FINAL REPORT

V/STOL TILT ROTOR STUDY - VOLUME VI HOVER, LOW SPEED AND CONVERSION TESTS OF A TILT ROTOR AEROELASTIC MODEL

MODEL 300

REPORT 301-099-002

NASA CONTRACT NAS 2-6599

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V/STOL TILT ROTOR STUDY - VOLUME VI
HOVER, LOW SPEED AND CONVERSION TESTS OF A
TILT ROTOR AEROELASTIC MODEL

by

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Ames Research Center
and

United States Army Air Mobility Research and
Development Laboratory, Ames Directorate

Foreword

This report is one of a series prepared by the Bell Helicopter Company, Fort Worth, Texas, for the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California, under Contract NAS2-6599. These Tilt Rotor Research Aircraft studies were jointly funded by NASA and the U. S. Army Air Mobility Research and Development Laboratory, Ames Directorate.

The Administrative Contracting Officer was Mr. Richard J. Abbott. The Technical Monitor was Mr. Martin D. Maisel, Tilt Rotor Research Aircraft Project Office. Mr. Gary B. Churchill, Tilt Rotor Research Aircraft Project provided technical support to the effort reported in Volume V.

Volume I   -- V/STOL Tilt Rotor Study - Conceptual Design -- CR114441

Volume II   -- V/STOL Tilt Rotor Study - Research Aircraft Design -- CR114442

Volume III -- V/STOL Tilt Rotor Study - Research Aircraft Project Plan -- CR114443

Volume IV   -- V/STOL Tilt Rotor Study - Wind Tunnel Investigation Plan - CR114444

Volume V   -- V/STOL Tilt Rotor Study - A Mathematical Model for Real Time Flight Simulation of the Bell Model 301 Tilt Rotor Research Aircraft -- CR114614

Volume VI   -- V/STOL Tilt Rotor Study - Hover, Low Speed and Conversion Tests of a Tilt Rotor Aeroelastic Model -- CR114615

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Symbol	Description	Units
a_1	F/A Flapping	deg
a_H	Horizontal Stabilizer Lift Curve Slope	1/deg
b	Wing Span	ft
b_1	Lateral Flapping	deg
B_{1S}	F/A Cyclic, ref. to shaft	deg
c	Wing chord	ft
D	Rotor diameter	ft
GW	Gross Weight	lb
FS	Fuselage Station	in
h	height of rotor disc above the ground	ft
h/D	height to diameter ratio	–
I_{XX}	Aircraft rolling inertia	$slug\text{-}ft^2$
I_{XZ}	Aircraft product of inertia	$slug\text{-}ft^2$
I_{YY}	Aircraft pitching inertia	$slug\text{-}ft^2$
I_{ZZ}	Aircraft yawing inertia	$slug\text{-}ft^2$
IGE	In-ground-effect	–
iH	Horizontal Stabilizer Incidence, +L. E. up	deg
$K_{R/W}$	Induced velocity constant at wing due to rotor	–
$K_{R/H}$	Induced velocity constant at hor. stab. due to rotor	–
L_H	Lift of horizontal stabilizer	lb
PM	Pitching Moment	ft-lb
q	Free stream dynamic pressure	lb/ft^2
Q	Rotor Torque (per rotor)	in-lb
RM	Rolling Moment	ft-lb
R_N	Reynolds Number	–
RPM	Rotor RPM	RPM
S	Area, W-wing, H-horizontal	ft^2
SF	Side Force	lb
T	Rotor Thrust	lb

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List of Symbols

Symbol	Description	Units
V	Full scale aircraft airspeed	kt
V_{iR}	Induced velocity at rotor disk	ft/sec
YM	Yawing Moment	ft-lb
WL	Waterline location	in
α_F	Fuselage angle of attack	deg
α_H	Angle of attack at horizontal stab.	deg
α_m	Mast angle (ref. from airplane mode)	deg
α_{TRIM}	Fuselage angle of attack for trimmed flight	deg
β	Sideslip angle	deg
β_m	Mast angle ($90 - \alpha_m$)	deg
δ_e	Elevator deflection angle, + dn	deg
δ_F	Flapperon deflection angle, + dn	deg
$\epsilon_{W/H}$	Wing wake deflection at horiz. stab.	deg
$\epsilon_{R/H}$	Rotor wake deflection at horiz. stab.	deg
ϵ_T	Wing + Rotor wake deflection at horiz. stab.	deg
$\theta_{3/4r}$	Rotor blade collective pitch setting at three quarter radius	deg
Ψ	Fuselage yaw angle, + nose right	deg
τ_e	Elevator effectiveness	
η_{HWB}	Dynamic pressure ratio at the horizontal stabilizer due to wing-body interference	
η_{HT}	Dynamic pressure ratio at the horizontal stabilizer due to wing-body-rotor interference	
$\partial\sigma/\partial\beta$	Sidewash factor	
C_L	Aircraft Lift Coefficient, L/qS	
C_m	Aircraft Pitching Moment Coefficient, PM/qSc	
$C_{m_{i_H}}$	Aircraft Pitching Moment Coefficient, due to horizontal stabilizer incidence	
C_{m_H}	Aircraft Pitching Moment Coefficient, due to horizontal stabilizer	

List of Symbols

Symbol	Description	Units
$C_{m_{\delta_e}}$	Aircraft Pitching Moment Coefficient, due to elevator	
C_ℓ	Aircraft Rolling Moment Coefficient, RM/qSb	
C_n	Aircraft Yawing Moment Coefficient, YM/qSb	
C_y	Aircraft Side Force Coefficient, SF/qS	

Subscripts

e	elevator
F	Fuselage
H	Horizontal Stabilizer
R	Rotor
R/H	Due to rotor at horizontal stabilizer
T	Total, includes wing wake and rotor wake
V	Vertical stabilizer
WB	Wing-Fuselage
W/H	Due to wing at horizontal stabilizer

I. SUMMARY

Two series of tests evaluated the stability and control characteristics of a one-fifth scale powered aeroelastic model of the Model 300 Tilt Rotor Research Aircraft in hover, low speed helicopter, and conversion flight. Hover tests took place in the Bell Helicopter Company rotor whirl cage from September 13, 1972 to December 22, 1972. Helicopter and conversion mode tests took place in the Vought low speed wind tunnel from January 15, 1973 to March 30, 1973. This testing was performed under NASA Contract NAS 2-6599.[1]

The hover testing was directed at measuring hover performance and roll static stability characteristics. During part of this testing, the model was secured to a mount which simulated free flight to investigate stability characteristics and control response.

The helicopter and conversion mode testing was in two phases, with a different mount for each. The first phase had the model mounted on the vertical rod shown in Figure I-1. This mount gave the model freedom to pitch, roll, and yaw and to translate vertically. Model "pilots" could adjust cyclic pitch, collective pitch, and elevator position to make the model fly in a trimmed flight condition. Trimmed flight conditions tested included level flight (OGE and IGE), climbs, descents, autorotation, and rearward and sideward flight. At selected flight conditions the model was disturbed from trim to investigate dynamic stability characteristics.

The second phase had the model mounted on a sting support, as shown in Figure I-2. Forces and moments were recorded with the model in the trim attitudes and its controls in the positions established during the first phase. Pitch and yaw sweeps about the trim condition determined the static stability characteristics. Tail-on, tail-off, rotors-on, and rotors-off configurations were tested to determine the influence of the rotor wake on the empennage.

The principal results of the tests are as follows:

(1) The hover test confirmed the presence of a static roll instability in an IGE hover. Figure I-3 shows the roll instability in terms of inches of lateral stick required to balance the unstable rolling moment. The roll instability vanishes at airspeeds above approximately 20 knots.

(2) The hover test showed the wing download to be greater than that estimated from previous test results. The measured download was 11.4 percent, compared to the 7

(2) Continued

percent estimated for the full-scale aircraft.
Analysis indicates this discrepancy may be caused
by the model's very low Reynolds number.

(3) Flow visualization techniques confirmed that the
wakes from the two rotors do not merge in the flight
conditions tested, and an analysis of empennage
balance data indicates that the wakes produce a net
upwash at the horizontal stabilizer. The upwash is
greater IGE than OGE. Vortex lift produced by the
rolled up rotor wakes is a significant factor in the
effect of the rotor wake on the empennage.

(4) At mast angles of 90 and 75 degrees, the longitudinal
stick gradient is shallow between hover and 60 knots
with a slight stick reversal occurring between 20 and
40 knots (see Figure I-4). The shallow gradient and
reversal are caused by the rotor wake upwash on the
horizontal stabilizer.

(5) The interaction between the rotor wake and the hori-
zontal stabilizer causes a nose-up pitching moment when
the aircraft is yawed. The longitudinal cyclic required
to compensate for the pitching moment, shown in Figure
I-5, is easily within a pilot's ability. On the Model
301 aircraft, with the SCAS on, the attitude-hold loop
will make the longitudinal cyclic input.

(6) At mast angles 90, 75 and 60 degrees, the rotor wake
acts on the vertical fins in such a manner as to reduce
directional stability for sideslip angles less than
about 12 degrees and to increase it for larger sideslip
angles. At mast angles 90 and 75 degrees and at speed
less than 60 knots, the rotor wake causes this air-
craft to be directionally unstable between ±4 degrees
of sideslip.

(7) In autorotation at the speeds tested (80 and 90 knots),
the model was very stable but had a rate of descent in
excess of 4000 fpm, compared to the 2200 fpm predicted
for the Model 301. Analysis indicates that the model's
low Reynolds number is responsible for its higher rates
of descent. The rate of descent predicted for the
Model 301 is still considered to be correct.

(8) On the rod the model was difficult to fly at speeds
below 30 knots, but was controllable. Above 30 knots,
the model was relatively easy to fly. Controllability
was adequate for rearward and sideward flight at
speeds up to 35 knots. It was not possible to fly the

(8) Continued

model in a sustained controlled hover in the hover test rig. Failure to accomplish this task is attributed to the faster response characteristics of the model (2.24 times full scale) and because of the lack of many of the normal pilot cues. It should be noted that the model was not equipped with a stability augmentation system (SAS).

(9) Correlation between theory[2] and measured trim attitudes and control positions is generally good when Reynolds numbers effects on the model are accounted for.

(10) Lateral flapping in helicopter mode was approximately 50 percent higher than predicted, as shown in Figure I-6, but was within flapping limits for all conditions tested. Analysis shows that the computer representation of the longitudinal distribution of rotor induced velocity was not representative of the actual longitudinal distribution. By modifying the computer representation good correlation with the measured flapping was obtained. The modified method is satisfactory for design purposes.

(11) There was no evidence of rotor or rotor-pylon-wing instability during the tests.

(12) Scaled rotor and control system loads were significantly lower than those predicted for the full-scale aircraft.

(13) Airframe vibration levels were higher than predicted but within design limits.

Note: In referring to the Model 301 throughout the report, the Model 301 is of similar configuration as the Model 300. The Model 300 data is therefore considered applicable to the Model 301.
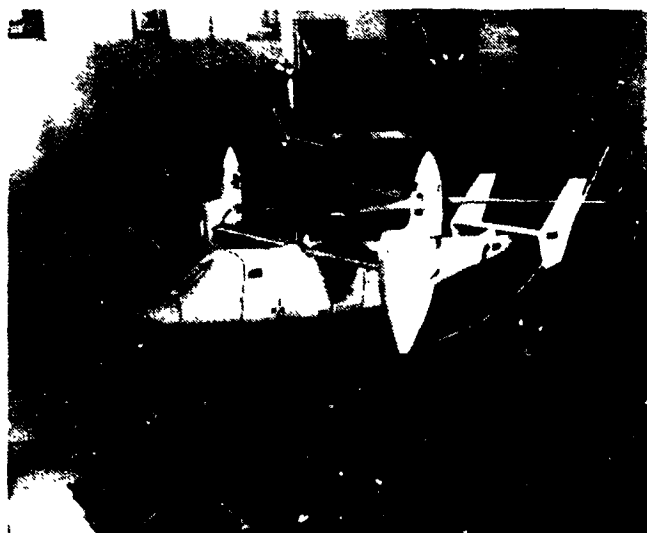
Figure I-1.   Powered Aeroelastic Model on Vertical Rod
              Mount, LTV LSWT Test 418, January 1973.
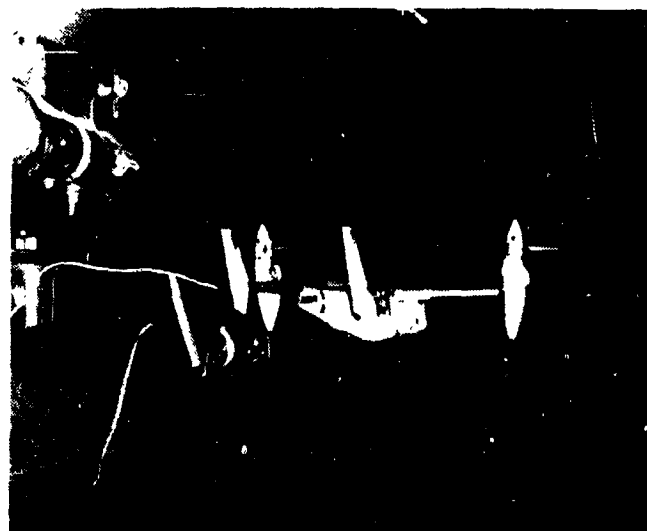
Figure I-2.   Powered Aeroelastic Model on Sting Mount,
              LTV LSWT Test 421 March 1973.

HOVER TEST
MAST ANGLE ≈ 90°
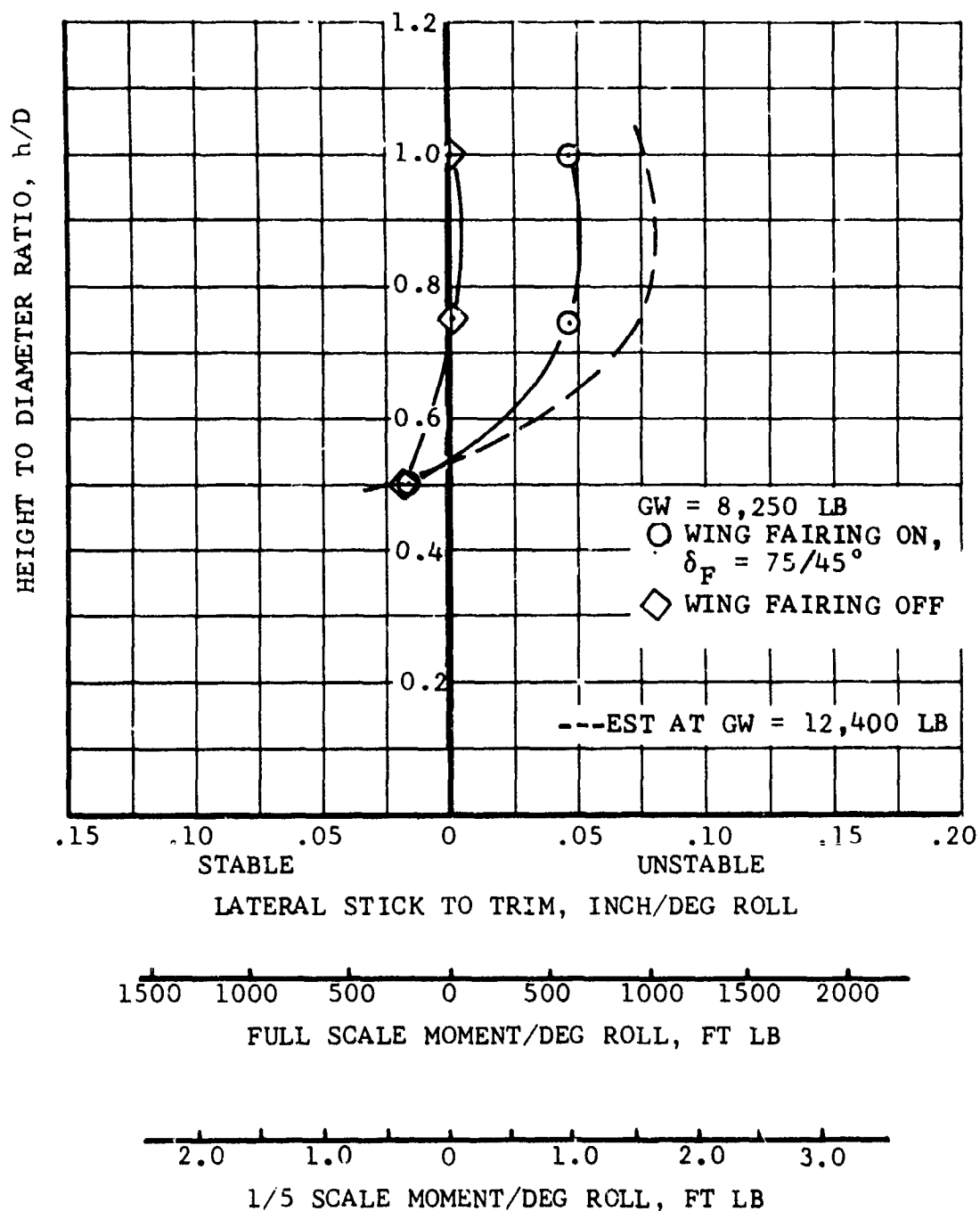
Figure I-3. Summary of Roll Stability Characteristics
in IGE Hover.

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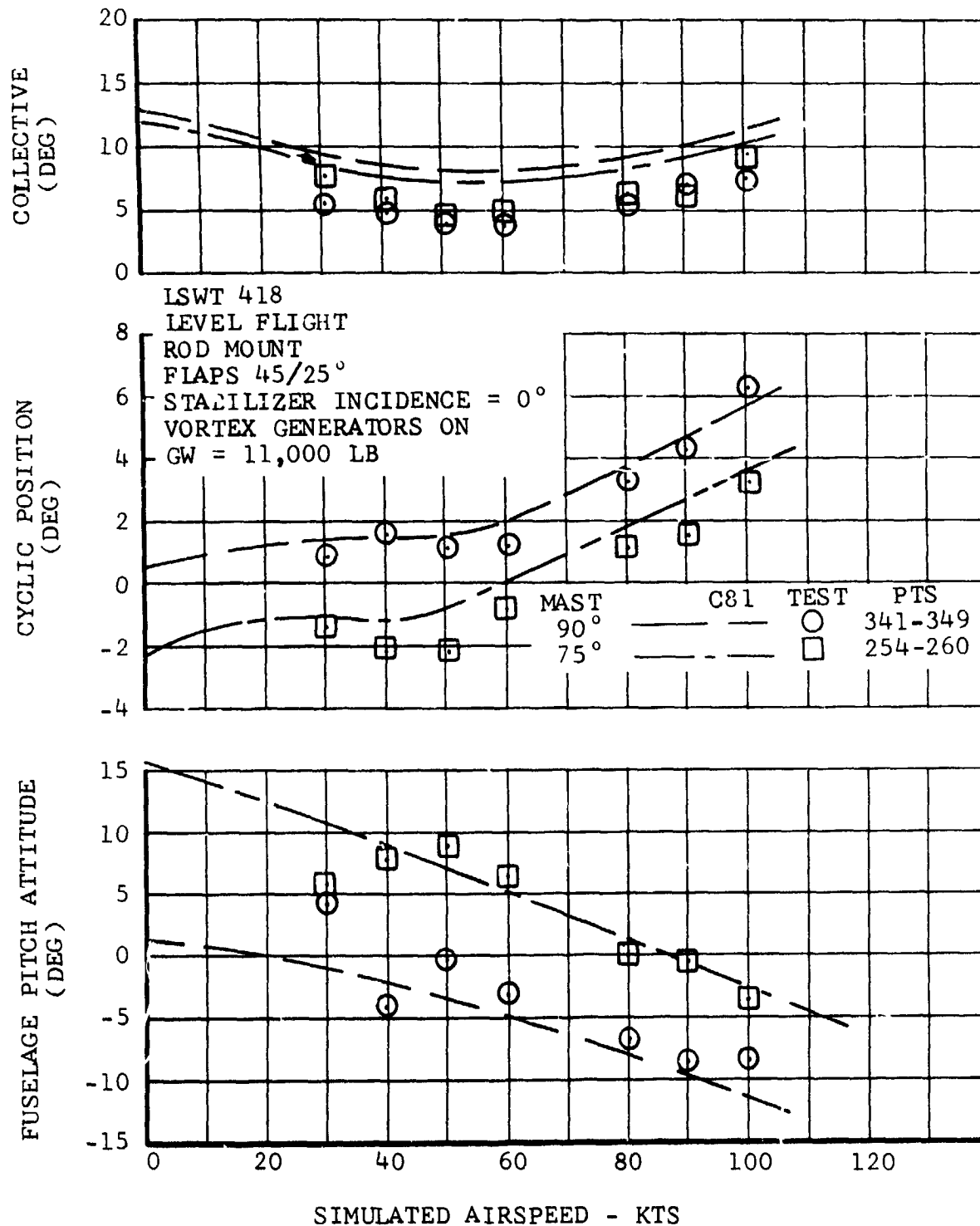
Figure I-4. Level Flight Trim Attitude and Control Positions
Versus Airspeed, Mast Angles 90° and 75°.

LSWT 418
YAW SWEEP
ROD MOUNT
FLAPS = 45/25°
STABILIZER INCIDENCE = 0°
VORTEX GENERATORS ON
GW = 11,000 LB

	MAST	PTS
O	90°	370-372
□	75°	268-273

AIRSPEED 40 KTS
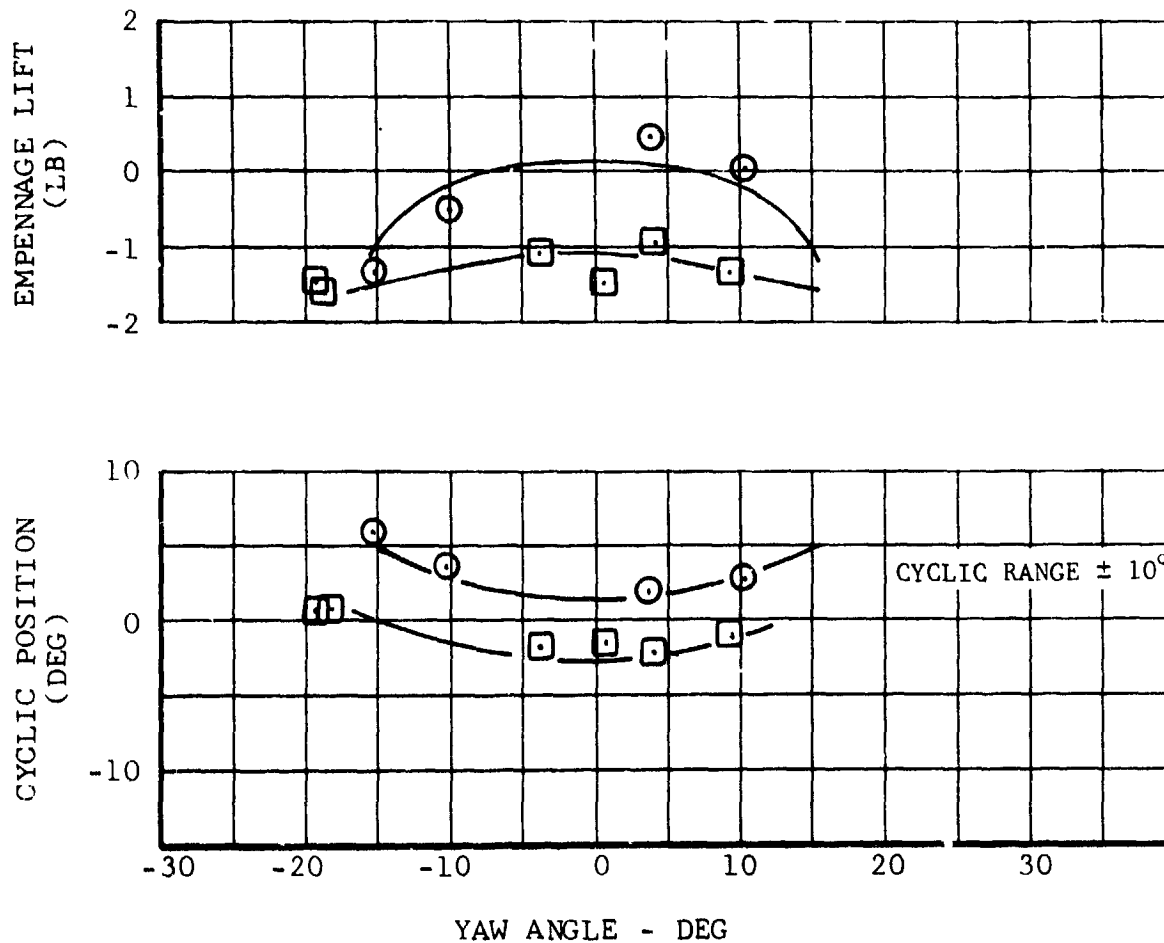
Figure I-5. Level Flight Trim Longitudinal Cyclic Versus
Yaw Angle, Mast Angles 90° and 75°.

LSWT 418
LEVEL FLIGHT
ROD MOUNT
FLAPS 45/25°
STABILIZER INCIDENCE = 0°
VORTEX GENERATORS ON
GW = 11,000 LB
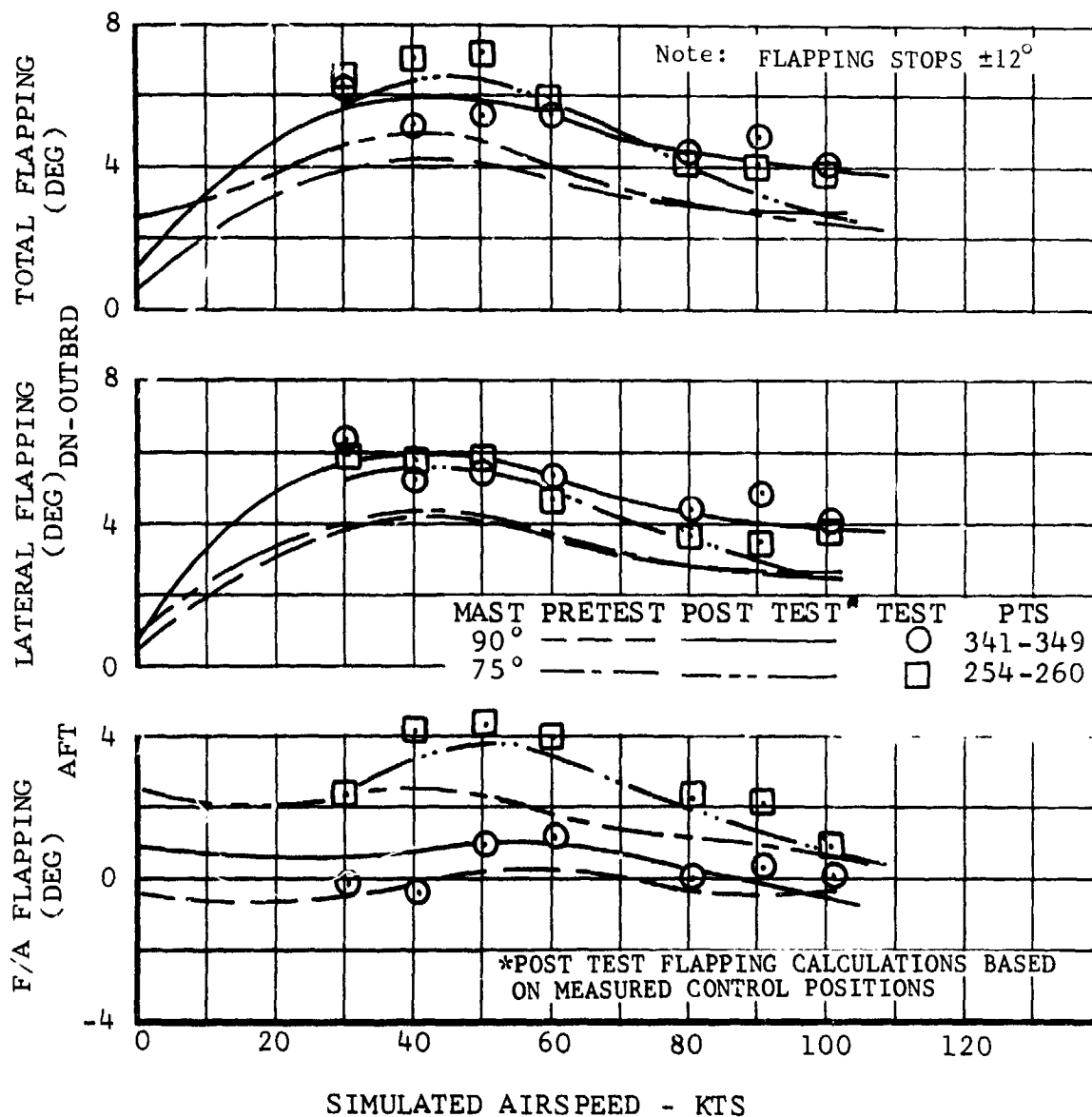
Figure I-6. Correlation Between Calculated and Measured Flapping, Mast Angles 90° and 75°.

II.   INTRODUCTION

This report presents the results of an investigation into the
hover, low speed helicopter and conversion flight characteristics
of the Model 301 Tilt Rotor Research Aircraft. A one-fifth scale,
powered aeroelastic model of the Bell Model 300 tilt rotor aircraft
was tested to determine performance, stability and control, and
aeroelastic characteristics. The Model 300 and 301 are nearly
identical designs differing primarily in the power plant instal-
lations; therefore, the results of the tests are readily applicable
to the Model 301. The principle objectives were to identify if
any significant problems existed and to obtain data on rotor wake/
airframe aerodynamic interference for use in the Model 301 real-
time flight simulation mathematical model. The investigation
was conducted under NASA Contract NAS 2-6599.

A.   Previous Tests

Scale model testing of the Model 300, under way since 1968,
has been directed at verifying that the aircraft met design
requirements. These tests were a part of Bell Helicopter
Company's tilt rotor IR & D program. One-fifth scale aero-
dynamic and aeroelastic models were designed and fabricated
to confirm airframe performance, stability characteristics,
dynamic design criteria, and demonstrate the aircraft to be
free from flutter or other aeroelastic instability.[3-11] Air-
plane configuration tests of the aeroelastic model (unpowered)
in the NASA Langley 16-foot Transonic Dynamic Tunnel were suc-
cessfully completed in April 1972. This test confirmed the
Model 300 to be free of rotor-pylon-wing instability and
other forms of aeroelastic instability at all speeds up to
1.2 times limit dive airspeed, that the aircraft short period
and Dutch Roll modes are adequately damped beyond the limit
dive airspeed of the aircraft, and that loads and vibration
levels are within design limits. The results of the airplane
test program are discussed in Reference 12.

B.   Technical Background

The results of the XV-3 convertiplane tests[13], recent tests
of a powered force model[14], and analysis of the Model 301
stability and control characteristics were the basis for plan-
ning the current series of investigations. During the XV-3
flight test program several stability and control problems
were noted in hover and low speed helicopter flights. No
serious flying problems were encountered when operating at
intermediate conversion angles or when converting. In air-
plane flight the only significant problems were light damping
of the Dutch Roll and longitudinal short period modes and
excessive transient flapping during maneuvers. A full scale
wind tunnel test later indicated the potential of rotor-pylon-
wing aeroelastic instability. The model tests, discussed
above, have verified that the airplane flight problems have

B. (Continued)

been solved in the Model 300 and 301 designs. Problems en-
countered in hover and low speed helicopter flight during
the XV-3 program, some of which have been addressed during
this test program, are summarized in Table II-1.

Items 6 and 7 in Table II-1 have been corrected in the Model
301 design by providing adequate control power about all
three axes. The pitch control power is 4.5 times, and the
yaw control 2.74 times that of the XV-3. Item 3, likewise
has been corrected in that the Model 301 has considerable
excess power, whereas the XV-3 had insufficient power to
hover. A blade-pitch governor, rather than pilot coordina-
tion, will automatically maintain rotor speed. As a result,
the pilot effort required in the Model 301 to cope with the
power rise is reduced to a level comparable to typical
turbine-powered helicopters. The other items listed are
inherent in the tilt rotor concept and are expected to
occur to some degree in the Model 301 as discussed in the
section noted. The subject investigation was directed at
evaluating the magnitude of these problems and in obtaining
data for use in real time flight simulation of the Model 301.
The latter objective is significant since the real time
flight simulation will be used to guide the design of the
Model 301's Stability and Control Augmentation System which
provides a substantial improvement in the flight character-
istics in hover and in low speed helicopter flight. (The
XV-3 did not have a stability and control augmentation system.)

C. Scope of the Program

The test program was accomplished in four tasks.

1. Task I Pretest Activities

Modifications were made to the model to meet the test
requirements as follows:

- Addition of rotor drive system and motors

- Remote controlled rotor cyclic, differential
  cyclic, collective, and differential collec-
  tive. Controls were designed with frequency
  response characteristics required to "fly"
  the model in semifree flight.

- Install a strain gage balance in the
  empennage to measure horizontal stabilizer
  normal force and rolling moment.

A hover test rig was designed for the semifree flight
hover tests.

2. Task II Hover Test

The hover test was accomplished in two phases:

Phase I - Obtain performance and static stability data

Phase II - Obtain controllability data

3. Task III Wind Tunnel Test

Wind tunnel testing was in the low speed helicopter and conversion flight configurations. The model was tested in the Vought Aeronautics Low Speed Wind Tunnel located in Grand Prairie, Texas.[15] The Vought tunnel has a 15 by 20-foot test section that can operate through a speed range of 9 to 77 feet per second and was ideal for this test program. Two types of mounting systems were used during this testing.

Phase I - Rod mount test which allowed the model to pitch, roll, yaw, and translate vertically. During this test trim flight data was obtained, dynamic stability characteristics were determined, and controllability was evaluated.

Phase II - Sting mount test with a six-component internal balance. This testing provided necessary data to define rotor wake/airframe aerodynamic interference.

4. Task IV Documentation

This task consisted of technical reports on program progress and analysis of test results.

D. Objectives of the Program

The investigation had the following specific objectives:

1. Measure roll static stability in hover as a function of hovering height (h/D).

2. Examine controllability in hover by "flying" the model in near free flight conditions.

3. Determine control settings for trimmed level flight, climbs, descents, rearward, and sideward flight, both in and out of ground effect.

4. Measure dynamic stability characteristics in low speed helicopter and conversion flight.

5. Investigate controllability in low speed helicopter and conversion flight.

6. Measure autorotational rate of descent and control positions.

7. Measure rotor wake effects on the empennage in sufficient detail to develop a mathematical model for real-time flight simulation.

8. Measure rotor and control system loads, airframe vibration, and aeroelastic stability during hover, low speed helicopter, and conversion flight.

All of these objectives were accomplished and the results are reported and discussed in the following sections.

