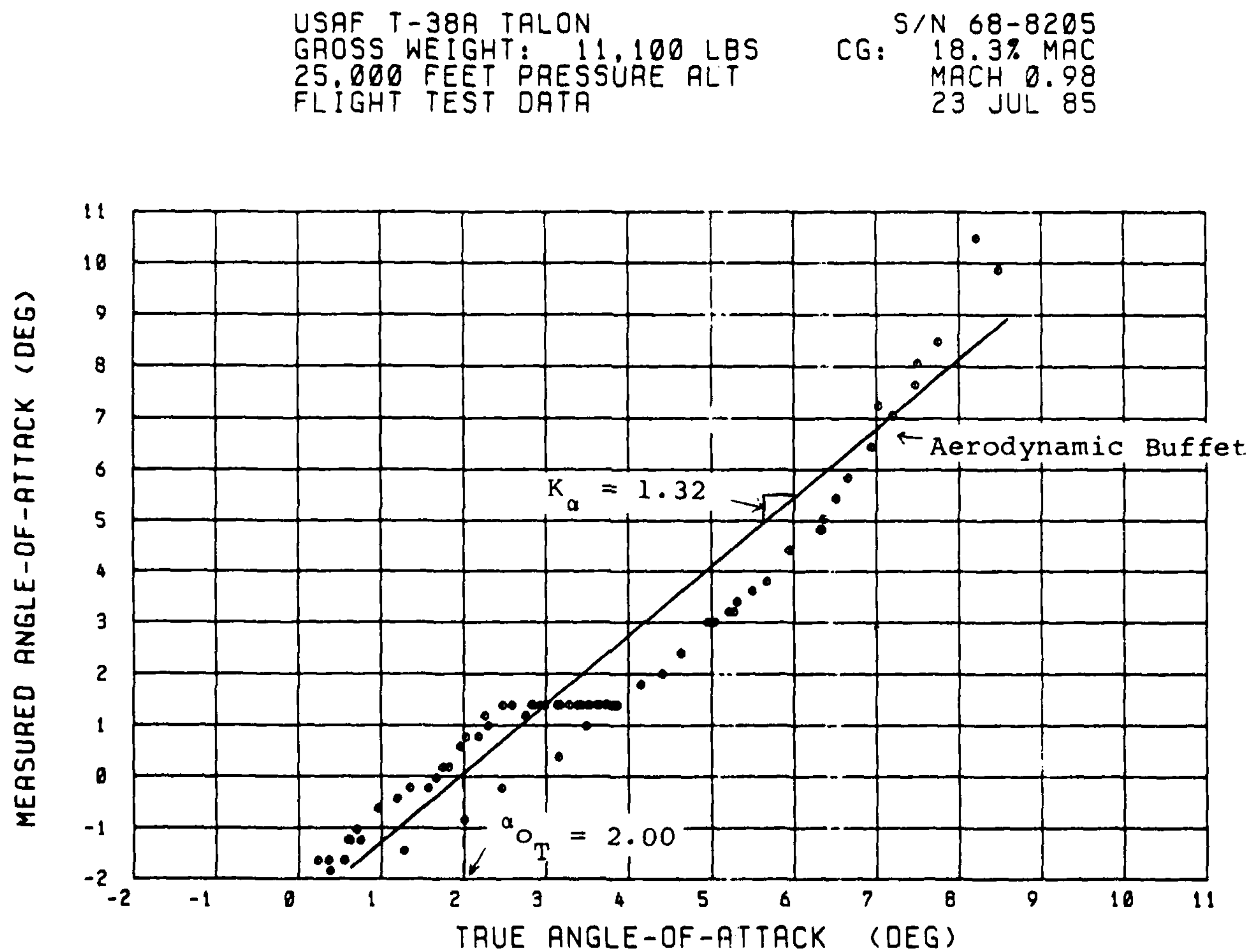


Figure 14. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.98

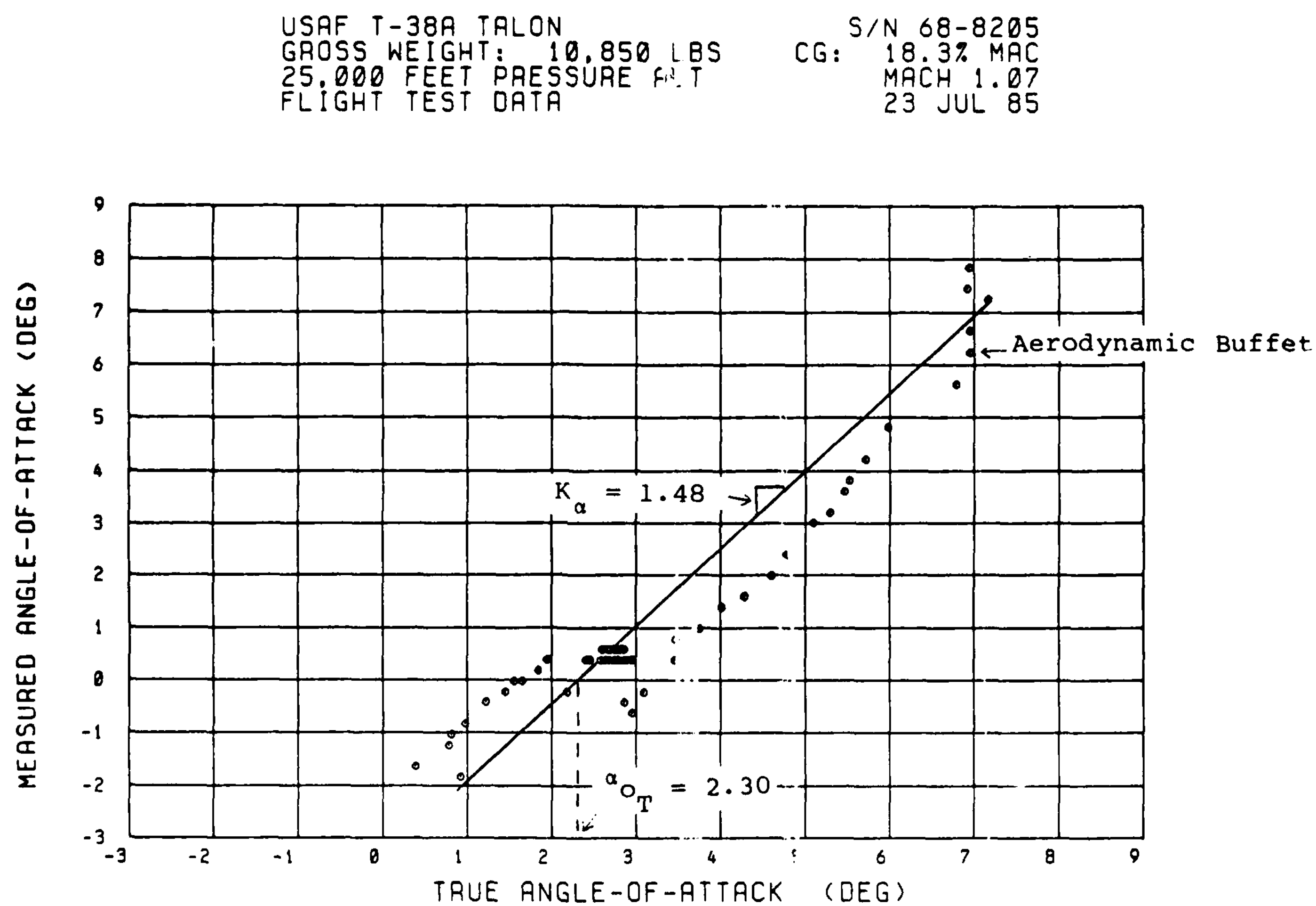


Figure 15. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 1.07

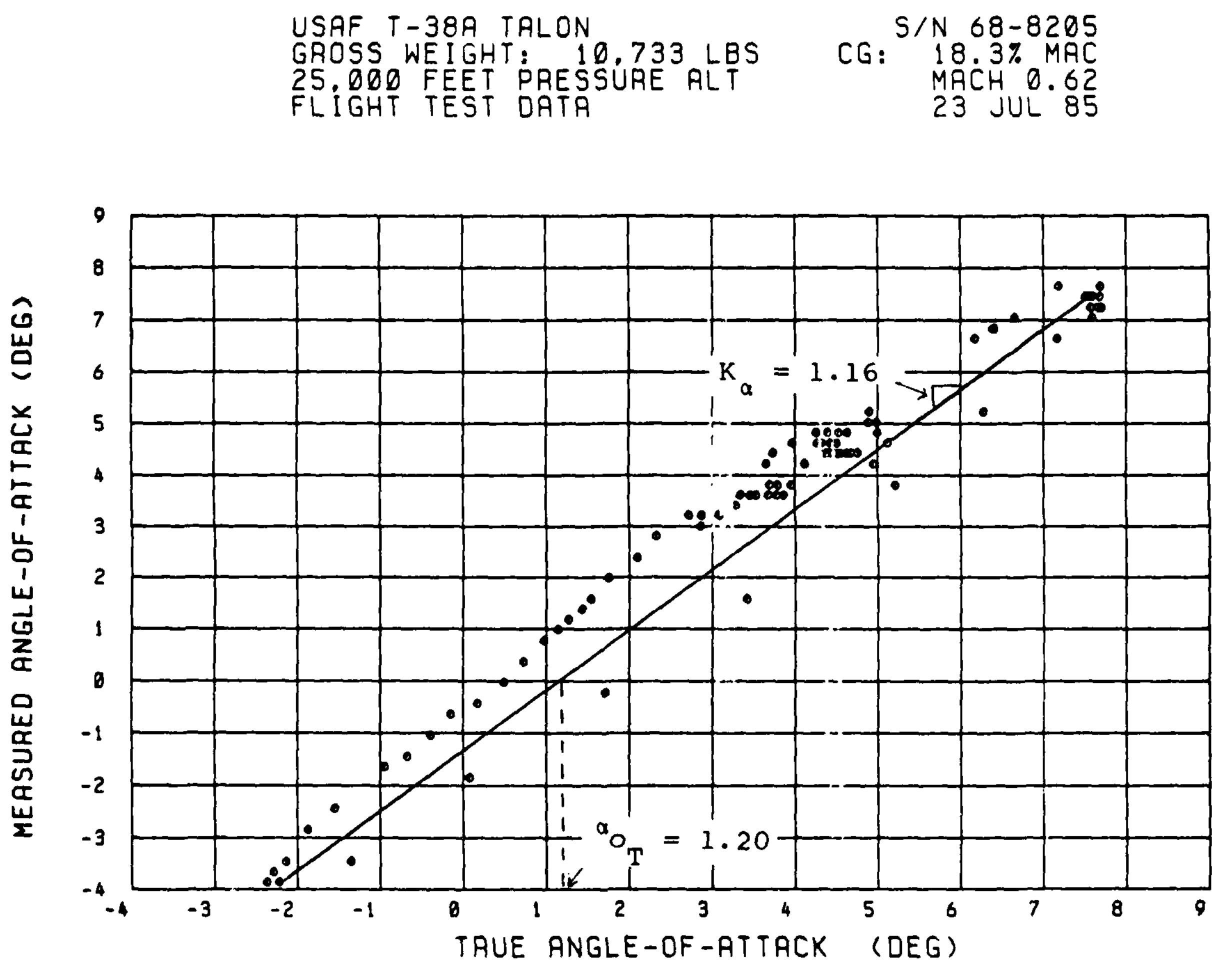


Figure 16. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.62

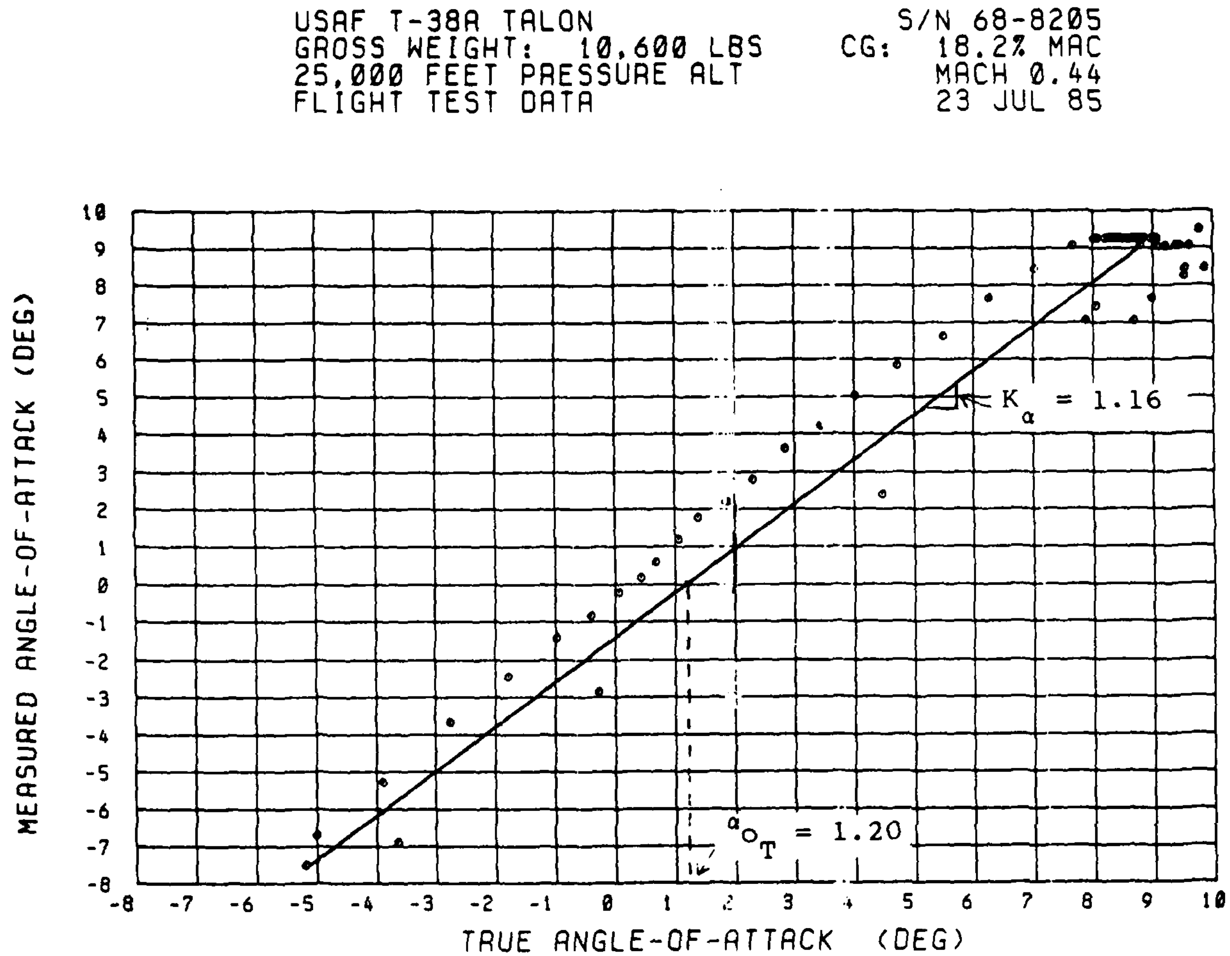


Figure 17. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom AT 25,000 feet H_c, Mach 0.44

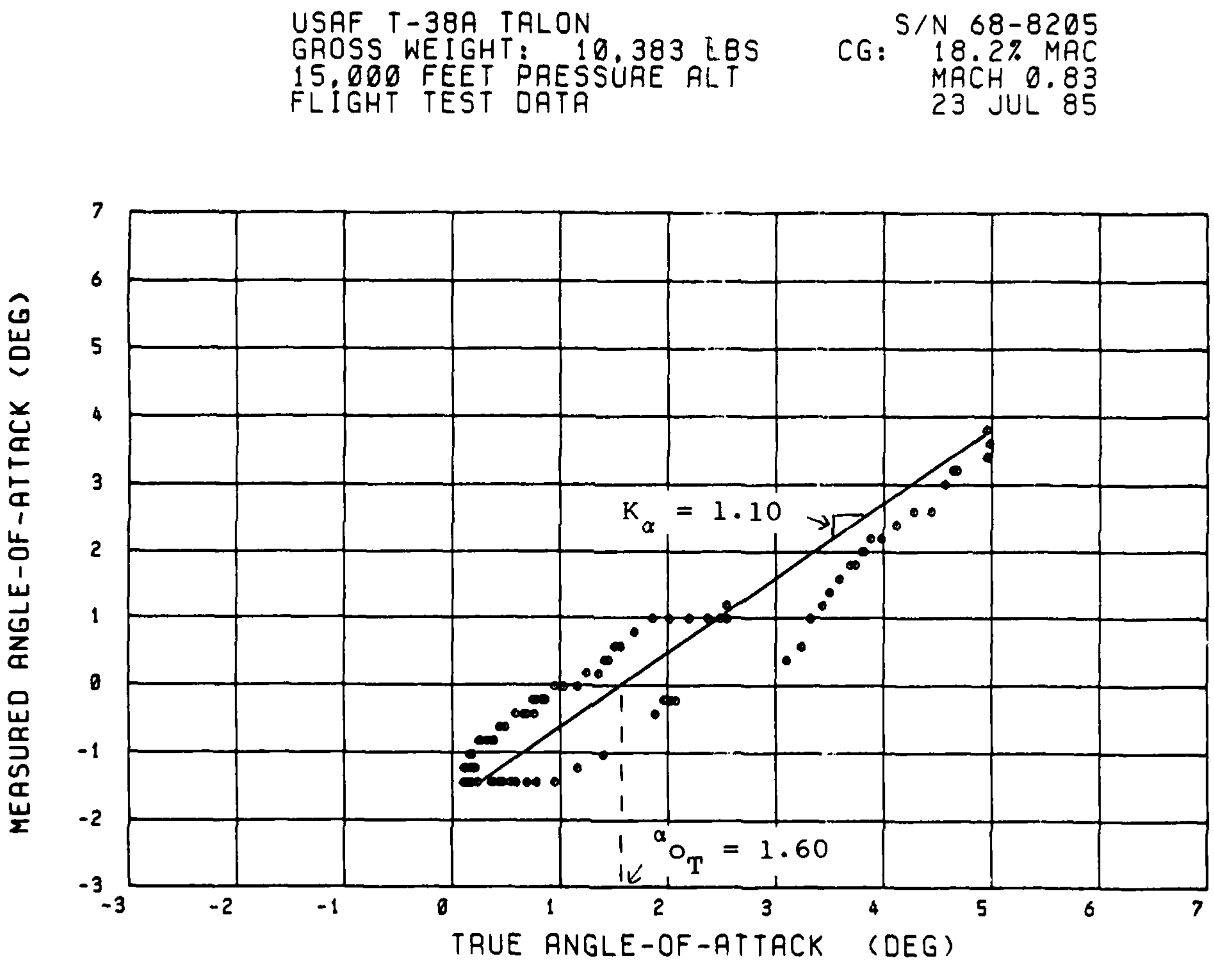


Figure 18. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 15,000 feet H_c, Mach 0.83

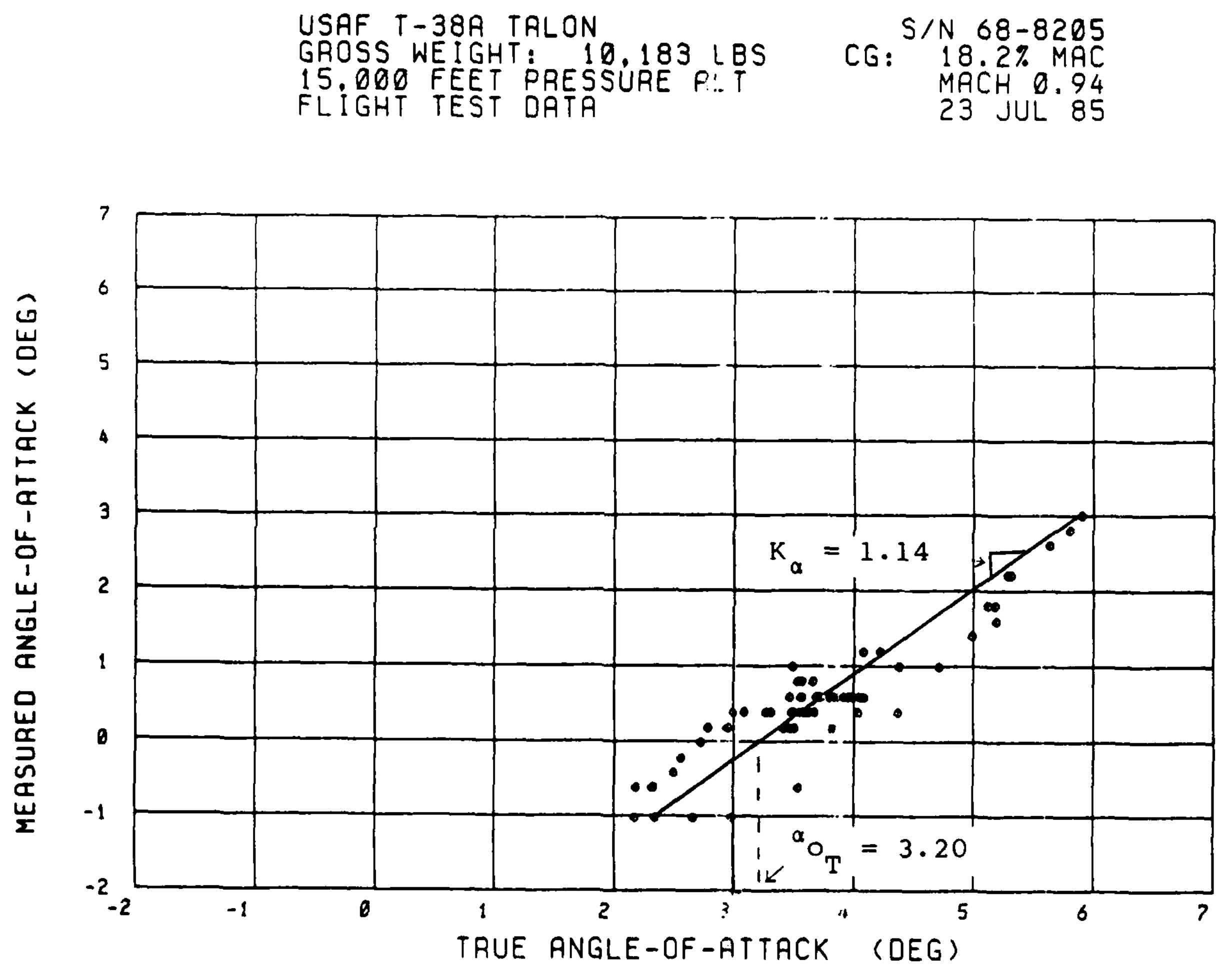


Figure 19. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 15,000 feet H_c, Mach 0.94

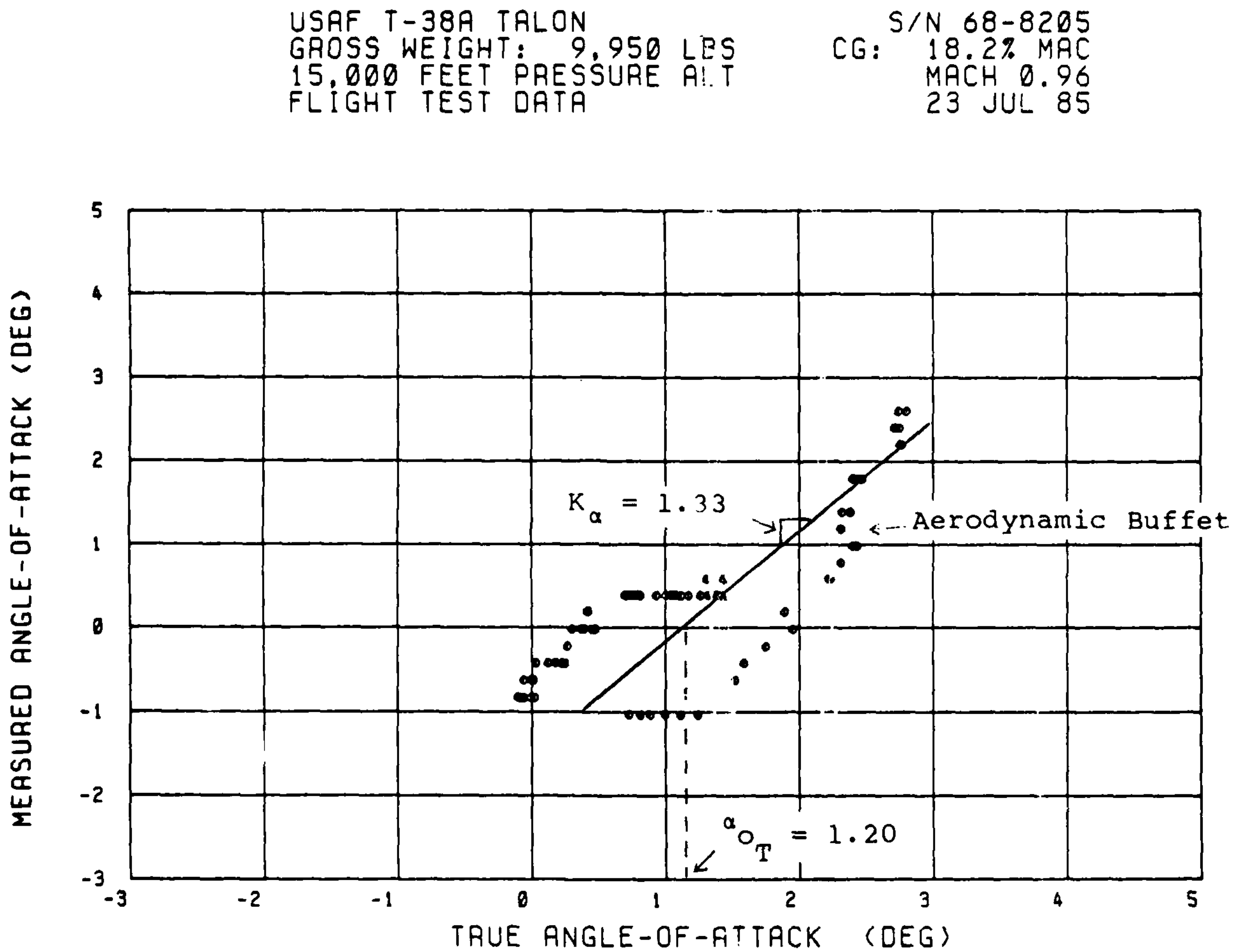


Figure 20. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 15,000 feet H_c, Mach 0.96