TABLE II-1. XV-3 FLYING QUALITIES PROBLEMS

Problem Encountered During the XV-3 Test Program[13]	Action Taken During The Test Program and Resolution of Problem in M301 Design
1. Roll Instability, IGE "When hovering within 5 feet of the ground the XV-3 exhibited an erratic tendency to dart laterally. These characteristics were considered unacceptable. Above a 5 foot skid height, these characteristics disappeared and a stabilized hover over a spot was accomplished without difficulty. ••• When approaching the power required for hovering lift-off the aircraft reacted with a lateral oscillation that was disturbing to the pilot."	**Action:** Hover test to measure roll stability, Section V-A **Resolution:** Improved control power and addition of SCAS which provides additional roll damping
2. Low Speed Stick Reversal "At low forward speeds (15 to 20 knots CAS) and small conversion angles ($\alpha_m = 90°$ to $75°$), there was a longitudinal stick position reversal. This characteristic had to be anticipated for each take-off."	**Action:** Rod test, cyclic stick position for level flight, Section V-B **Resolution:** Improved control power with small stick reversal (within MIL-SPEC Requirements)
3. Increase in Power Required as Hovering Flight is Approached "At speeds between hover and best climb speed there was a large reduction in power required, which provided good forward acceleration and STOL capability. ••• When approaching for a zero speed landing the change in power was even more pronounced. As hovering flight was approached (approximately 10 knots CAS) there was a sudden requirement for an additional 9 to 10 inches Hg manifold pressure. Even though	**Action:** Rod test demonstrates the effect of power (torque) change with airspeed, Section V-B **Resolution:** Increased power available and blade-pitch governor to give pilot effort

TABLE II-1. (Continued)

Problem Encountered During the XV-3 Test Program[13]	Action Taken During The Test Program and Resolution of Problem in M301 Design
3. (Continued) anticipating this condition, it was nearly impossible to prevent the aircraft from suddenly losing 2 or 3 feet of altitude before sufficient power for stabilized hover could be obtained. This characteristic was undesirable and was considered a safety of flight item."	comparable to typical turbine-powered helicopter.
4. Dynamic Directional Stability and Pitch With Yaw "In general, the stability and controllability characteristics in the helicopter level flight regime were satisfactory except at speeds below 35 knots CAS. At low speeds a divergent long-period (6 seconds per cycle) directional oscillation was apparent. The amplitude of this oscillation increased to the point where the tail swung into the proprotor wash, promptly inducing nose up pitch."	Action: Dynamic stability check during rod test, Section V-B, static directional stability, sting test, Section V-C, and analysis of rotor wake during sideslip, Section VI-A Resolution: Addition of SCAS
5. Dynamic Longitudinal Stability "Longitudinal dynamic stability in configuration H and C15 was excellent above 40 knots CAS. Response to a pulse input of the longitudinal control was essentially deadbeat. At 25 knots or lower an unacceptable pitching oscillation was present. It was virtually impossible to stabilize the aircraft longitudinally in this low speed regime."	Action: Dynamic stability check during rod test, Section V-B Resolution: Addition of SCAS
6. Longitudinal Controllability "The longitudinal maneuverability was unacceptable in the low speed (30 knots CAS or less), small conversion angle (C15 or Less) regime.	No testing of control power, Model 301 control power higher than XV-3

TABLE II-1. (Continued)

Problem Encountered During the XV-3 Test Program[13]	Action Taken During The Test Program
6. (Continued) The primary problem was in the poor response to control inputs. For example, in C15 at 30 knots CAS, after 0.4 inch aft control pulse, 2.3 seconds were required to obtain the maximum pitch rate of 3 degrees per second. ··· At a higher speed, 42 knots, 1.9 degrees per second was recorded after 0.7 seconds after a similar stick displacement. The 0.7 second response time was acceptable."	
7. Directional Controllability "The directional maneuverability was unacceptable primarily because of poor response characteristics. At 30 knots CAS in configuration H, a maximum rate of yaw per inch pedal displacement of 12.4 degrees per second was recorded. However, more than 3 seconds were required to reach this maximum rate of yaw. ··· The excessive delay time is especially irritating in the XV-3 because of the poor lateral directional stability characteristics in the low speed regime."	No testing of control power. Model 301 control power higher than XV-3

III. DESCRIPTION OF THE MODEL

The model tested is a one-fifth length scale, dynamically and
aerodynamically similar model of the Bell Model 300 Tilt Rotor
Research Aircraft. The model was designed and fabricated under
Bell's Tilt Rotor research program and prior to this test pro-
gram had been used for tests involving over 900 hours of test
section occupancy. As noted earlier in Section II the purpose
of previous testing was to investigate aeroelastic and flight
stability characteristics in airplane mode. The tests were con-
ducted with the model unpowered, since the windmilling thrust
condition is the most critical from the standpoint of rotor-
pylon-wing stability. These tests were made using remote trim
type collective pitch and elevator controls. For the hover,
low speed helicopter, and conversion tests the model was powered
and had proportional type controls.

A. Model Scaling

Scale factors for the aeroelastic model are given in Table
III-1. All components of the model have been weighed and
swung to determine inerti properties and, where possible,
stiffness properties have been measured. Vibration sur-
veys have been conducted to verify dynamic similitude
(see Reference 12). A force and moment wind tunnel test
of the airframe less rotors has verified that the aero-
dynamic characteristics are the same as the Model 300 force
and moment model.

B. Construction and Design Parameters

The basic dimensional parameters for the aeroelastic model
are given in Table III-2 along with the corresponding full
scale design parameters. A description of the model com-
ponents are as follows:

1. Fuselage

The fuselage is an aluminum box spar, partially foam
filled, which represents the fuselage stiffness both
in bending and torsion. Figure III-1 shows the fuselage
spar during buildup for the sting mount test. Non-
structural fiberglass fairings, shown partially in-
stalled in Figure III-2, form the fuselage contour.
The fuselage houses the instrumentation package, the
motors, as well as balance weights used to simulate
gross weight and center of gravity configurations.

2. Empennage

The horizontal stabilizer is composed of an aluminum
spar which provides the required beam, chord, and
torsional stiffness, and nonstructural segmented
balsa and fiberglass fairings which give the aero-
dynamic shape. The elevator is remotely controlled.
The mass balance of the elevator is duplicated. The
vertical stabilizers, made entirely of balsa, were not
designed to give the correct stiffness properties but
have the correct mass, center of gravity and inertia.
The rudders are not provided on the model.

A two component empennage balance was designed and
fabricated for the helicopter and conversion portions
of this program. The balance measures empennage lift
and rolling moment. Figure III-3 shows a close up of
the empennage balance. Empennage incidence is ground
adjustable.

3. Wing

The bending and torsional stiffness characteristics of
the wing are scaled by an aluminum spar filled with
aluminum honeycomb. Aluminum-covered, nonstructural,
segmented fairings provide the aerodynamic contour.
Lead weights attached to the spar simulate fuel weight.

The wing spar is made in three parts: left and right
sections (with 6.5 degrees forward sweep) and a carry-
through center section with zero sweep. This arrangement
maintains the chordwise location of the wing elastic
axis. The attachments of the outer segments to the
center section are at buttline 5.6 (model scale). This
is also the location of the wing fuselage attachment,
simulating to some degree the full scale wing-to-
fuselage load paths.

The wing-pylon attachment is at buttline 35.85 (model
scale). The full scale load paths are maintained through
the ribs which carry the conversion spindle hanger bear-
ings.

4. Nacelles

The distributed stiffness properties of the nacelle
structure is not scaled, but mass, center of gravity,
inertias, and mounting stiffnesses are matched to
give the correct frequencies. Non-structural fiberglass
fairings house the transmission drive system, collective
drive system and dummy engine mounts. The engine mounts
are designed to give the correct engine pitch and yaw
natural frequencies. The transmissions have full-scale

4. (Continued)

gear ratios, therefore, the interconnect torque characteristics are scaled. (However, the torsional stiffness of the interconnect shafting is not scaled. It is several times the full-scale equivalent stiffness.)

Conversion struts with scaled axial stiffness provide the correct wing torsion (pylon pitch) natural frequencies in helicopter and conversion modes. The struts allow for manual adjustment of the pylon conversion angle from 95° (helicopter mode) to 0° (airplane mode) in fifteen degree increments.

5. Rotors

The blades are constructed to scale the appropriate material modulus, beamwise, chordwise, and torsional stiffness. Both the blade spar and the outer shell are fiberglass. The aluminum blade grips, which carry the pitch-change bearings, are bonded to the blade structure at the blade root. The aluminum hub yoke rings are machined to represent the yoke inplane and out-of-plane stiffness. Stainless steel spindles are assembled with a thermal fit to the hub. The retention system for attaching the blades to the hub duplicates the full scale design. The rotor system is shown in Figure III-4.

6. Drive System

Power for the rotors is provided by two task motors (3.4 horsepower each) mounted aft of the center wing section in the fuselage as shown in Figure III-5. The motors drive forward to individual right angle gearboxes which in turn power the rotor interconnect drive shaft. A two layer fiberglass coupling to relieve misalignment is located between each motor and gearbox. At the beginning of the test a coupling between the motor gearbox and the interconnect shafting was also used. Following failures of this coupling during testing, the drive system was redesigned to eliminate misalignment between the gearboxes and the interconnect shaft and couplings were added between the left and right gearboxes. Figure III-5 shows motors and gearboxes following a coupling failure prior to the redesign of the drive system. Additional couplings are used along the interconnect shaft outboard of the motors and at the outboard section of the wing to relieve misalignment. The interconnect shaft connects to 90 degree gearboxes mounted on the pylons. The pylon shaft is mounted to the pylon axis and drives two stage planetary gearboxes. The output shaft of the planetary gearboxes has a

6. (Continued)

coupling which enables the rotors to be readily dis-
connected from the drive system. The gear ratio of the
transmission is 17.6 to 1; thus with the rotors at 1260
rpm, the normal operating speed, the interconnect shaft
is turning at 22170 rpm.

C. Model Controls

A dc servo feedback system was used for the rotor controls
because of the advantages of simplicity and reliability
offered by the dc system. To determine design requirements
for the rotor controls a mathematical model of the IGE hover
dynamics, including the roll static instability, was derived
and programmed on an analog computer. Several of the model
"pilots" flew the math model using a control stick similar
to that used with the aeroelastic model. Each pilot evalu-
ated control systems having varying amounts of hysteresis,
various time constants and values of rate limiting. The
following design parameters were established: (1) hysteresis
of two percent maximum, (2) time constant of 0.15 seconds
maximum, and (3) rate limit of 3.5 degrees per second minimum.
These are applicable to the model-pilot system and cannot be
directly related to full-scale. A dc servo system which met
these requirements was designed and fabricated. The system
provided collective pitch, differential collective (for roll
control), longitudinal cyclic, and differential cyclic (for
yaw control).

The elevator and ailerons were controlled using "beep" type
controls. Flaps and empennage incidence were ground adjustable.

D. Natural Frequencies

Prior to wind tunnel testing of the model in airplane mode a
complete shake test was performed to verify placement of
natural frequencies[12]. Comparison of the frequencies ob-
tained at that time with those calculated for the full-scale
design indicated a good dynamic similitude. Therefore,
prior to testing the model in helicopter mode it was only
necessary to confirm the location of modes affected by
nacelle conversion.

As noted earlier, conversion struts having scaled axial
stiffness were used to provide the correct pitching mode
frequency of the pylons. As shown in Table III-3 the mea-
sured pitching mode frequency is slightly above the full-
scale equivalent frequency. However, this provided ade-
quate margin with respect to one-per-rev resonance which
was a major concern. The other important frequencies which
were affected by the change to helicopter mode are also
summarized.

E. Instrumentation

1. General

The model was instrumented to provide 61 channels of
data as shown in Table III-4. The locations of the
transducers were chosen to provide the required informa-
tion on loads and stability.

A maximum of 39 of the 61 available channels was multi-
plexed from the model to the Bell Offsite Data Acquisition
Package (ODAP). From the ODAP the data was recorded
on magnetic tape and on direct write oscillographs,
with selected data monitored on panel meters.

A schematic of the ODAP system as used on the model is
shown in Figure III-6. The components of the system
are identified as follows:

- Model Patch Panel - This panel, in the
  model, allows various trandsucers to
  be selected for signal conditioning and
  read out.

- Signal Conditioners - These amplify the
  transducer outputs to an analog output
  of ± 2.5 volts full scale. There are
  39 signal conditioners available in the
  model.

- Voltage Controlled Oscillator - The VCO
  uses the conditioned analog output of
  each transducer to modulate the output
  frequency of an oscillator. These out-
  puts are mixed in a linear mixer amplifier
  to create a composite signal containing up
  to 13 channels of data. There are 39
  VCO's in the model which provide two channels
  of multiplexed data. This minimizes model
  umbilical cross-section requirements for
  carrying instrumentation data.

- Discriminators - Twenty-six (26) discriminators
  are available to return the multiplexed signal
  to an analog form. These are mounted in the
  portion of the equipment located in the model
  control room (Figure III-7). Selected data
  may be enlarged or reduced in amplitude at
  this point for display on panel meters
  (Figure III-8) or oscillograph recorders
  (Figure III-9). All composite signals are
  also recorded on magnetic tape, also shown
  in Figure III-7.

1. (Continued)

   As shown in Figure III-8 a closed circuit TV was used
   to monitor the test so that problems during a run
   could be reviewed. The equipment used to monitor
   temperatures and the power supply to the motors is
   shown in Figure III-10.

2. Model Rotating System

   One blade on the right hand rotor was fully instru-
   mented with beam, chord, and torsion gages. In addi-
   tion the hub, rotating controls, and shaft were gaged
   for pitch link loads, blade flapping and shaft torque.
   Both rotors were gaged for cyclic and collective pitch.
   The right rotor was fitted with a 28-ring slipring.

   An encoder wheel was mounted on each rotor shaft to
   generate rpm and azimuth signals.

3. Non-Rotating System

   Each wing spar was gaged for beam, chord and torsion.
   Accelerometers were mounted on each pylon as well as
   along the fuselage to obtain dynamic data. The gimbal-
   ring mount to the vertical rod was instrumented to
   provide fuselage pitch, roll, yaw and vertical posi-
   tion data. In addition a three axis gyro was used
   to obtain pitch, roll and yaw rates. The horizontal
   tail was gaged for beam and torsion bending loads
   and the vertical tail had provisions for mounting
   two accelerometers.

4. Balance Systems

   On the rod mount a strain gage balance was placed in
   the model gimbal slider to measure the drag or thrust
   of the model. During the sting mount test, six com-
   ponent internal balance data referenced to the aircraft
   center of gravity was recorded. The empennage was
   mounted on a balance (Figure III-3) which measured
   empennage lift and rolling moment.

TABLE III-1   SCALE FACTORS FOR ONE-FIFTH SCALE MODEL IN AIR

Parameter	Scale Factor (Model/Full Scale)
Froude Number	1.0
Locke Number	1.0
Mach Number	0.447
Reynolds Number	0.0894
Length	0.2
Density	1.0
Velocity	0.447
Time	0.447
Mass	0.008
Frequency	2.24
Force	0.008
Power	0.00358
Bending Moment	0.0016
Stiffness	0.00032
Bending Spring Rate	0.04
Torsional Spring Rate	0.0016

TABLE III-2. MODEL DESIGN PARAMETERS

		Full Scale	Model
Aircraft:			
Overall Length	IN.	505	101.0
Overall Width (Rotor Turning)	IN.	686	137.2
Overall Height	IN.	300	60.0
Design Gross Weight	LB	13000	104
Model Flying Weight	LB	11125	89
Rotors:			
Blades Per Rotor		3	3
Diameter	FT	25	5
Blade Chord (Basic)	IN.	14	2.8
Hub Precone Angle	DEG	2.5	2.5
δ_3 (Minimum)	DEG	-15	-15
Flapping Clearance	DEG	±12	±12
Wing:			
Span (Pylon Edge to Pylon Edge)	IN.	412.80	82.56
Span (Pylon Centerline to Pylon Centerline)	IN.	386	77.2
Area (Total)	SQ FT	181	7.25
Mean Aerodynamic Chord	FT	5.255	1.045
Airfoil Section (Constant)		NACA 64A223 Modified	NACA 64A223 Modified

TABLE III-2. MODEL DESIGN PARAMETERS CONT'D.

		Full Scale	Model
Wing Cont'd:			
Aspect Ratio		6.6	6.6
Forward Sweep Angle	DEG	6.5	6.5
Dihedral	DEG	2.0	2.167
Aileron:			
Span (Along Hingeline)	IN.	94.32	19.0
Chord/Wing Chord		.25	.25
Flap:			
Span (Along Hingeline)	IN.	51.0	10.5
Chord/Wing Chord		.25	.25
Fuselage:			
Length	IN.	492	98.45
Maximum Breadth	IN.	66	13.2
Maximum Depth	IN.	74	14.8
Vertical Stabilizer:			
Span	IN.	92.16	18.44
Total Area Per Side	SQ FT	50.5	2.00
Aspect Ratio		1.18	1.18
Root Chord	IN.	49.04	9.808

TABLE III-2. MODEL DESIGN PARAMETERS CONT'D.

		Full Scale	Model
Vertical Stabilizer Cont'd:			
Tip Chord (Both Ends)	IN.	28.75	5.75
Mean Aerodynamic Chord	IN.	44.7	8.94
1/4 Chord at F.S. (Total)		570.015	114.003
1/4 Chord at W.L. (Total)		115.69	23.13
Sweep of 1/4 Chordline (Upper)	DEG	31.6	31.5
Horizontal Stabilizer:			
Span	IN.	153.96	30.8
Total Area	SQ FT	50.25	2.0
Aspect Ratio		3.28	3.28
Angle of Incidence	DEG	+20 to -10	+25 to -25
Sweep of 1/4 Chordline	DEG	0.0	0.0
Root Chord	IN.	47.04	9.40
Tip Chord	IN.	47.04	9.40
Mean Aerodynamic Chord	IN.	47.04	9.40
1/4 Chord at F.S.		560.0	112.0
1/4 Chord at W.L.		103.0	20.6
Elevator:			
Area	SQ FT	13.0	.52
Chord/Stabilizer Chord		.30	.30

TABLE III-3. HELICOPTER MODE NATURAL FREQUENCIES

Mode	Frequencies - cps	
	Full-Scale Equivalent	Model Measured
Wing Symmetric Beam	7.17	6.8
Wing Chord	11.20	12.0
Wing Torsion (Pylon Pitching)	15.90	17.0
Horizontal Tail Beam	24.0	26.0
Pylon Yaw	25.0	26.0
Horizontal Tail Torsion	37.63	38.0

301-099-002

TABLE III-4. MODEL INSTRUMENTED ITEMS

Right-Hand Rotor Blade	Beam 31.7 Percent
Right-Hand Rotor Blade	Chord 31.7 Percent
Right-Hand Rotor Blade	Torsion 31.7 Percent
Right-Hand Rotor Blade	Beam 50 Percent
Right-Hand Rotor Blade	Chord 50 Percent
Right-Hand Rotor Blade	Beam 50 Percent
Right-Hand Rotor Blade	Chord 50 Percent
Right-Hand Yoke Chord	
Right-Hand Pitch Link	
Right-Hand Rotor Shaft	Torque
Right-Hand Rotor Flapping	
Right-Hand Rotor Tachometer	
Left-Hand Rotor Tachometer	
Right-Hand Rotor Collective Position	
Right-Hand Rotor Cyclic Position	
Left-Hand Rotor Collective Position	
Left-Hand Rotor Cyclic Position	
Right-Hand Wing Pod Conversion Strut	
Right-Hand Wing Pod Accelerometer	Longitudinal (Conv. Axis)

TABLE III-4. MODEL INSTRUMENTED ITEMS CONT'D.

Right-Hand Wing Pod Accelerometer	Axial (Conv. Axis)
Right-Hand Wing Pod Accelerometer	Longitudinal (Xmsn Case)
Right-Hand Wing Pod Accelerometer	Lateral (Xmsn Case)
Right-Hand Wing Spar	Beam 27.2 Percent
Right-Hand Wing Spar	Chord 27.2 Percent
Right-Hand Wing Spar	Torsion 27.2 Percent
Right-Hand Wing Spar	Beam 52 Percent
Right-Hand Wing Spar	Chord 52 Percent
Right-Hand Wing Spar	Torsion 52 Percent
Left-Hand Wing Pod Conversion Strut	
Left-Hand Wing Pod Accelerometer	Longitudinal (Xmsn Case)
Left-Hand Wing Spar	Beam 27.2 Percent
Left-Hand Wing Spar	Chord 27.2 Percent
Left-Hand Wing Spar	Torsion 27.2 Percent
Left-Hand Wing Spar	Beam 52 Percent
Left-Hand Wing Spar	Chord 52 Percent
Left-Hand Wing Spar	Torsion 52 Percent
Aft Fuselage Spar Vertical	Beam Station 71.79
Aft Fuselage Spar Horizontal	Beam Station 71.79
Aft Fuselage Spar	Torsion Station 71.79
Fuselage Crew Station Accelerometer	Vertical

TABLE III-4. MODEL INSTRUMENTED ITEMS CONT'D.

Fuselage Center of Gravity Accelerometer	Vertical
Aircraft Drag	
Aircraft Position	Pitch Rate
Aircraft Position	Roll Rate
Aircraft Position	Yaw Rate
Aircraft Position	Pitch Attitude
Aircraft Position	Roll Attitude
Aircraft Position	Yaw Attitude
Elevator Position	
Aileron Position	
Right-Hand Empennage Horizontal Spar	Inboard Beam
Right-Hand Empennage Horizontal Spar	Inboard Chord
Right-Hand Empennage Horizontal Spar	Inboard Torsion
Left-Hand Empennage Horizontal Spar	Inboard Beam
Left-Hand Empennage Horizontal Spar	Inboard Chord
Left-Hand Empennage Horizontal Spar	Inboard Torsion
Right-Hand Vertical Stabilizer Accelerometer	Fore and Aft
Left-Hand Vertical Stabilizer Accelerometer	Fore and Aft
Empennage Balance	Lift
Empennage Balance	Rolling Moment

Figure III-1. Fuselage ... and Rotors ...
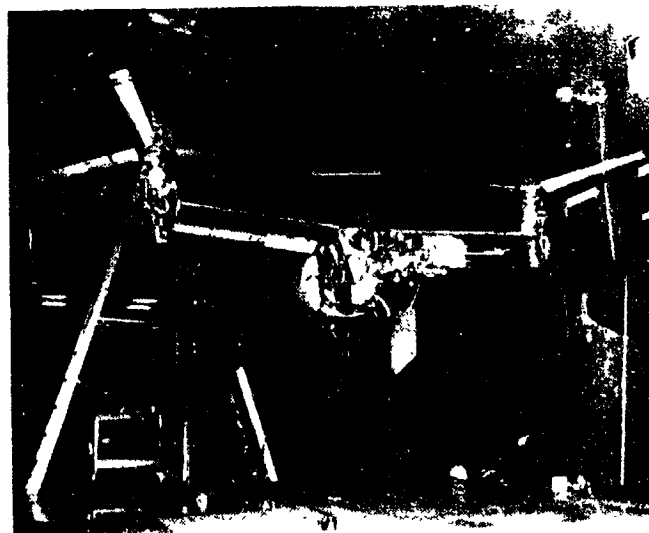
Figure III-2. Fuselage Fairings.
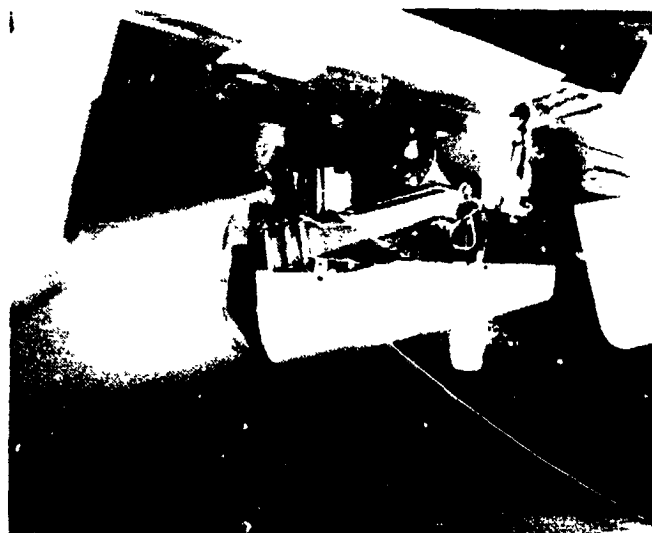
Figure III-3. Empennage Balance and Variable
Incidence Adjustment.

Figure III-4. Rotor-Hub.
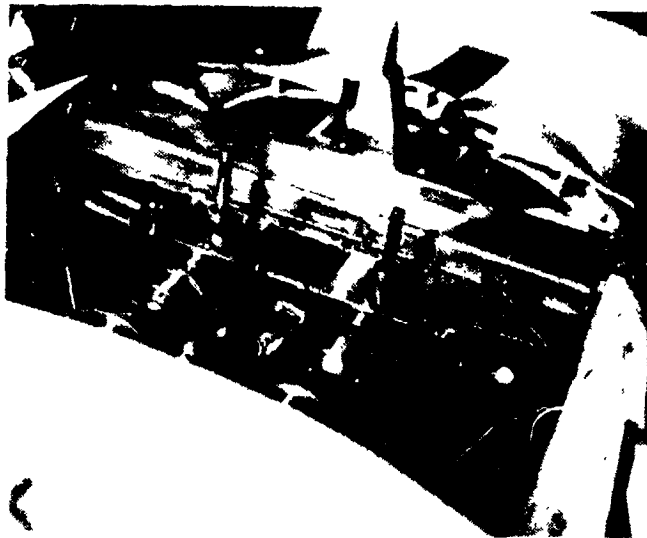
Figure III-5. Model Motors, Couplings, and
Interconnect Shaft (Following
Coupling Failure).
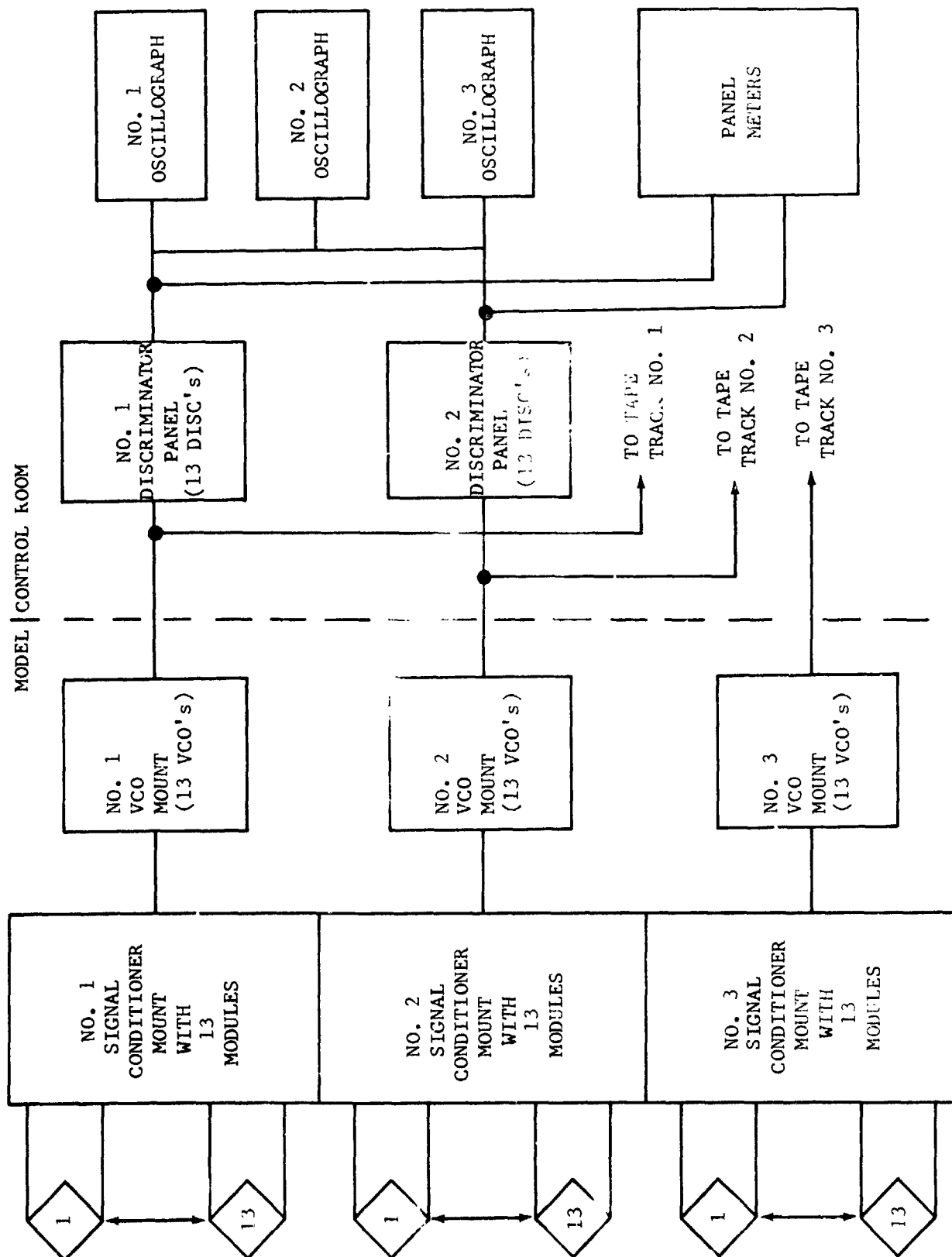
Figure III-6. Schematic of Offsite Data Acquisition Package.
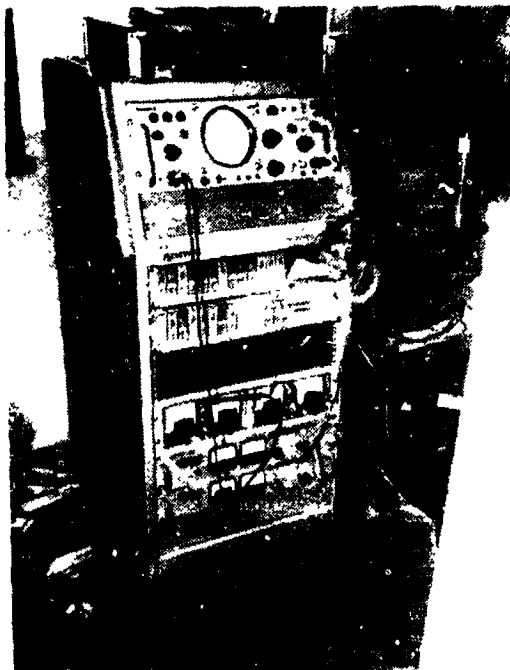
Figure III-7.   Instrumentation, Discriminator
Panel and Magnetic Tape.

Figure III-8. Instrumentation, Model Pilot Controls
Panel Meters and Closed Circuit TV.
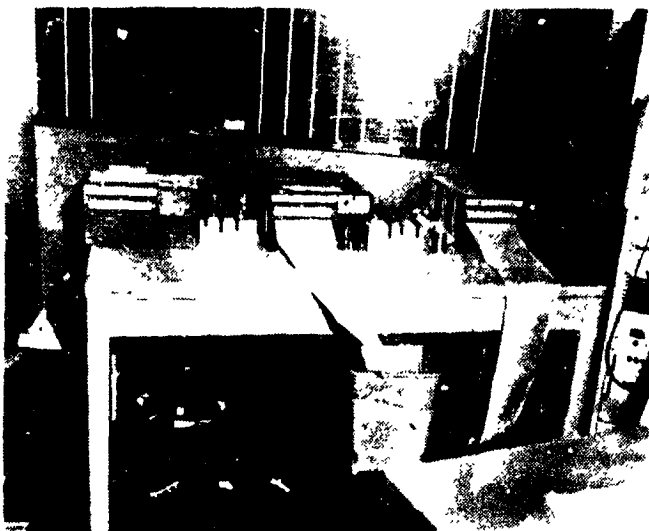
Figure III-9. Instrumentation, Oscillographs.

Figure III-10. Instrumentation, Power Supply and
Motor Temperature.

IV. DESCRIPTION OF TESTS

Tests of the Model 301 one-fifth scale powered aeroelastic
model were conducted during the period September 13, 1972
through March 30, 1973 in four phases:

Phase I, Hover Test, September 13 - October 3, 1972 (19 hours
rotors turning) - This test was conducted in the Bell
Helicopter Company whirl cage, on a rod-mount, to investigate hover performance and stability.

Phase II, Hover Test, December 18-22, 1972 (7 hours rotors
turning) - This test was conducted in the Bell Helicopter
Company whirl cage, in semi-free flight, to investigate
stability and control response.

Phase III, Rod Test, January 15 - March 9, 1973 (220 hours
occupancy, 34.5 hours rotors-turning) - This test was con-
ducted in the Vought Aeronautics low speed wind tunnel, on
a rod mount, to investigate trimmed flight control position,
performance, and evaluate overall flying qualities.

Phase IV, Sting Test, March 15-30, 1973 (107 hours occupancy,
16.7 hours rotors-turning, 11.0 hours rotors-off testing) -
This test was conducted in the Vought Aeronautics low speed
wind tunnel, with an internal balance and mounted on a sting,
to investigate static stability in yaw and pitch and measure
rotor wake/airframe interference effects.

A. Hover Test

   1. Phase I Hover Test

      a. Roll Stability Investigation

         A vertical rod mounting system was used which pro-
         vided the model with ± 6 degrees of roll freedom
         at height/diameter (h/D) ratios from 0.5 to 1.0.
         The model control was with rotor cyclic pitch,
         collective-pitch and differential collective-pitch.
         The test rig shown in Figure IV-1, permitted the
         rolling moment required to hold the model at a
         given roll angle to be measured at any roll angle.
         The required moment was measured with the rotors
         stopped and with the rotors thrusting. The model
         was tested with the wing panels on and with the
         panels removed (Figures IV-2 and IV-3) at h/D
         ratios of 0.5, 0.75 and 1.0 to determine the
         influence of the wing/rotor interactions.

b. **Wing Download Investigation**

The wing download due to rotor downwash was investigated at h/D ratios from 0.5 thru 1.0 with the wing panels on and with the panels removed. The model was counterbalanced so that hover performance could be investigated at net-thrust values of 50 pounds to 100 pounds (6250 pounds to 12500 pounds full-scale.

c. **Optimum Flap Setting to Minimize Hover Download**

Various flap settings were tested at h/D = 1.0 to investigate hover download. The flap/aileron setting tested were 0/0°, 45/25°, 60/45°, 75/45°, and 75/75°.

2. **Phase II Hover Test**

a. **Stability and Control Response**

The model was mounted on a translational hover test rig such that free flight was nearly simulated. The model was free in pitch ($\pm12°$), roll ($\pm8°$), yaw ($\pm30°$), vertical translation (h/D = .5 to 1.5) and horizontal translational within a five foot diameter circle. The model could be restrained in yaw and in vertical and horizontal translation by tensioning snubber cables. The test rig is shown in Figure IV-4.

Three "pilots" were used to control the model. Pilot A controlled vertical height and roll, Pilot B controlled pitch, and Pilot C controlled yaw. The dc servo feedback control system discussed in Section IIIC was used for this portion of the test. The model pilots were stationed at the most convenient azimuth with respect to the pilots control (i.e., the roll and yaw pilots were behind the model and the pitch pilot to the side).

B. **Semi-Free-Flight, Rod Mount Test**

The model was mounted on a vertical rod and was free in pitch (-12° to +10°), roll ($\pm6°$), yaw (generally $\pm15°$) and vertical translation (±18 inches). The model control system was the same as for the Phase II hover test.

Realistic "flight" data was obtained at each level flight trim point by flying the model clear of the vertical restraint limits and by nulling the horizontal drag on the rod to zero, such that thrust equalled drag. Climb

B. (Continued)

and descent conditions were simulated by trimming the axial
force on the rod to the value of thrust or drag required to
simulate the gravity component acting parallel to the flight
path. At selected data points the model was disturbed in
pitch and yaw, separately, to obtain frequency and damping
data.

Data was obtained in helicopter, conversion and auto-
rotative flight to evaluate aircraft characteristics in
the performance, dynamics, stability and control areas.
Testing covered forward flight from hover to 100 knots
full-scale at mast angles of 90 degrees (Figure IV-5),
75 degrees (Figure IV-6), 60 degrees (Figure IV-7), and 30
degrees (Figure IV-8). Sideward flight testing with a
mast angle of 90 degrees was investigated from hover to 30
knots (Figure IV-9). Rearward flight was tested from
hover to 35 knots (Figure IV 10) at yaw angles up to 45
degrees. IGE testing was at h/D = .5 to .75 from 16 to
60 knots using the moving belt ground plane (Figure IV-11).

C. Sting Mount Test

The model was mounted on a sting support system with an
internal, six component strain gage balance to measure force
and moment data. Fuselage pitch attitude was generally
varied from -18 to +20 degrees and yaw angle varied from
-2 to +20 degrees. Only forward flight was investigated.

Control position determined at the trim points during the
rod test were used to set fuselage attitude, rotor control
positions and elevator position. Static stability data
were obtained during pitch and yaw sweeps from the trim
condition. Control settings were held constant during
the sweeps. Horizontal stabilizer incidence was varied
to obtain stabilizer effectiveness and elevator sweeps
were made for elevator effectiveness. Tests were also
conducted with the empennage and/or the rotors removed
(Figures IV-12 and IV-13).

A tuft grid (3 ft x 4 ft) was mounted on the sting and placed
directly behind the empennage (Figure IV-14) to visually
determine flow patterns in the vicinity of the empennage
during low speed helicopter flight (airspeed from 16 to
40 knots). Photographs were taken with the rotors stopped
and at trimmed level flight conditions for yaw angle of 0
and 10 degrees.

D. Data Reduction

Corrections for Reynolds number effects must be made to make
the model data representative of the full-scale aircraft char-
acteristics. Trends and rotor wake effects on the airframe
are believed representative of what might be expected on
the full-scale aircraft. Rotor and airframe aerodynamic
parameters used in the correlation study have been modified
to account for the model Reynolds numbers. Reynolds number
during the test ranged from 80,000 to 500,000 referenced
to the wing chord of 1.045 feet. Airspeed used in this
report is the equivalent full-scale airspeed. Vortex
generators were placed on the wing quarter-chord to better
simulate full scale lift characteristics at high angles of
attack. The data reference center for all data obtained
during the sting test is stationline 60.0, waterline 16.7
and buttline 0.0 of the model. This corresponds to an
equivalent full scale stationline 300.0 and waterline 83.5.
Center of gravity positions tested during the rod test were
equivalent to an aft cg for mast angles of 90 and 75 degrees.
As the nacelles were tilted forward the weight of the model
rotor gear box caused a forward shift in cg, more than for
the full scale aircraft. This resulted in mast angles 60
and 30 degrees being tested at forward cg. Center of gravity
locations for the rod test at the mast angles tested are as
follows:

Mast Angle	Fuselage* Station	Waterline	
90	60.0	16.7	
75	59.4	16.6	
60	59.0	16.5	
30	58.2	16.2	*Model Scale

Force and moment data obtained during the sting test was
measured on the wind tunnel six-component internal balance
and data reduction was by a Vought Aeronautics Wind Tunnel
data reduction program. Dynamic and static model data
(control position, rotor parameters, and empennage lift)
obtained during all phases of testing were recorded on
magnetic tape and three oscillographs. Calibration of all
strain gages were completed prior to testing with additional
calibration made frequently during the tests. Magnetic
tape data was put in digitized form at Bell Helicopter's
Data Reduction Center. Comparisons and verification have
been made with hand-reduced data from the oscillographs.
The force and moment sign convention used for the rotor and
airframe is shown in Figure IV-15.

VERTICAL ROD

46½"

FUSELAGE FREE
TO ROLL ± 5°,
VERTICAL FREEDOM
LOCKED OUT

ROLL SPRING

ROLL FORCE
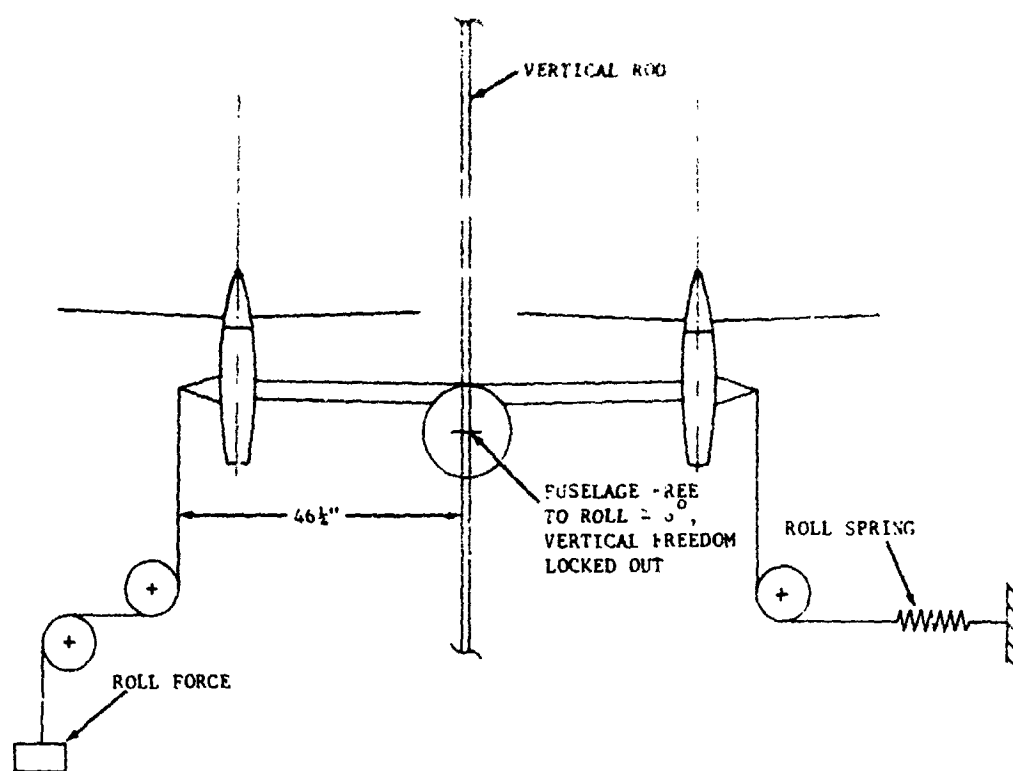
Figure IV-1. Roll Test Rig Schematic.
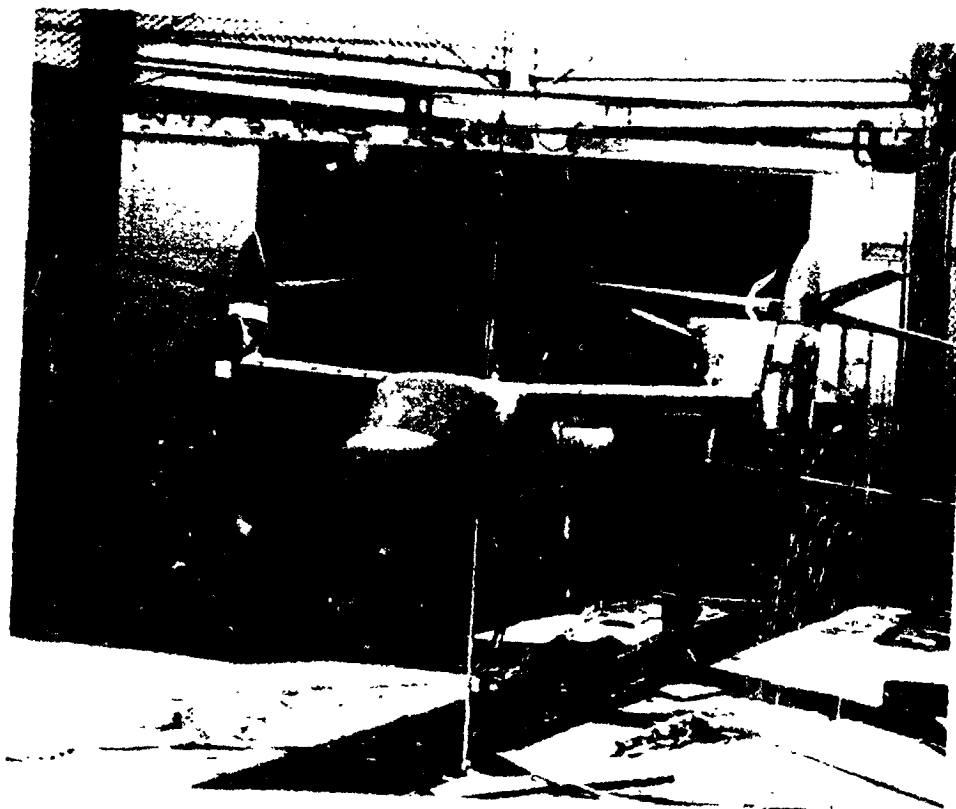
Figure IV-2.   Hover Test, Wing Fairings On.

Figure IV-3. Hover Test, Wing Fairings Removed.
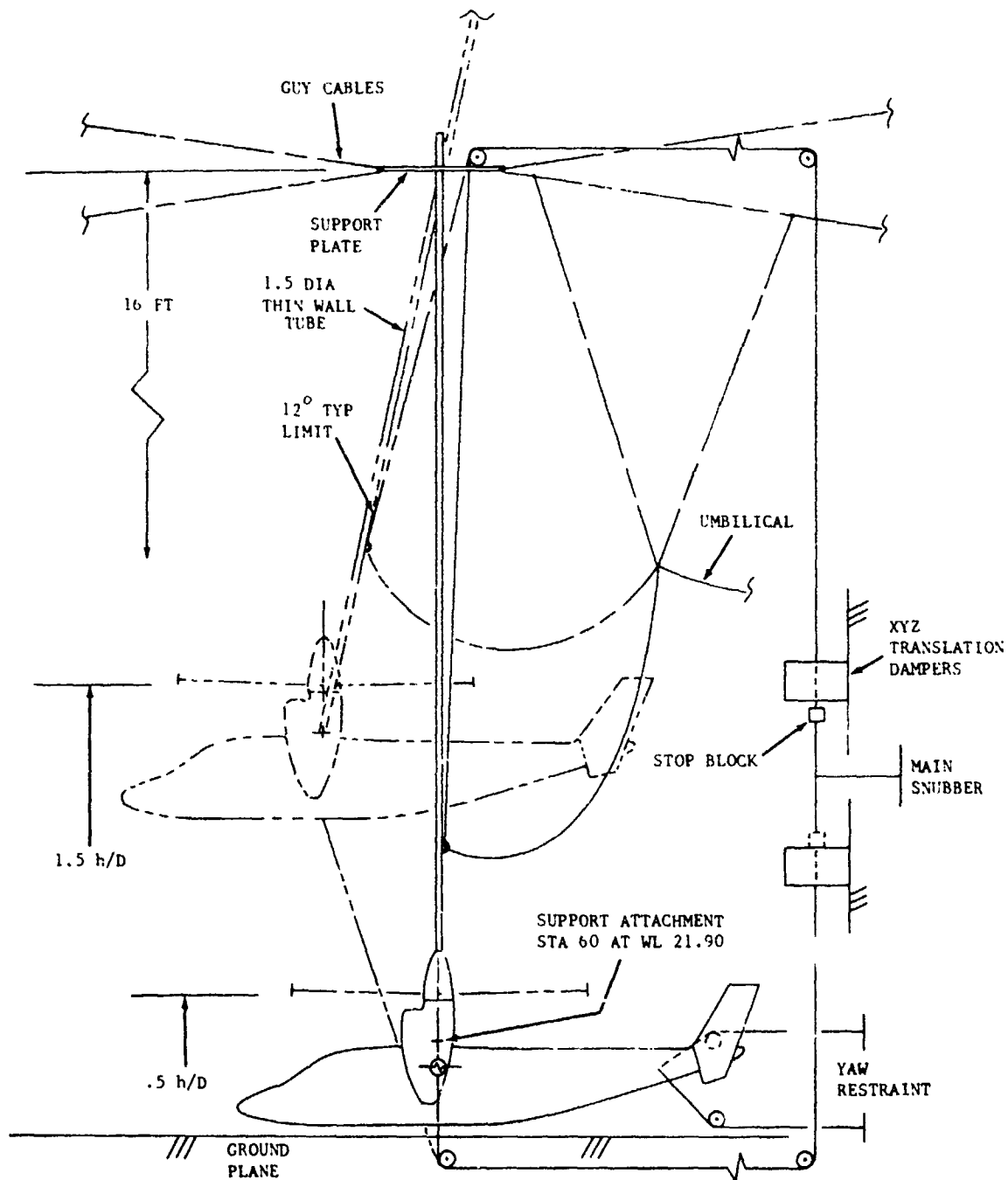
Figure IV-4. Free Flight Test Rig Schematic.

Figure IV-5.   Rod Test, Mast 90 Degrees.

Figure IV-6.   Rod Test, Mast 75 Degrees.
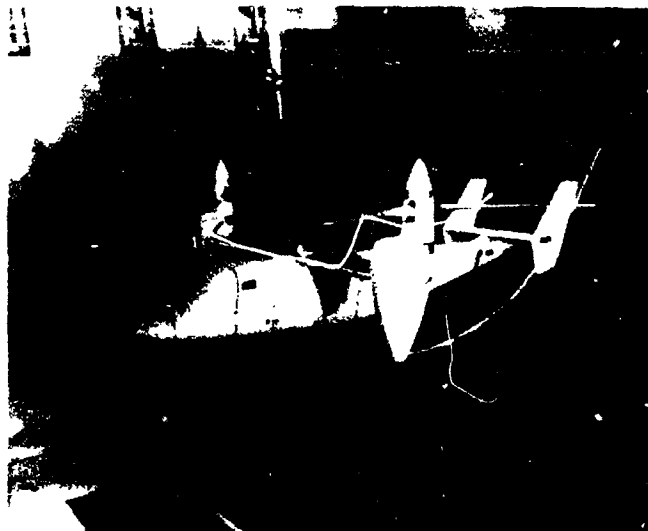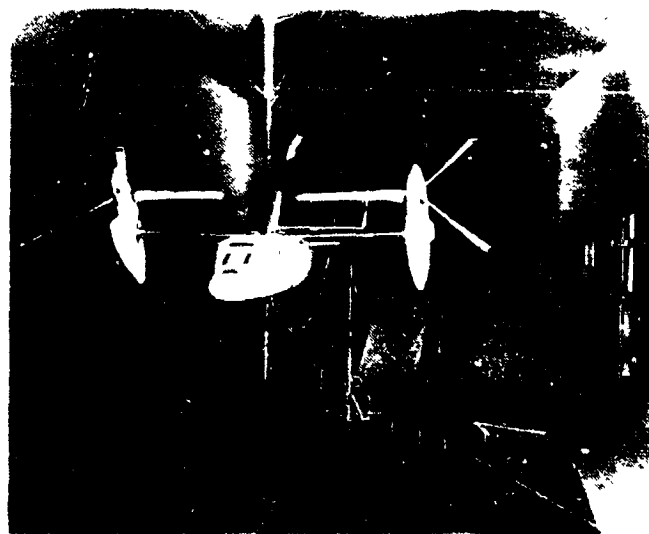
Figure IV-7. Rod Test, Mast 60 Degrees.
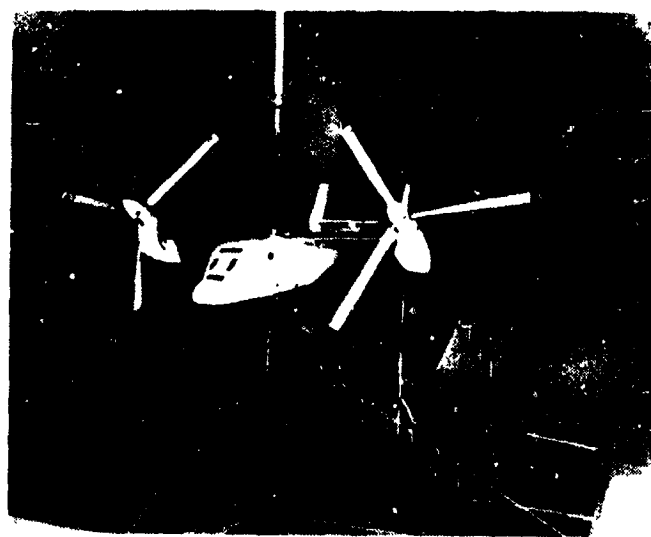
Figure IV-8. Rod Test, Mast 30 Degrees.
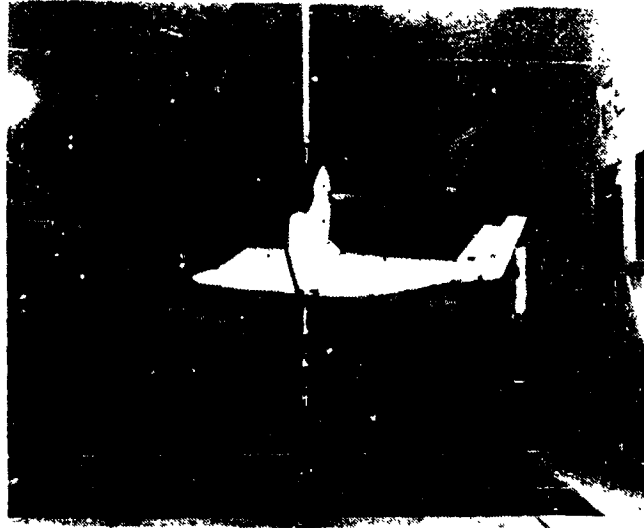
Figure IV-9. Rod Test, Sideward Flight
Mast 90 Degrees.
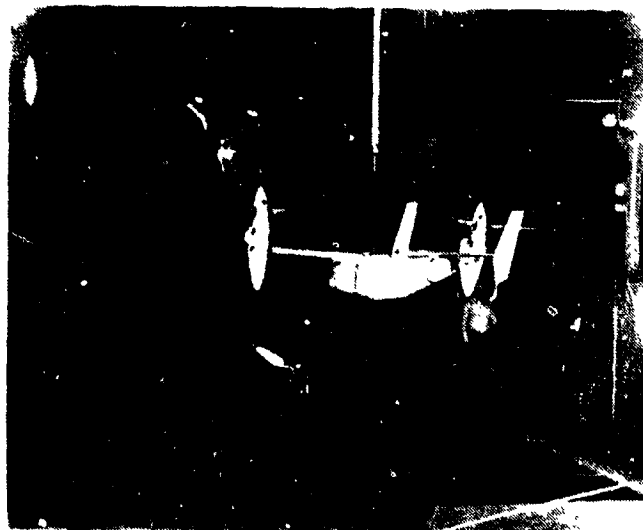
Figure IV-10. Rod Test, Rearward Flight
Mast 90 Degrees.
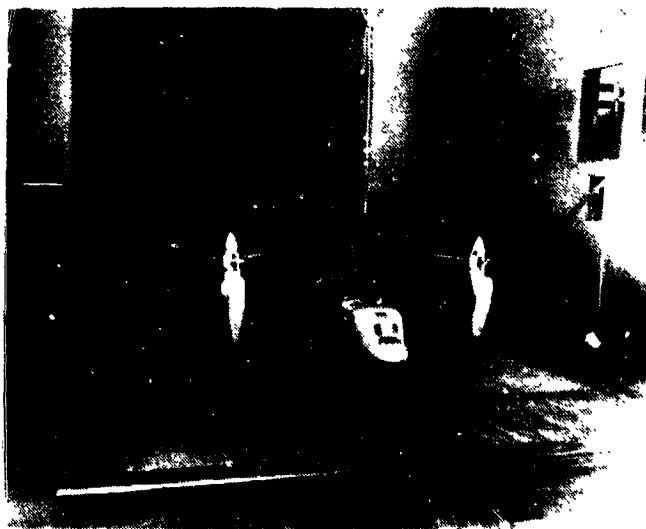
Figure IV-11.   Rod Test, IGE, Mast 90 Degrees.
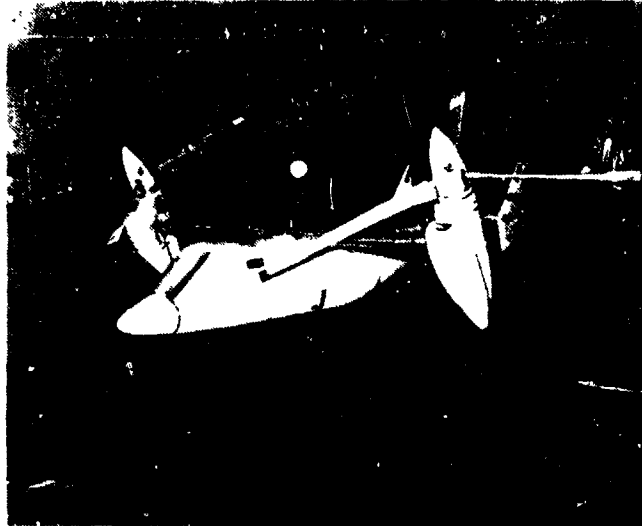
Figure IV-12.   Sting Test, Mast 60 Degrees
Rotors-On.

Figure IV-13.   Sting Test, Mast 60 Degrees
Rotors-Off.
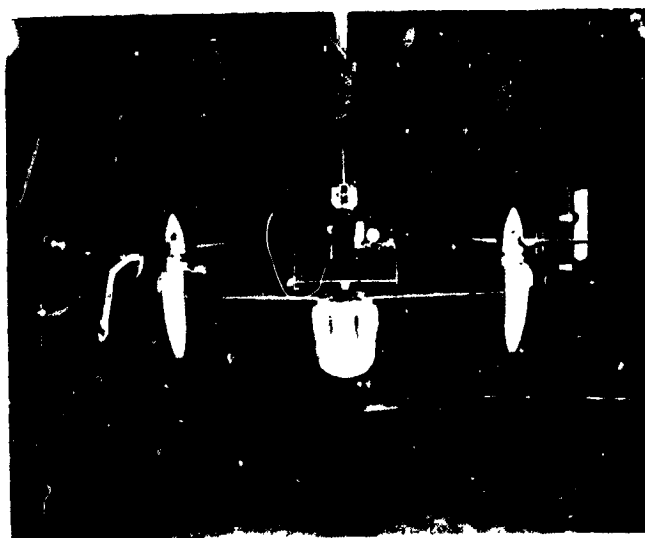
Figure IV-14.   Sting Test, Tuft Grid.

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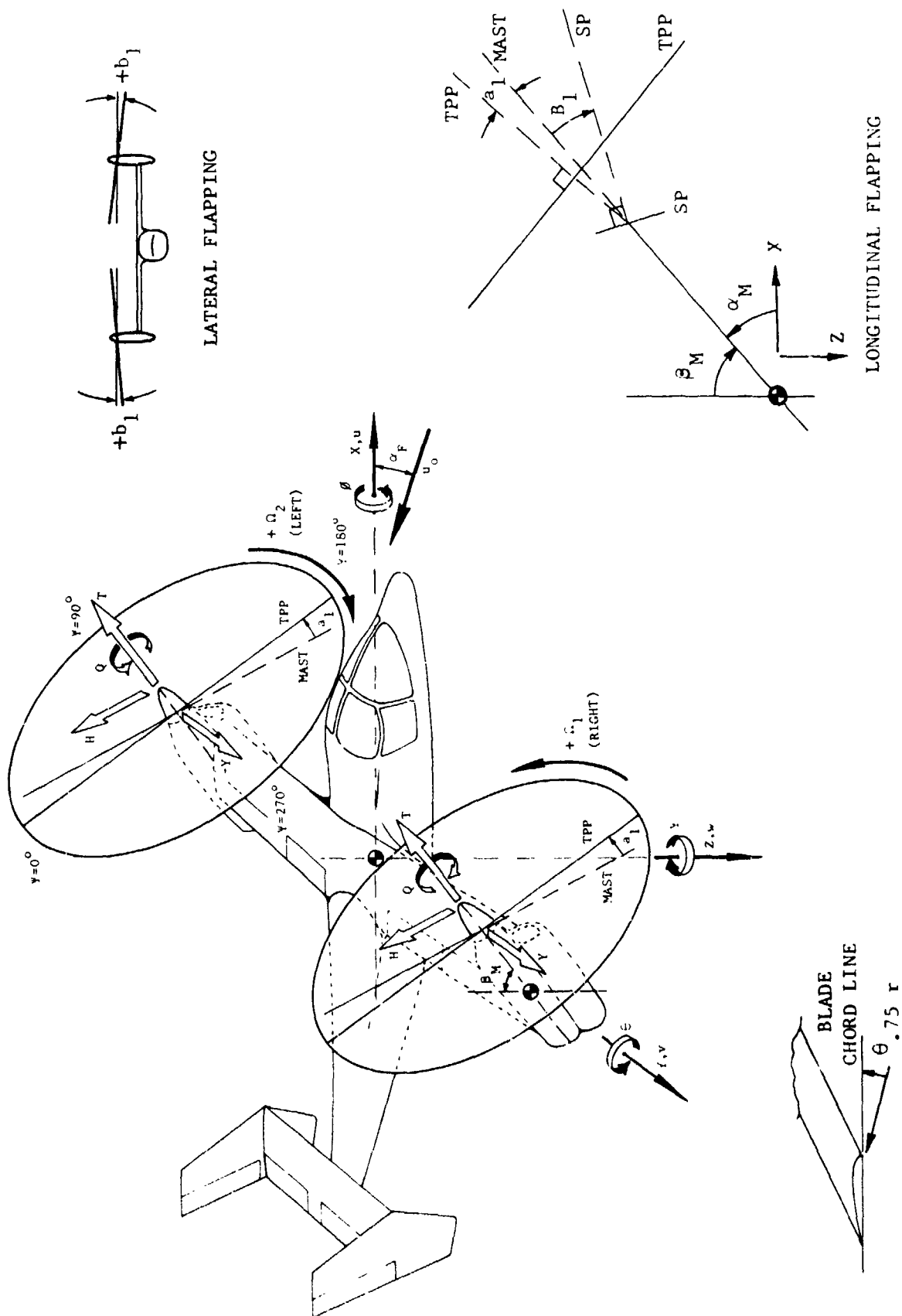
Figure IV-15. Force and Moment Sign Convention.

V. RESULTS OF TESTS

A. Hover Test

1. Roll Static Stability

The model was tested at h/D ratios of 0.5, 0.75, and
1.0. The moment required to hold the model at a given
roll angle was measured first with the rotors stopped
and then with the rotors turning and at 1g thrust.
The difference between the recorded moments is the con-
tribution of the rotor wake-wing-ground aerodynamic
interference. Data recorded in terms of weight at the
wing tip required to maintain a given roll angle is
given in Figure V-1 for h/D = 0.50. At h/D = 0.5 the
roll moment is stabilizing. Figure V-2 shows the
weight versus roll angle for h/D = 1.0 where the roll
moment is destabilizing. Tests were also made with the
wing aerodynamic fairings removed.

Figure V-3 summarizes the roll static stability data
As indicated, the model was found to have positive roll
stability at touchdown, h/D = 0.50. Above an h/D = 0.54
the model showed negative roll stability which was lin-
ear with roll angle, over ±6 degrees. The maximum
instability was found to occur at h/D = 0.85. The full-
scale control input required to trim this instability
was found to be 0.051 inches of lateral stick per degree
of roll at an equivalent full-scale gross weight of
8,250 pounds. The lateral control requirement is ex-
pected to increase linearly with the rotor downwash
dynamic pressure. Thus at the maximum VTOL gross weight
of the Model 301 (15,000 pounds) the maximum lateral
control requirement is predicted to be 0.093 inches of
stick per degree of roll. For a ten-degree wing drop
in-ground-effect the lateral stick for trim would be
0.93 inches. Maximum lateral stick travel available is
±4.8 inches.

2. Performance

The model was tested with wing panels on and off, at
h/D ratios from 0.5 through 1.0. The model had vertical
freedom, ±4° roll freedom and ±20° yaw freedom. Pitch
freedom was locked out. The model was counterbalanced
by weights so that hover performance could be investi-
gated at net-thrust values of 50 pounds through 100
pounds (6250 pounds through 12,500 pounds full-scale).

2. (Continued)

The optimum flap setting to minimize wing download was investigated at h/D = 1.0. The baseline data point was the model hovering at a net thrust of 75 pounds (9375 pounds full-scale), flaps/ailerons at zero settings. Resetting the flaps/ailerons to 45 25°, 60 45° and 75/45° showed in each case an increase of 6.0 percent in net thrust compared to the baseline case. A setting of 75/75° showed only a 4.7 percent increase compared to the baseline case. The 75/45° setting was selected for further performance testing since it was apparently near optimum. This setting was also used for the roll stability testing. The wing panels were also removed to determine the difference in power required. (The exposed wing spar is a rectangular section so wing download was not completely eliminated.) Results at h/D = 1.0 are shown in coefficient form in Figure V-4.

The model rotor hovering performance was predicted by correcting the full scale blade aerodynamic characteristics for Reynolds number and Mach number effects. The blade profile drag coefficient at zero lift was increased and the section lift curve slope and maximum lift decreased. The computed model performance is compared to full scale and to the model data in Figure V-4. The calculated model performance compares reasonably well with the performance measured with the wing aerodynamic fairings removed, which approximates a zero download/upload condition. The model power coefficient is calculated to be about 12 percent higher than the full scale power coefficient at the same weight coefficient.

Model performance data at h/D = 0.75 and 0.50 are shown in Figures V-5 and V-6.

3. Wing Download

The wing download was measured by comparing wing-fairings on and off data. At thrust/sigma prime = 84.5 pounds (10,600 pounds full-scale) the wing-panel download was 11.4 percent of thrust at h/D = 1.0, as shown in Figure V-7. This is higher than the 7 percent predicted for full-scale and it is believed that a Reynolds number effect may explain the difference. Hoerner shows that the drag of cross-sectional shapes with rounded edges and flat sides are strongly dependent on Reynolds number as shown in Figure V-8. The sections tested by Hoerner are considered to be comparable to a wing, with flaps deflected, at an angle of attack of -90 degrees. The drag coefficients of Figure V-8 drop by at least 50 percent as Reynolds number is increased from that of the model (2×10^5) to that of full-scale (2×10^6). Thus the model download of 11.4 percent thrust can be predicted to reduce to 5-7 percent of full-scale thrust.

3. (Continued)

Also shown on Figure V-7 are the results of a recent test
of the one-tenth-scale Bell C100 F1B tilt rotor model.
This test was conducted at Reynolds numbers around
4×10^5 and showed a wing download under 7 percent.
This Reynolds number is close to the "drag-bucket" of
Figure V-8 and may be more representative of full-scale
values. The reduction in power required in-ground-effect
is shown in Figure V-9. The reduction is 12 percent at
high model weights, increasing to over 19 percent at low
model weights.

The vertical "spring" in ground effect is shown in Fig-
ure V-10. Collective pitch was not instrumented on this
test so the thrust change is shown at constant torque.
The vertical "spring" is virtually the same with wing-
panels on or off. At medium model weights, the net
thrust increased by 19 percent as h/D reduced from 1.0
to 0.5.

4. <u>Controllability in Semi-Free Flight</u>

The model was tested in a transitional hover test rig
(h/D = 1.0) to investigate free flight stability and con-
trol response. Control of the model was found to be so
difficult that most of the test period was spent developing
piloting skill. The pilot's skill level at the end of the
test period was much higher than at the beginning but was
still less than that required for precision control of
the model. Hover over a spot could generally be achiev
until a wind gust or an input caused a roll disturbance
The model then translated rapidly until the transitiona
limits were reached. The tendency was aggrevated by the
horizontal translational restraint system causing a
rolling moment with horizontal displacement.

Yaw control proved to be easy once the pilot recognized
the need to lead yaw rate in order to stop on the desired
heading. Pitch control was not difficult except where
roll and lateral translation motion became large, then
pitch could not be controlled.

The wing panels were removed from the model to see if
rotor-wing aerodynamic interference contributed to the
roll-lateral translation control problem. However, no
significant change was observed. The model was also
flown at h/D = 0.75. At that height it was not con-
trollable. When the snubber was released, the model
would translate laterally so quickly that the pilots
did not have time to regain control over the model
before it reached the stops.

4. (Continued)

It was apparent from the hover controllability test that given sufficient time to develop piloting proficiency, the model could be successfully hovered. However, this task was felt to be of low priority and was terminated in order to prepare for the wind tunnel tests.

B. Semi-Free Flight, Rod Mount Test

Tests on the vertical rod mount covered the mast angle range from 95 degrees to 30 degrees and airspeeds from hover to 100 knots. (Airspeeds and gross weights given in this report are equivalent full scale values. All other parameters shown are in model scale.) Tests were performed to determine control positions for trimmed flight and to evaluate stability and flying qualities about trim.

Airframe and rotor parameters monitored during the test included fuselage pitch attitude, elevator position, empennage lift, cyclic position, collective setting at 0.75 radius of the blade, rotor torque, and rotor flapping. Rotor loads and vibration levels were also monitored to prevent exceeding limits. There is some scatter in the test data which must be attributed to the model not being in trim flight. A trimmed level flight condition on the rod was obtained when the thrust/drag indicator on the rod indicated zero. Accuracy of the indicator and its reading, and the low dynamic pressure at the low airspeeds tested made it difficult to determine the precise trim point. Any off-trim thrust or drag is equivalent to flying in a climb or descent. Also, small roll and yaw angles were difficult to detect. Every attempt was made to maintain correct rigging between cyclic and elevator through the test, but some data points were taken with improper elevator settings. Combinations of these factors resulted in off-trim conditions for some recorded test points. Therefore, general trends should receive more emphasis than studying each individual data point.