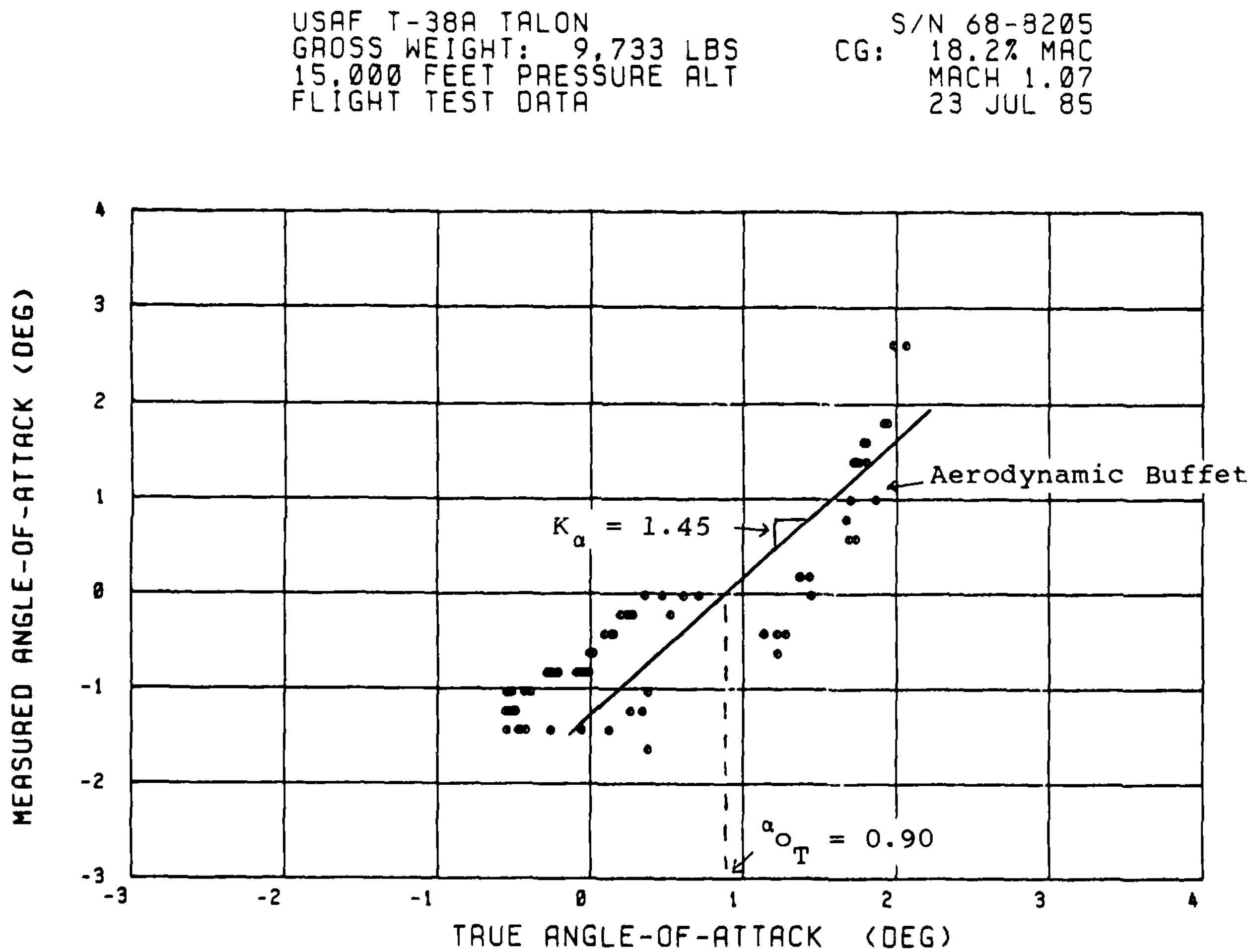


Figure 21. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 15,000 feet H_c, Mach 1.07

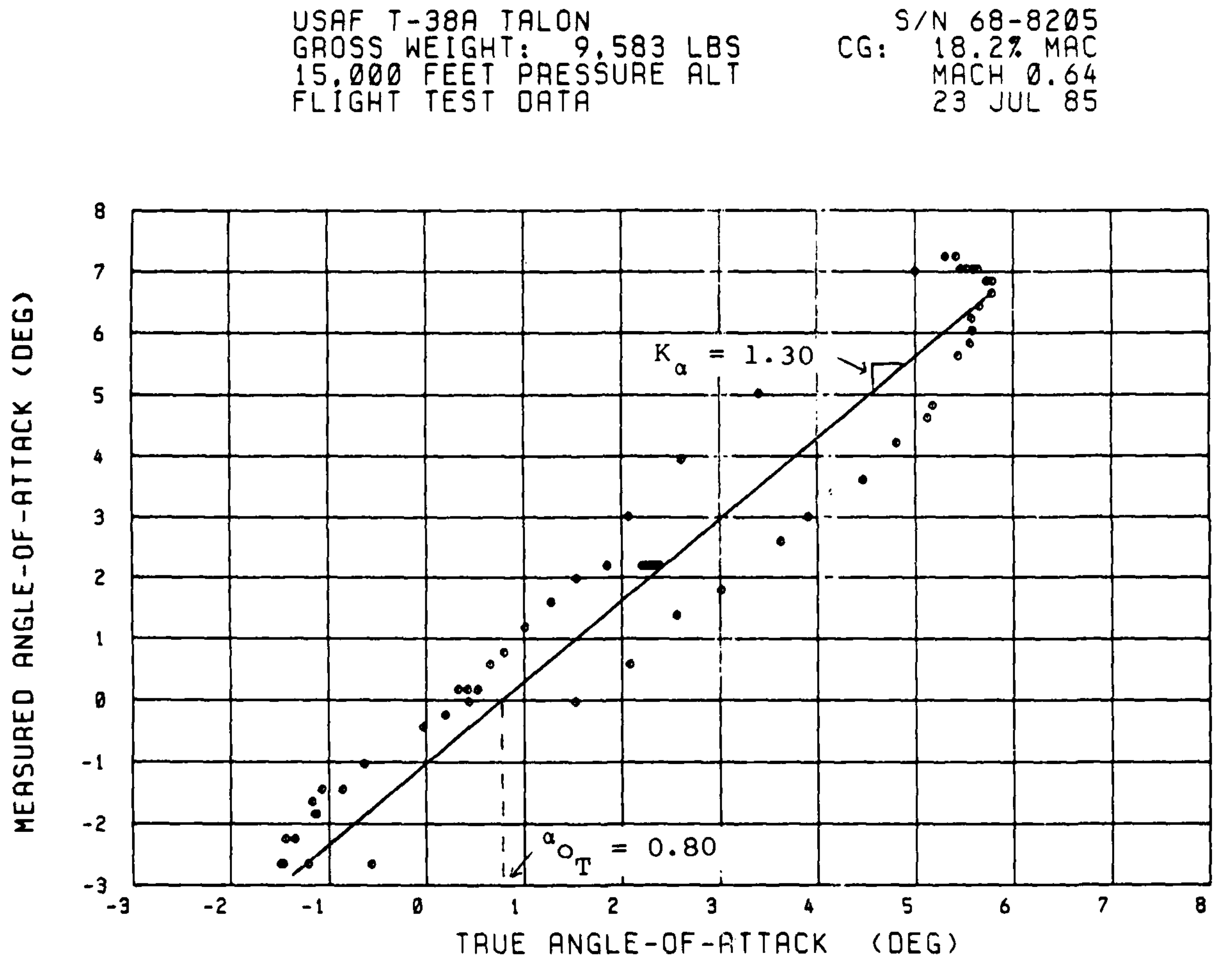


Figure 22. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 15,000 feet H_c, Mach 0.64

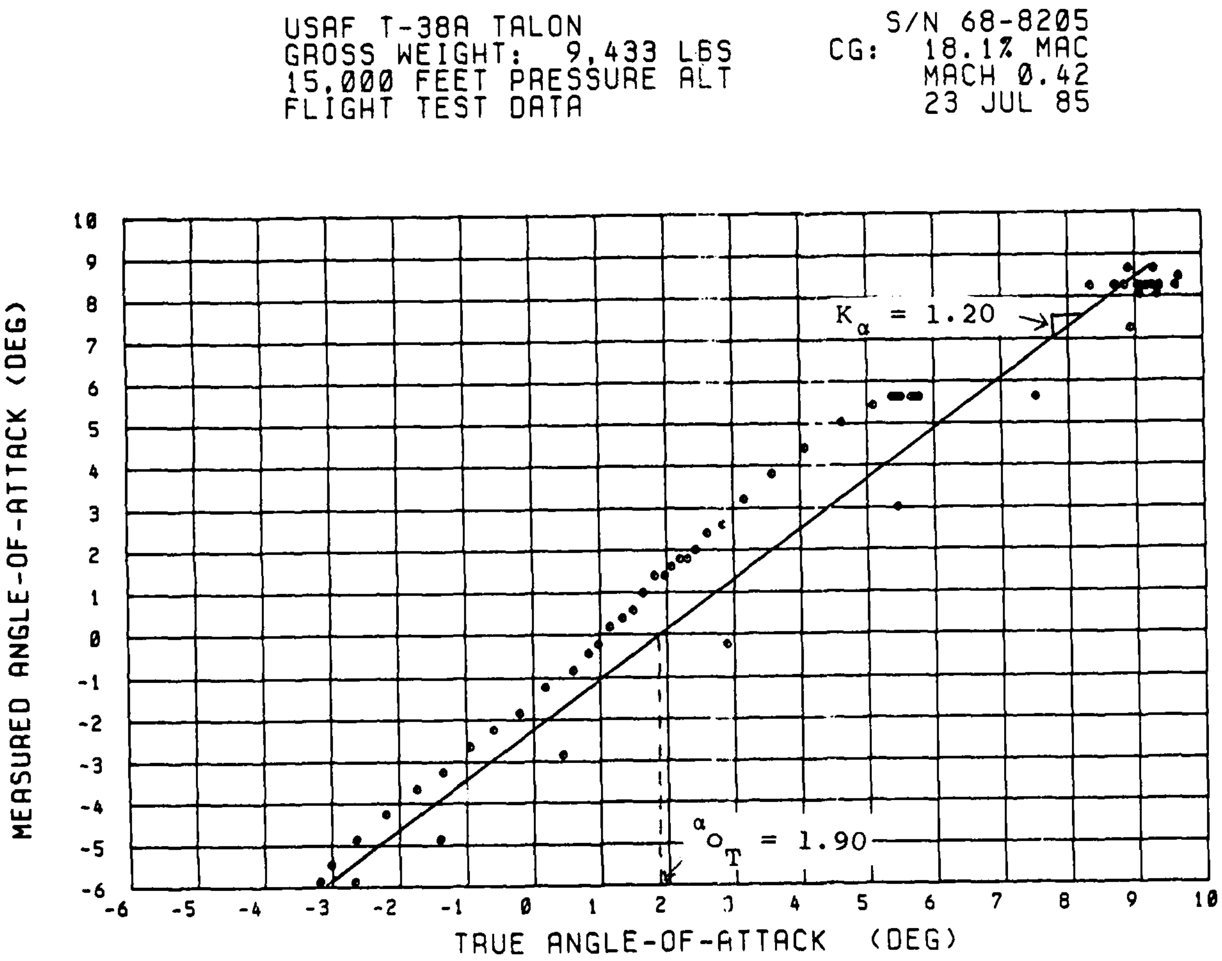


Figure 23. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 15,000 feet H_c, Mach 0.42