Comparison between test data and parameters calculated using BHC computer program C81 is shown in each figure. Airframe and rotor aerodynamic inputs to program C81 were corrected to reflect Reynolds number effects. Reynolds number effects are significant and must be kept in mind when using the test data to predict full-scale characteristics. The recommended approach is to validate the computer program using test data and then scale up to predict full-scale characteristics using aerodynamic data appropriate to the full-scale Reynolds numbers.

1. **Trimmed Flight Conditions**

   a. **Mast angle 90°**

      (1) **Level flight**

Figure V-11 shows the model parameters for
trimmed level flight versus airspeed. Vortex
generators have been used during low Reynolds
number scaled model testing of the Model 300
to obtain lift characteristics that approxi-
mate full scale[12]. As shown in Figure V-11,
vortex generators do not have a significant
effect on the trim parameters. The largest
difference occurred in the rotor torque. In
analyzing the data for points 80 through
89 with the vortex generators off, it was
noted that a small amount of drag was in-
dicated on the rod which is equivalent to
having a rate of descent. This may explain
why the vortex generators off data required
less power.

Cyclic position and empennage lift are a pri-
mary indicator of rotor wake effects on the
horizontal stabilizer. The shallow stick
gradient between 20 and 50 knots is the result
of the rotor wake having the net effect of an
upwash; positive empennage lift tends to con-
firm that the rotor wake acts as an upwash.
As speed increases above 50 knots wing down-
wash becomes more effective, providing a stable
stick gradient with airspeed. The shallow
gradient between 20 and 50 knots is within the
MIL-SPEC requirements and should not present
a problem. During recent simulator studies
of the Model 301 the shallow gradient did not
present a problem for the pilots.

Calculated rotor torque (power) was found to
be in very good agreement with test. Torque
readings remained consistant throughout the
test and were later used as a trim indicator
during the sting test.

In general, calculated empennage lift is in
close agreement with test data although the
empennage lift data had the largest amount of
scatter of all the parameters measured. Several
items that affect empennage lift which contri-
bute to this variation are fuselage pitch atti-
tude, elevator position, and rotor power. The

(1) (Continued)

trend generally remained the same and shows
a variation initially from a download to an
upload. As speed increases, the upwash from
the rotor decreases and the downwash from the
wing increases. Another reason for the dif-
ference between theory and test is the present
inadequate representation of the rotor wake in
program C81. This will be discussed later in
Section VI.

Rotor flapping was found to be up to 40 percent
higher than calculated. This difference is
primarily due to lateral flapping. Measured
fore-and-aft flapping is in good agreement with
that estimated. By refining the original theory,
analysis shows that better agreement can be made
with the measured flapping as discussed in
detail in Section VI. Although flapping was
higher than calculated, flapping does not appear
to restrict any flight condition.

During the initial portion of the rod test the
roll degree of freedom was locked out to reduce
the model pilots work load and to allow a pre-
liminary evaluation of flying qualities. Once
the model was checked out and found to be easy
to control the roll restraint was removed. No
significant difference was observed in trim
characteristics between having the roll fixed
or free.

(2) Effect of Horizontal Stabilizer Incidence on
Level Flight Trim

Figure V-12 illustrates the effect of horizontal
incidence ($\pm 4°$) on level flight trim parameters.
The trends and agreement between theory and
test data are similar to those discussed above.
The main effect of changing stabilizer incidence
is in the longitudinal cyclic stick trim posi-
tion. Positive incidence (leading edge up)
tends to reduce the gradient with airspeed
between 20 and 60 knots. For the four degrees
tested, the cyclic stick position showed a
stick reversal. With negative incidence, the
cyclic stick gradient with airspeed improves.

Changing incidence in either direction has
advantages and disadvantages. Positive inci-
dence produces a stick reversal between 20 and

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(2)   (Continued)

60 knots, but does not require as much forward
stick at high speeds. High speed helicopter
would then be limited by power instead of stick
travel. Positive incidence also tends to pitch
the aircraft more nose down which would be
undesirable from a pilot's point of view (fuse-
lage angle of attack with +4 degrees incidence
is -15 degrees at 100 knots) and may also result
in increased blade loads. Negative incidence
on the other hand, although steepening the
stick gradient, requires more cyclic for high
speed helicopter flight and thereby restricts
the speed in this mode.

(3)   Effect of Gross Weight on Level Flight Trim

The effect of gross weight on level flight trim
parameters is shown in Figure V-13. The para-
meters which are significantly affected by
gross weight are collective pitch, mast
torque, and blade flapping. Computed collec-
tive pitch and mast torque are in close agree-
ment with test data. A significant difference
was again found in computed lateral flapping
due to the simplified math model used for the
induced velocity distribution.

(4)   Effect of Wing Lift on Trim Parameters

The aerodynamic fairings were removed from the
wing spar to investigate the effect of wing
lift on the level flight trim parameters. Fig-
ure V-14 shows the trim parameters versus air-
speed, with the fairings removed. Comparison
of the wing fairings off and fairings on data,
Figure V-11, does not indicate a significant
influence.

Early in the test program wing aerodynamic
interference was suspected to be one of the
reasons for the higher than calculated lateral
flapping. However, flapping with the fairings
removed was the same as with the fairings on.

(5)   Descent and Autorotation Trim Parameters

Rate of descent or climb was determined from
the magnitude of the model's drag or thrust
on the vertical rods. The method is illustrated
in Figure V-15.

(5) (Continued)

Descent and autorotation trim parameters were
determined over a range of airspeed (50 to 90
knots), for flap settings of 0° and 45°, and
mast angles of 90° and 95°. Descent and auto-
rotation trim data is summarized in Figure V-16.
Autorotation was achieved at an equivalent full-
scale rate of descent of 4000 feet per minute
at 80 knots airspeed. The calculated full-
scale autorotation rate of descent at 80 knots
is 2400 feet per minute. Reynolds number
effects are considered to be responsible for
this difference between the model and full-
scale rates of descent.

Airframe and rotor characteristics were corrected
to test Reynolds numbers and used to compute the
rate of descent. Reasonable agreement with re-
gard to fuselage attitude, rotor power and con-
trol positions was obtained for rates of descent
up to about 2500 feet per minute. However, at
higher rates of descent the calculated fuselage
attitude was signficantly higher than measured.
Investigation showed that the wing was not
stalled in the computer program whereas during
the test tufts on the wing indicated flow over
the wing was separated at the higher rates of
descent. The inputs to the program were modified
to reflect the model's wing stall characteristics,
including the wing downwash above stall. Once
this was done the calculated and measured rates
of descent were in good agreement.

(6) Climb Trim Parameters

Structural interference limited the nose down
pitch attitude with respect to the rod to -10
degrees. Consequently, very little data on
climb trim parameters was obtained for mast
angle 90°. What data was obtained is presented
with the static stability data in sec-
tion V.B.2.

(7) Forward Flight IGE, Sideward and Rearward
Flight OGE

Tests were made in ground effect (IGE) at h/D =
.75 and .50 with a moving ground plane and bound-
ary layer suction. The moving ground plane was
operated at speeds corresponding to the tunnel's

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subject to the restriction on the title page.

(7) (Continued)

air velocity. The test data for forward flight
shown in Figure V-17 indicates the model to have
a more nose down fuselage pitch attitude IGE
than OGE. The cyclic stick position is more
aft than OGE indicating a significant upload on
the horizontal stabilizer. Torque was lower
during hover IGE, but was about the same as OGE
between 16 and 30 knots. More torque was re-
quired at 40 and 60 knots IGE than OGE.

Sideward flight was accomplished at airspeeds
up to 30 knots. As shown in Figure V-18,
approximately 4.6 degrees of differential cyclic
pitch was required to hold heading at 30 knots.
This is higher than the design value of 4.0.
However, the model did not have rudders, which
will tend to relieve the differential cyclic
requirement. One significant factor observed
during sideward flight was the increase in
power required resulting from the tandem rotor
effect. (The trailing rotor is in the downwash
of the leading rotor and therefore has a higher
induced power loss.)

Rearward flight was accomplished at speeds up
to 35 knots at both forward and aft cg positions.
Figure V-19 shows the trim parameters versus
airspeed. Eight degrees of aft cyclic was
required at 35 knots, forward cg. (Elevator
settings were not set at the proper value for
the cyclic position because of a physical limit on
up-elevator. The proper value would tend to
reduce the cyclic requirement slightly.) Blade
flapping at forward cg was high because of the
aft cyclic required to trim out the weight
moment.

b. Mast angle 75°

Figure V-20 shows the level flight trim parameters for
mast angle 75 degrees. Comparison between theory
(C81) and measured parameters is similar to chat
discussed for mast angle 90 degrees.

The longitudinal cyclic gradient with airspeed shows
a reversal between 30 knots and 60 knots that is more
pronounced than the reversal at mast angle 90 degrees.
The lateral flapping trend with airspeed is similar
to that for mast angle 90 degrees.

b. (Continued)

Several gross weights were tested at mast angle 75 degrees; trim positions are shown in Figure V-21. The most significant variation with gross weight was in power and flapping.

Figure V-22 shows the effect of flap setting on trim parameters. During runs at 75 degrees (and at 90 degrees) it was noticed that the tufts on the flaps indicated separation and spanwise flow. This was considered as a possible unrealistic representation of the wing wake. Vortex generators were added to the wing at the flap hinge line, but did not change the appearance of the tufts. The flaps were then raised to eliminate the separation. However, little difference was noted in the trim attitude or cyclic stick position, as is shown in Figure V-22.

c. Mast angles 60° and 30°

Figures V-23 and V-24 show trim parameters for mast angle 60 degrees and 30 degrees respectively. Only a limited amount of data was taken at these mast angles since analysis and test (the XV-3) indicate the significant handling qualities problems are at mast angle 90 and 75 degrees.

In comparing computed and test for mast angles of 60 and 30 degrees, caution should be exercised in using these as realistic trimmed flight conditions. Incorrect cyclic/elevator gearing used for these mast angles caused large differences in empennage lift. Proper elevator settings at aft cyclic positions were not obtained because of limits on up elevator deflections. In addition are the model scale effects (low $C_{L_{max}}$ and low test airspeeds) as discussed earlier, which make these trim conditions unrepresentative of the full scale aircraft trim.

2. Static Stability Characteristics

a. Longitudinal

Longitudinal static stability at mast angles 90 and 75 degrees was investigated by trimming the model over a range of fuselage trim attitudes. Figures V-25 and V-26 show the trim parameters versus fuselage pitch attitude for mast angles 90 and 75 degrees respectively. These trim points correspond to climbing or descending flight.

a. (Continued)

At mast angle 90 degrees the longitudinal cyclic
stick position gradient with fuselage pitch attitude
indicates positive static stability. However, at
mast angle 75 degrees and 30 knots, the stick
gradient shows approximately neutral static sta-
bility. The model pilot found it very difficult to
trim the model in this condition. Positive static
stability was indicated at the higher speeds tested
for mast angle 75 degrees.

b. Lateral directional

Lateral directional static stability was investigated
at mast angles 90 and 75 degrees by yawing the model
with differential cyclic and trimming roll and pitch
with differential collective and longitudinal cyclic
respectively. Figures V-27 and V-28 show the trim
parameters versus yaw angle for mast angles 90 and
75 degrees respectively.

The differential cyclic required to maintain a given
yaw angle was essentially linear over the range of
speed and yaw angle tested. There was a significant
variation in the longitudinal cyclic required for
trim with yaw angle. This variation reflects the
change in upload to download on the horizontal
stabilizer due to the rotor wake with yaw angle.

3. Dynamic Stability Characteristics

At each trim point the model was disturbed in pitch and
yaw to obtain dynamic stability data. The frequency
and damping of the pitching and yawing modes were
extracted from the time history of the model response.

Table V-1 compares calculated frequency and damping
with that measured in the test. In general, agreement
is good. Note that at the lower speeds the measured
frequency and damping are not given because the model
response was so sluggish that the frequency and damping
could not be accurately calculated from the time history.

4. Controllability

The evaluation given in Table V-2 with regard to model
controllability on the rod mount was recorded by the
model pilots: (all comments pertain to the 11,000
pound gross weight configuration).

5. <u>Aeroelastic Stability, Oscillatory Loads, and Vibration</u>

a. <u>Aeroelastic stability</u>

The aeroelastic stability characteristics were investigated by exciting the coupled rotor-pylon-wing modes and observing the decay of the motion. The model was excited by plucking wires attached to the left hand nacelle and by bouncing the model on the vertical restraint cable. The damping was monitored visually and on an oscillograph.

Rotor-pylon-wing motions were well damped with no evidence of instability at any conversion angle or speed. It was not possible to accurately reduce the damping using conventional decay analysis because the steady-state forced vibration partially obscured the decay. Since aeroelastic stability has not been identified as a problem in helicopter and conversion mode, no attempt was made to reduce the data using other methods.

b. <u>Oscillatory loads</u>

Rotor and control system oscillatory loads were monitored during the test to avoid exceeding limits. Figure V-29 shows typical waveforms of the rotor loads and flapping. Loads were predominantly one-per-rev at all mast angles and airspeeds. One-per-rev loads were also predominant in the 25-foot prop-rotor test in the 40- by 80-foot wind tunnel.

Figures V-30 through 32 show measured beamwise and chordwise bending at blade station 52.5, and the pitch link load versus airspeed, for the four conversion angles tested. Blade and control loads were well below the equivalent full scale endurance limit and show a reduction in magnitude as the nacelles convert from helicopter to airplane mode. Figure V-33 shows the variation in blade and control loads with gross weight at constant airspeed. The load trend does not indicate the occurrence of stall flutter or other aeroelastic instability.

It should be noted that blade and control loads are well below the loads predicted for the Model 301[17]. This is caused by (1) the model rotor not being Mach scaled when tested in air (scale factor is 0.447) so that drag rise at high lift coefficients is not correct, and (2) the model blade first inplane natural frequency ratio is slightly higher than that for the full-scale blade.

c. Airframe vibration

Vibration levels were measured at seven locations:
three on the right hand nacelle, one on the left
hand nacelle, one on each of the vertical fins and
one at the aircraft center of gravity. The nacelle
accelerometers could be moved to monitor vibrations
in either the fore-and-aft or the lateral sense.
The vertical fin accelerometers measured in the
fore-and-aft sense and the accelerometer at the
aircraft center of gravity measured in the vertical
sense.

Figure V-34 shows the frequency content of the vibra-
tion of four stations. The dominant frequency is
one-per-rev (21 cps) and is caused by rotor out of
balance and/or out of track. The operating toler-
ance on out of balance and out of track is much
greater for the model than full-scale. Hence, the
one-per-rev vibration is not considered representa-
tive of full-scale.

The two-per-rev vibration is caused by the Hooke's
joint effects of the rotor gimbals in combination
with rotor flapping. With a gimballed rotor, flap-
ping induces a two-per-rev torque and moment at the
hub which are proportional to the square of the
flapping angle. Figure V-35 shows that the maximum
two-per-rev vibration occurs at approximately 40
knots which is the airspeed at which flapping was
the maximum.

The three-per-rev vibration results from two-per-
rev and three-per-rev airloads. The three-per-rev
vibration level, shown in Figure V-36 has a peak
near 40 knots where the rotor near wake has the
most significant effect and then begins to increase
again about 80 knots as the rotor advance ratio be-
comes signfiicant. The variation in vibration level
with conversion angle is shown in Figures V-37 and
V-38 for two-per-rev and three-per-rev respectively.
In general the level of vibration decreases as the
nacelles are converted forward.

Both the two-per-rev and three-per-rev vibration
are representative of the full-scale aircraft. The
levels are within design limits. Analysis of the
airframe oscillatory load level corresponding to the
measured vibration levels, indicate the loads are low
with respect to structural allowables.

c. (Continued)

Empennage vibration levels were monitored to deter-
mine if the rotor wake excited empennage natural
frequencies. While some natural frequency excita-
tion was evident it was very low in mag..tude. The
dominant vibration frequency was one-per-rev but
as noted earlier the one-per-rev level is not repre-
sentative of full-scale.

C. **Sting Mount Test**

The sting mount test covered the same ranges of mast angle
and airspeed covered in the rod mount test. Control posi-
tions and pitch attitude were based on the trim values
determined during the rod mount test. The strain gage
balance was used to verify that lift and pitching moment
were in trim; when indicated, collective pitch and/or
longitudinal cyclic was adjusted to improve the trim condi-
tion. In some cases climb and descent trim were estimated
by extrapolating trim data from the rod test since only a
limited amount of data for these conditions had been obtained.

A tuft grid was placed in the vicinity of the empennage in
order to observe the rotor wake. Photographs of the tuft
grid patterns are given in Appendix C. An analysis of the
rotor wake observations is presented in Section VI-A.

1. **Comparison of Airframe Characteristics with Previous**
   Test Data

   At the low Reynolds number tested, the stall character-
   istics are significantly different than estimated for
   the full scale aircraft. Stall occurred at 12 degrees
   which is 4 degrees lower than that obtained on the force
   model at a higher Reynolds number. The maximum lift
   coefficient for the aeroelastic model was 0.44 less
   than the force model. A comparison of the aeroelastic
   model with the force model showing the effect of Reynolds
   number is presented in Figure V-39. The reduction in
   maximum lift coefficient had a significant effect on
   simulation of autorotation as discussed in Section V-B.

   There was very little difference in the lateral-direc-
   tional characteristics as shown in Figure V-40.

2. **Comparison of Rod and Sting Mount Trim Parameters**

   Figure V-41 shows comparison of typical trimmed level
   flight attitudes and control positions used during the
   rod and sting tests. Fuselage pitch attitude, cyclic
   position, and torque settings used during the sting

2. (Continued)

test were preset based on trim settings determined during
the rod test. During the sting test torque was found to
be the quickest and best indicator of the model being in
trim. Higher collective pitch settings were required
during the sting test than the rod test for the same
torque readings. Checks made of model lift on the tunnel
balance verified the model to be in trim and that the
torque to be indicating properly. A difference in collec-
tive pitch calibration is suspected to be the reason
for the difference between the tests. Collective pitch
settings obtained during the sting test are in better
agreement with the theoretically computed values.
Empennage lift and rotor flapping, the two independent
parameters in the two tests, are in close agreement.

3. Static Stability Characteristics

Force and moment data was obtained for rotors and/or
empennage on and off to evaluate rotor wake effects.
Pitch and yaw sweeps were made about the trim attitude
for level, climb, and descent flight configurations.
Lift, pitching moment, and yawing moment data are
summarized in this section for level flight from 40 to
100 knots. A complete set of force and moment data
at lower airspeeds, and climb and descent are given in
Appendix B.

a. Lift coefficient, mast angle 90 degrees

A comparison between rotors-off and rotors-on lift
coefficients at 40, 60, and 80 knots is shown in
Figures V-42, V-43, and V-44. (Lift coefficient
is based on wing area and freestream dynamic pres-
sure.)

b. Pitching moment coefficient, mast 90 degrees

Pitching moment characteristics for the same speed
range are shown in Figures V-45, V-46, and V-47.
The change in pitching moment with angle of attack
(C_{m_α}), rotors-off, does not change with airspeed.
Wing downwash and pitching moment at zero pitch
($C_{m_{\alpha = 0}}$), empennage off, changes due to Reynolds
number effects. These effects combine causing the
shift in $C_{m_{\alpha = 0}}$, empennage on. At the low Reynolds
numbers, the lift curve slope of the horizontal tail

b. (Continued)

$(C_{L_{\alpha_H}})$ is reduced and results in lower static longi-

tudinal stability (C_{m_α}) than for the full scale

airframe. The comparison of the aeroelastic model with the force model, given in Figure V-39 shows the effect of Reynolds number on C_{m_α}. The break in

the pitching moment curve at 12 degrees is due to wing stall.

The effect of the rotor wake on the pitching moment is apparent when the rotors-on and rotors-off pitch- ing moments are compared. At 40 knots the model is unstable for negative angles of attack and stable at positive angles. At 60 and 80 knots the model is stable on either side of trim. The change in pitching moment due to the empennage indicates that the wake changes from a downwash with rotors-off to an upwash with rotors-on (see Section VI for a more detailed discussion of rotor wake effects).

The pitching moment characteristics may be explained as follows: at 40 knots and at negative angles of attack, the rate of change of the upwash at the hor- izontal stabilizer with angle of attack $(d_{\epsilon_T}/d_\alpha)$ is

greater than 1.0, which cancels the stabilizing effect of the horizontal stabilizer. At positive angle of attack, d_{ϵ_T}/d_α is nearly zero so the slope of the pitching moment is about the same as rotors- off. At 60 and 80 knots, d_{ϵ_T}/d_α is lower than

$d_{\epsilon_{W/H}}/d_\alpha$ (rotors-off wing downwash) making the air-

craft more stable rotors-on than rotors-off. The unstable bump in the pitching moment coefficient around 8 degrees angle of attack at 60 and 80 knots is not understood. However, it may be due to the rotor vortices impinging on the horizontal stabilizer and causing the flow to separate.

c. <u>Yawing moment coefficient, mast 90 degrees</u>

Directional stability with rotors-on and -off at 40, 60, and 80 knots is shown in Figures V-49 through V-50 for up to 20 degrees of yaw. Again, the rotor wake effect on the empennage is significant at 40 knots and decreases with increasing airspeed.

c. (Continued)

Very little change in rotors-off directional sta-
bility is noted with airspeed for either empennage-
off or on. A change in slope occurs at about 6
degrees yaw which appears to be a Reynolds number
effect on the vertical fins.

At 40 knots, rotors-on, directional stability is
neutral for yaw angles less than 4 degrees and in-
creases in stability between 4 and 20 degrees yaw.
As speed increases, stability improved between ±4°
and decreases between 4 and 20 degrees. For the air-
speeds tested, directional stability above 12 degrees
yaw is greater rotors-on than rotors-off.

d. Lift coefficient, Mast angle 75 degrees

Lift characteristics at mast angle 75 degrees are
very similar to those for mast angle 90 degrees and
are shown in Figures V-51, V-52, and V-53 respec-
tively.

e. Pitching Moment Coefficient, Mast angle 75 degrees

Pitching moment characteristics are shown in Figures
V-54, V-55, and V-56. Rotors-off characteristics
with airspeed and angle of attack are similar to
mast angle 90 degrees.

At 40 knots, rotors-on, empennage-on, the aircraft
pitching moment shows a stable slope below trim,
changing to an unstable slope above trim and then
returns to a stable slope after wing stall. The
pitch up which is also present at 60 knots, makes
the aircraft unstable about level flight trim. The
pitch up tendency disappears at 80 knots, when the
pitching moment becomes slightly stable. This trend
is different than at mast angle 90 degrees. At mast
angle 90 degrees, rotors-on stability was better
than rotors off (except at 40 knots). At mast angle
75 degrees the pitching moment varies with angle of
attack and airspeed, and is less stable at the
higher airspeed.

The unstable region in the pitching moment curve
also exists at mast angle 90 degrees, but does not
occur near the level flight trim attitude. Level
flight trim for mast angle 90 degrees ranges from
0 to -8 degrees, several degrees below the pitch-
up, whereas trim for mast 75 degrees is at angles
of attack ranging from 8 to 2 degrees, right in the

e. (Continued)

middle of the pitch up. Referring to climb and
descent data presented in Appendix B, it is seen
that the pitch up occurs near these trim attitudes
and would also have an effect on stability for these
configurations. This change in stability with pitch
attitude is caused by the relationship of the rotors
tip vortices to the horizontal stabilizer. At small
and negative angles of attack the horizontal sta-
bilizer moves closer to the core of the vortices.
The rate of change of the tail angle of attack with
change in pitch attitude $(\partial\alpha_H/\partial\alpha_F)$ is generally

less in this region because of the strength of the
rotor wake relative to the freestream. At positive
angles of attack the horizontal stabilizer moves
farther away from the vortices. In this region
the total wake changes more rapidly with angle of
attack, in some cases at nearly the same rate as
the change aircraft pitch attitude eliminating the
stabilizing contribution of the horizontal stabilizer
$(\partial\alpha_H/\partial\alpha_F = 0)$. It should be noted that this pitch up

is dependent on the relationship of mast angle and
airspeed and will not occur at the same angle of
attack for all cases. The strength of the rotor wake,
wing wake, and freestream must all be included. As
speed increases this trend continues but the rotor
wake weakens until the main influence on stability
is the wing downwash. The rotors give an upwash in
the 40 to 80 knots speed range.

f. **Yawing Moment Coefficient, Mast angle 75 degrees**

Directional stability for mast angle 75 degrees at
40, 60, and 80 knots is shown in Figures V-57, V-58,
and V-59 respectively. Rotors-off, the yawing moment
is very similar in magnitude and characteristics as
for mast angle 90 degrees.

Rotors-on tests showed increased directional stability
at yaw angles greater than six degrees. As airspeed
increases directional stability rotors-on reduces at
high yaw angles to nearly the level of rotors-off.
At 40 knots, rotors-on, the rotor wake effects are
more severe than for mast angle 90 degrees. Between
yaw angles of ±2.0 degrees the model was directionally
unstable. As airspeed increased the model becomes
more stable.

g. **Lift Coefficient, Mast Angles 60 and 30 degrees**

Lift coefficients at 80 and 100 knots for mast
angles of 60 and 30 degrees are shown in Figures
V-60 and V-61.

h. **Pitching Moment Coefficient, Mast Angle 60 and 30
degrees**

Pitching moment coefficient versus angle of attack
for the same speeds are shown in Figures V-62 and
V-63. The rotors-off pitching moment is similar to
that at the other mast angles. The rotors-on pitch-
ing moment has a pitch up region at 80 knots, mast
60, as it did at mast angles 90 and 75 degrees at
lower airspeeds. The pitch up does not occur at
100 knots or at mast angle of 30 degrees.

Rotors-on, empennage off data, shows very little
destabilizing effect due to the rotor. Empennage
on, the rotor wake effect is stabilizing in that
the rotors-on pitching moment is more stable than
the rotors-off.

i. **Yawing Moment Coefficient, Mast angle 60 and 30
degrees**

Directional stability for 80 and 100 knots is very
good as shown in Figures V-64 and V-65. Rotor wake
effects on the vertical fins' aerodynamics are still
indicated to be destabilizing but do not reduce the
directional stability between ±4 degrees as much as
at mast angles 90 and 75 degrees. Directional sta-
bility is still increased, rotors-on, at yaw angles
greater than 4 degrees.

TABLE V-1.   CORRELATION BETWEEN CALCULATED AND MEASURED
DYNAMIC STABILITY CHARACTERISTICS

Pitch

Mast Angle	Airspeed Knots	Measured[1] Period/Damping	Calculated[1] Period/Damping
90°	80	3.6 sec/ζ = .41	3.2 sec/ζ = .66 (3.62 sec/ζ = .41)[2]
	90	2.7 sec/ζ = .45	2.98 sec/ζ = .64
	100	2.34 sec/ζ = .35	2.7/ζ = .62
75°	60	3.42 sec/ζ = .33	4.4 sec/ζ = .67
60°	—	(Data At This Mast Angle Not Usable)	—
30°	100	3.03 sec/ζ = .42	2.0 sec/ζ = .47

Yaw

Mast Angle	Airspeed Knots	Measured Period/Damping	Calculated Period/Damping
90°	100	5.6 sec/ζ = .190	4.6 sec/ζ = .14
75°	100	5.6 sec/ζ = .17	4.3 sec/ζ = .17
60°	80	7.05 sec/ζ = .25	5.6 sec/ζ = .24
30°	—	(Data At This Mast Angle Not Usable)	—

1.   Full-scale equivalent

2.   Calculated used measured static derivatives and with
     dynamic derivatives corrected for Reynolds number

TABLE V-2 EVALUATION OF MODEL CONTROLIABILITY

Conversion Angle	Airspeed	Evaluation
90°	Hover (IGE and OGE)	Very difficult to control and impossible to stabilize on trim. (Recirculation was evident in the test section.)
90°	16 to 20 Knots	Roll and pitch control very difficult but could be stabilized on trim. Directional control fairly easy. Height control easy.
90°	20 to 30 Knots	Roll control difficult but pitch control much easier. Directional and height control easy.
90°	35 Knots	All axes fairly solid at this speed. Easy to obtain trim condition.
90°	50 to 100 Knots	Model handles very well, trim flight is easy to obtain and can be maintained hands off.
90°	Sideward Flight	Roll control difficult. Pitch control easy and directional control fairly easy. Difficult to control height.
90°	Rearward Flight	Roll control difficult up to 20 knots, then less difficult up to 35 knots. Pitch control fairly easy. Yaw control power adequate to maintain directional control.

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TABLE V-2 EVALUATION OF MODEL CONTROLLABILITY (CONT)

Conversion Angle	Airspeed	Evaluation
90°	Descent and Auto-rotation (50 to 80 Knots)	Easy to control up to the point where wing stall occurs, then porpoising in pitch was present.
75°	All Speeds	*Comments are essentially the same as for mast angle 90°.
60° and 30°	80 to 100 Knots	Easy to control about all axes.

*Except for trim at 30 knots where the model was difficult to hold in trim.

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SEE FIGURE IV-1 FOR
TEST RIG SCHEMATIC

HOVER TEST
MAST ANGLE = 90°
FLAPS = $75/45^\circ$
WING FAIRING ON
h/D = 0.5
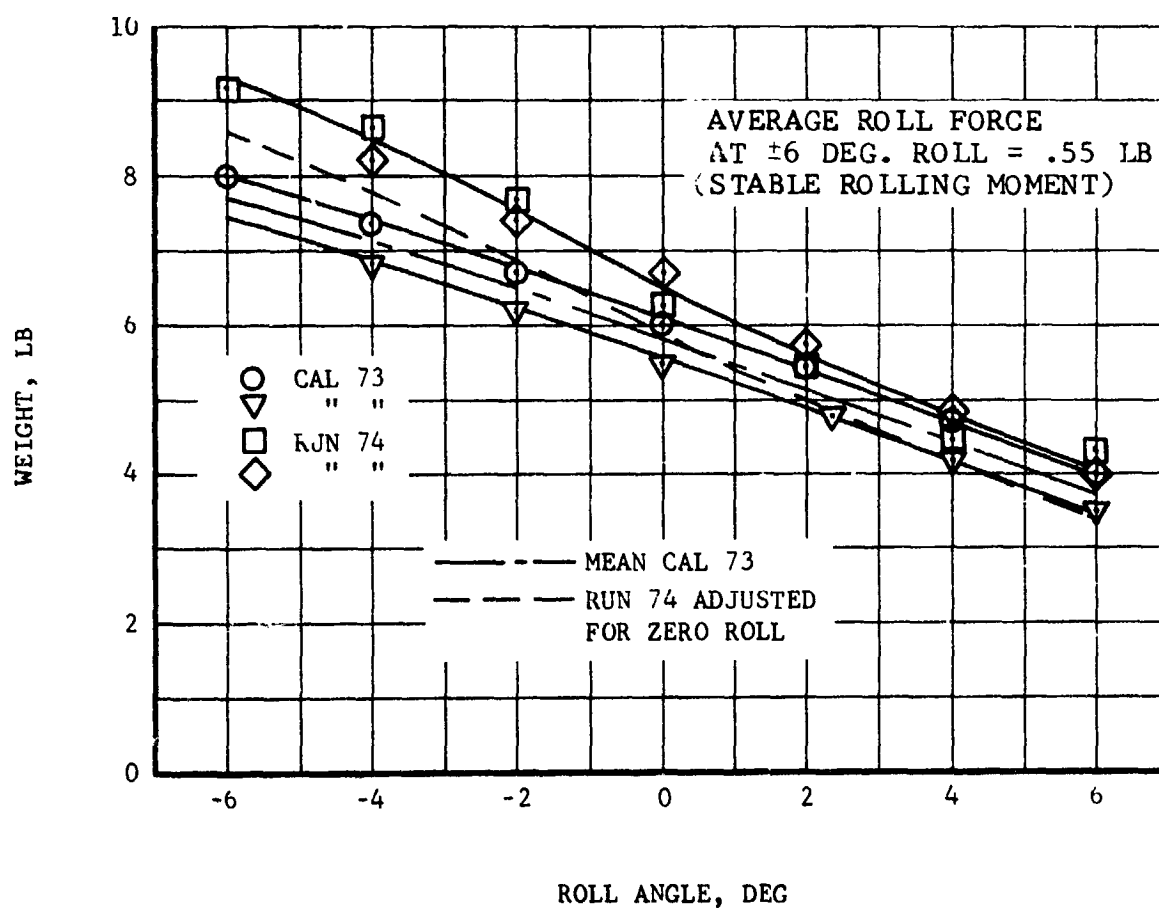
Figure V-1. Weight Required to Maintain a Given
Roll Angle versus Roll Angle, h/D = 0.5,
Wing Fairings On.

HOVER TEST
MAST ANGLE = 90°
FLAPS = $75/45^\circ$
WING FAIRING ON
h/D = 1.0
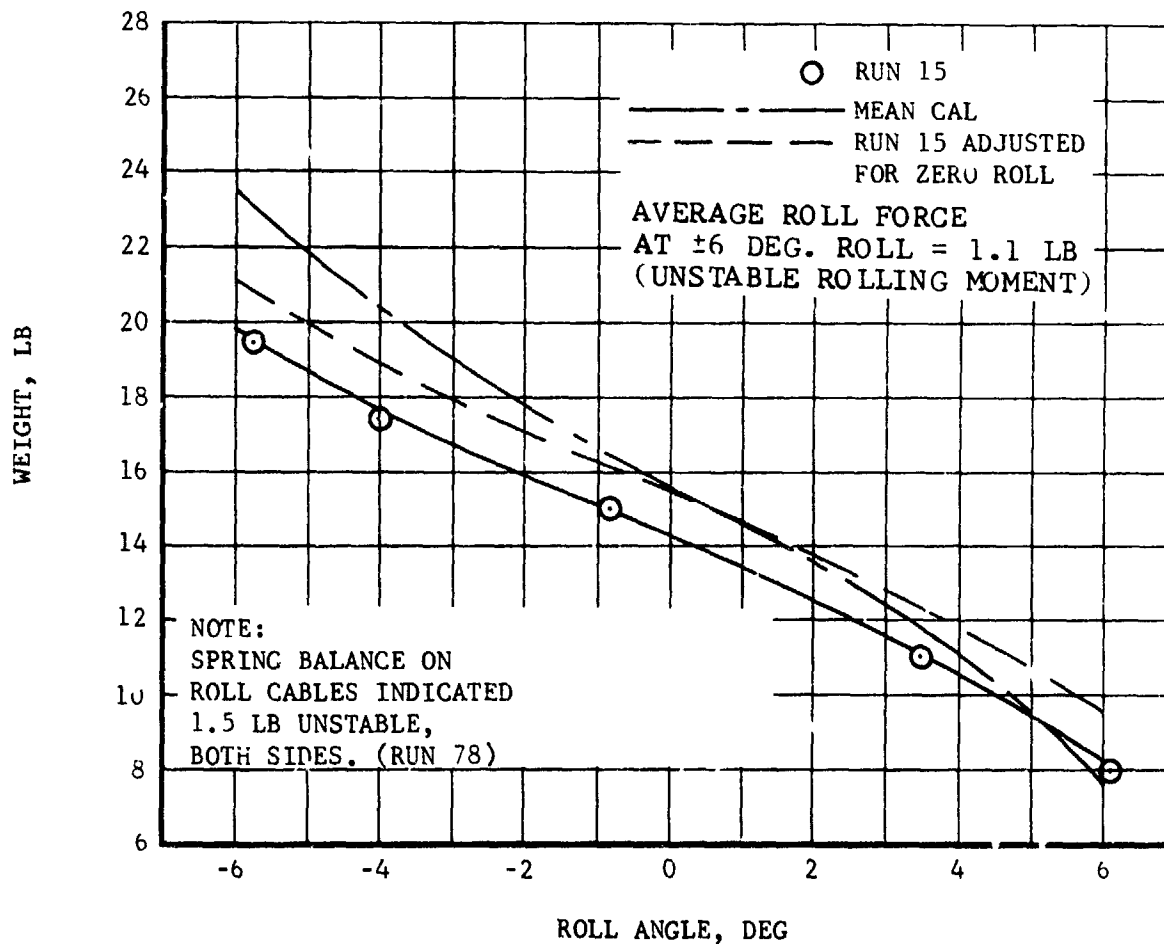
Figure V-2.   Weight Required to Maintain a Given
              Roll Angle versus Roll Angle, h/D = 0.75,
              Wing Fairings On.

HOVER TEST
MAST ANGLE = 90°

GW = 8,250 LB
○ WING FAIRING ON,
δ_F = 75/45°
◇ WING FAIRING OFF

---EST AT GW = 12,400 LB

LATERAL STICK TO TRIM, INCH/DEG ROLL

STABLE   UNSTABLE

FULL SCALE MOMENT/DEG ROLL, FT LB

1/5 SCALE MOMENT/DEG ROLL, FT LB
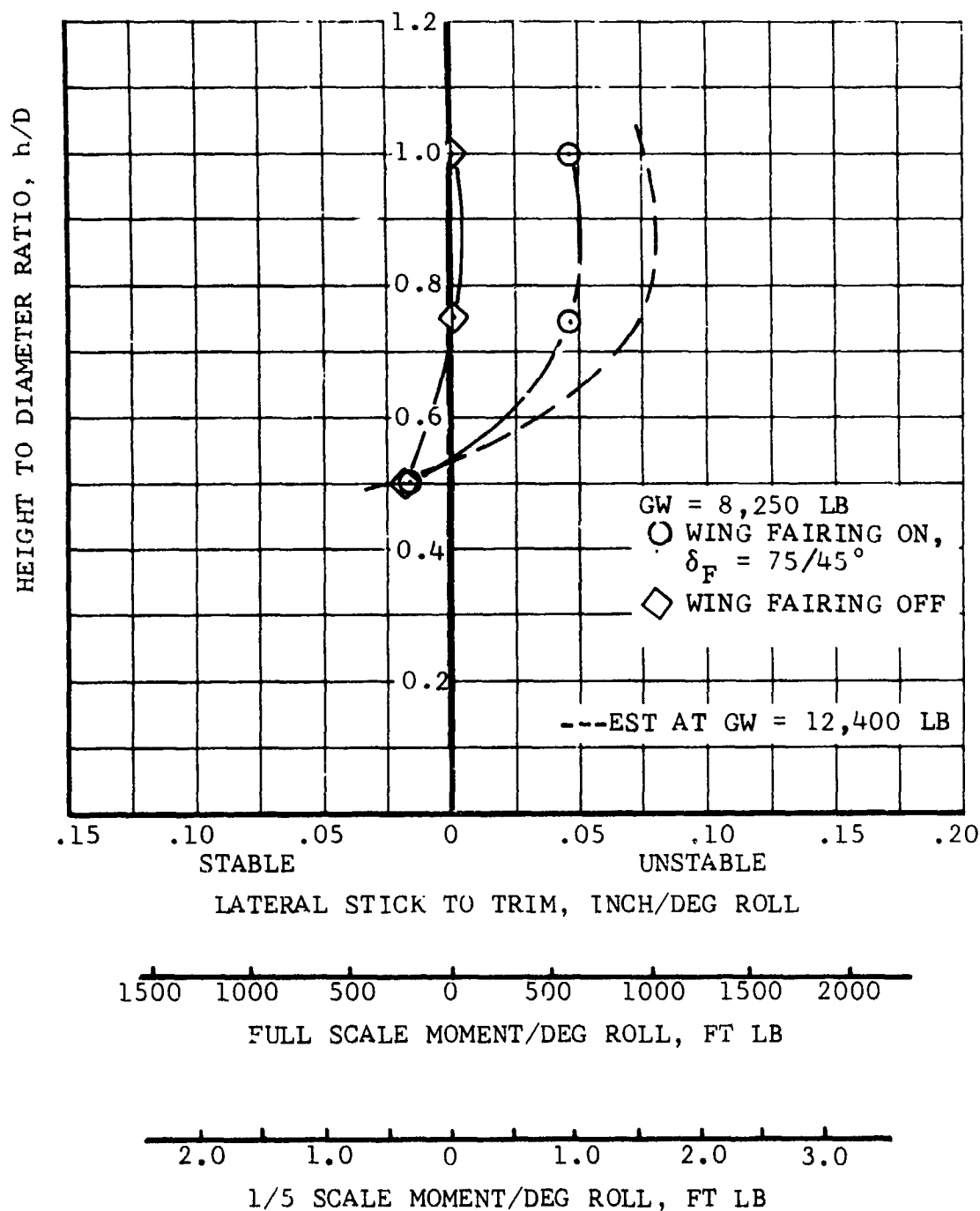
Figure V-3.   Summary of Roll Stability Characteristics
in IGE Hover.

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HOVER TEST
MAST ANGLE = 90°
h/D = 1.0
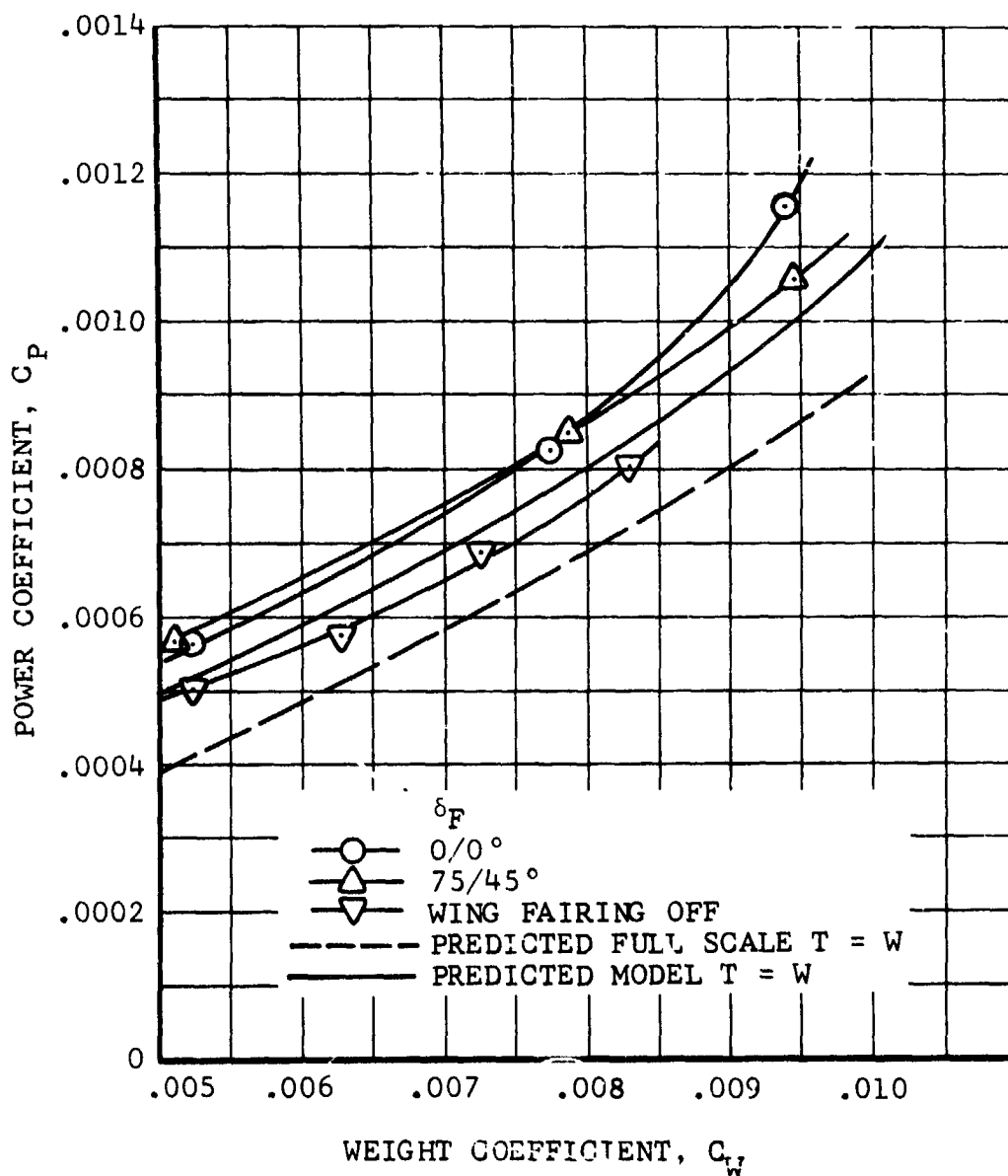
Figure V-4. Hover Performance, h/D = 1.0.

HOVER TEST
MAST ANGLE = 90°
h/D = .75
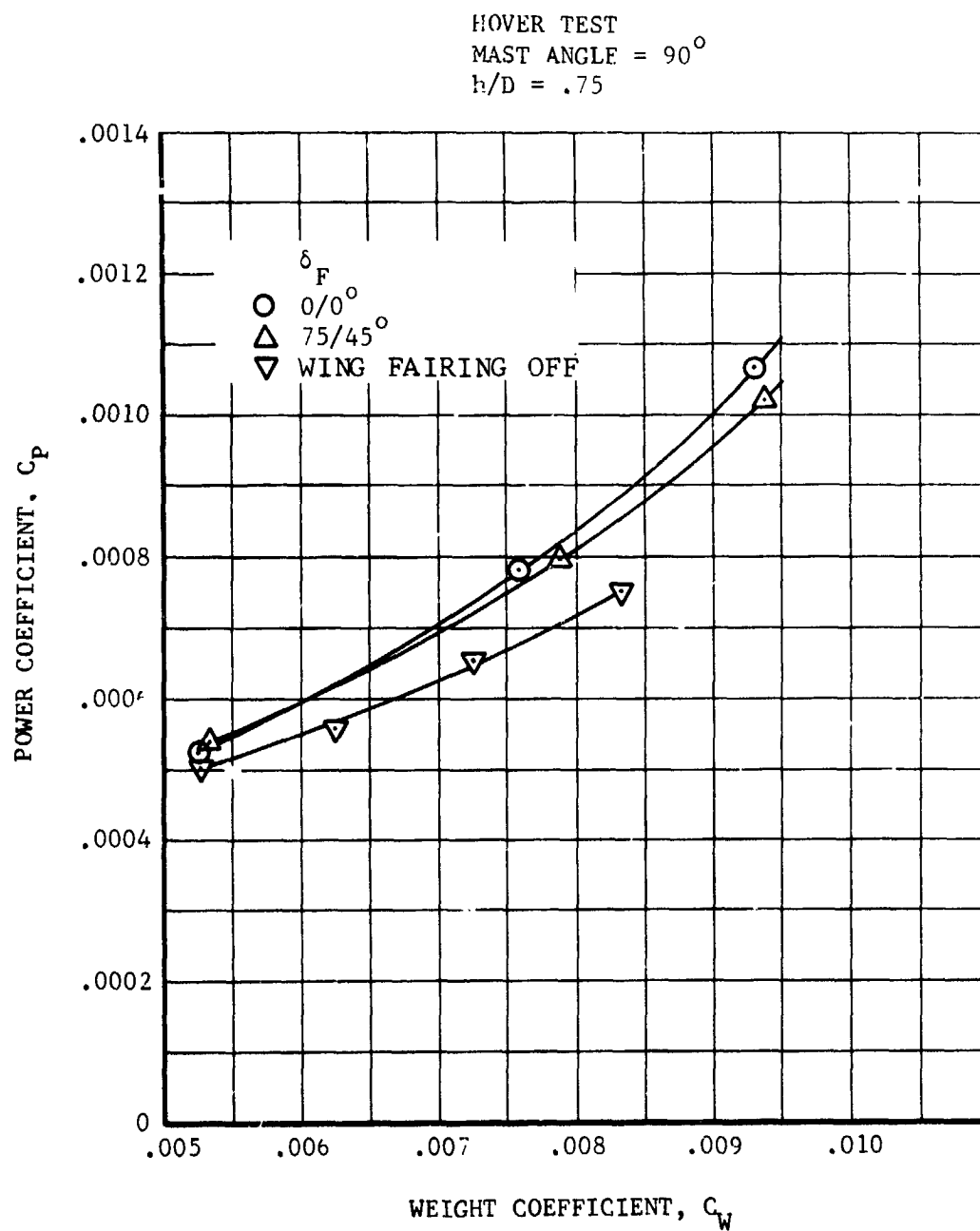
Figure V-5. Hover Performance, h/D = .75.

HOVER TEST
MAST ANGLE = 90°
h/D = .50
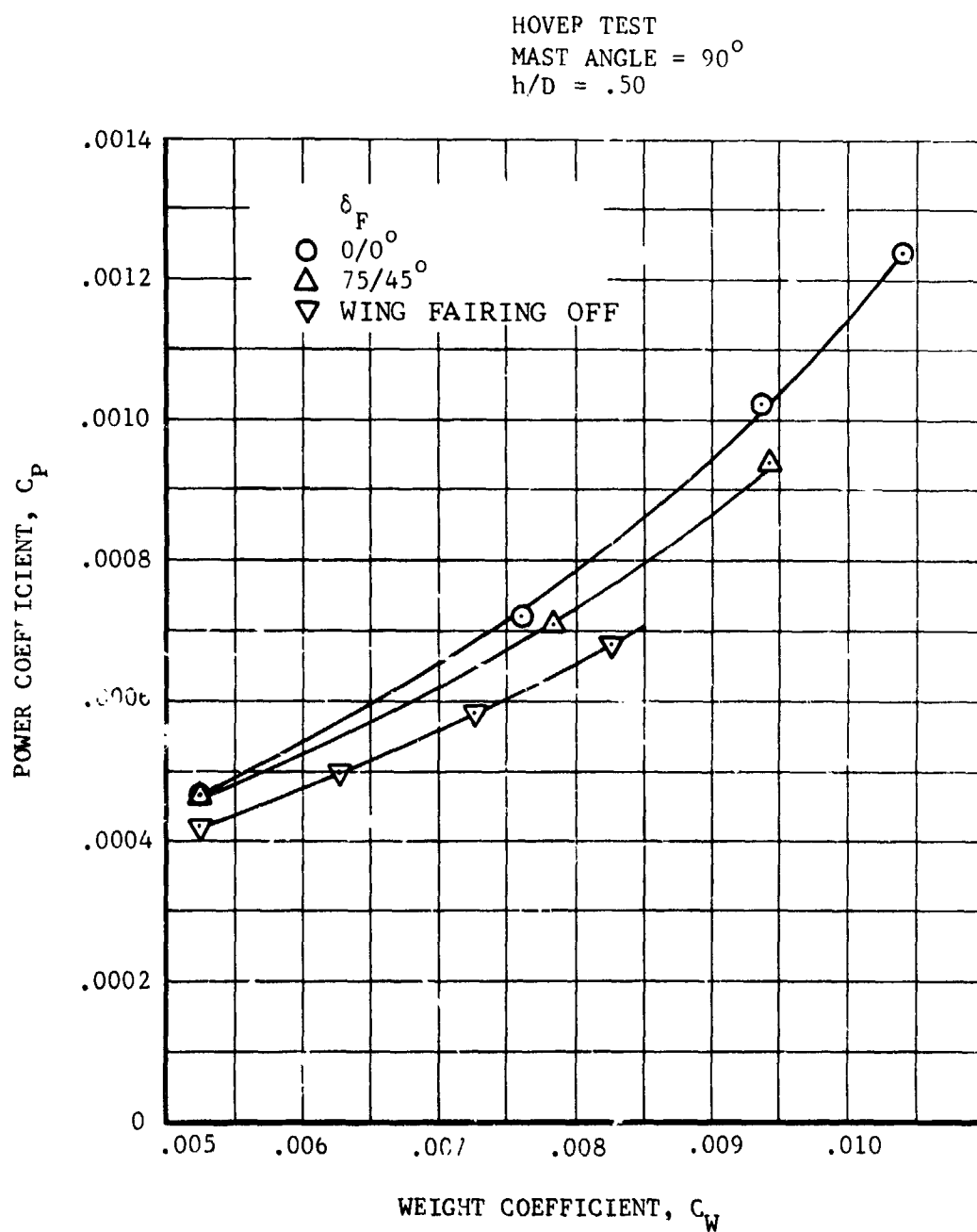
Figure V-6. Hover Performance, h/D = 0.5

○ 1/5 SCALE MODEL, RN < 2 X 10^5
▽ 1/10 SCALE C100F1B, RN = 4 X 10^5
▨ MODEL 301 FULL SCALE, RN > 2 X 10^6

HEIGHT TO DIAMETER RATIO, h/D

1.2

1.0

0.8

0.6

0.4

0.2

-8   -4   0   4   8   12   16

$\dfrac{\text{WING DOWNLOAD}}{\text{ROTOR THRUST}}$, %
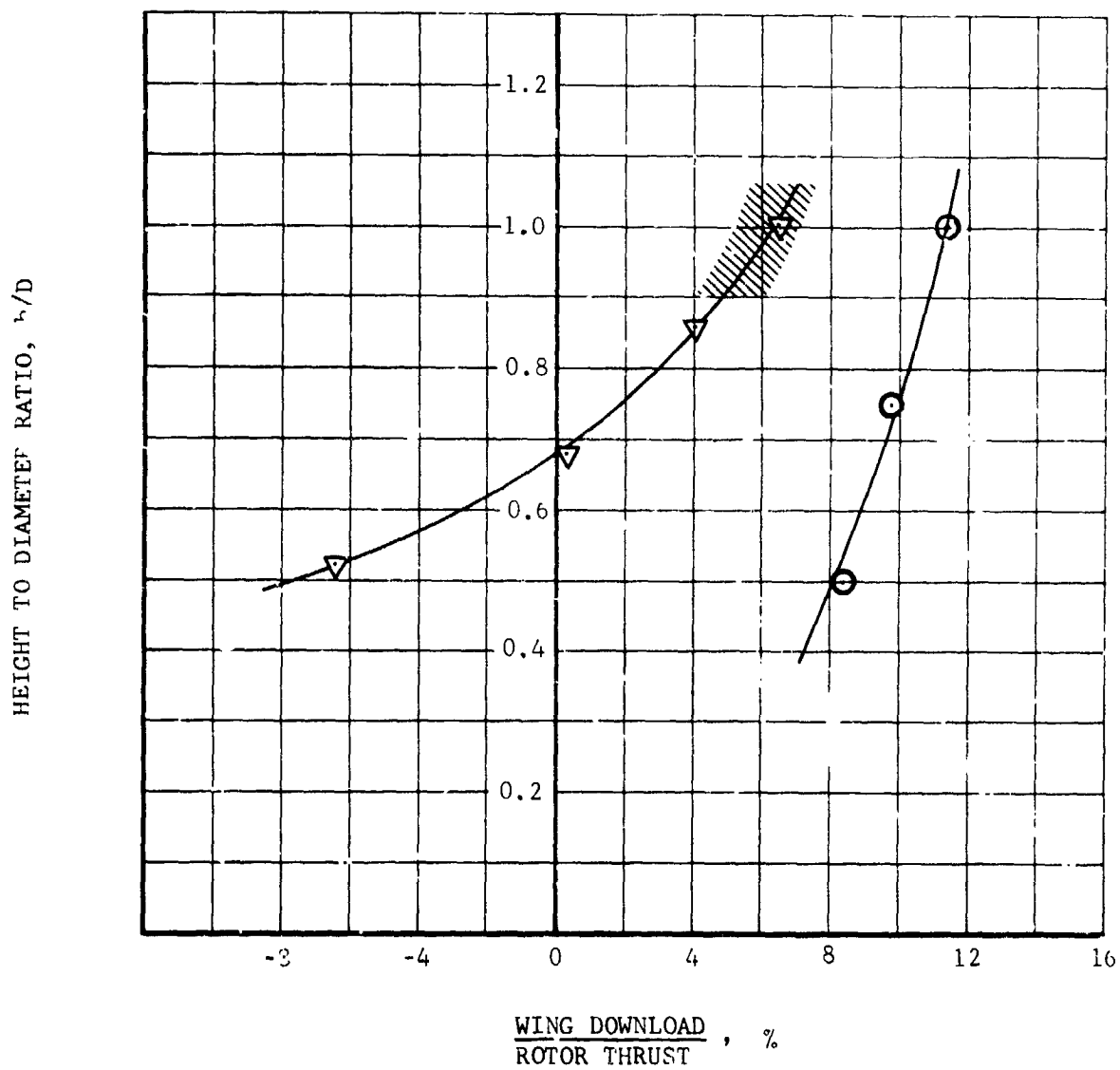
Figure V-7. Wing Down Load Summary, IGE.

BASED ON CROSS-SECTIONAL SHAPES
WITH ROUNDED EDGES AND FLAT SIDES
REF: FIG 13, SEC 3-9, HOERNER (1965)

1/5-SCALE AEROELASTIC MODEL
(C300A2B)

C100F1B

MODEL 301

NOTE: RN BASED ON WING CHORD
AND DOWNWASH VELOCITY
IN HOVER
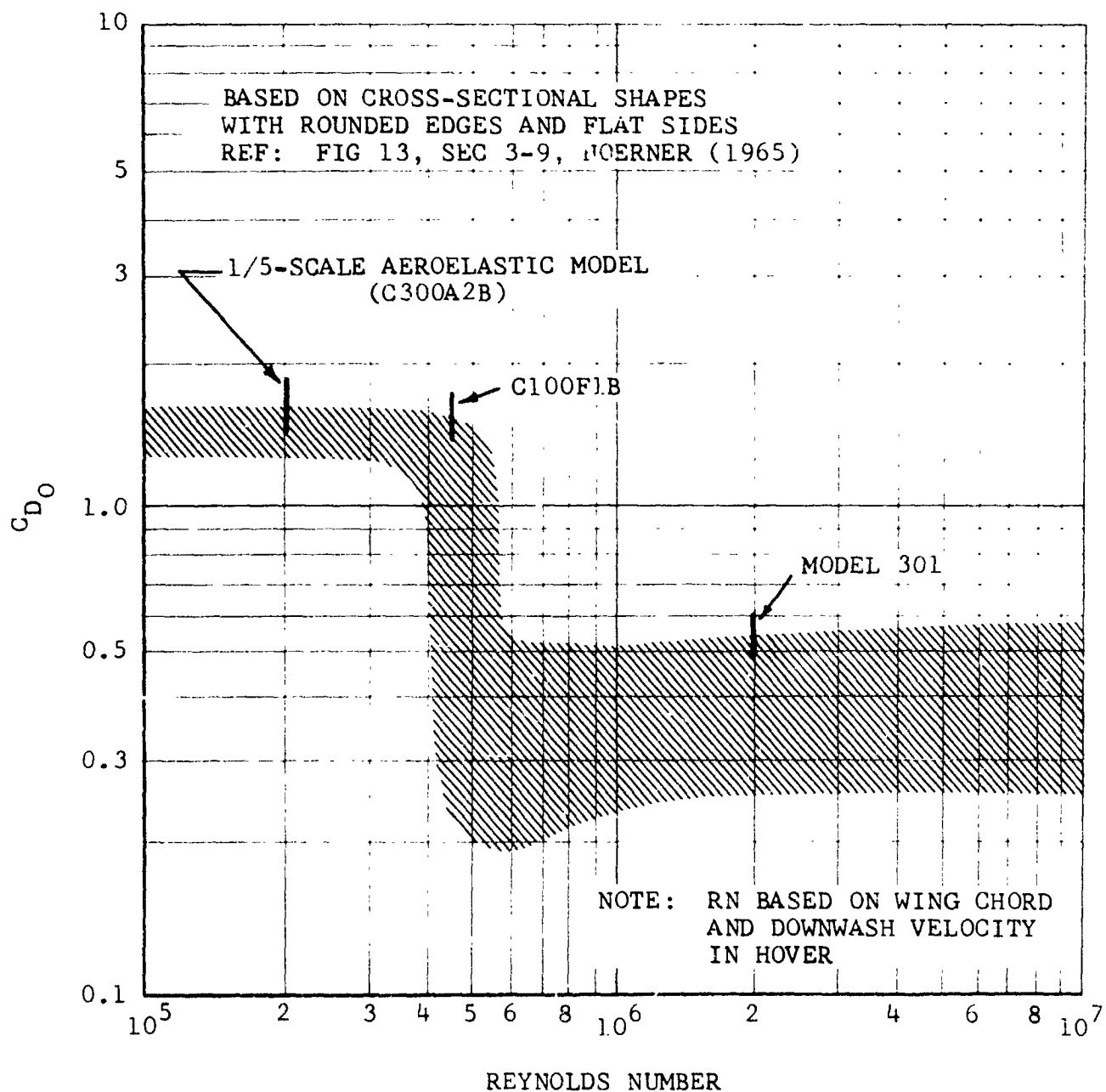
REYNOLDS NUMBER

Figure V-8.   Drag Coefficient of Wing at $\alpha_w = -90°$
verus Reynolds Number.

HOVER TEST
MAST ANGLE = 90°
FLAPS = $75/45^o$
WING FAIRING ON

$$\text{POWER REDUCTION} = \frac{HP_{h/D = 1.0} - HP_{h/D = 0.5}}{HP_{h/D = 1.0}}$$

POWER REDUCTION, %

GW/σ', LB
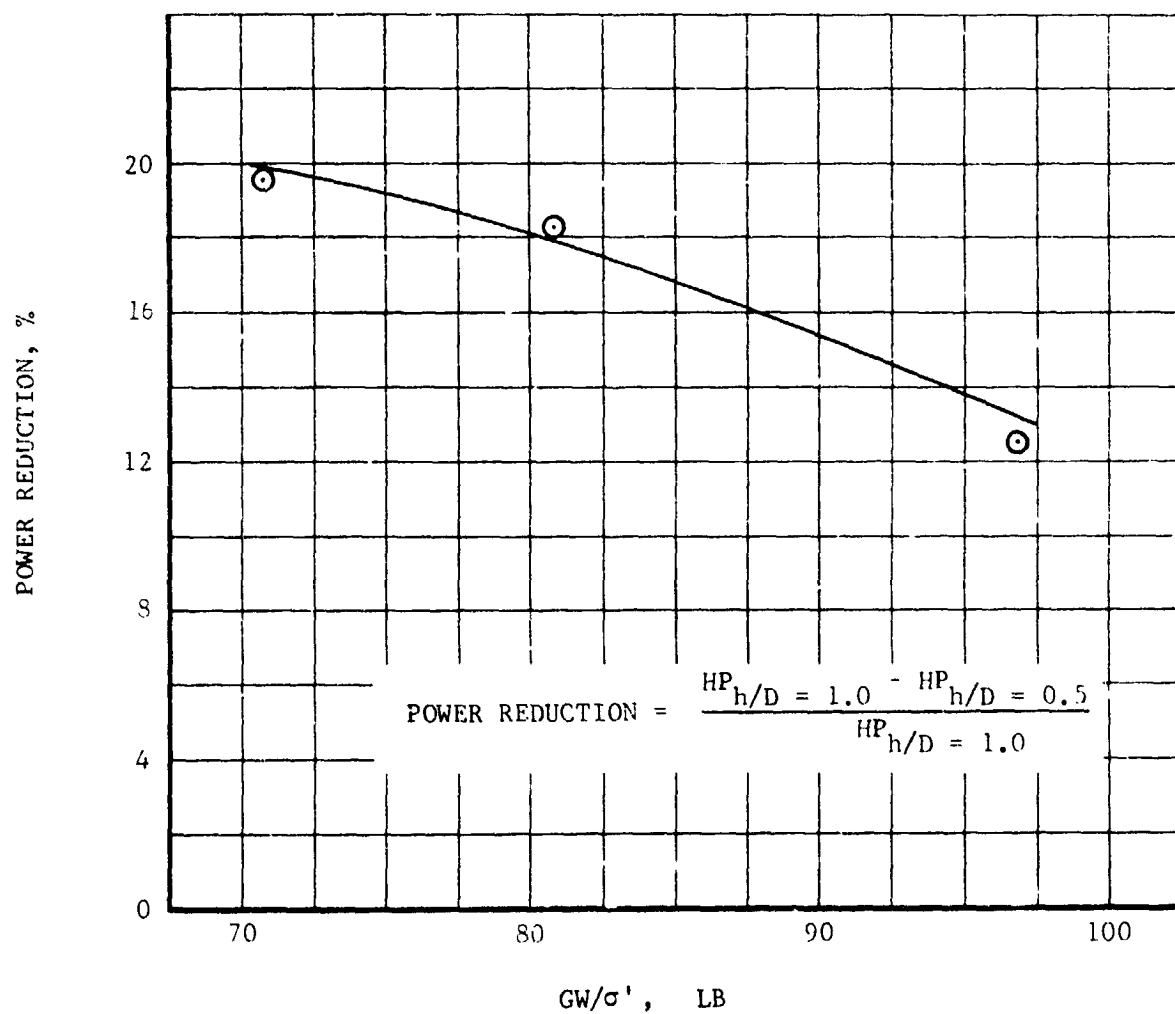
Figure V-9. Power Reduction in Ground Effect.
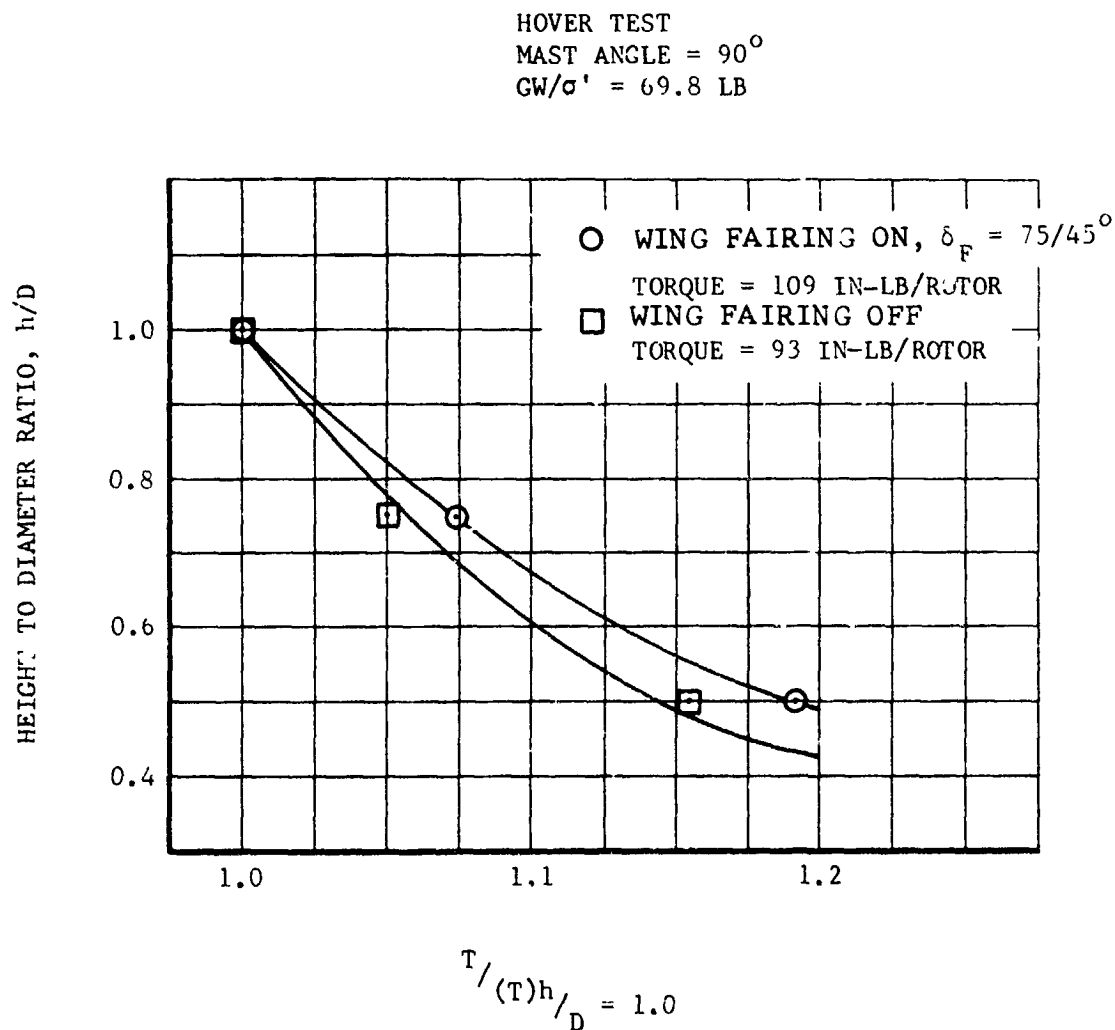
Figure V-10.   Vertical "Spring" in Ground Effect.
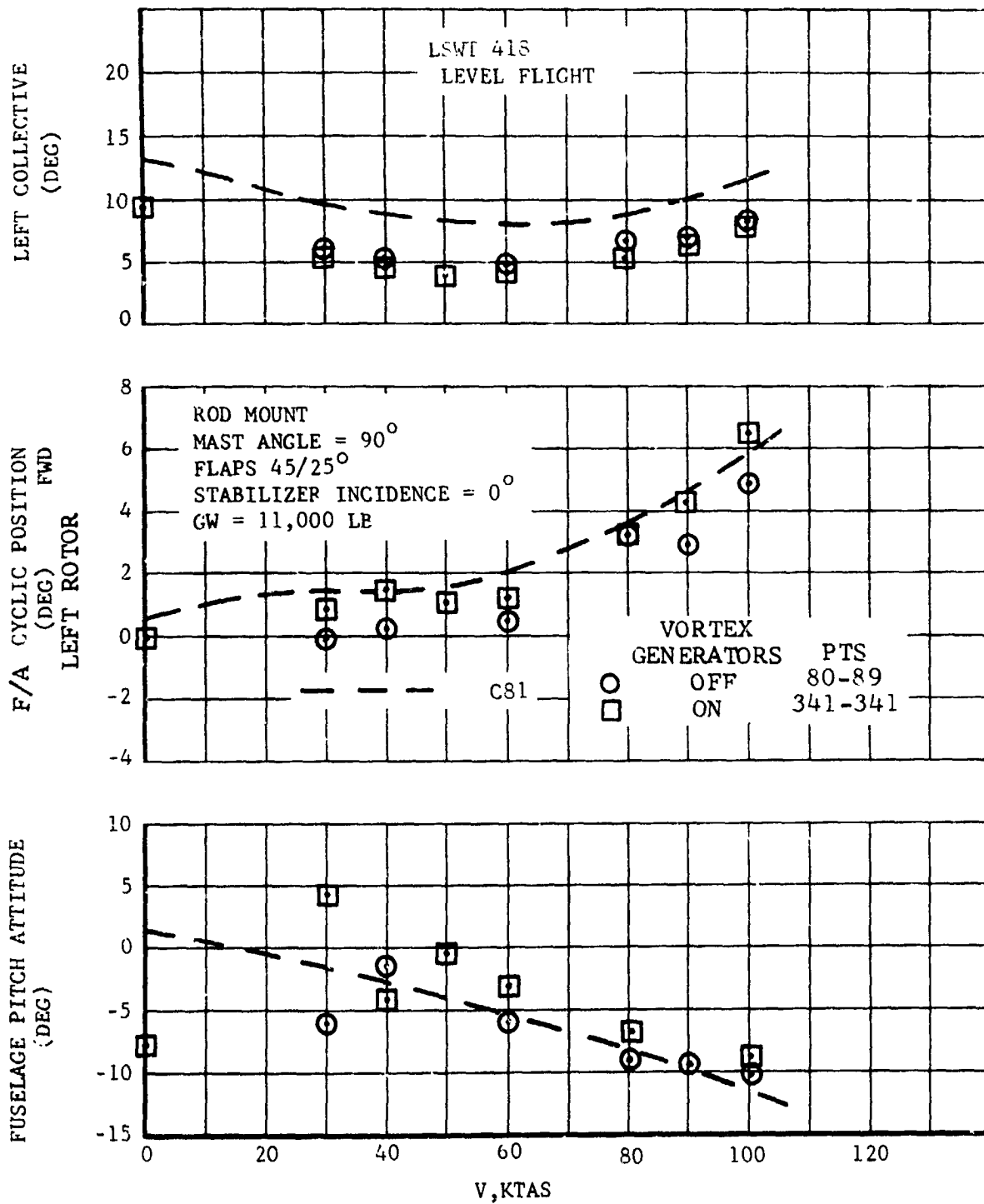
Figure V-11. Level Flight Trim Parameters,
Mast Angle 90°.