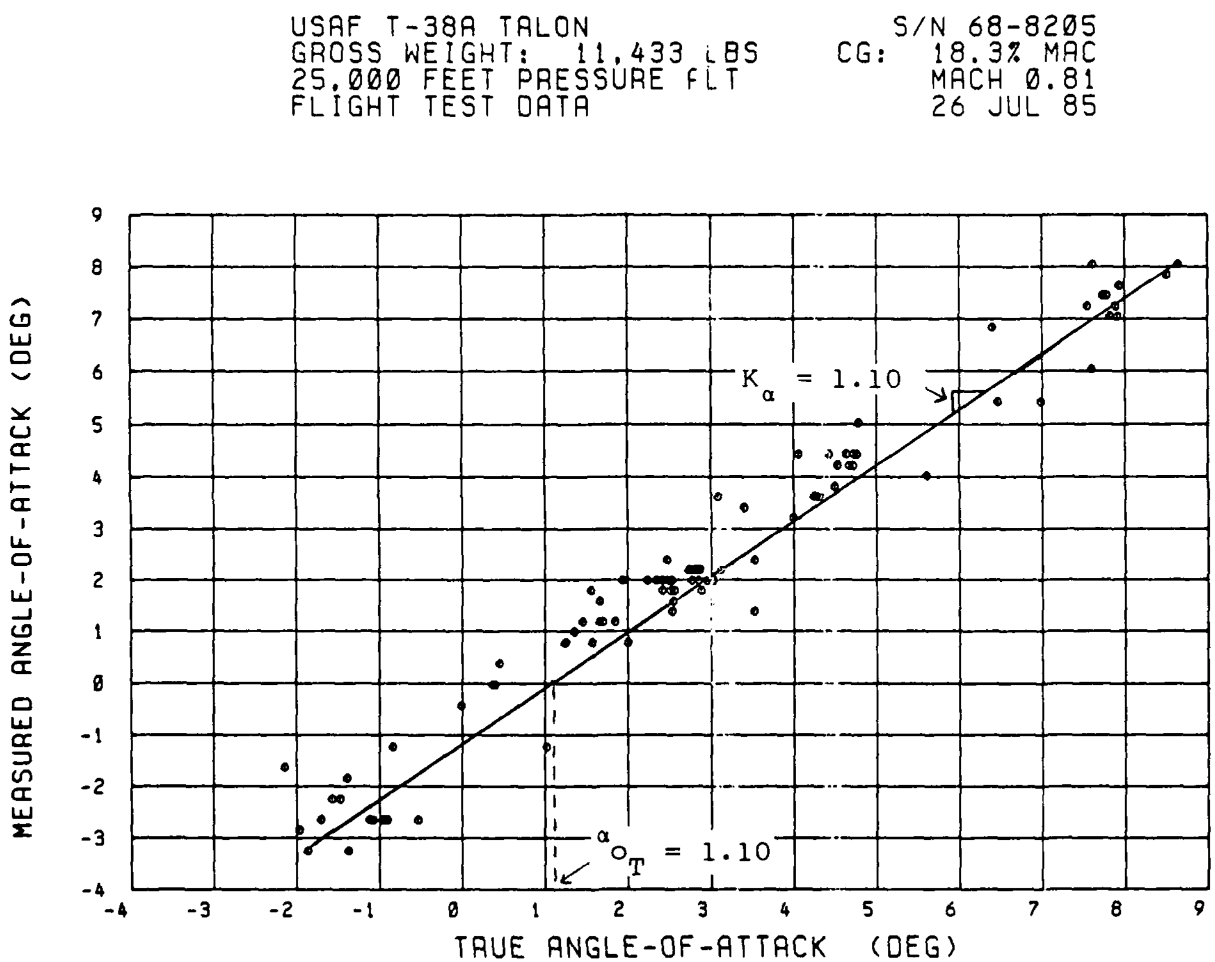


Figure 24. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.81

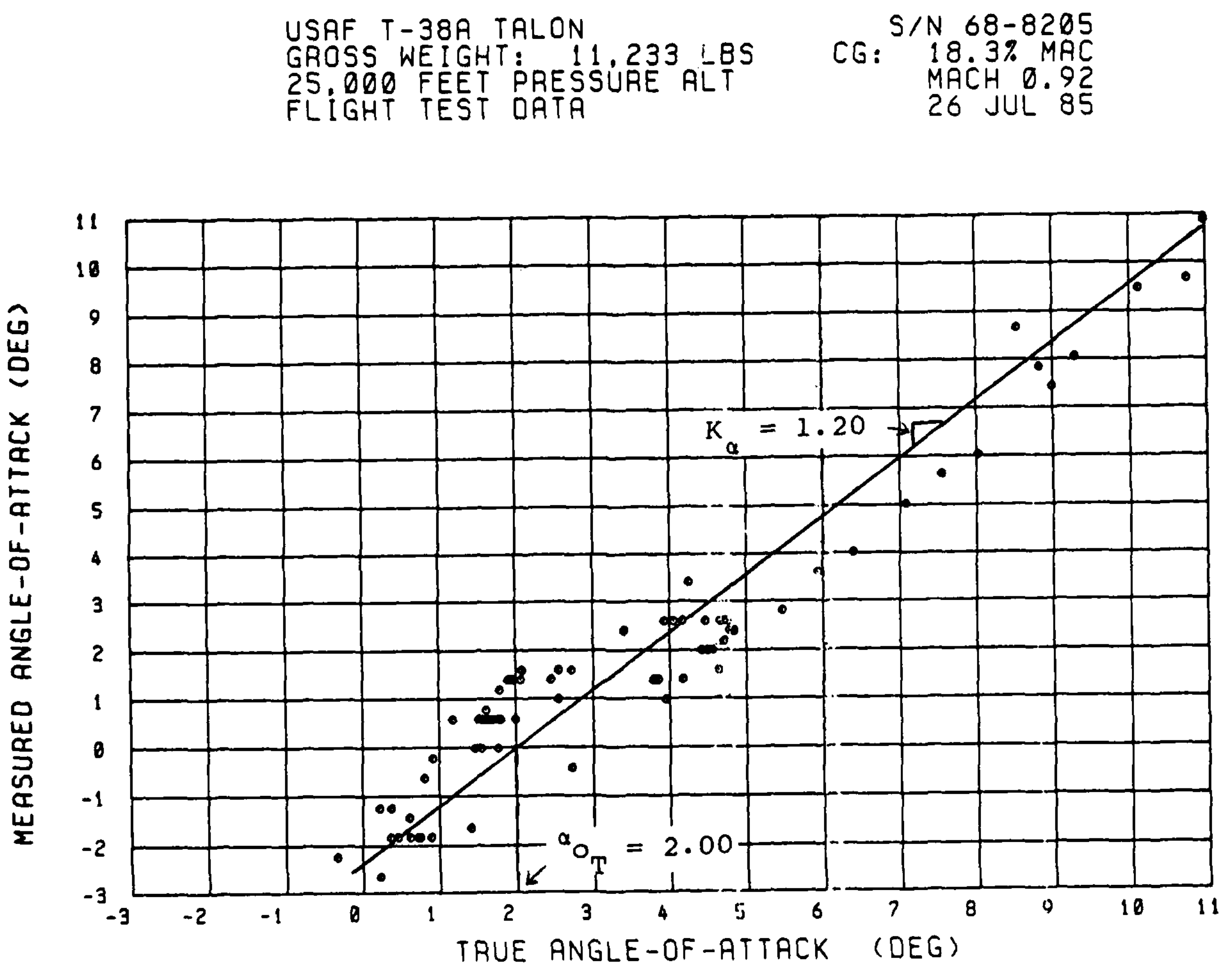


Figure 25. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.92

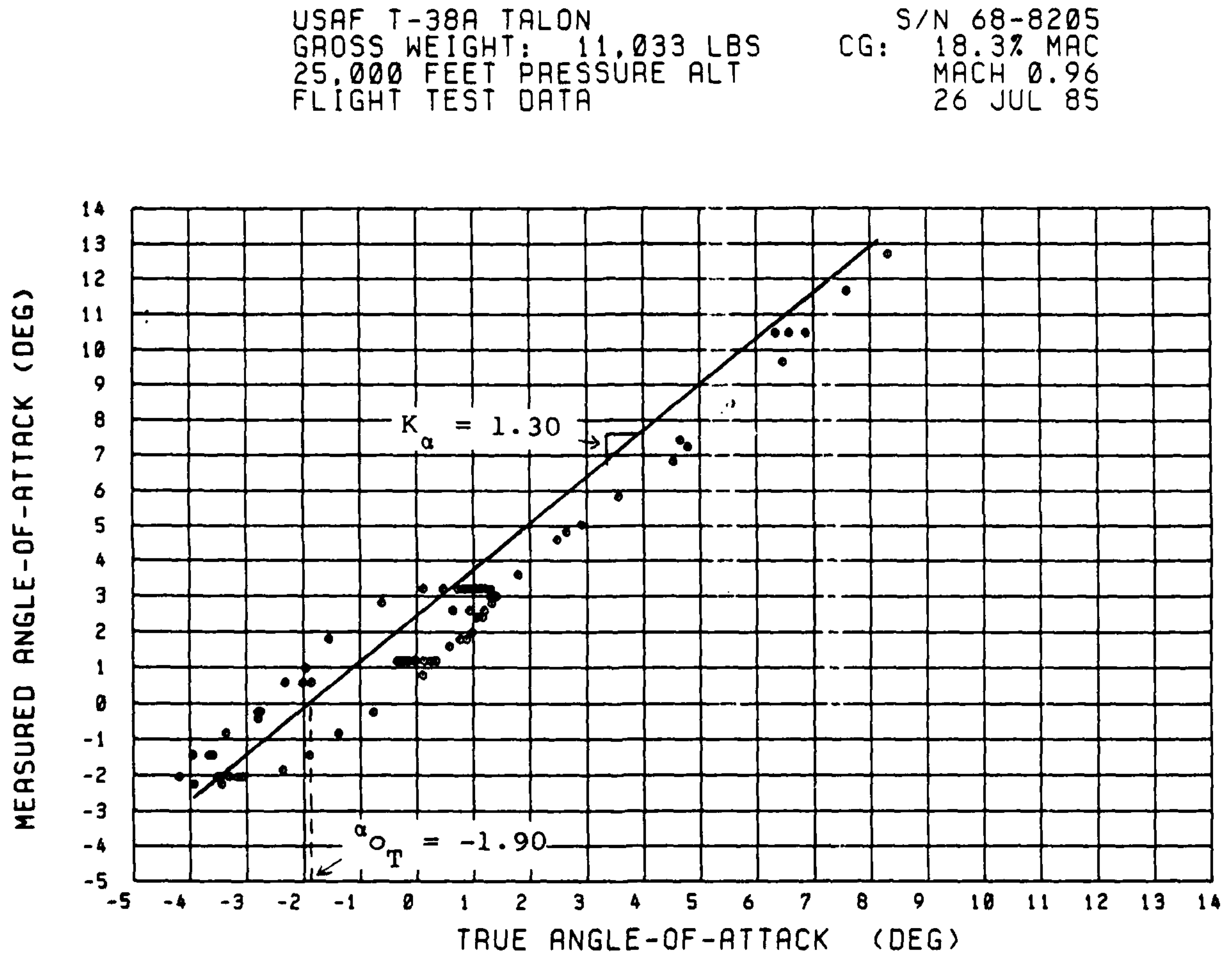


Figure 26. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.96