Figure V-11.   Continued

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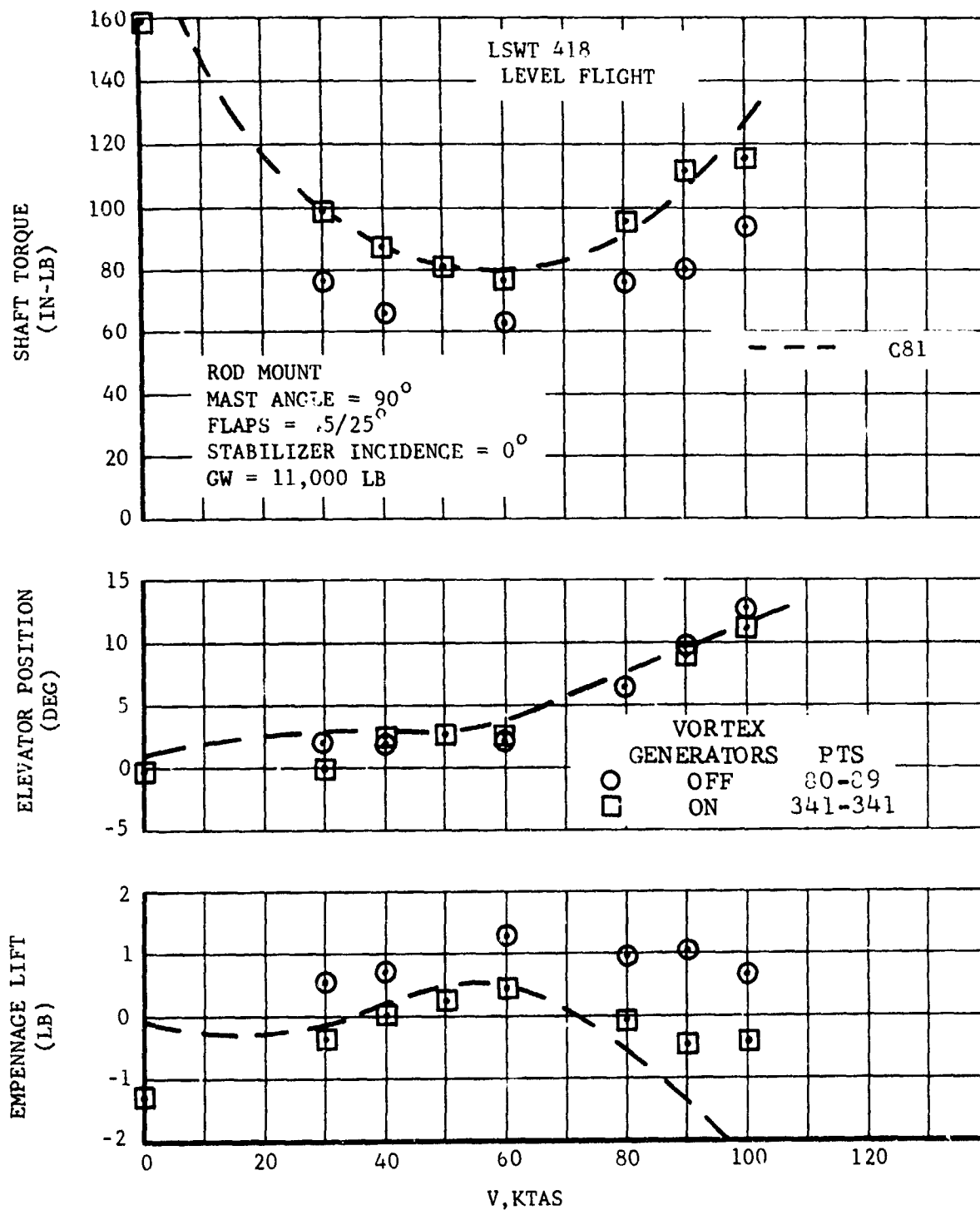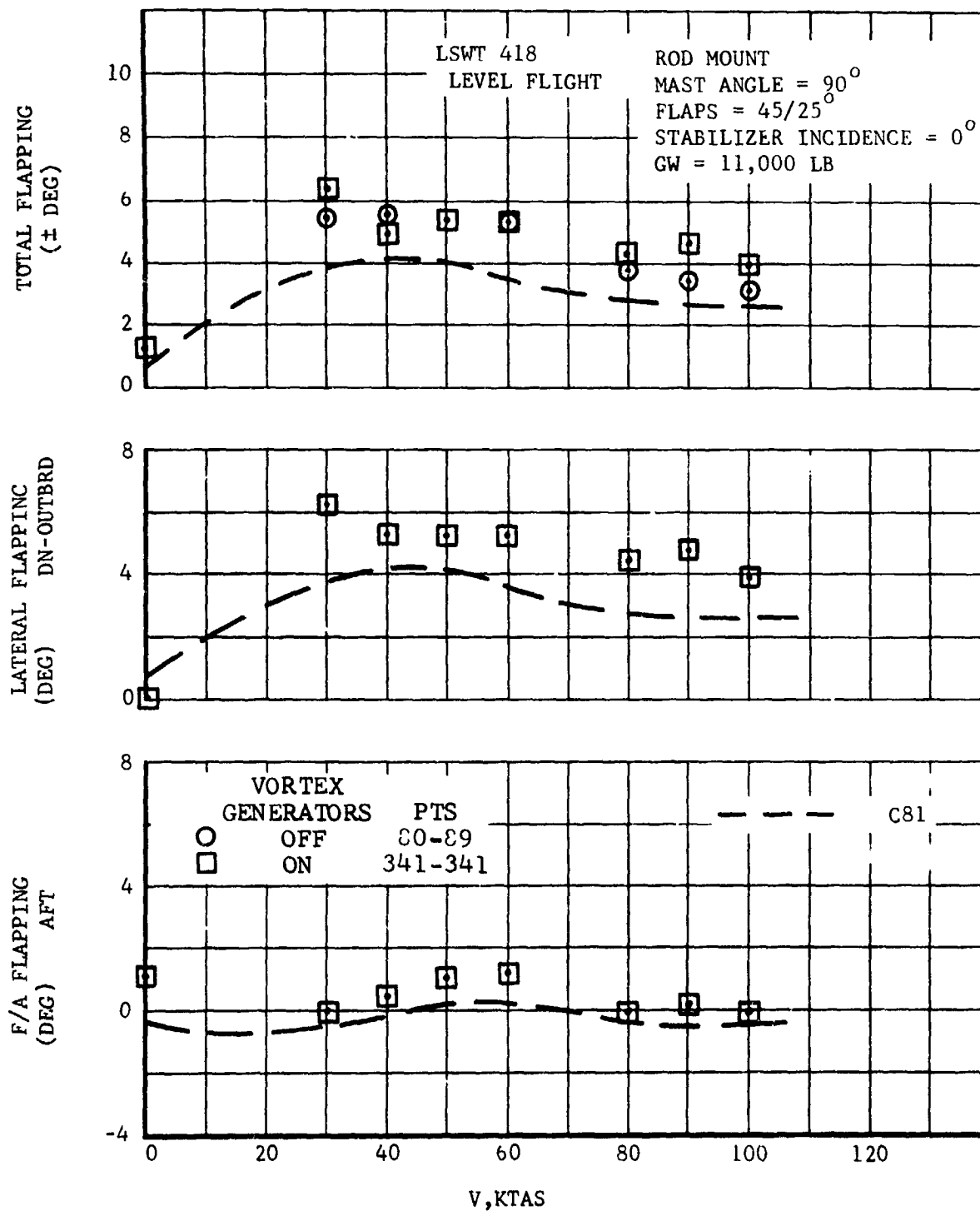
Figure V-11. Concluded
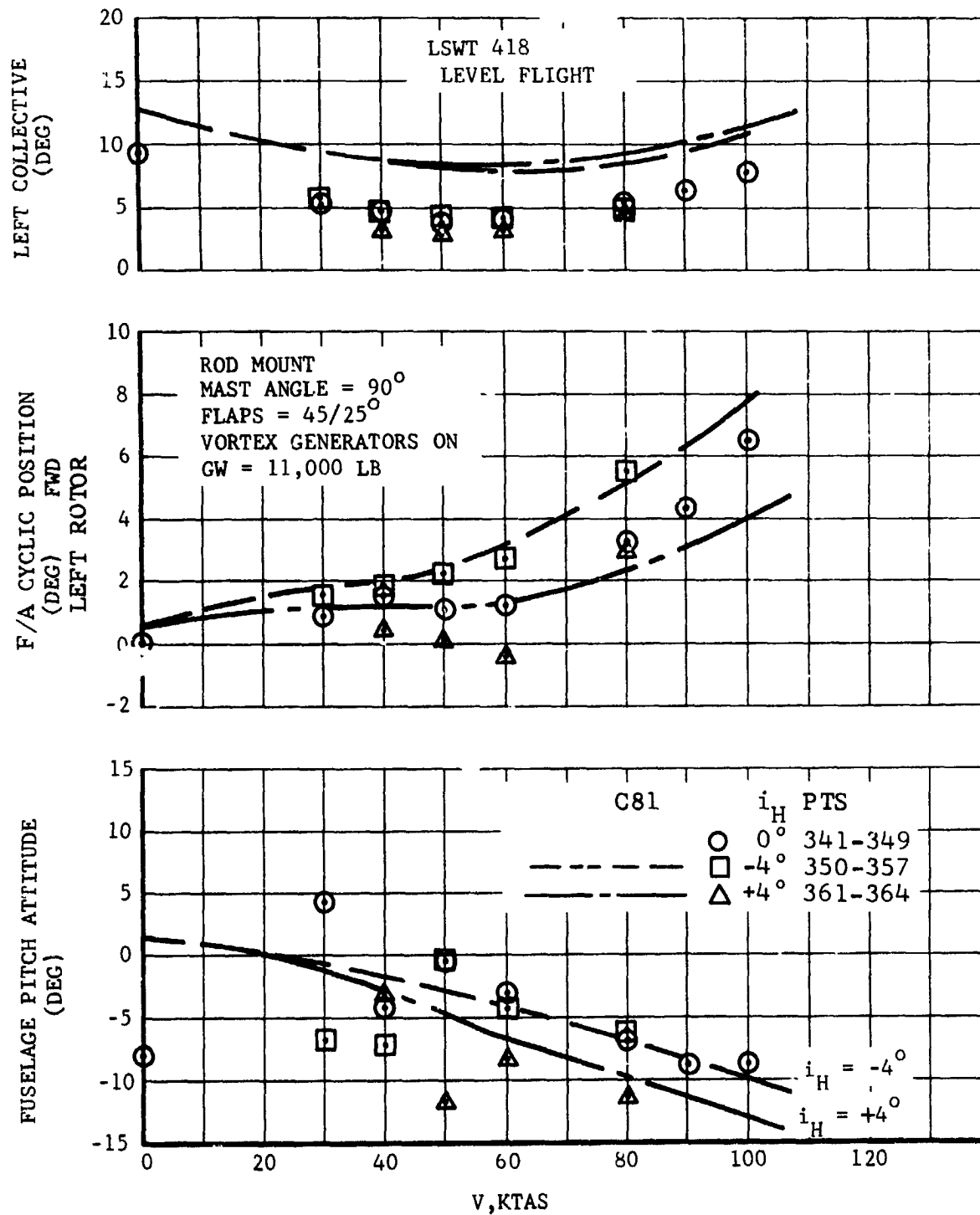
Figure V-12. Effect of Horizontal Stabilizer Incidence
on Level Flight Trim Parameters,
Mast Angle 90°.

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Figure V-12.   Continued

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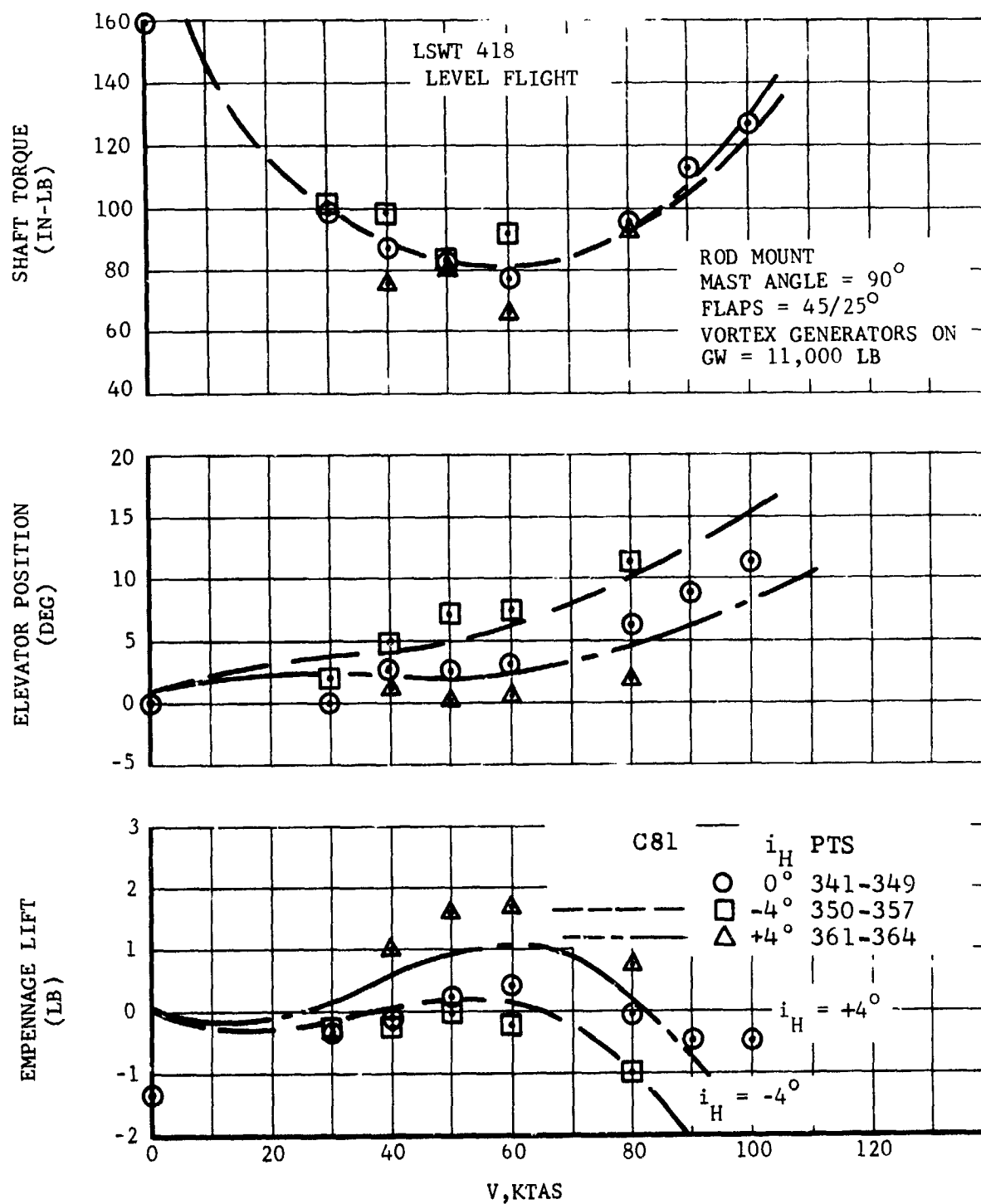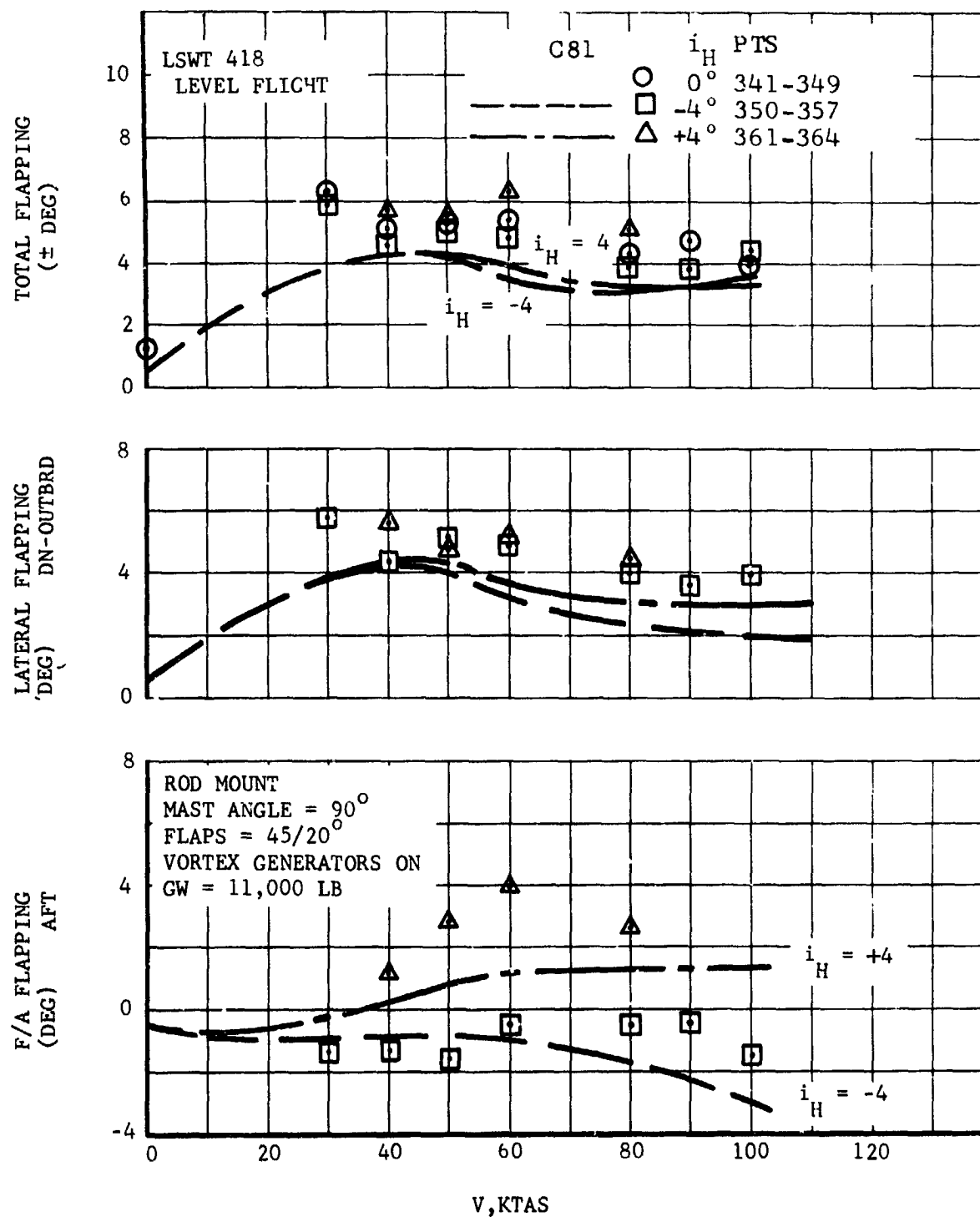
Figure V-12. Concluded
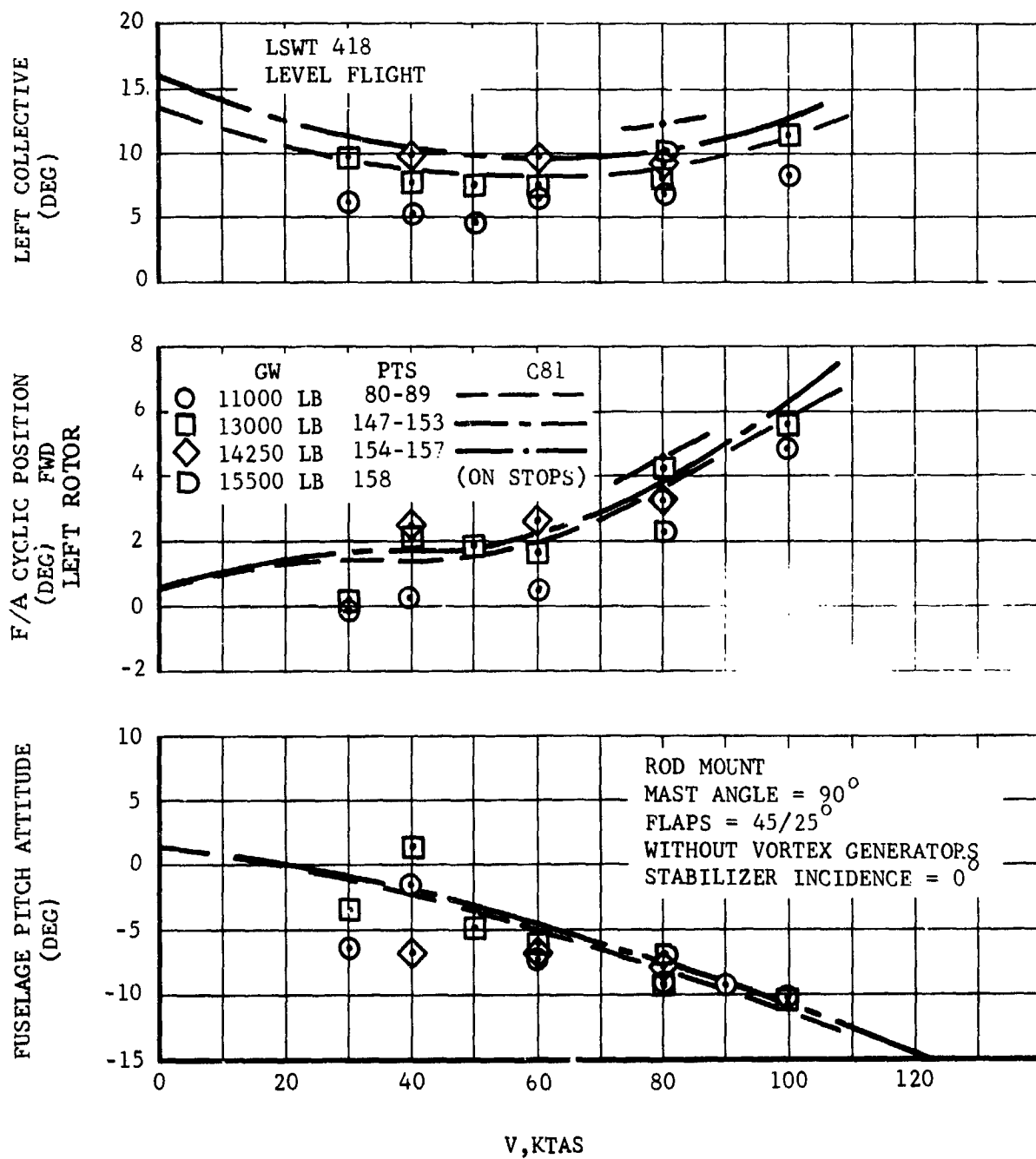
Figure V-13. Effect of Gross Weight on Level Flight
Trim Parameters, Mast Angle 90°.

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LSWT 418
LEVEL FLIGHT

C81		PTS	GW
– – –	◯	80-89	11000 LB
——	☐	147-153	13000 LB
—·—	◇	154-157	14250 LB
(ON STOPS)	⬠	158	15500 LB

ROD MOUNT
MAST ANGLE = $90°$
FLAPS = $45/25°$
WITHOUT VORTEX GENERATORS
STABILIZER INCIDENCE = $0°$

Figure V-13. Continued

BELL HELICOPTER COMPANY
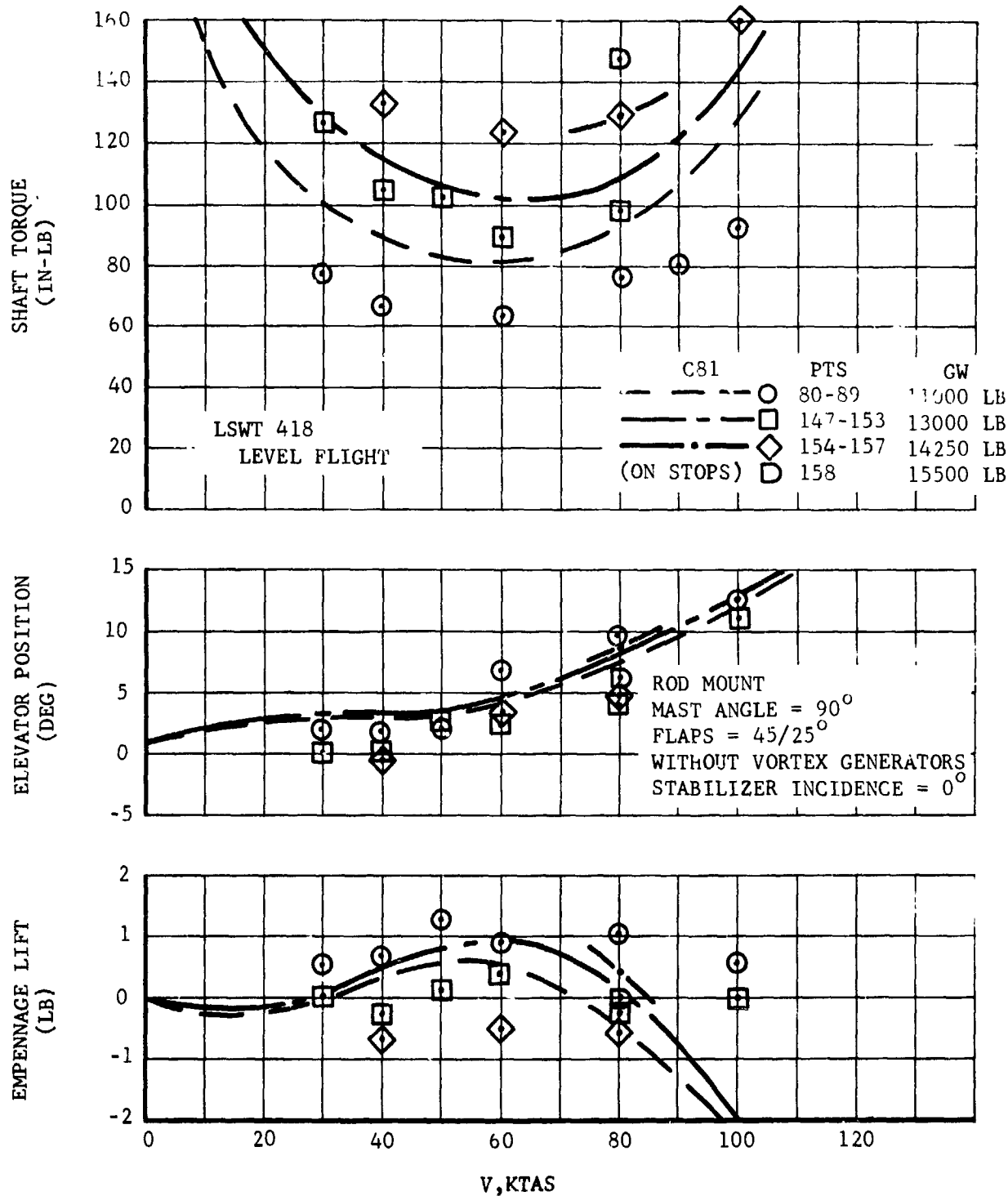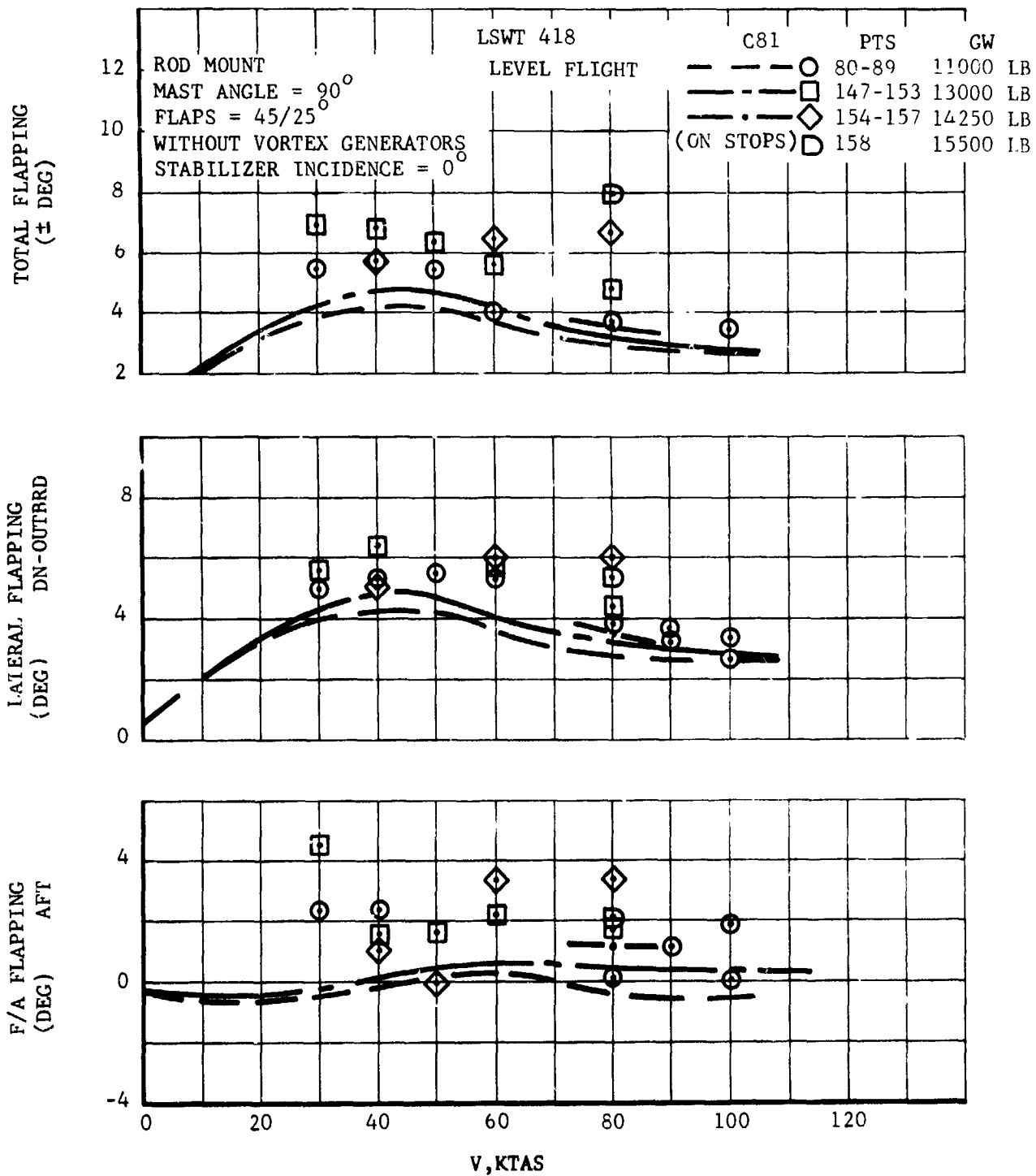
Figure V-13. Concluded

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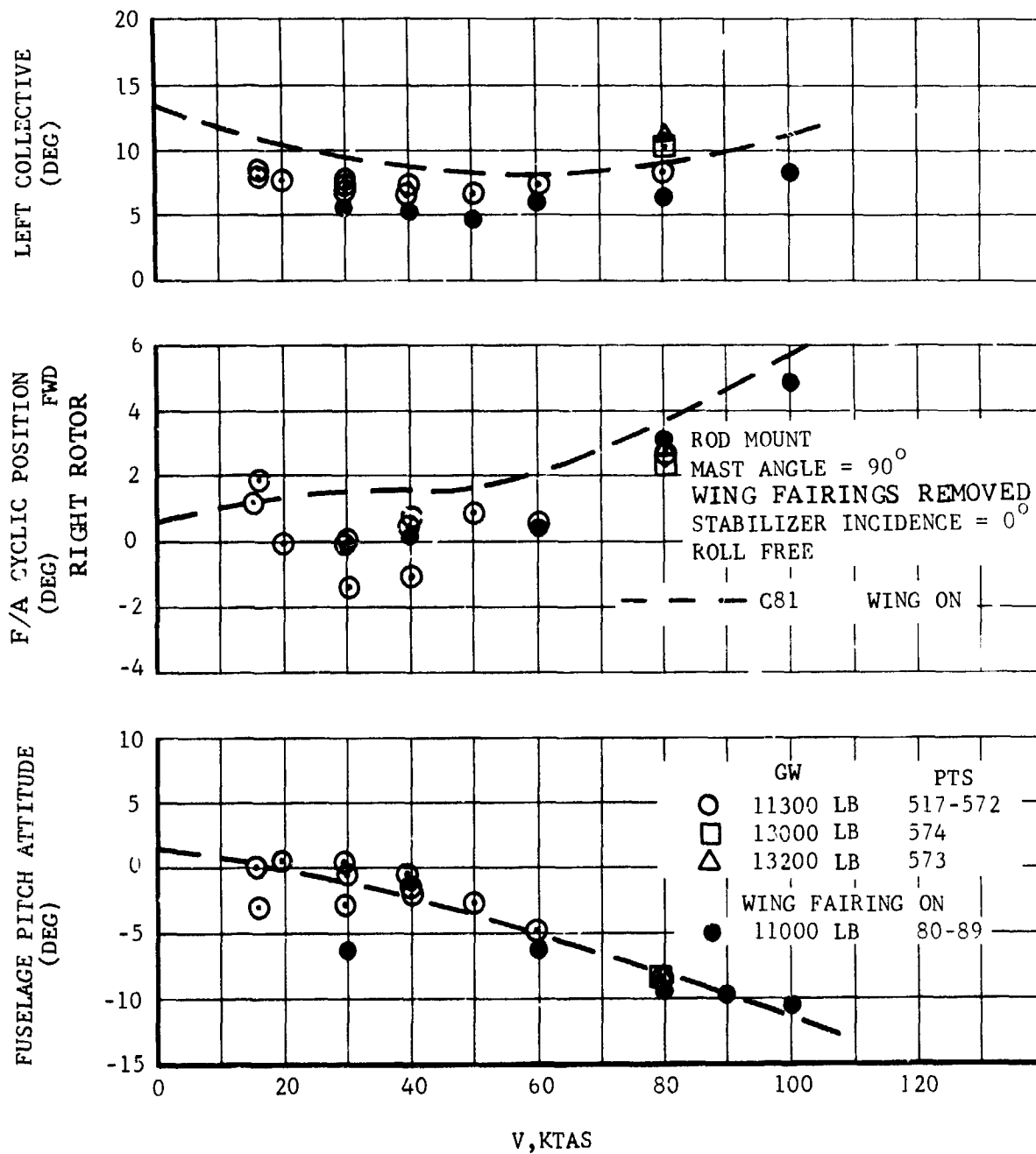
Figure V-14. Effect of Wing Lift on Level Flight Trim
Parameters, Mast Angle 90°.

Figure V-14 Continued

ROD MOUNT
MAST ANGLE = 90°
WING FAIRINGS OFF
STABILIZER INCIDENCE = 0°
ROLL FREE
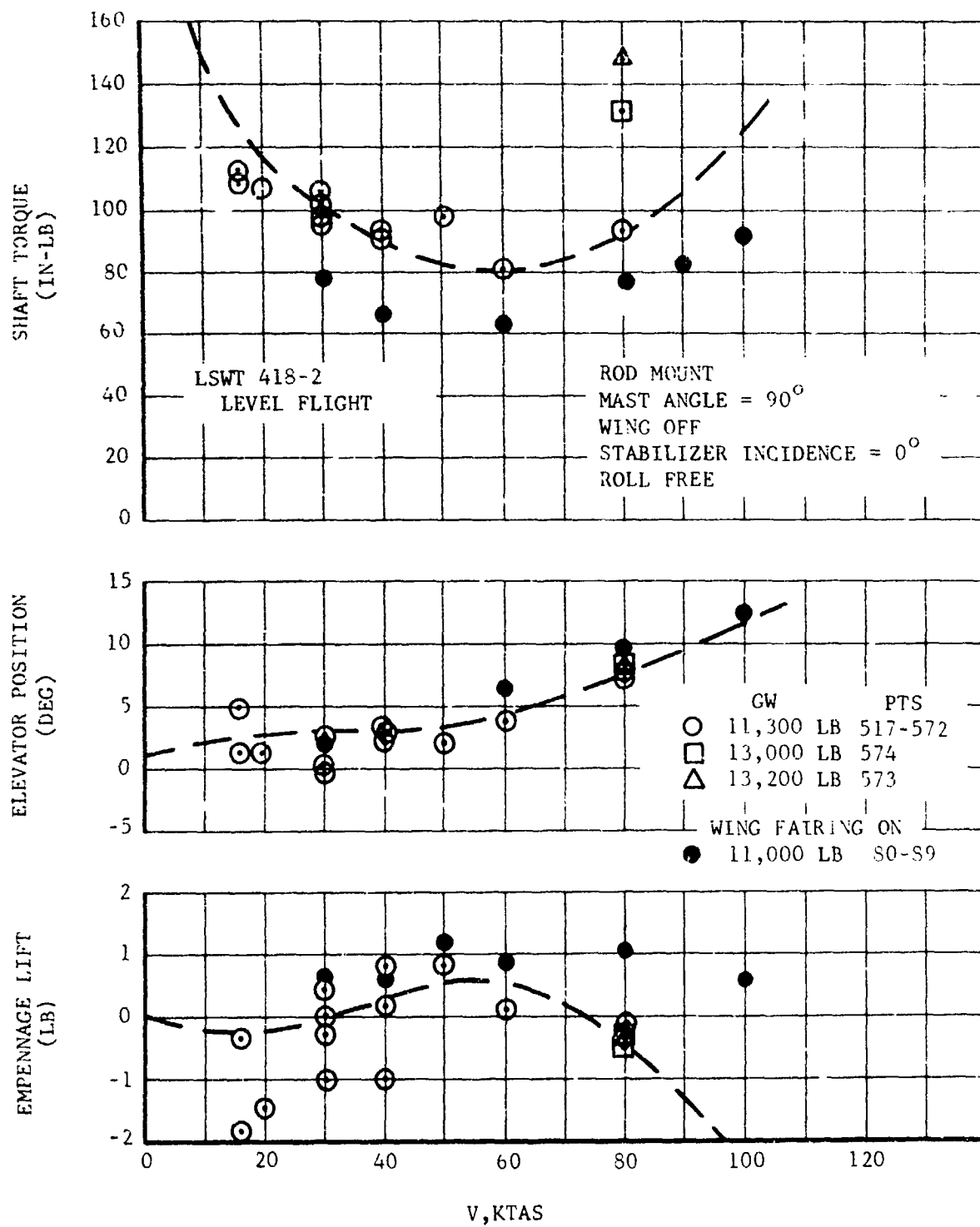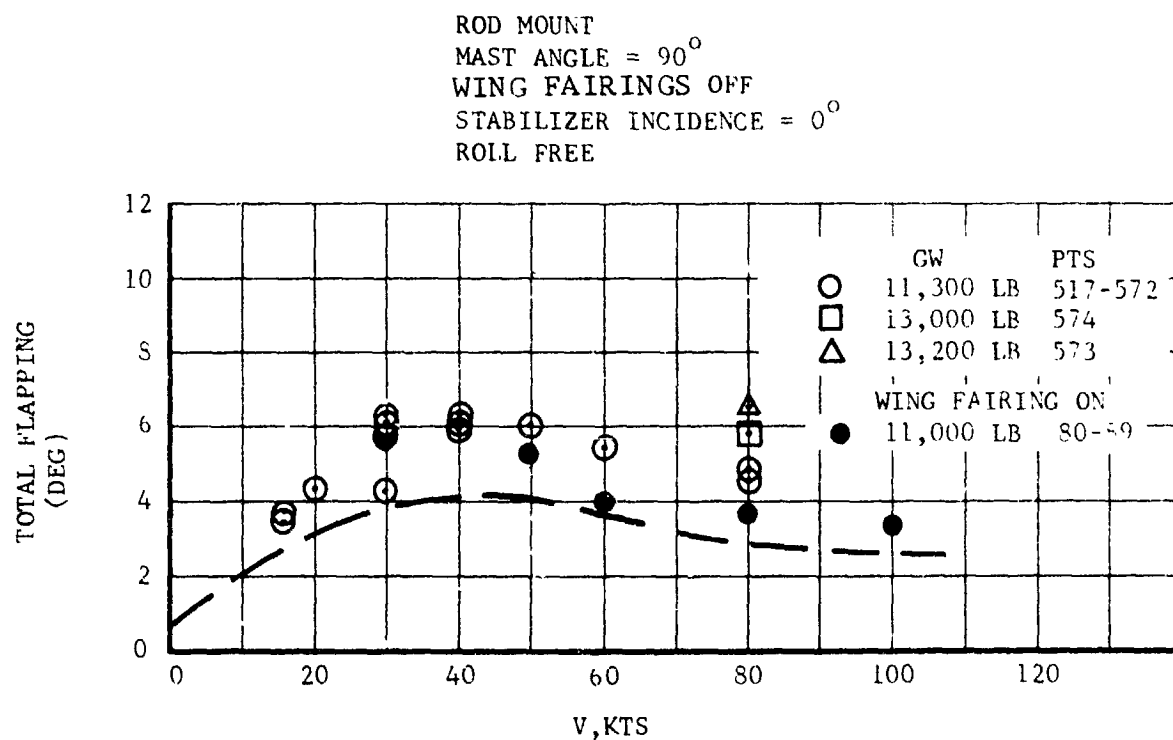
Figure V-14. Concluded

Flight

$R = W$

$R/D = V \sin\gamma$
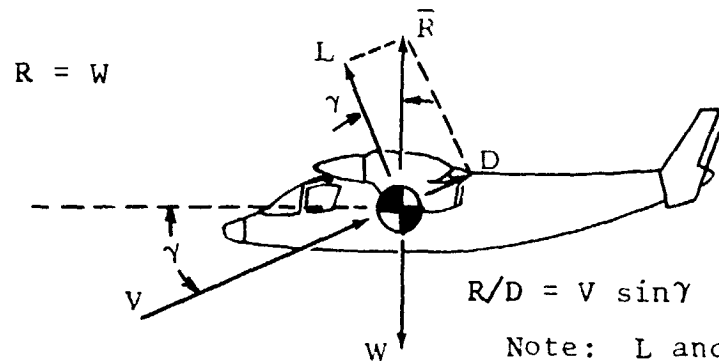
Note: L and D represent Net Lift and Drag With Respect to Relative Wind V.

Wind Tunnel

$\overline{R} = \overline{W} = W/\cos\gamma$

Measured on Rod Balance

Rod
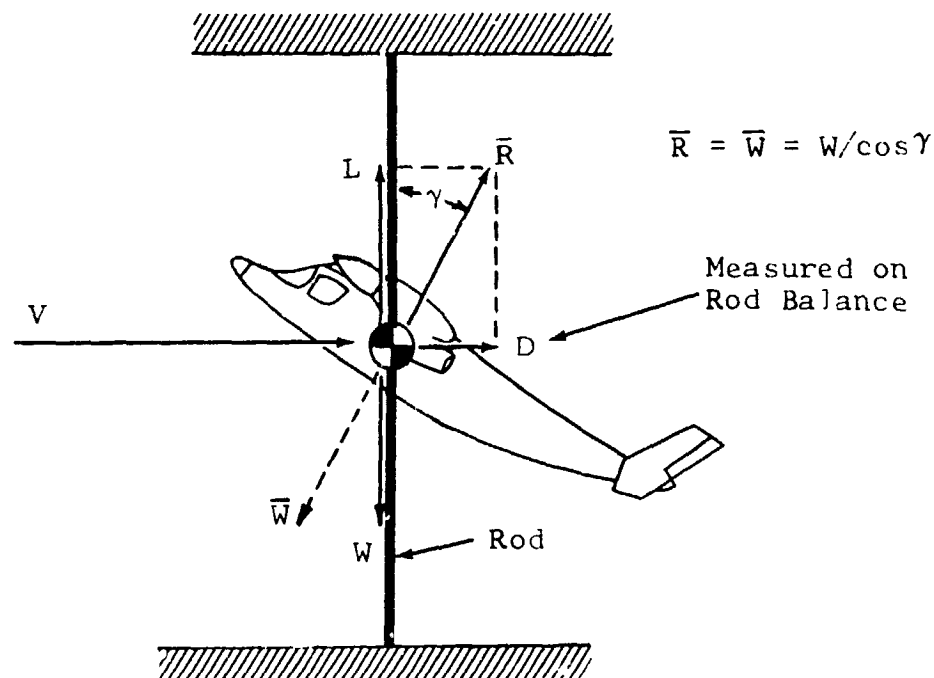
$$R/D = V \sin\gamma = V \tan^{-1}\left(\frac{D}{L}\right) = V \tan^{-1}\left(\frac{D}{W}\right)$$
Simulated Weight $\overline{W} = W/\cos\gamma$

Figure V-15. Method of Determining Simulated Rate of Descent or Climb from Rod Thrust or Drag.

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LEFT COLLECTIVE (DEG)

10
5
0
-5
-10
-20

ROD TEST
GW = 11,000 LB
STABILIZER INCIDENCE = 0°

LSWT 41S
AUTOROTATION

	V,KTS	MAST	FLAPS
	50	95°	45/25°
	60	95°,90°	"
	70	95°	"
	80	95°,90°	"
	"	90°	0/0
	90	95°	45/25°

F/A CYCLIC POSITION LEFT ROTOR (DEG)

FWD

10
5
0
-5
-20

— — — C81
V = 80 KNOTS
MAST = 95°

FUSELAGE PITCH ATTITUDE (DEG)

20
10
0
-10
-20

PTS
○ 82,163,164
□ 85,168
◇ 173,174
△ 322,324,325
◁ 228
◳ 330
⬓ 327
⬟ 332,334

*VORTEX GENERATORS ON

0   1000   2000   3000   4000   5000   6000

R/D, FT/MIN

Figure V-16. Trim Parameters versus Rate of Descent in
Descents and Autorotation, Mast Angle 90°.

Figure V-16. Continued

PTS

○ 82,163,164
□ 85,168
◇ 173,174
△ 322,324,325
◁ 328
◁ 330
□ 327
◇ 332,334

LSWT 418
AUTOROTATION

ROD TEST
GW = 11,000 LB
STABILIZER INCIDENCE = $0°$
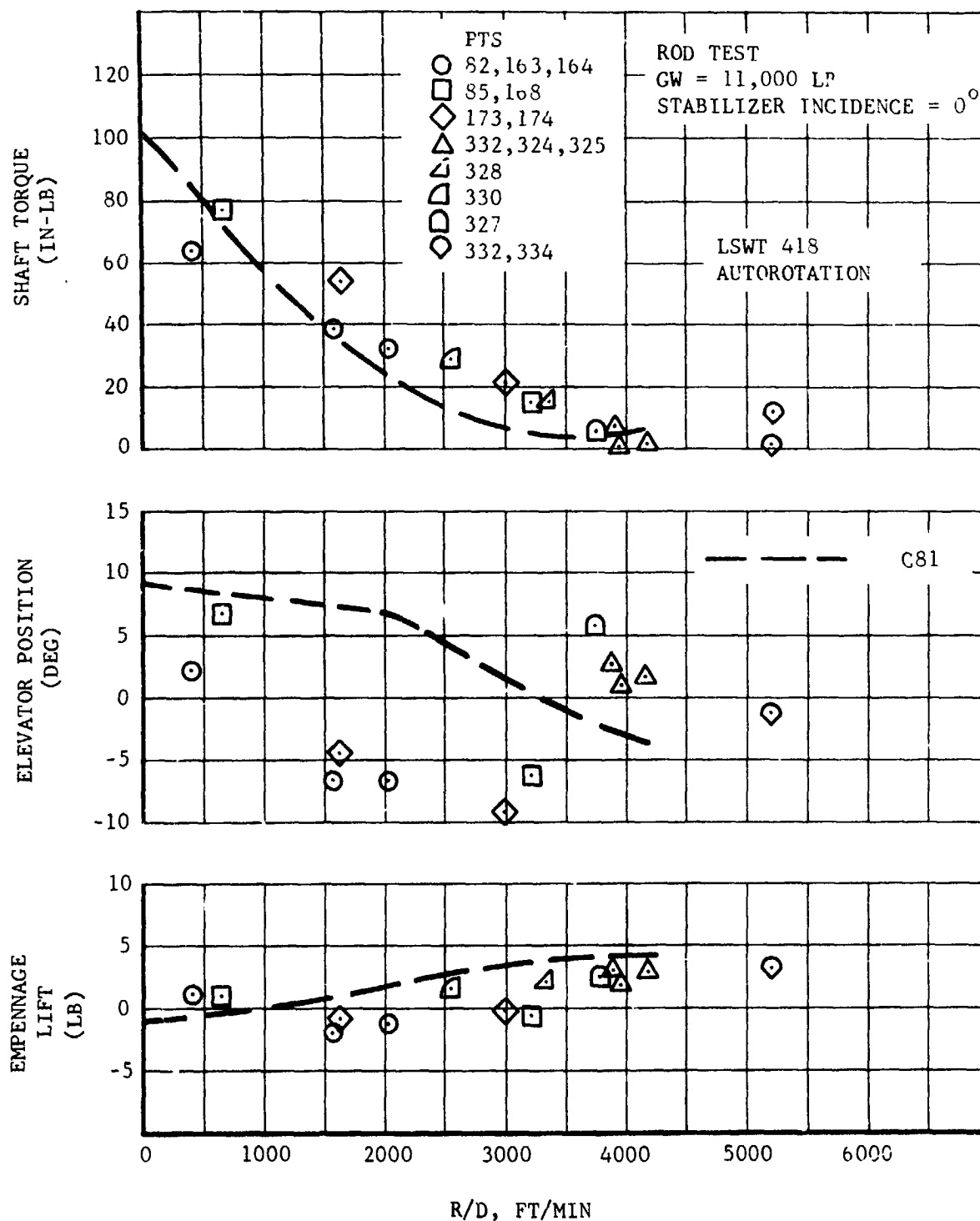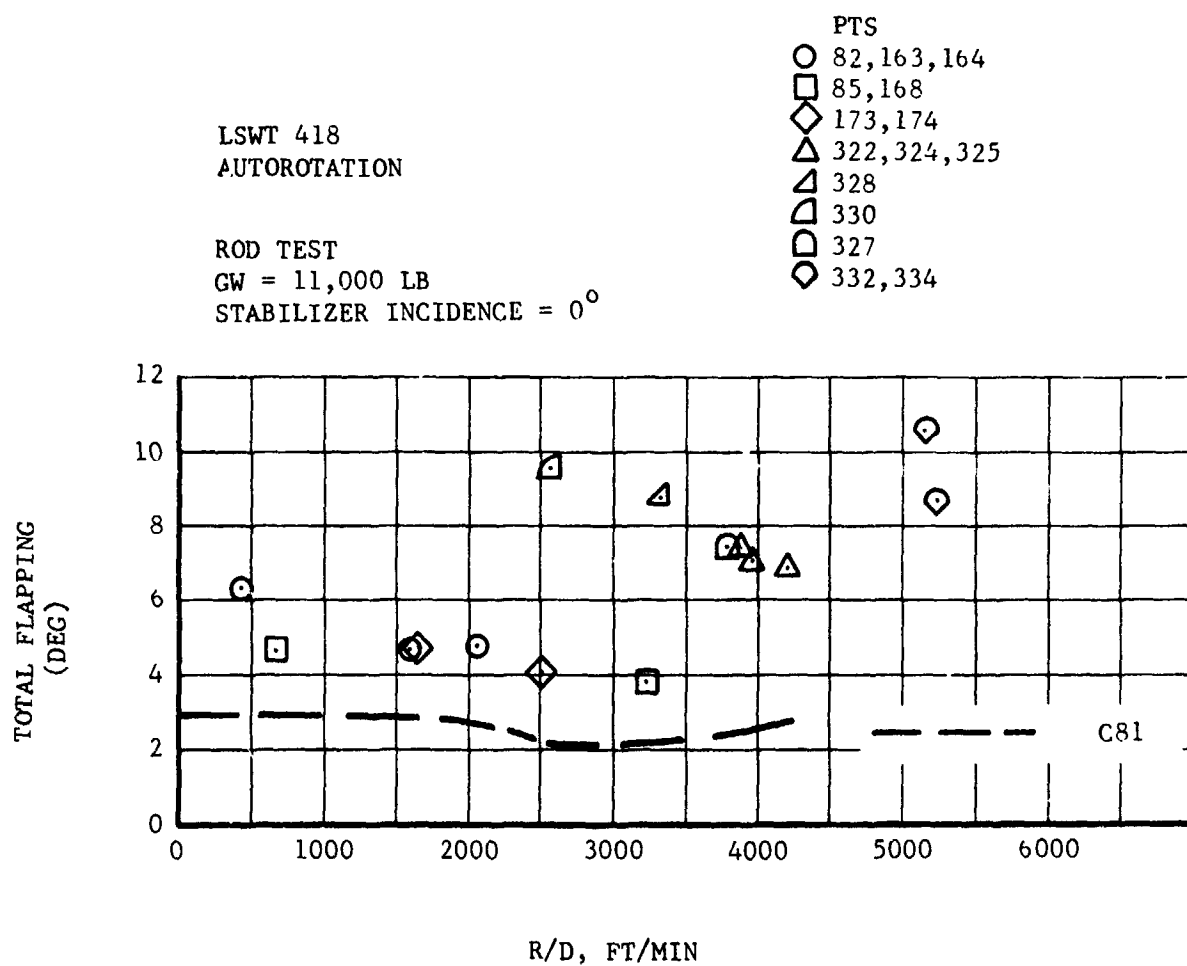
Figure V-16. Concluded
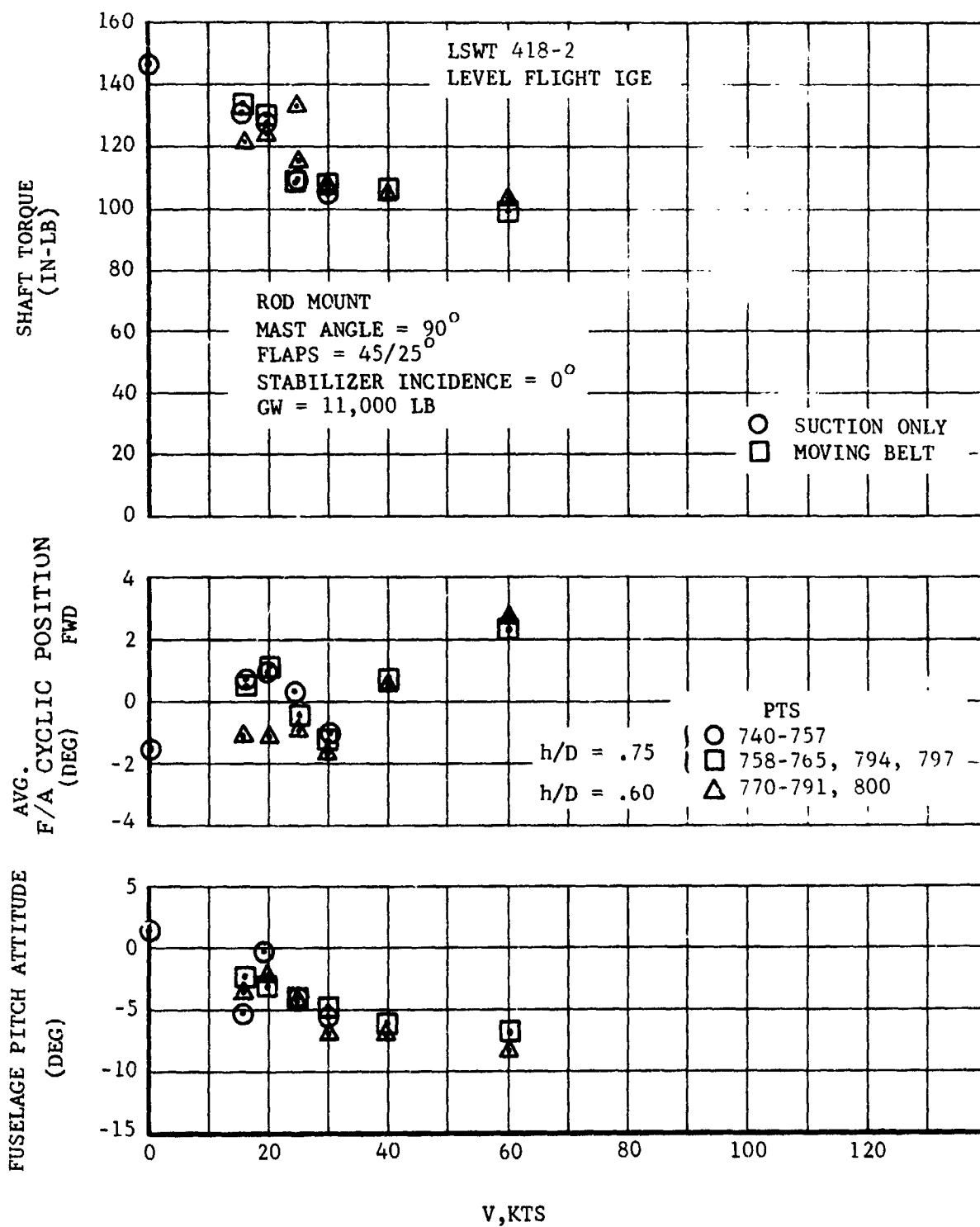
Figure V-17.   Trim Parameters for Level Flight IGE,
Mast Angle 90°.

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LSWT 418-2
LEVEL FLIGHT-IGE

ROD MOUNT
MAST ANGLE = 90°
FLAPS = $45/25^{\circ}$
STABILIZER INCIDENCE = 0°
GW = 11,000 LB

PTS
$h/D = .75 \begin{cases} \bigcirc \; 740\text{-}757 \\ \square \; 758\text{-}765, 794, 797 \end{cases}$
$h/D = .60 \quad \triangle \; 770\text{-}71, 800$

V, KTS
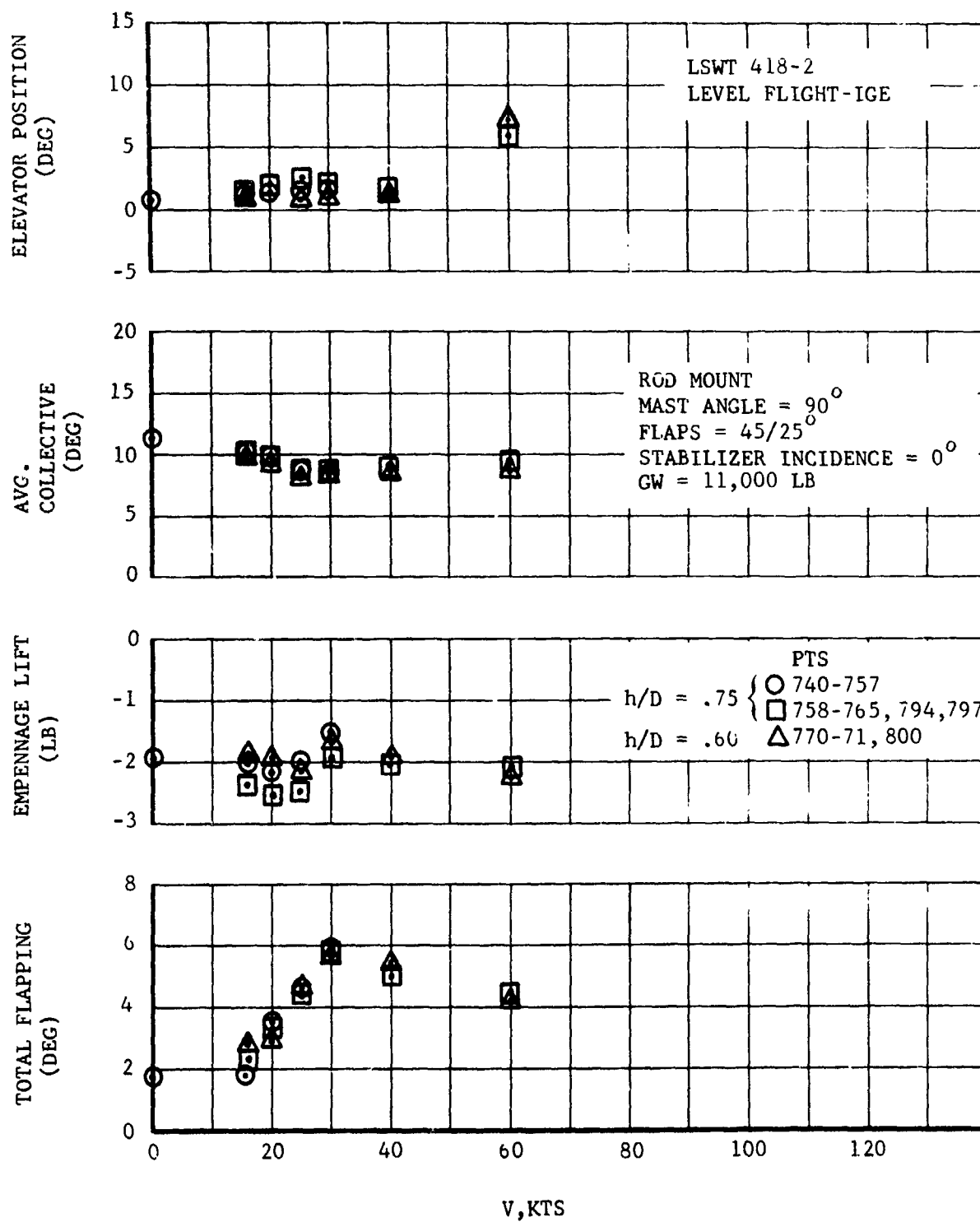
Figure V-17. Concluded
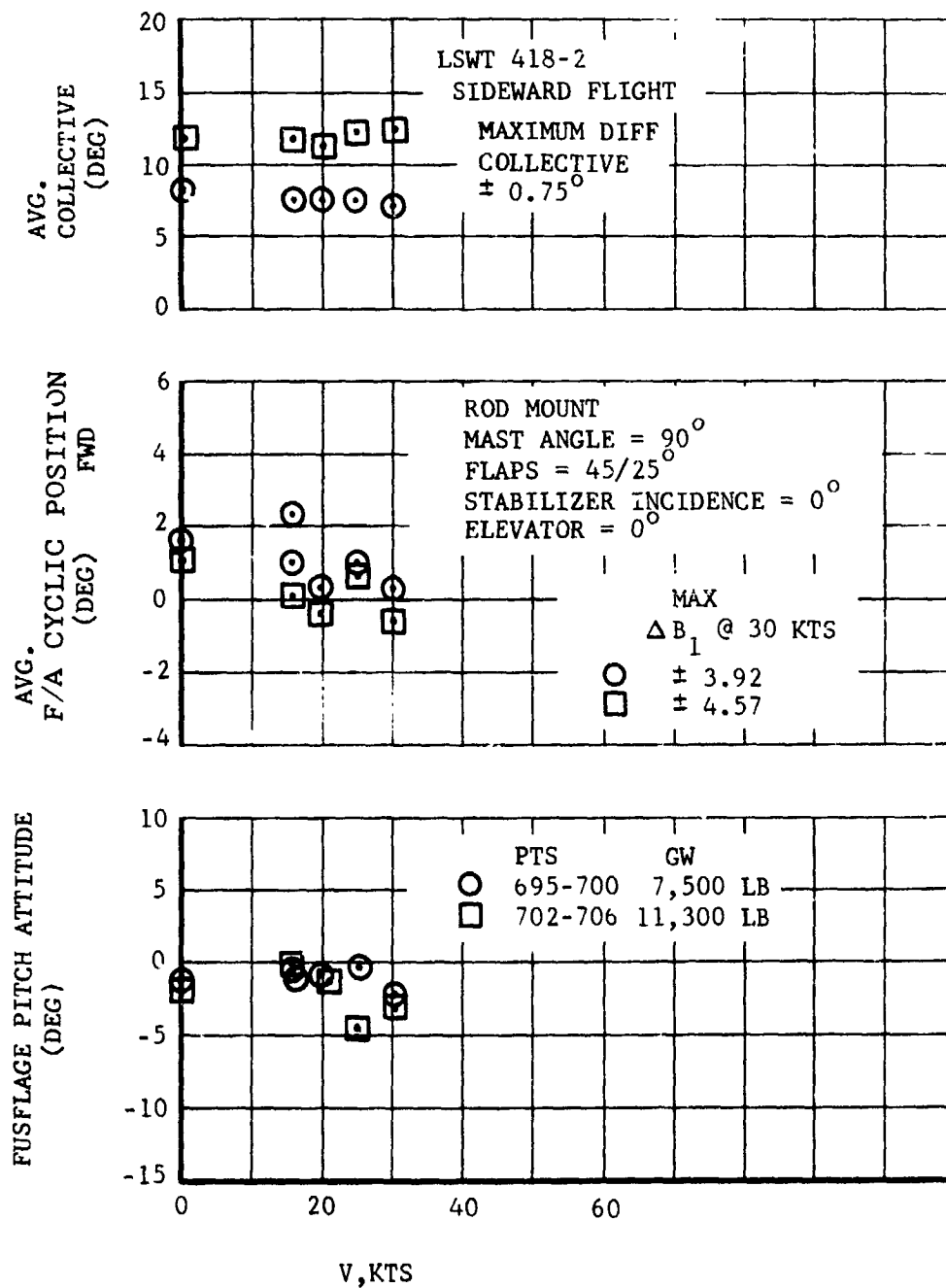
Figure V-18   Sideward Flight Trim Parameters