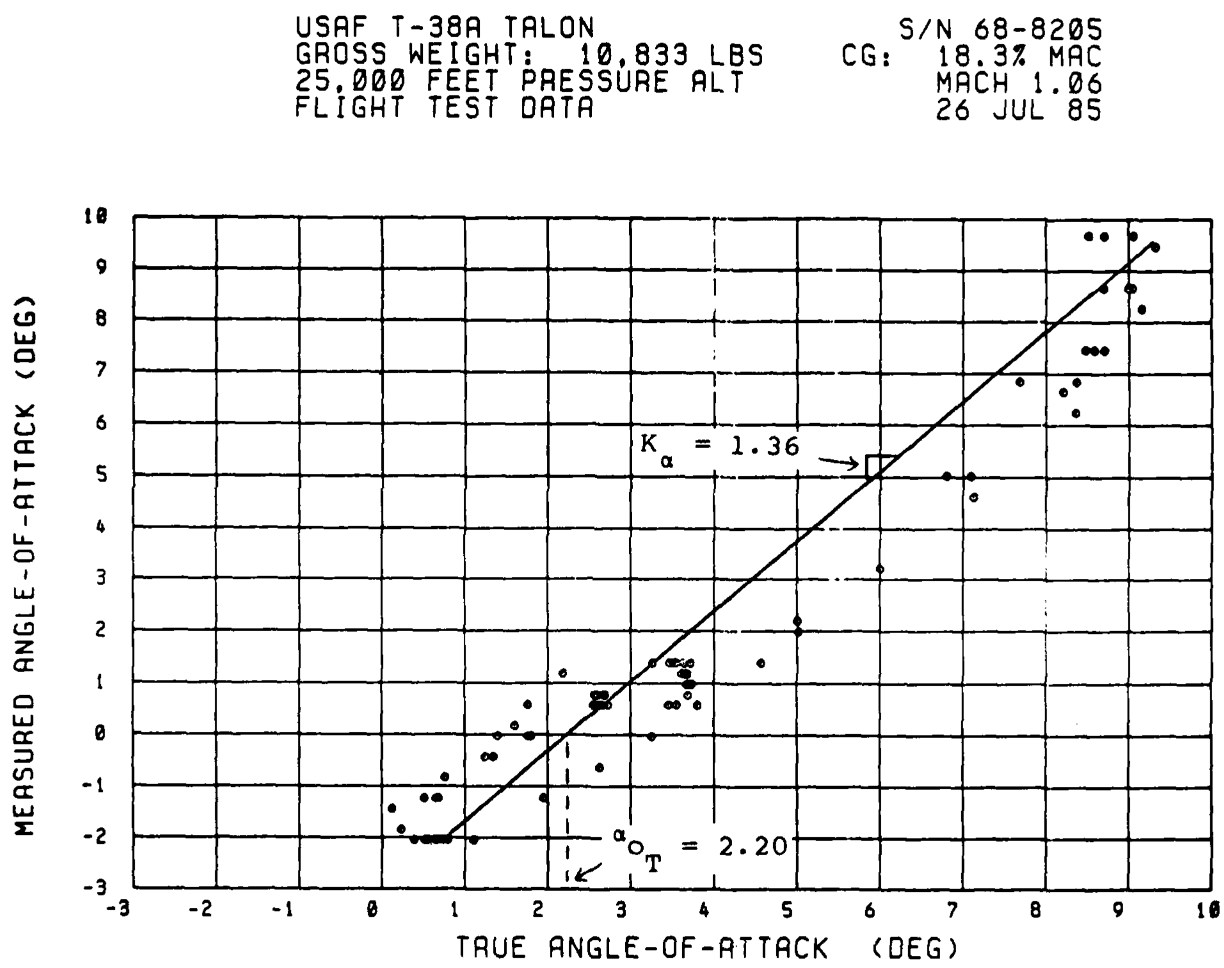


Figure 27. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 1.06

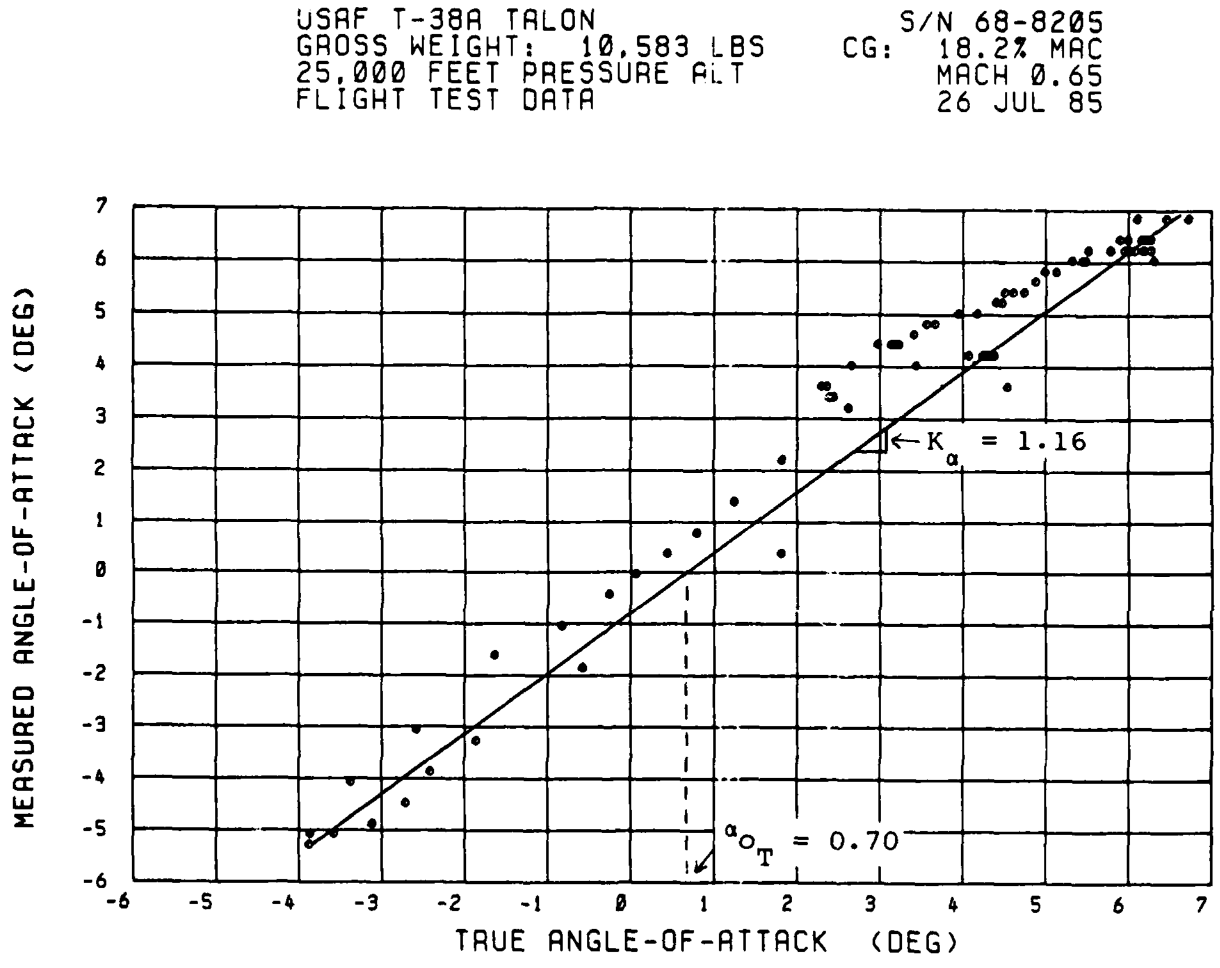


Figure 28. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.65

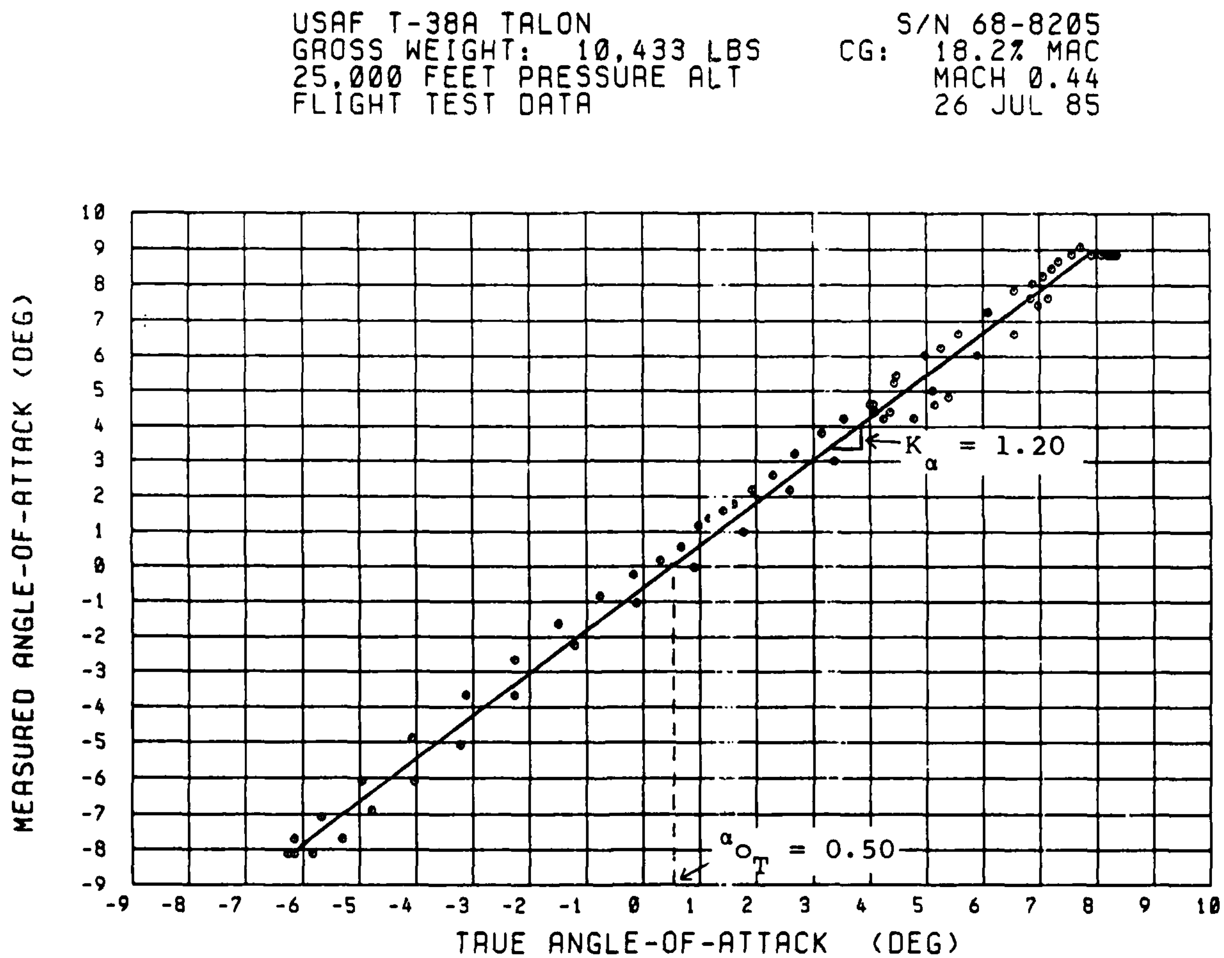


Figure 29. T-38A Angle-of-Attack Position Error for a Vought YAPS Noseboom at 25,000 feet H_c, Mach 0.44

Appendix B

Computer Program AOAOPT

```
      PROGRAM AOAOPT
C
C    PROGRAM AOAOPT IS A FORTRAN PROGRAM DESIGNED TO CALCULATE
C      ANGLE-OF-ATTACK FROM PARAMETERS RECORDED INFLIGHT ON AN
C    AYDIN-VECTOR DATA ACQUISITION SYSTEM.  AOAOPT USES TIME (TI),
C      IAS (AS), INDICATED ALT (ALT), MEASURED AOA (AOA), PITCH
C    ANGLE (TH), PITCH RATE (Q), NORMAL ACCELERATION (ZN), SIDESLIP
C      ANGLE (AOSS), AND BANK ANGLE (BA) OBTAINED FROM DYNAMICS EUP
C    AND EUS FILES.  AOAOPT CORRECTS FOR PITOT-STATIC ERRORS TO
C      DETERMINE PRESSURE ALTITUDE (HC), MACH NUMBER (AMC), AND
C    TRUE AIRSPEED (VTAS).  THE PROGRAM CALCULATES TRUE AOA BY TWO
C      METHODS: ANGLE RELATIONSHIP WHERE AOA = PITCH - FLIGHT PATH,
C    AND ITERATIVE METHOD WHERE AOA-DOT = PITCH RATE - NORMAL G.
C      PROGRAM CAN BE USED WITH T-38A OR RF-4C DATA.
C
      BYTE PNAME(12),ENAME(12),DOF(7),ATYPE(5),ATAIL(3)
      DIMENSION TI(300),AS(300),ALT(300),AOA(300),TH(300),Q(300),
     1ZN(300),AOSS(300),BA(300),HC(300),AMC(300),VTAS(300),AOA2(300),
     2AOA3(300),HDOT(300),FPANG(300),AOA1(300),ERROR(300),ADOT(300),
     3ADOTM(300),ERROR1(300),ERROR2(300)
      BYTE FILE1(15),OFILE1(17)
      BYTE FILE2(15),OFILE2(17)
      BYTE FILE3(15),OFILE3(17)
      BYTE FILE4(15),OFILE4(17)
      BYTE FILE6(15),OFILE6(17)
      DATA FILE1/'E','S',':',8*0,'.','E','U','P'/
      DATA FILE2/'E','S',':',8*0,'.','E','U','S'/
      DATA FILE3/'E','S',':',8*0,'.','D','A','T'/
      DATA FILE4/'E','S',':',8*0,'.','D','A','T'/
      DATA FILE6/'E','S',':',8*0,'.','D','A','T'/
C
      WRITE (5,100)
100   FORMAT(' ENTER NAME OF DYNAMICS FILE: NNNNN.EUP',/)
      ACCEPT 110,(FILE1(I),I=4,11)
110   FORMAT (8A1)
      OPEN(UNIT=1,NAME=FILE1,TYPE='UNKNOWN')
      WRITE (5,120)
120   FORMAT(' ENTER NAME OF DYNAMICS FILE: NNNNN.EUS',/)
      ACCEPT 110,(FILE2(I),I=4,11)
      OPEN(UNIT=2,NAME=FILE2,TYPE='UNKNOWN')
      WRITE(5,130)
130   FORMAT(' ENTER NAME FOR DATA FILE 1:  NNNNNN.DAT',/)
      ACCEPT 110,(FILE3(I),I=4,11)
      OPEN(UNIT=3,NAME=FILE3,TYPE='UNKNOWN')
      WRITE(5,140)
140   FORMAT(' ENTER NAME FOR DATA FILE 2:  NNNNNN.DAT',/)
      ACCEPT 110,(FILE4(I),I=4,11)
      OPEN(UNIT=4,NAME=FILE4,TYPE='UNKNOWN')
      WRITE(5,150)
150   FORMAT(' ENTER NAME FOR DATA FILE 3:  NNNNNN.DAT',/)
      ACCEPT 110,(FILE6(I),I=4,11)
      OPEN(UNIT=6,NAME=FILE6,TYPE='UNKNOWN')
      WRITE(5,160)
160   FORMAT (' ENTER NUMBER OF DATA POINTS DESIRED ' ,/)
      READ(5,170) N
170   FORMAT (I5)
```

```
      WRITE(5,180)
180   FORMAT (' ENTER FIRST LINE OF DATA DESIRED ' ,/)
      READ(5,190) M
190   FORMAT (I5)
      WRITE(5,192)
192   FORMAT (' ENTER NUMBER OF INITIAL CONSTANT AOA LINES ' ,/)
      READ(5,195) NA
195   FORMAT(I5)
      WRITE(5,200)
200   FORMAT(' ENTER CORRECTIONS FOR PITCH RATE AND NZ: XXX,YYY',/,
     1'     CORR ARE FROM GROUND BLOCK, ABOVE/BELOW 0 FOR Q',/,
     2'                                 ABOVE/BELOW 1 FOR NZ',/)
      READ(5,210)QCORR,ZNCORR
210   FORMAT(2F10.3)
C
C   THIS PORTION READS DATA FROM THE DYNAMICS EUP AND EUS FILES
C     AND FORMATS THREE DATA FILES (XXXX.DAT) WHERE THE PROGRAM
C   RESULTS WILL BE SENT.  DATA FILE #1 RECORDS THE RAW DATA FROM
C     THE EUP AND EUS FILES.  DATA FILE #2 CONTAINS AOA COMPUTED
C   BY THE ANGLE METHOD.  DATA FILE #3 CONTAINS AOA COMPUTED BY
C     THE ITERATIVE METHOD.
C
      READ (1,220)PNAME,ENAME,DOF
220   FORMAT (8X,12A1,15X,12A1,21X,7A1)
      WRITE (3,230)PNAME,ENAME,DOF
      WRITE (4,230)PNAME,ENAME,DOF
      WRITE (6,230)PNAME,ENAME,DOF
230   FORMAT (' PILOT:',1X,12A1,5X,'ENGINEER:',1X,12A1,5X,
     1'DATE OF FLIGHT:',1X,7A1)
      READ (1,240)ATYPE,ATAIL
240   FORMAT (11X,5A1,13X,3A1)
      WRITE (3,250)ATYPE,ATAIL
      WRITE (4,250)ATYPE,ATAIL
      WRITE (6,250)ATYPE,ATAIL
250   FORMAT(' A/C TYPE:',1X,5A1,5X,'TAIL #:',1X,3A1)
      WRITE (3,260)
260   FORMAT (//,7X,'LINE',6X,'TIME',2X,'AIRSPEED',2X,'ALTITUDE',7X,
     1'AOA',5X,'PITCH',2X,'PITCH RT',8X,'NZ',6X,'AOSS',2X,'ROLL ANG')
      WRITE (4,270)
270   FORMAT (//,7X,'LINE',6X,'TIME',1X,'PRESS ALT',6X,'MACH',3X,'TRUE
     1 AS',7X,'ROC',2X,'FLT PATH',5X,'AOA-1',5X,'AOA-M',5X,'ERROR')
      WRITE (6,280)
280   FORMAT (//,7X,'LINE',6X,'TIME',4X,'A-DOT1',4X,'A-DOTM',5X,'AOA-2',
     15X,'AOA-M',5X,'ERROR',5X,'AOA-3',5X,'AOA-T',5X,'ERROR',3X,
     2'K-ALPHA')
      WRITE (3,290)
290   FORMAT (16X,'(SEC)',3X,'(KNOTS)',4X,'(FEET)',5X,'(DEG)',
     15X,'(DEG)',1X,'(DEG/SEC)',7X,'(G)',5X,'(DEG)',5X,'(DEG)',/)
      WRITE (4,300)
300   FORMAT (16X,'(SEC)',4X,'(FEET)',12X,'(FT/SEC)',2X,
     1'(FT/SEC)',5X,'(DEG)',5X,'(DEG)',5X,'(DEG)',5X,'(DEG)',/)
      WRITE(6,310)
310   FORMAT (16X,'(SEC)',1X,'(DEG/SEC)',1X,'(DEG/SEC)',5X,'(DEG)',
     15X,'(DEG)',5X,'(DEG)',5X,'(DEG)',5X,'(DEG)',5X,'(DEG)',/)
      READ (1,320)
320   FORMAT (/,/,/)
```

```
      READ (2,330)
330   FORMAT (/,/,/,/,/)
      DO 350 J=1,M
      READ(1,340)
      READ(2,340)
340   FORMAT ( )
350   CONTINUE
      DO 390 I=1,N
      READ(1,360)TI(I),AS(I),ALT(I),AOA(I),TH(I),Q(I),ZN(I)
360   FORMAT (31X,F10.3,F10.3,F10.3,F10.3,F10.3,F10.3,20X,F10.3)
      READ(2,370)AOSS(I),BA(I)
370   FORMAT (31X,F10.3,10X,F10.3)
C
C   THE NEXT TWO STEPS CORRECT PITCH RATE AND NORMAL G FOR BIAS
C     FOUND ON THE GROUND (BIAS COMPUTED FROM 0 DEG/SEC FOR PITCH
C   RATE AND 1 G FOR NORMAL G)
C
      Q(I)=Q(I)-QCORR
      ZN(I)=ZN(I)-ZNCORR
      WRITE(3,380)I,TI(I),AS(I),ALT(I),AOA(I),TH(I),Q(I),ZN(I),AOSS(I),
     1BA(I)
380   FORMAT(1X,I10,9F10.3)
390   CONTINUE
C
C   THIS PORTION PERFORMS THE PITOT-STATIC CORRECTIONS TO COMPUTE
C     PRESSURE ALT, MACH NUMBER, AND TRUE AIRSPEED.  IT USES THE
C   STANDARD PITOT-STATIC EQUATIONS AND ERROR COEFFICIENTS FROM THE
C     USAF TEST PILOT SCHOOL FILES.
C
      DO 710 I=1,N
      IF(ALT(I) .GT. 36089)GO TO 500
      DELTA=(1-(6.87559E-6*ALT(I)))**5.2561
      THETA=1-(6.87559E-6*ALT(I))
      GO TO 510
500   DELTA=.22337*EXP(-4.80635E-5*(ALT(I)-36089))
      THETA=.751874
510   IF(AS(I) .GT. 661.48)GO TO 520
      QCICPA=((1+(.2*((AS(I)/661.48)**2)))**3.5)-1
      GO TO 530
520   QCICPA=((166.922*((AS(I)/661.48)**7))/(((7*((AS(I)/661.48)**2))
     1-1)**2.5))-1
530   QCICPS=QCICPA/DELTA
      AMIC=SQRT(5*(((QCICPS+1)**(.2857143))-1))
      IF(ATYPE(1) .EQ. 'T')GO TO 600
C
      DHPC=(-907+(10270*AMIC)+(-44495*(AMIC**2))+(93931*(AMIC**3))
     1+(-95540*(AMIC**4))+(37208*(AMIC**5)))*THETA
      IF(AMIC .GT. 1)DHPC=0
      GO TO 700
600   IF(AMIC .LT. .955)GO TO 620
      IF(AMIC .LT. .967)GO TO 640
      IF(AMIC .LT. 1.025)GO TO 660
      C0=-46325
      C1=82583
      C2=-36667
      GO TO 680
```

```
620   C0=118
      C1=-478
      C2=912
      GO TO 680
640   C0=2676675
      C1=-5636789
      C2=2968036
      GO TO 680
660   C0=-141471
      C1=308123
      C2=-166107
      GO TO 680
680   DHPC=(C0+(C1*AMIC)+(C2*(AMIC**2)))*THETA
700   HC(I)=ALT(I)+DHPC
      DPPPS=(3.61382E-5*DHPC)/THETA
      DMPC=((1+(.2*(AMIC**2)))*DPPPS)/(1.4*AMIC)
      AMC(I)=AMIC+DMPC
      VTAS(I)=1.6878*AMC(I)*38.96763*SQRT(THETA*288.15)
710   CONTINUE
C
C    THIS PORTION COMPUTES TRUE AOA BY THE ANGLE METHOD.  FLIGHT
C      PATH ANGLE IS CALCULATED BY:
C    FLIGHT PATH ANGLE (FPANG) = INV SIN [VERTICAL VEL (HDOT) /
C      TRUE AIRSPEED (VTAS)]
C    TRUE AOA IS CALCULATED BY:  AOA = PITCH ANGLE - FLT PATH
C      AOA IS THEN AVERAGED OVER A TIME INTERVAL FOR USE AS
C    THE INITIAL AOA FOR THE ITERATIVE METHOD
C
      DO 750 I=1,NA
      IF(I .EQ. 1)GO TO 720
      IF(I .EQ. NA)GO TO 720
      DTIME=TI(I+1)-TI(I-1)
      DALT=HC(I+1)-HC(I-1)
      HDOT(I)=DALT/DTIME
      FPANG(I)=ASIN(HDOT(I)/VTAS(I))
      FPANG(I)=FPANG(I)*57.29578
      AOA1(I)=(TH(I)-FPANG(I))/COS(BA(I)/57.29578)
      ERROR(I)=AOA1(I)-AOA(I)
720   WRITE(4,740)I,TI(I),HC(I),AMC(I),VTAS(I),HDOT(I),FPANG(I),
     1AOA1(I),AOA(I),ERROR(I)
740   FORMAT(1X,I10,9F10.3)
750   CONTINUE
C
C    THIS PORTION AVERAGES THE ALTITUDE, MACH, PITCH,
C      PITCH RATE, NZ, AOA-MEAS, AOA-CALC, AND ERROR
C
      DO 752 I=2,NA-1
      DTIME=TI(I+1)-TI(I-1)
      TALT=TALT+(HC(I)*DTIME)
      TMACH=TMACH+(AMC(I)*DTIME)
      TTH=TTH+(TH(I)*DTIME)
      TQ=TQ+(Q(I)*DTIME)
      TNZ=TNZ+(ZN(I)*DTIME)
      TAOA=TAOA+(AOA(I)*DTIME)
      TAOA1=TAOA1+(AOA1(I)*DTIME)
      TERROR=TERROR+(ERROR(I)*DTIME)
```

```
      TTIME=TTIME+DTIME
752   CONTINUE
      AALT=TALT/TTIME
      AMACH=TMACH/TTIME
      ATH=TTH/TTIME
      AQ=TQ/TTIME
      ANZ=TNZ/TTIME
      AAOA=TAOA/TTIME
      AAOA1=TAOA1/TTIME
      AERROR=TERROR/TTIME
      DO 755 I=2,NA-1
      XALT=AALT-HC(I)
      XMACH=AMACH-AMC(I)
      XTH=ATH-TH(I)
      XQ=AQ-Q(I)
      XNZ=ANZ-ZN(I)
      XAOA=AAOA-AOA(I)
      XAOA1=AAOA1-AOA1(I)
      XERROR=AERROR-ERROR(I)
      XALT=ABS(XALT)
      XMACH=ABS(XMACH)
      XTH=ABS(XTH)
      XQ=ABS(XQ)
      XNZ=ABS(XNZ)
      XAOA=ABS(XAOA)
      XAOA1=ABS(XAOA1)
      XERROR=ABS(XERROR)
      IF(XALT .GT. YALT)YALT=XALT
      IF(XMACH .GT. YMACH)YMACH=XMACH
      IF(XTH .GT. YTH)YTH=XTH
      IF(XQ .GT. YQ)YQ=XQ
      IF(XNZ .GT. YNZ)YNZ=XNZ
      IF(XAOA .GT. YAOA)YAOA=XAOA
      IF(XAOA1 .GT. YAOA1)YAOA1=XAOA1
      IF(XERROR .GT. YERROR)YERROR=XERROR
755   CONTINUE
      WRITE (4,758)AALT,YALT,AMACH,YMACH,ATH,YTH,AQ,YQ,ANZ,YNZ,
     1AAOA1,YAOA1,AAOA,YAOA,AERROR,YERROR
758   FORMAT(/,' AVG ALT = ',F10.3,'  +/- ',F8.3,'     AVG MACH = ',
     1F10.3,'  +/- ',F6.3,/,' AVG PITCH = ',F10.3,'  +/- ',F6.3,
     2'    AVG PITCH RATE = ',F10.3,'  +/- ',F6.3,'    AVG NZ = ',
     3F10.3,'  +/- ',F6.3,/,' AVG AOA-1 = ',F10.3,'  +/- ',F6.3,
     4'    AVG AOA-M = ',F10.3,'  +/- ',F6.3,'    AVG ERROR = ',
     5F10.3,'  +/- ',F6.3,//)
C
C    THIS PORTION COMPUTES THE AOA OF THE REMAINING
C      POINTS BY COMPUTING FLIGHT PATH ANGLE, THEN
C    USING:  AOA = PITCH ANGLE - FLIGHT PATH ANGLE
C
      IF(N .EQ. NA)GO TO 772
      DO 770 I=NA+1,N
      IF(I .EQ. N)GO TO 760
      DTIME=TI(I+1)-TI(I-1)
      DALT=HC(I+1)-HC(I-1)
      HDOT(I)=DALT/DTIME
      FPANG(I)=ASIN(HDOT(I)/VTAS(I))
```

```
      FPANG(I)=FPANG(I)*57.29578
      AOA1(I)=(TH(I)-FPANG(I))/COS(BA(I)/57.29578)
      ERROR(I)=AOA1(I)-AOA(I)
760   WRITE(4,765)I,TI(I),HC(I),AMC(I),VTAS(I),HDOT(I),FPANG(I),
     1AOA1(I),AOA(I),ERROR(I)
765   FORMAT(1X,I10,9F10.3)
770   CONTINUE
C
C    THIS PORTION COMPUTES ANGLE-OF-ATTACK BY
C      COMPUTING ALPHA-DOT, THEN ADDING IT TO
C    THE PREVIOUS AOA TO COMPUTE A NEW AOA
C
772   DO 800 I=1,N
      AOA2(1)=AOA(1)
      AOA3(1)=AAOA1
      IF(I .EQ. 1)GO TO 775
      AVGZN=(ZN(I)+ZN(I-1))/2
      AVGBA=(BA(I)+BA(I-1))/2
      AVGTH=(TH(I)+TH(I-1))/2
      AVGQ=(Q(I)+Q(I-1))/2
      AVGTAS=(VTAS(I)+VTAS(I-1))/2
      AZ=AVGZN-(COS(AVGBA/57.29578)*COS(AVGTH/57.29578))
      ADOT(I)=AVGQ-(AZ*32.2*57.29578/AVGTAS)
      DTIME=TI(I)-TI(I-1)
      AOA2(I)=AOA(I-1)+(ADOT(I)*DTIME)
      AOA3(I)=AOA3(I-1)+(ADOT(I)*DTIME)
C
C    THIS PORTION CORRECTS AOA CALCULATED AT CG FOR
C      PITCH RATE:  AOA CG = AOA VANE + (Q * X / VTAS)
C    WHERE X IS THE DISTANCE BETWEEN THE VANE AND THE
C      ACCELEROMETER LOCATION ON THE AIRCRAFT
C
      X=17
      IF(ATYPE(1) .EQ. 'T')X=25
      AOAT=AOA3(I)-(Q(I)*X/VTAS(I))
      ALPHAK=AOA(I)/AOAT
      ERROR1(I)=AOA2(I)-AOA(I)
      ERROR2(I)=AOAT-AOA(I)
      ADOTM(I)=(AOA(I)-AOA(I-1))/DTIME
775   WRITE(6,780)I,TI(I),ADOT(I),ADOTM(I),AOA2(I),AOA(I),ERROR1(I),
     1AOA3(I),AOAT,ERROR2(I),ALPHAK
780   FORMAT(1X,I10,10F10.3)
800   CONTINUE
C
      STOP
      END
```

Appendix C

Computer Program KALOPT

```
      PROGRAM KALOPT
C
C     PROGRAM KALOPT IS A FORTRAN PROGRAM DESIGNED TO OPTIMIZE
C       ANGLE-OF-ATTACK USING KALMAN FILTER EQUATIONS.  THE
C     PROGRAM USES DATA PARAMETERS RECORDED INFLIGHT ON AN
C       AYDIN-VECTOR DATA ACQUISITION SYSTEM.  KALOPT USES TIME (TI),
C     IAS (AS), INDICATED ALT (ALT), MEASURED AOA (AOA), PITCH
C       ANGLE (TH), PITCH RATE (Q), NORMAL ACCELERATION (ZN), SIDESLIP
C     ANGLE (AOSS), AND BANK ANGLE (BA) OBTAINED FROM DYNAMICS EUP
C       AND EUS FILES.  AOAOPT CORRECTS FOR PITOT-STATIC ERRORS TO
C     DETERMINE PRESSURE ALTITUDE (HC), MACH NUMBER (AMC), AND
C       TRUE AIRSPEED (VTAS).  THE PROGRAM CALCULATES AN "OPTIMAL" TRUE
C     AOA BY COMBINING STATE ESTIMATION OF AOA AND PITCH ANGLE WITH
C       ACTUAL PITCH ANGLE MEASUREMENTS.  THE PROGRAM WEIGHS THE PITCH
C     ANGLE MEASUREMENT AND ADDS THE WEIGHTED VALUE TO THE ESTIMATED
C       AOA AND PITCH ANGLE.  THE UPDATED AOA IS THE "OPTIMAL" TRUE AOA.
C     THIS PROGRAM CAN BE USED WITH T-38A OR RF-4C DATA.
C
      BYTE PNAME(12),ENAME(12),DOF(7),ATYPE(5),ATAIL(3)
      DIMENSION TI(300),AS(300),ALT(300),AOA(300),TH(300),Q(300),
     1ZN(300),AOSS(300),BA(300),HC(300),AMC(300),VTAS(300)
      DIMENSION B(2,2),G(2,2),QA(2,2),P(2,2),AK(2,1),XA(2,1),BT(2,2),
     1U(2,1),XP(2,1),GT(2,2),QGT(2,2),GQGT(2,2),PP(2,2),H(1,2),
     2HT(2,1),PPHT(2,1),AKZMXH(2,1),HPP(1,2),AKHPP(2,2)
      BYTE FILE1(15),OFILE1(17)
      BYTE FILE2(15),OFILE2(17)
      BYTE FILE3(15),OFILE3(17)
      BYTE FILE4(15),OFILE4(17)
      BYTE FILE6(15),OFILE6(17)
      BYTE FILE7(15),OFILE7(17)
      BYTE FILE8(15),OFILE8(17)
      DATA FILE1/'E','S',':',8*0,'.','E','U','P'/
      DATA FILE2/'E','S',':',8*0,'.','E','U','S'/
      DATA FILE3/'E','S',':',8*0,'.','D','A','T'/
      DATA FILE4/'E','S',':',8*0,'.','D','A','T'/
      DATA FILE6/'E','S',':',8*0,'.','D','A','T'/
      DATA FILE7/'E','S',':',8*0,'.','D','A','T'/
      DATA FILE8/'E','S',':',8*0,'.','I','N','P'/
C
      WRITE (5,100)
100   FORMAT(' ENTER NAME OF DYNAMICS FILE: NNNNN.EUP',/)
      ACCEPT 110,(FILE1(I),I=4,11)
110   FORMAT (8A1)
      OPEN(UNIT=1,NAME=FILE1,TYPE='UNKNOWN')
      WRITE (5,120)
120   FORMAT(' ENTER NAME OF DYNAMICS FILE: NNNNN.EUS',/)
      ACCEPT 110,(FILE2(I),I=4,11)
      OPEN(UNIT=2,NAME=FILE2,TYPE='UNKNOWN')
      WRITE (5,125)
125   FORMAT(' ENTER NAME OF INPUT FILE: NNNNN.INP'/)
      ACCEPT 110,(FILE8(I),I=4,11)
      OPEN(UNIT=8,NAME=FILE8,TYPE='UNKNOWN')
      WRITE(5,130)
130   FORMAT(' ENTER NAME FOR DATA FILE 1:  NNNNNN.DAT',/)
      ACCEPT 110,(FILE3(I),I=4,11)
      OPEN(UNIT=3,NAME=FILE3,TYPE='UNKNOWN')
```

```
      WRITE(5,140)
140   FORMAT(' ENTER NAME FOR DATA FILE 2:  NNNNNN.DAT',/)
      ACCEPT 110,(FILE4(I),I=4,11)
      OPEN(UNIT=4,NAME=FILE4,TYPE='UNKNOWN')
      WRITE(5,150)
150   FORMAT(' ENTER NAME FOR DATA FILE 3:  NNNNNN.DAT',/)
      ACCEPT 110,(FILE6(I),I=4,11)
      OPEN(UNIT=6,NAME=FILE6,TYPE='UNKNOWN')
      WRITE(5,155)
155   FORMAT(' ENTER NAME FOR DATA FILE 4:  NNNNNN.DAT',/)
      ACCEPT 110,(FILE7(I),I=4,11)
      OPEN(UNIT=7,NAME=FILE7,TYPE='UNKNOWN')
      WRITE(5,160)
160   FORMAT (' ENTER NUMBER OF DATA POINTS DESIRED ' ,/)
      READ(5,170) N
170   FORMAT (I5)
      WRITE(5,180)
180   FORMAT (' ENTER FIRST LINE OF DATA DESIRED ' ,/)
      READ(5,170) M
      WRITE(5,190)
190   FORMAT(' ENTER NUMBER OF INITIAL CONSTANT AOA LINES ' ,/)
      READ(5,170)NA
      WRITE(5,200)
200   FORMAT(' ENTER CORRECTIONS FOR PITCH RATE AND NZ: XXX,YYY',/,
     1'     CORR ARE FROM GROUND BLOCK, ABOVE/BELOW 0 FOR Q',/,
     2'                                 ABOVE/BELOW 1 FOR NZ',/)
      READ(5,210)QCORR,ZNCORR
210   FORMAT(2F10.3)
C
C   THIS PORTION READS DATA FROM THE DYNAMICS EUP AND EUS FILES
C     AND FORMATS FOUR DATA FILES (XXXX.DAT) WHERE THE PROGRAM
C   RESULTS WILL BE SENT.  DATA FILE #1 RECORDS THE RAW DATA FROM
C     THE EUP AND EUS FILES.  DATA FILE #2 RECORDS THE OUTPUT OF
C   THE KALMAN FILTER PROGRAM:  "OPTIMAL" VALUES FOR AOA AND PITCH
C     ANGLE, AND THE ERROR BETWEEN THE "OPTIMAL" AOA AND MEASURED
C   AOA.  DATA FILE #3 RECORDS THE VALUES OF THE P(T)- AND K(T)
C     MATRICES.  DATA FILE #4 RECORDS THE VALUES OF THE P(T)+
C   MATRIX AND THE TRUE RMS ERROR.
C
      READ (1,220)PNAME,ENAME,DOF
220   FORMAT (8X,12A1,15X,12A1,21X,7A1)
      WRITE (3,230)PNAME,ENAME,DOF
      WRITE (4,230)PNAME,ENAME,DOF
      WRITE (6,230)PNAME,ENAME,DOF
      WRITE (7,230)PNAME,ENAME,DOF
230   FORMAT (' PILOT:',1X,12A1,5X,'ENGINEER:',1X,12A1,5X,
     1'DATE OF FLIGHT:',1X,7A1)
      READ (1,240)ATYPE,ATAIL
240   FORMAT (11X,5A1,13X,3A1)
      WRITE (3,250)ATYPE,ATAIL
      WRITE (4,250)ATYPE,ATAIL
      WRITE (6,250)ATYPE,ATAIL
      WRITE (7,250)ATYPE,ATAIL
250   FORMAT(' A/C TYPE:',1X,5A1,5X,'TAIL #:',1X,3A1)
      WRITE (3,260)
260   FORMAT (//,7X,'LINE',6X,'TIME',2X,'AIRSPEED',2X,'ALTITUDE',7X,
```

```
      1'AOA',5X,'PITCH',2X,'PITCH RT',8X,'NZ',6X,'AOSS',2X,'ROLL ANG')
       WRITE (4,270)
270    FORMAT (//,7X,'LINE',6X,'TIME',1X,'PRESS ALT',6X,'MACH',3X,'TRUE
      1 AS',5X,'PITCH',3X,'PITCH +',5X,'AOA +',5X,'Q*X/V',2X,'AOA-TRUE',
      22X,'AOA-VANE',5X,'ERROR',3X,'K-ALPHA')
       WRITE (6,280)
280    FORMAT (//,7X,'LINE',6X,'TIME',5X,'P-(1)',5X,'P-(2)',5X,'P-(3)',
      15X,'P-(4)',6X,'K(1)',6X,'K(2)')
       WRITE (7,285)
285    FORMAT (//,7X,'LINE',6X,'TIME',5X,'P+(1)',5X,'P+(2)',5X,'P+(3)',
      15X,'P+(4)',4X,'ZABOVE',4X,'ZBELOW',4X,'ZA-TOT',4X,'ZB-TOT',6X,
      2'DIFF',7X,'SUM',5X,'PSQRT')
       WRITE (3,290)
290    FORMAT (16X,'(SEC)',3X,'(KNOTS)',4X,'(FEET)',5X,'(DEG)',
      15X,'(DEG)',1X,'(DEG/SEC)',7X,'(G)',5X,'(DEG)',5X,'(DEG)',/)
       WRITE (4,300)
300    FORMAT (16X,'(SEC)',4X,'(FEET)',12X,'(FT/SEC)',5X,
      1'(DEG)',5X,'(DEG)',5X,'(DEG)',5X,'(DEG)',5X,'(DEG)',
      25X,'(DEG)',5X,'(DEG)',/)
       WRITE(6,310)
310    FORMAT (16X,'(SEC)',/)
       WRITE(7,310)
       READ (1,320)
320    FORMAT (/,/,/)
       READ (2,330)
330    FORMAT (/,/,/,/,/)
       DO 350 J=1,M
       READ(1,340)
       READ(2,340)
340    FORMAT ( )
350    CONTINUE
       DO 390 I=1,N
       READ(1,360)TI(I),AS(I),ALT(I),AOA(I),TH(I),Q(I),ZN(I)
360    FORMAT (31X,F10.3,F10.3,F10.3,F10.3,F10.3,F10.3,20X,F10.3)
       READ(2,370)AOSS(I),BA(I)
370    FORMAT (31X,F10.3,10X,F10.3)
C
C    THE NEXT TWO STEPS CORRECT PITCH RATE AND NORMAL G FOR BIAS
C      FOUND ON THE GROUND (BIAS COMPUTED FROM 0 DEG/SEC FOR PITCH
C    RATE AND 1 G FOR NORMAL G)
C
       Q(I)=Q(I)-QCORR
       ZN(I)=ZN(I)-ZNCORR
       WRITE(3,380)I,TI(I),AS(I),ALT(I),AOA(I),TH(I),Q(I),ZN(I),AOSS(I),
      1BA(I)
380    FORMAT(1X,I10,9F10.3)
390    CONTINUE
C
C    THIS PORTION PERFORMS THE PITOT-STATIC CORRECTIONS TO COMPUTE
C      PRESSURE ALT, MACH NUMBER, AND TRUE AIRSPEED.  IT USES THE
C    STANDARD PITOT-STATIC EQUATIONS AND ERROR COEFFICIENTS FROM THE
C      USAF TEST PILOT SCHOOL FILES.
C
       DO 710 I=1,N
       IF(ALT(I) .GT. 36089)GO TO 500
       DELTA=(1-(6.87559E-6*ALT(I)))**5.2561
```

```
      THETA=1-(6.87559E-6*ALT(I))
      GO TO 510
500   DELTA=.22337*EXP(-4.80635E-5*(ALT(I)-36089))
      THETA=.751874
510   IF(AS(I) .GT. 661.48)GO TO 520
      QCICPA=((1+(.2*((AS(I)/661.48)**2)))**3.5)-1
      GO TO 530
520   QCICPA=((166.922*((AS(I)/661.48)**7))/(((7*((AS(I)/661.48)**2))
     1-1)**2.5))-1
530   QCICPS=QCICPA/DELTA
      AMIC=SQRT(5*(((QCICPS+1)**(.2857143))-1))
      IF(ATYPE(1) .EQ. 'T')GO TO 600
C
C    POSITION ERROR COEFFICIENTS COME FROM THE
C      USAF TEST PILOT SCHOOL PITOT-STATIC FILES
C
      DHPC=(-907+(10270*AMIC)+(-44495*(AMIC**2))+(93931*(AMIC**3))
     1+(-95540*(AMIC**4))+(37208*(AMIC**5)))*THETA
      IF(AMIC .GT. 1)DHPC=0
      GO TO 700
600   IF(AMIC .LT. .955)GO TO 620
      IF(AMIC .LT. .967)GO TO 640
      IF(AMIC .LT. 1.025)GO TO 660
      C0=-46325
      C1=82583
      C2=-36667
      GO TO 680
620   C0=118
      C1=-478
      C2=912
      GO TO 680
640   C0=2676675
      C1=-5636789
      C2=2968036
      GO TO 680
660   C0=-141471
      C1=308123
      C2=-166107
      GO TO 680
680   DHPC=(C0+(C1*AMIC)+(C2*(AMIC**2)))*THETA
700   HC(I)=ALT(I)+DHPC
      DPPPS=(3.61382E-5*DHPC)/THETA
      DMPC=((1+(.2*(AMIC**2)))*DPPPS)/(1.4*AMIC)
      AMC(I)=AMIC+DMPC
      VTAS(I)=1.6878*AMC(I)*38.96763*SQRT(THETA*288.15)
710   CONTINUE
C
C    THIS PORTION COMPUTES TRUE AOA BY THE RELATIONSHIP: TRUE AOA =
C      PITCH ANGLE - FLIGHT PATH ANGLE.  FLIGHT PATH ANGLE IS
C    COMPUTED BY:  FLIGHT PATH ANGLE (FPANG) = INV SIN [VERTICAL
C      VEL (HDOT) / TRUE AIRSPEED (VTAS)].  THE TRUE AOA IS THEN
C    AVERAGED OVER A TIME INTERVAL TO USE AS AOA(0) IN THE STATE
C      ESTIMATION.  THE DATA USED HERE IS UNFILTERED.
C
      IF(NA .EQ. 0)AAOA=AOA(1)
      IF(NA .EQ. 0)GO TO 790
```

```
      DO 750 I=2,NA-1
      DTIME=TI(I+1)-TI(I-1)
      DALT=HC(I+1)-HC(I-1)
      HDOT=DALT/DTIME
      FPANG=ASIN(HDOT/VTAS(I))
      FPANG=FPANG*57.29578
      AOA1=TH(I)-FPANG
C
C    THIS PORTION AVERAGES THE CALCULATED AOA TO
C      USE AS AOA(0) IN THE STATE ESTIMATION
C
      TTIME=TTIME+DTIME
      TAOA=TAOA+(AOA1*DTIME)
750   CONTINUE
      AAOA=TAOA/TTIME
C
C    THIS SECTION PRINTS OUT THE INITIAL LINE OF DATA
C
790   WRITE(4,800)TI(1),HC(1),AMC(1),VTAS(1),TH(1),TH(1),AAOA,AOA(1)
800   FORMAT(10X,'1',7F10.3,20X,F10.3)
      WRITE(6,810)TI(1)
810   FORMAT(10X,'1',F10.3)
      WRITE(7,810)TI(1)
C
C    THIS PORTION READS IN THE VALUES FOR THE MATRICES P(T0), Q(T),
C      AND R(T) [CALLED P,QA,AND R] FROM AN INPUT FILE CALLED XXXX.INP
C
      DO 910 I=1,2
      READ(8,900)P(I,1),P(I,2)
900   FORMAT(11X,F10.3,10X,F10.3)
910   CONTINUE
      DO 930 I=1,2
      READ(8,920)QA(I,1),QA(I,2)
920   FORMAT(11X,F10.3,10X,F10.3)
930   CONTINUE
      READ(8,940)R
940   FORMAT(11X,F10.3)
C
C    THIS PORTION INITIALIZES THE STATE MATRIX X(T0) AND THE
C      MATRICES B(T), G(T), H(T), AND U(T).
C
      XA(1,1)=AAOA
      XA(2,1)=TH(1)
      DO 2000 I=2,N
      T=TI(I)-TI(I-1)
      B(1,1)=1
      B(1,2)=(-32.2)*57.29578/VTAS(I)
      B(2,1)=1
      G(1,1)=1
      G(1,2)=(-32.2)*57.29578/VTAS(I)
      G(2,1)=1
      H(1,2)=1
      U(1,1)=Q(I)
      U(2,1)=ZN(I)-COS(TH(I)/57.29578)
C
C    THIS PORTION PROPOGATES THE STATE MATRIX FROM THE LAST UPDATE
```

```
C     X(T+) TO THE POINT PRIOR TO THIS UPDATE X(T-)
C
      DO 1020 J=1,2
      DO 1010 K=1,2
      BT(J,K)=B(J,K)*T
1010  CONTINUE
1020  CONTINUE
      DO 1030 J=1,2
      XP(J,1)=XA(J,1)+(BT(J,1)*U(1,1))+(BT(J,2)*U(2,1))
1030  CONTINUE
C
C   THIS PORTION PROPOGATES THE COVARIANCE MATRIX P(T) FROM THE
C     LAST UPDATE P(T+) TO THE POINT PRIOR TO THIS UPDATE, P(T-)
C
      DO 1050 J=1,2
      DO 1040 K=1,2
      GT(K,J)=G(J,K)
1040  CONTINUE
1050  CONTINUE
      DO 1070 J=1,2
      DO 1060 K=1,2
      QGT(K,J)=(QA(K,1)*GT(1,J))+(QA(K,2)*GT(2,J))
1060  CONTINUE
1070  CONTINUE
      DO 1090 J=1,2
      DO 1080 K=1,2
      GQGT(J,K)=(G(J,1)*QGT(1,K))+(G(J,2)*QGT(2,K))
1080  CONTINUE
1090  CONTINUE
      DO 1150 J=1,2
      DO 1140 K=1,2
      PP(J,K)=P(J,K)+(T*GQGT(J,K))
1140  CONTINUE
1150  CONTINUE
C
C   THIS PORTION COMPUTES THE GAIN MATRIX, K(T)
C
      DO 1210 J=1,2
      HT(J,1)=H(1,J)
1210  CONTINUE
      DO 1230 J=1,2
      PPHT(J,1)=(PP(J,1)*HT(1,1))+(PP(J,2)*HT(2,1))
1230  CONTINUE
      HPPHT=(H(1,1)*PPHT(1,1))+(H(1,2)*PPHT(2,1))
      HPPHTR=HPPHT+R
      DO 1270 J=1,2
      AK(J,1)=PPHT(J,1)/HPPHTR
1270  CONTINUE
C
C   THIS PORTION PERFORMS THE UPDATE OF THE STATE MATRIX, X(T),
C     TO TAKE IT FROM X(T-) TO X(T+) BY WEIGHING THE PITCH ANGLE
C   MEASUREMENT (Z) AND ADDING IT TO X(T-)
C
      Z=TH(I)
      HXP=(H(1,1)*XP(1,1))+(H(1,2)*XP(2,1))
      ZMXHP=Z-HXP
```

```
      DO 1320 J=1,2
      AKZMXH(J,1)=(AK(J,1)*ZMXHP)
      XA(J,1)=XP(J,1)+AKZMXH(J,1)
1320  CONTINUE
C
C   THIS PORTION UPDATES P(T) FROM P(T-) TO P(T+)
C
      DO 1400 K=1,2
      HPP(1,K)=(H(1,1)*PP(1,K))+(H(1,2)*PP(2,K))
1400  CONTINUE
      DO 1430 J=1,2
      DO 1420 K=1,2
      AKHPP(J,K)=(AK(J,1)*HPP(1,K))
      P(J,K)=PP(J,K)-AKHPP(J,K)
1420  CONTINUE
1430  CONTINUE
C
C   THIS PORTION CORRECTS AOA CALCULATED AT CG FOR
C     PITCH RATE:  AOA CG = AOA VANE + (Q * X / VTAS)
C   WHERE X IS THE DISTANCE BETWEEN THE VANE AND THE
C     ACCELEROMETER LOCATION ON THE AIRCRAFT
C
      X=17
      IF(ATYPE(1) .EQ. 'T')X=25
      AOAT=XA(1,1)-(Q(I)*X/VTAS(I))
      QXV=Q(I)*X/VTAS(I)
C
C   THIS PORTION COMPUTES THE SQUARE OF THE ERROR BETWEEN
C     PITCH ANGLE-CALC AND PITCH ANGLE-MEAS TO COMPUTE THE
C   TRUE RMS ERROR
C
      ZA=XA(2,1)-TH(I)
      IF(ZA .LT. 0)GO TO 1500
      ZA=ZA**2
      ZB=0
      ZAT=ZAT+ZA
      GO TO 1510
1500  ZB=ZA**2
      ZA=0
      ZBT=ZBT+ZB
1510  DIFF=ZAT-ZBT
      SUM=(ZA+ZB)**.5
      PSQRT=P(2,2)**.5
C
C   THIS PORTION COMPUTES THE ERROR BETWEEN THE "OPTIMAL" TRUE AOA
C     AND THE MEASURED AOA AND THEN PRINTS OUT THE RESULTS OF
C   THE KALMAN FILTER PROGRAM.
C
      ERROR=AOAT-AOA(I)
      ALPHAK=AOA(I)/AOAT
      WRITE(4,1800)I,TI(I),HC(I),AMC(I),VTAS(I),TH(I),
     1XA(2,1),XA(1,1),QXV,AOAT,AOA(I),ERROR,ALPHAK
1800  FORMAT(1X,I10,12F10.3)
      WRITE(6,1810)I,TI(I),PP(1,1),PP(1,2),PP(2,1),PP(2,2),
     1AK(1,1),AK(2,1)
1810  FORMAT(1X,I10,7F10.3)
```

```
      WRITE(7,1820)I,TI(I),P(1,1),P(1,2),P(2,1),P(2,2),
     1ZA,ZB,ZAT,ZBT,DIFF,SUM,PSQRT
1820  FORMAT(1X,I10,12F10.3)
2000  CONTINUE
      STOP
      END
```

Appendix D

Flight Test Summary

TABLE IV

T-38A Flight Test Points[1]

Cruise Configuration — No External Stores
Wings Level — 3 seconds per g min

Test Point	Altitude (feet)	Mach	Min G	Max G
1	25,000	0.8	-1	+4
2	25,000	0.9	-1	+6
3	25,000	0.95	-1	+6
4	25,000	1.05	-1	+6
5	25,000	0.6	-1	+2
6	25,000	0.45	-1	+1
7	15,000	0.8	-1	+6
8	15,000	0.9	-1	+6
9	15,000	0.95	-1	+6
10	15,000	1.05	-1	+6
11	15,000	0.6	-1	+4
12	15,000	0.45	-1	+2

[1] (11:3)

TABLE V

Summary of T-38A Test Flights

T-38A 68-8205 Edwards AFB, Cal.

Date of Flight	Pilot	Engineer	Flight Time	Test Points Flown
19 Jul 85	Eichhorn	Thacker	0.7 hrs	None - Bad Wx
23 Jul 85	Eichhorn	Thacker	1.0 hrs	1 through 12
26 Jul 85	Arnold	Thacker	0.8 hrs	1 through 6

Total Flight Time - 2.5 hrs

Appendix E

T-38A YAPS Noseboom Diagram

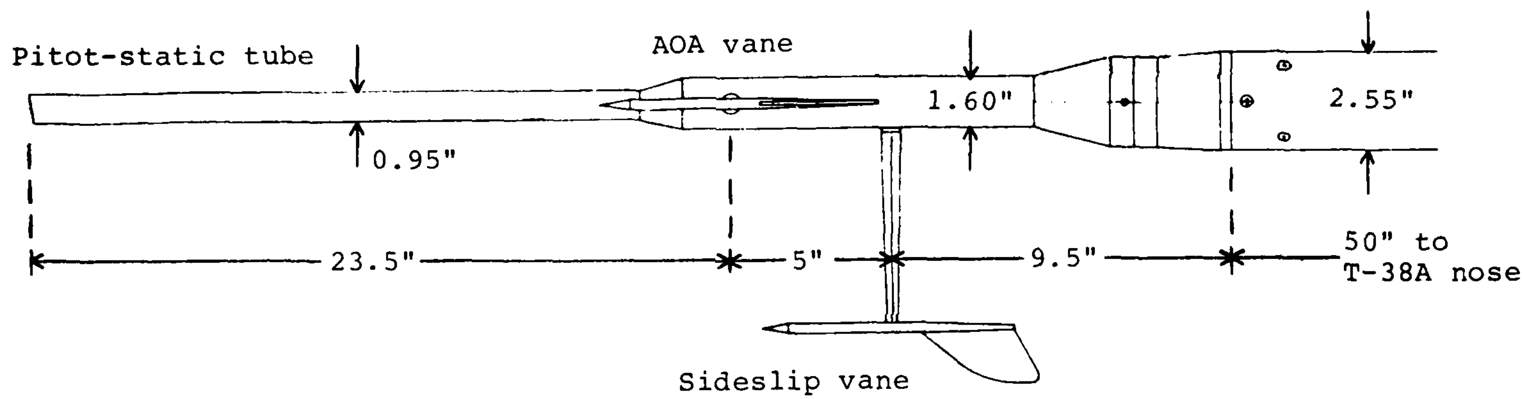


Figure 30. Vought Yaw and Pitch System Noseboom Diagram (Side View)

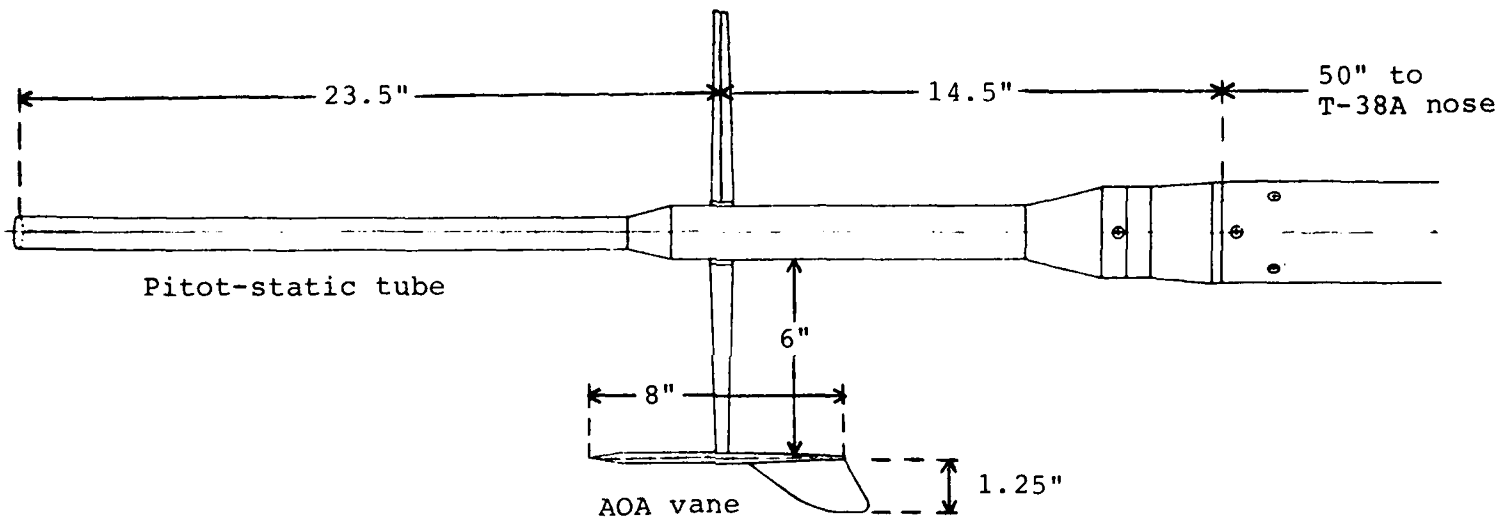


Figure 31. Vought Yaw and Pitch System Noseboom Diagram (Top View)

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AD-A163 962 USE OF STATE ESTIMATION TO CALCULATE ANGLE-OF-ATTACK POSITION ERROR FROM.. (U) AIR FORCE INST OF TECH WRIGHT-PATTERSON AFB OH SCHOOL OF ENGI.. T H THACKER 2/2

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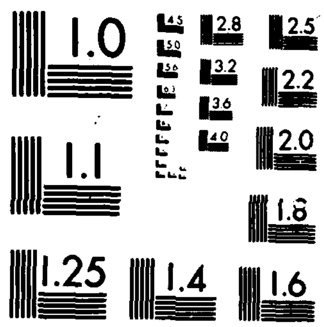

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Vita

Captain Thomas H. Thacker was born on 24 December 1955 in Casablanca, Morocco. He graduated from Beavercreek High School in Dayton, Ohio in 1973. Captain Thacker was a distinguished graduate of the United States Air Force Academy in 1978. He received the degree of Bachelor of Science in Aeronautical Engineering and a regular commission in the USAF from the Academy. He attended undergraduate navigator training and received his wings in February, 1979. He was a distinguished graduate and received the ATC Commander's Trophy as the top graduate in his class. Captain Thacker served as an F-111 instructor weapons systems officer in the 20th Tactical Fighter Wing, RAF Upper Heyford, United Kingdom, from November 1979 to March 1983. He graduated from the F-111 Fighter Weapons School in March, 1982. Captain Thacker was selected for the combined Air Force Institute of Technology/ Test Pilot School program in 1983. He attended the AFIT School of Engineering from June 1983 to June 1984. He graduated from the USAF Test Pilot School in June 1985 as a distinguished graduate and received the R. L. Jones Award as the top flight test engineer/navigator in his class. Captain Thacker is currently assigned with the 3247th Test Squadron, Eglin AFB, Florida.

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19. ABSTRACT (Continue on reverse if necessary and identify by block number)

This project determined the position errors of an aircraft's angle-of-attack (AOA) sensor using state estimation with flight test data. The position errors were caused by local flow and upwash and were found to be a function of AOA and Mach number. The test aircraft used in this project was a T-38A Talon supersonic trainer from the USAF Test Pilot School configured with a Vought yaw and pitch system noseboom and an internal Aydin-Vector data acquisition system (DAS).

The position errors were found by calculating the true AOA using equations of motion and DAS parameters. The data from the DAS were noise corrupted and had to be filtered. This was accomplished using state estimation in a Kalman filter. The estimated AOA was compared to the measured AOA from the noseboom sensor to obtain the position error. Accurate position errors were obtained, even in dynamic maneuvers. This method should be considered in future AOA (reverse)

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Block 11. (Cont) ANGLE-OF-ATTACK POSITION ERROR FROM FLIGHT TEST DATA

Block 19. (Cont) error testing. The method was accurate enough to identify a hysteresis error in the T-38A's AOA sensor of +/- 0.5 degrees, which was confirmed by ground calibration.

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