LSWT 418-2
SIDEWARD FLIGHT

ROD MOUNT
MAST ANGLE = 90°
FLAPS = $45/25^{\circ}$
STABILIZER INCIDENCE = 0°
ELEVATOR = 0°

	PTS	GW
○	695-700	7,500 LB
□	702-706	11,300 LB

V,KTS
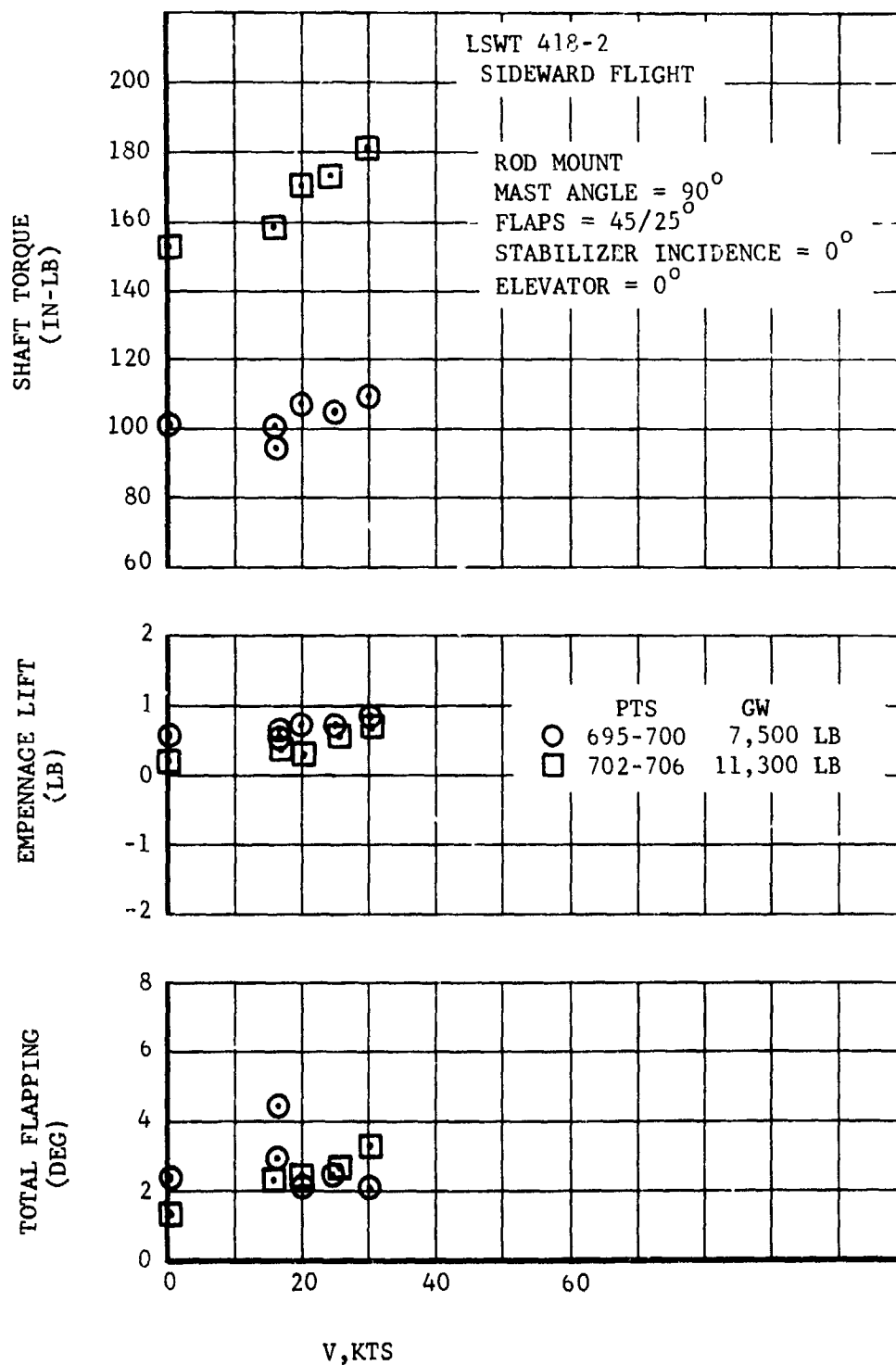
Figure V-18. Concluded
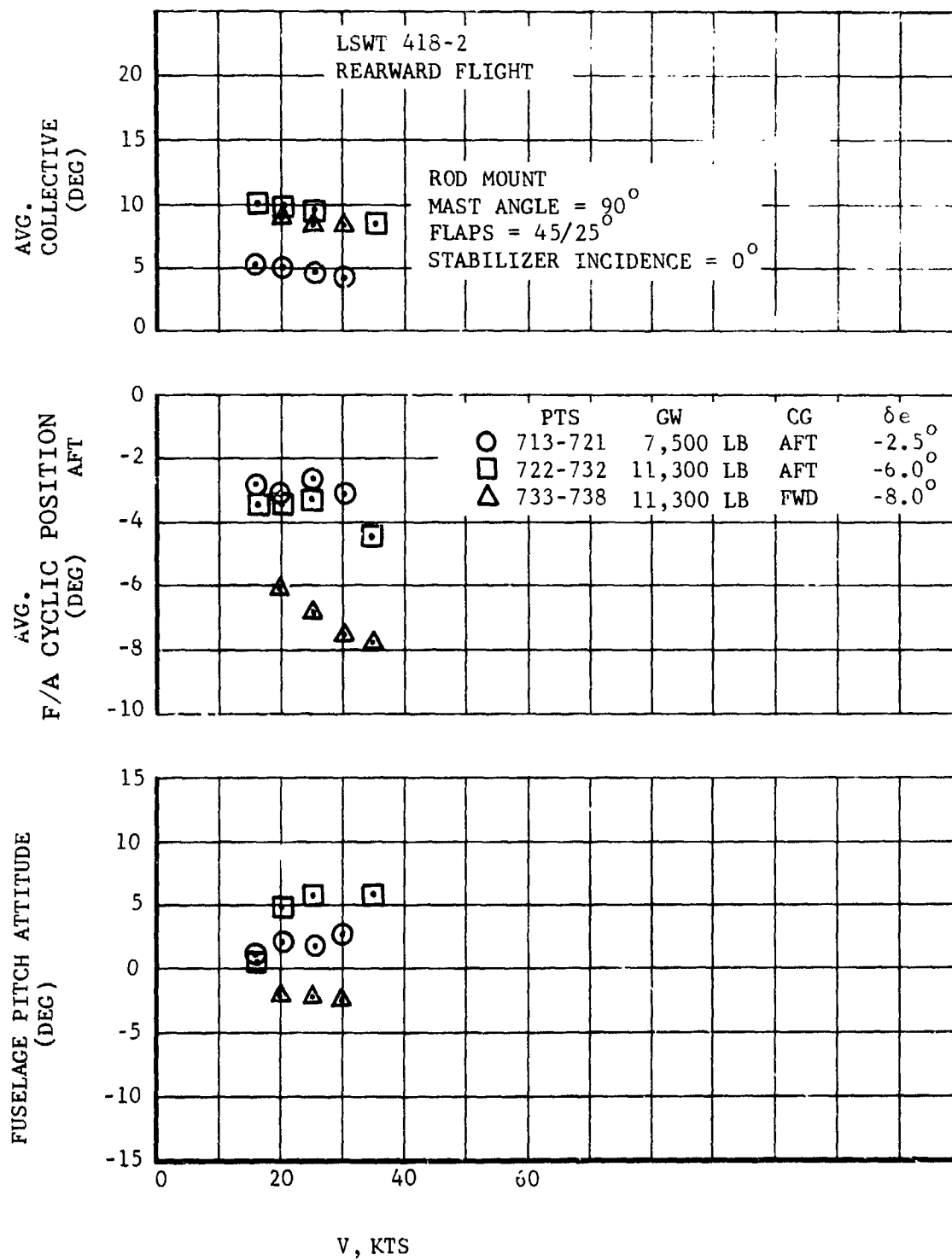
Figure V-19 Rearward Flight Trim Parameters

LSWT 418-2
REARWARD FLIGHT

ROD MOUNT
MAST ANGLE = 90°
FLAPS = $45/25^\circ$
STABILIZER INCIDENCE = 0°

PTS	GW	CG	δ_e
○ 713-721	7,500 LB	AFT	-2.5°
□ 722-732	11,300 LB	AFT	-6.0°
△ 733-738	11,300 LB	FWD	-8.0°
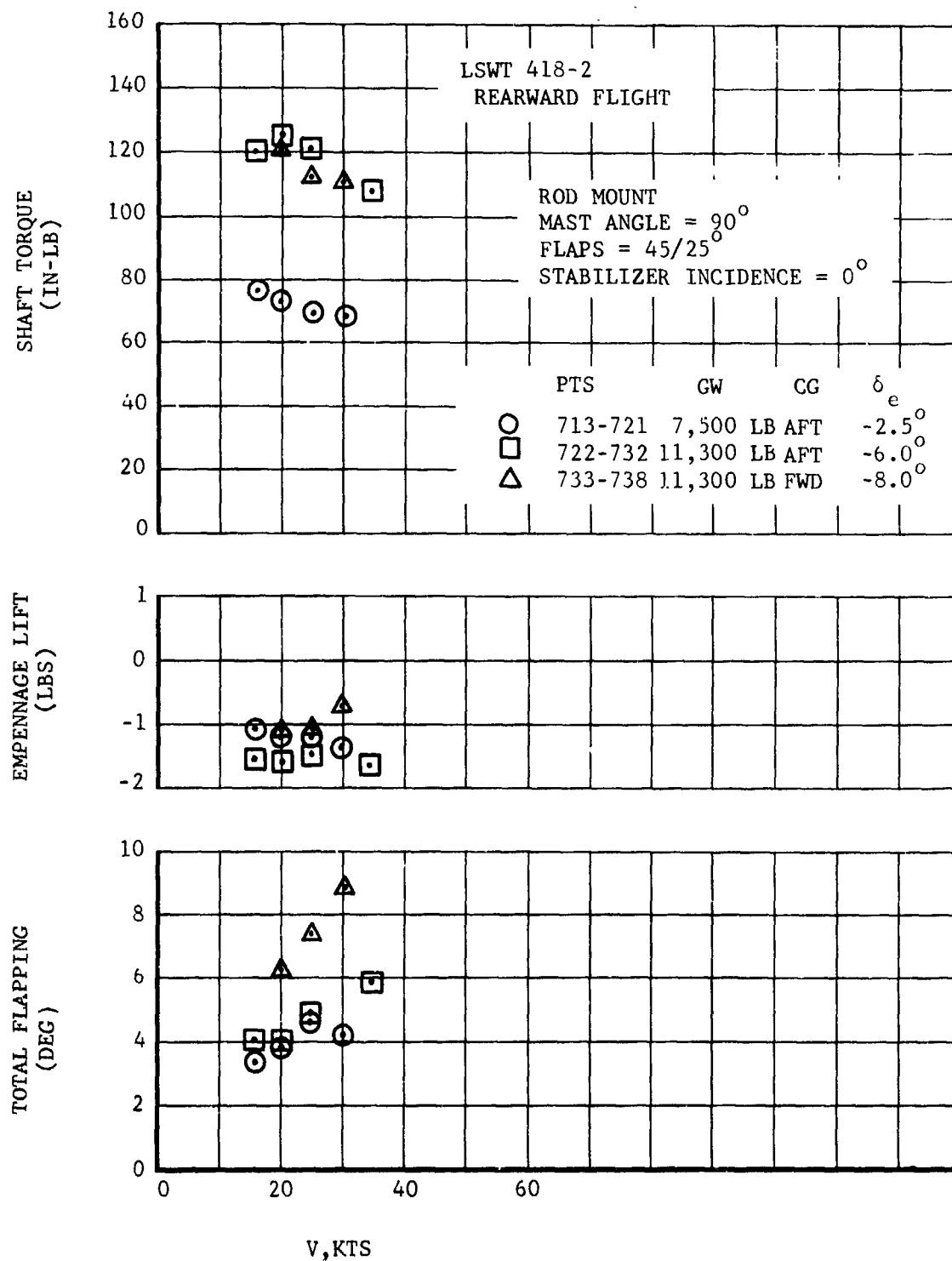
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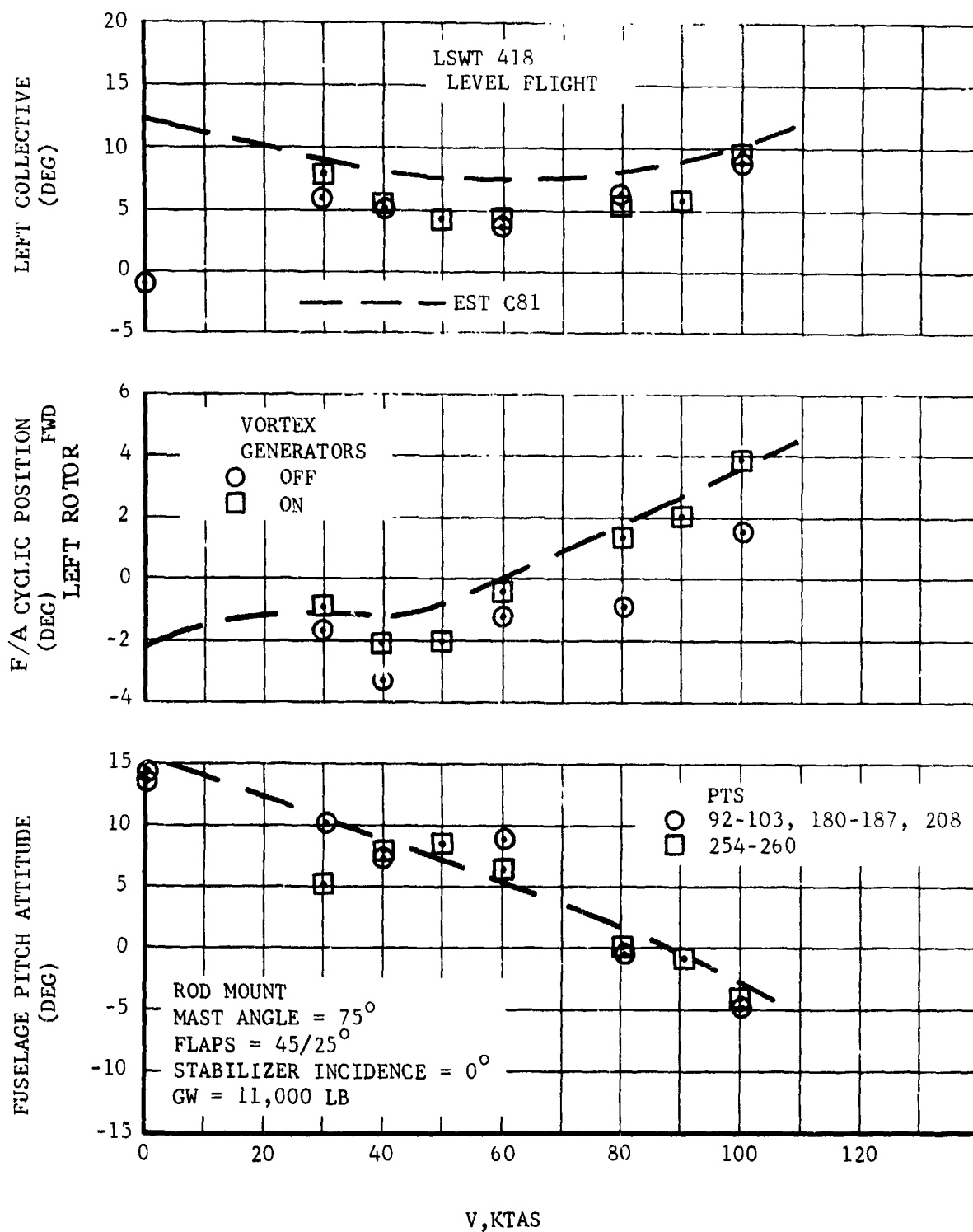
Figure V-20. Level Flight Trim Parameters, Mast Angle 75°.

LSWT 418
LEVEL FLIGHT

--- --- C81

ROD MOUNT
MAST ANGLE = 75°
FLAPS = 45/25°
STABILIZER INCIDENCE = 0°
GW = 11,000 LB

PTS
○ 92-103, 180-187, 208
□ 254-260

SHAFT TORQUE (IN-LB) — vertical axis, range 20 to 160

ELEVATOR POSITION (DEG) — vertical axis, range -10 to 10

EMPENNAGE LIFT (LB) — vertical axis, range -3 to 2

V, KTAS
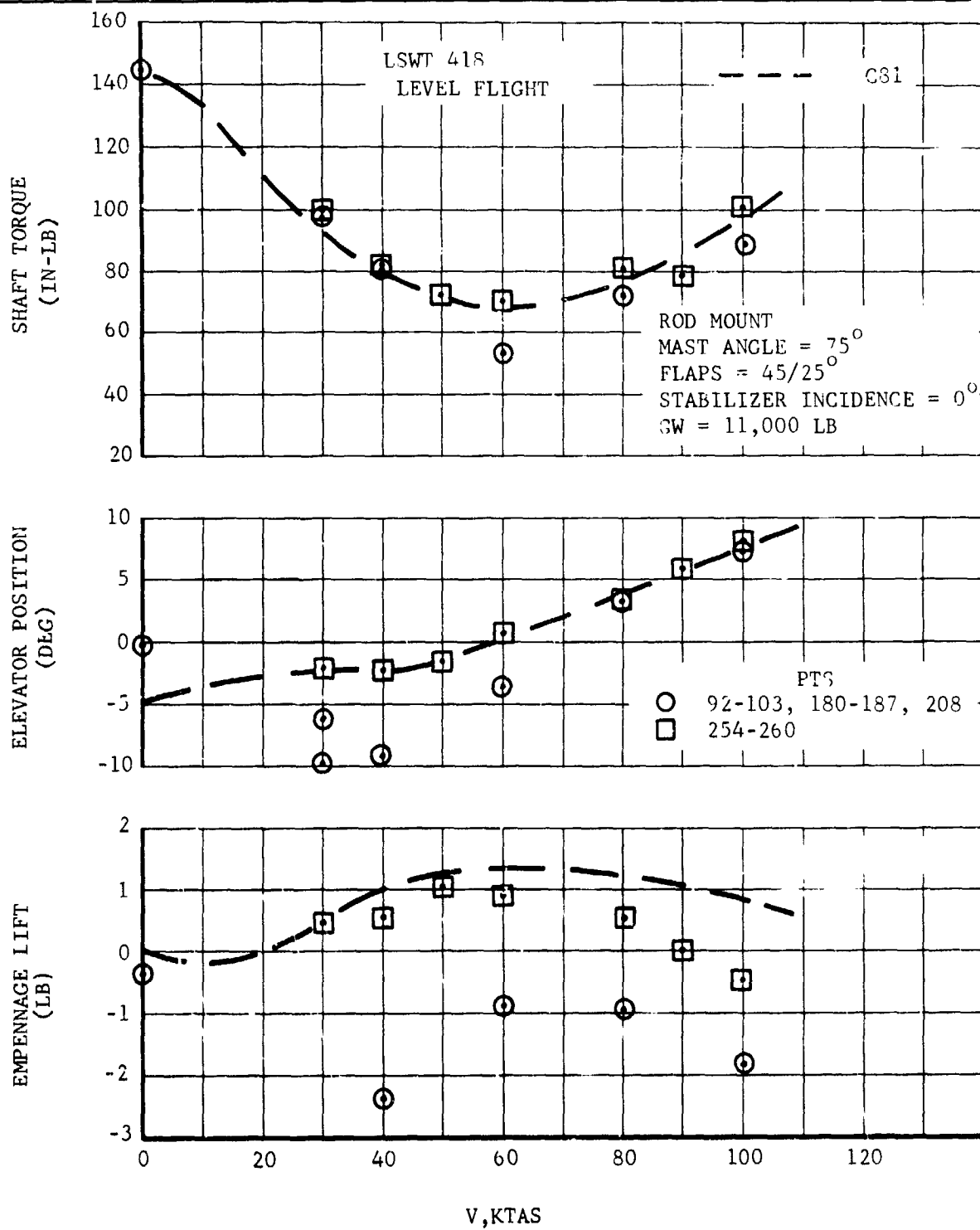
Figure V-20.   Continued

Figure V-20. Concluded

ROD MOUNT
MAST ANGLE = ⁻5°
FLAPS = 45/25°
STABILIZER INCIDENCE = 0°
VORTEX GENERATORS OFF

CR1	GW
——— O	11,000 LB
—·—· □	13,000 LB
—·— ◇	15,500 LB
—x— ⬠	16,100 LB

LSWT 415
LEVEL FLIGHT

LEFT COLLECTIVE (DEG)

F/A CYCLIC POSITION FWD (DEG) LEFT ROTOR

FUSELAGE PITCH ATTITUDE (DEG)

PTS
O 92-103/180-187
   208
□ 209-216
◇ 159-160
⬠ 161-162

V,KTAS
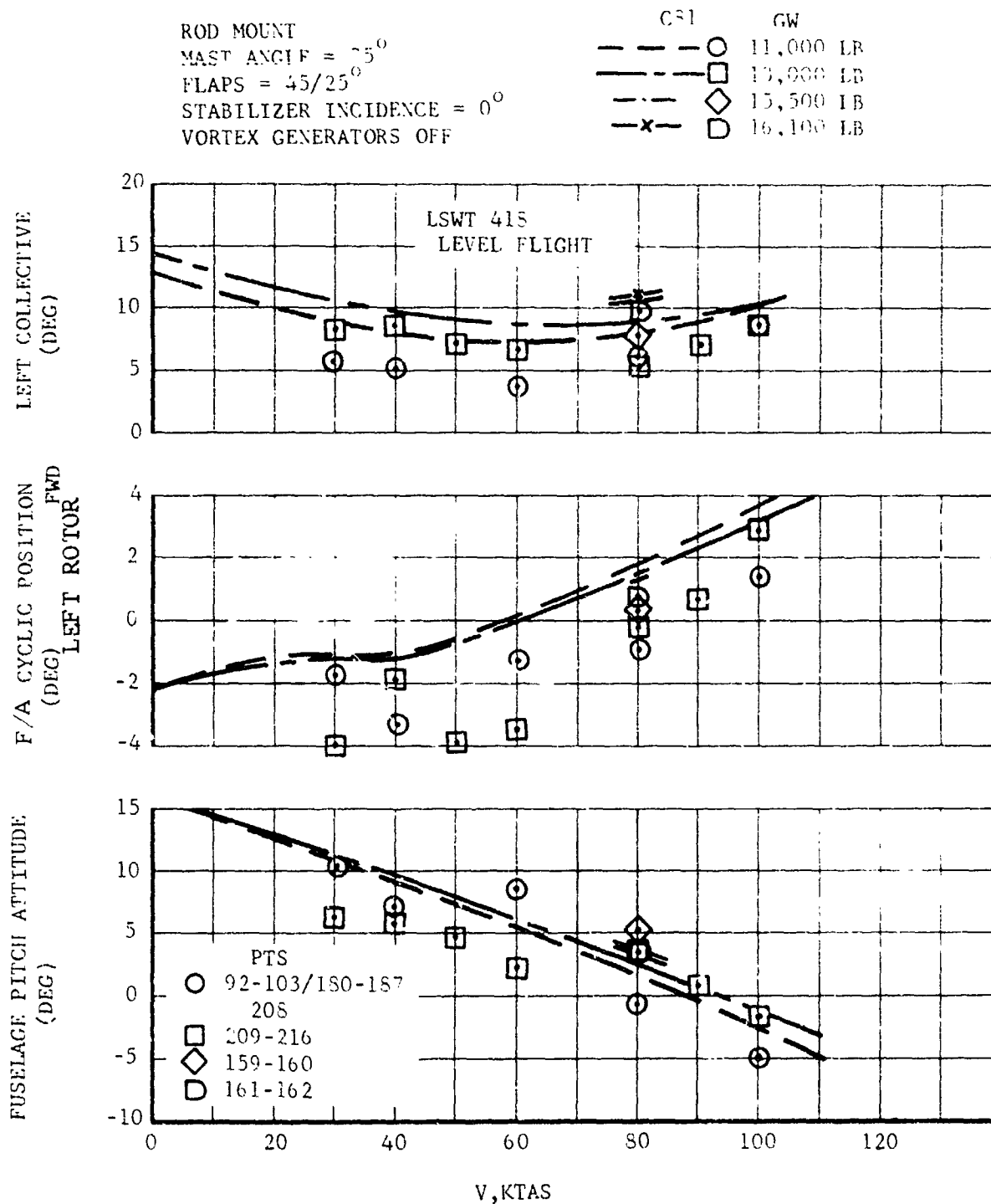
Figure V-21. Effect of Gross Weight on Level Flight
Trim Parameters, Mast Angle ⁻5°.

Figure V-21.   Continued

ROD MOUNT
MAST ANGLE = 75°
FLAPS = 45/25°
STABILIZER INCIDENCE = 0°
VORTEX GENERATORS OFF

C81	GW
— — —O	11,000 LB
— — —□	13,000 LB
— · —◇	15,500 LB
—✕—D	16,100 LB

LSWT 418
LEVEL FLIGHT

PTS
O 92-103/180-187/208
□ 209-216
◇ 159-160
D 161-162

TOTAL FLAPPING (± DEG)

V,KTAS
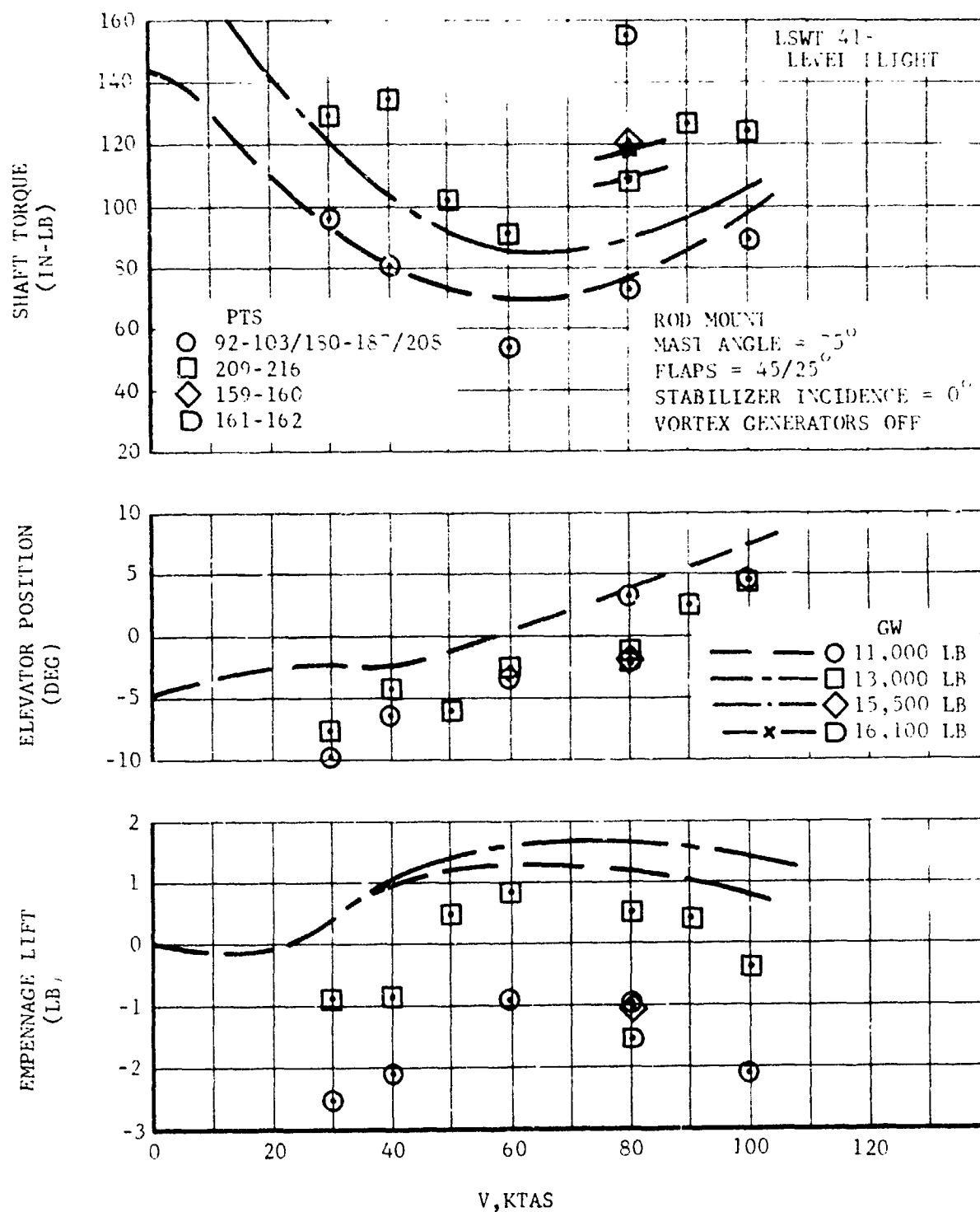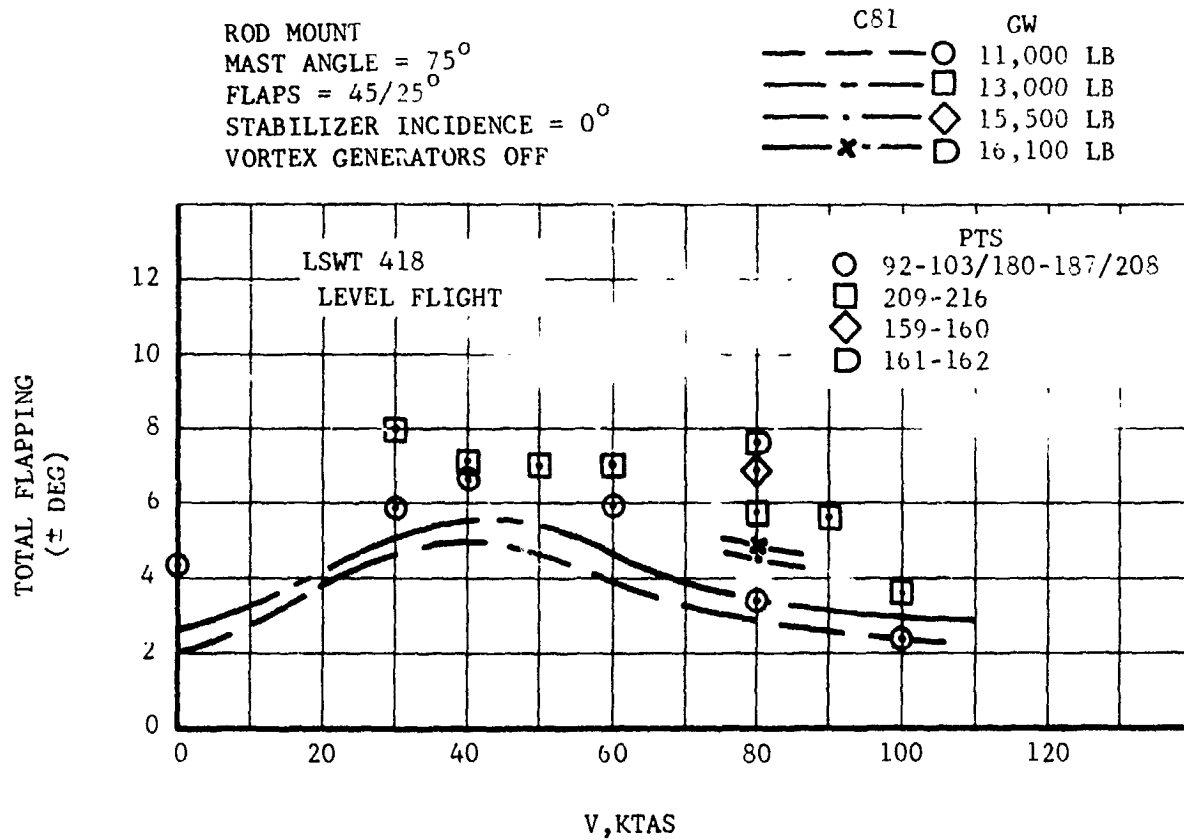
Figure V-21. Concluded
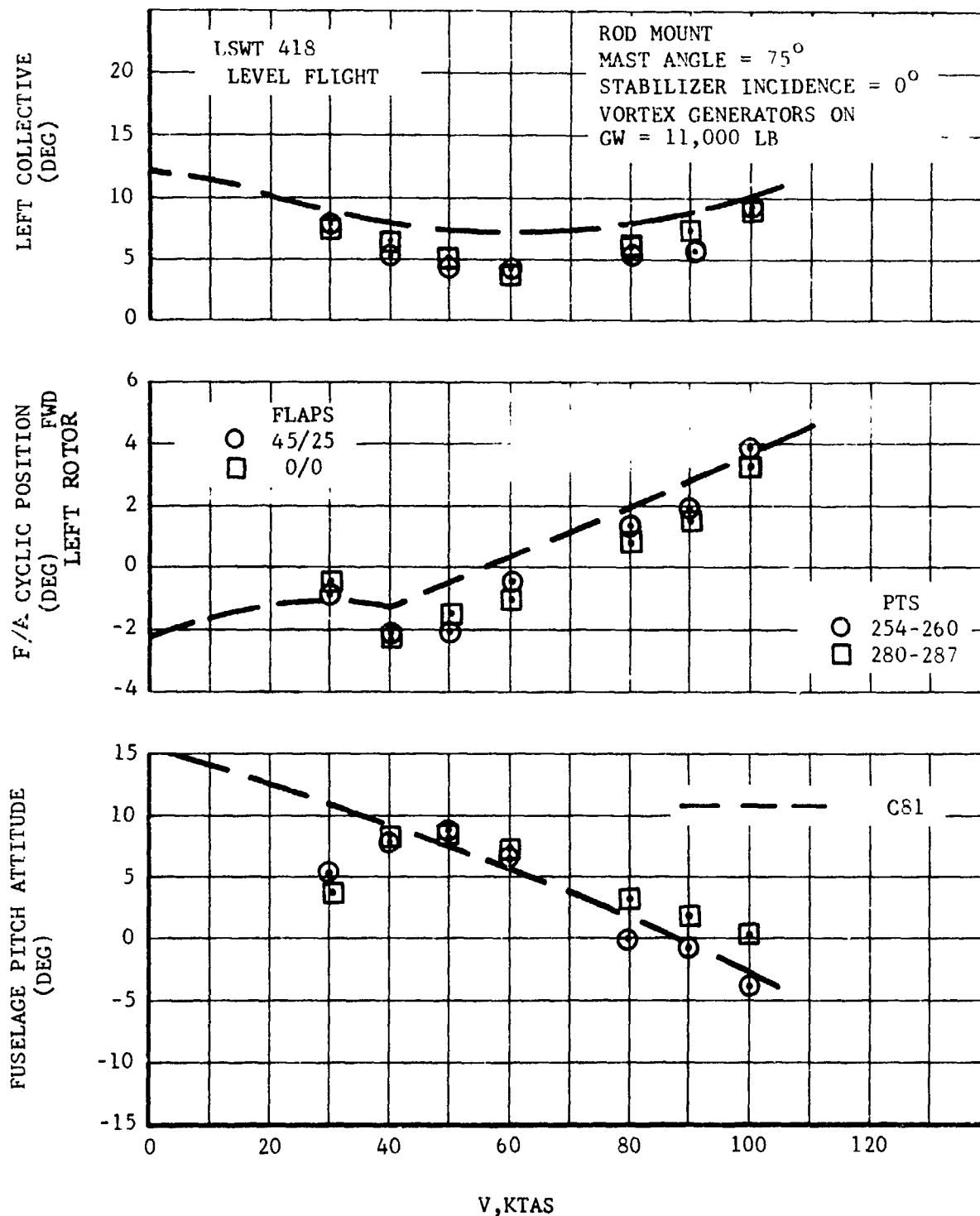
Figure V-22.   Effect of Flaps on Level Flight Trim
Parameters, Mast Angle 75°.

BELL HELICOPTER COMPANY

LSWT 418
LEVEL FLIGHT

ROD MOUNT
MAST ANGLE = 75°
STABILIZER INCIDENCE = 0°
VORTEX GENERATORS ON
GW = 11,000 LB

C81

PTS
○ 254-260
□ 280-287

V, KTAS

Figure V-22.   Continued

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ROD MOUNT
MAST ANGLE = 75°
STABILIZER INCIDENCE = 0°
VORTEX GENERATORS ON
GW = 11,000 LB

FLAPS
○ 45/25
□ 0/0
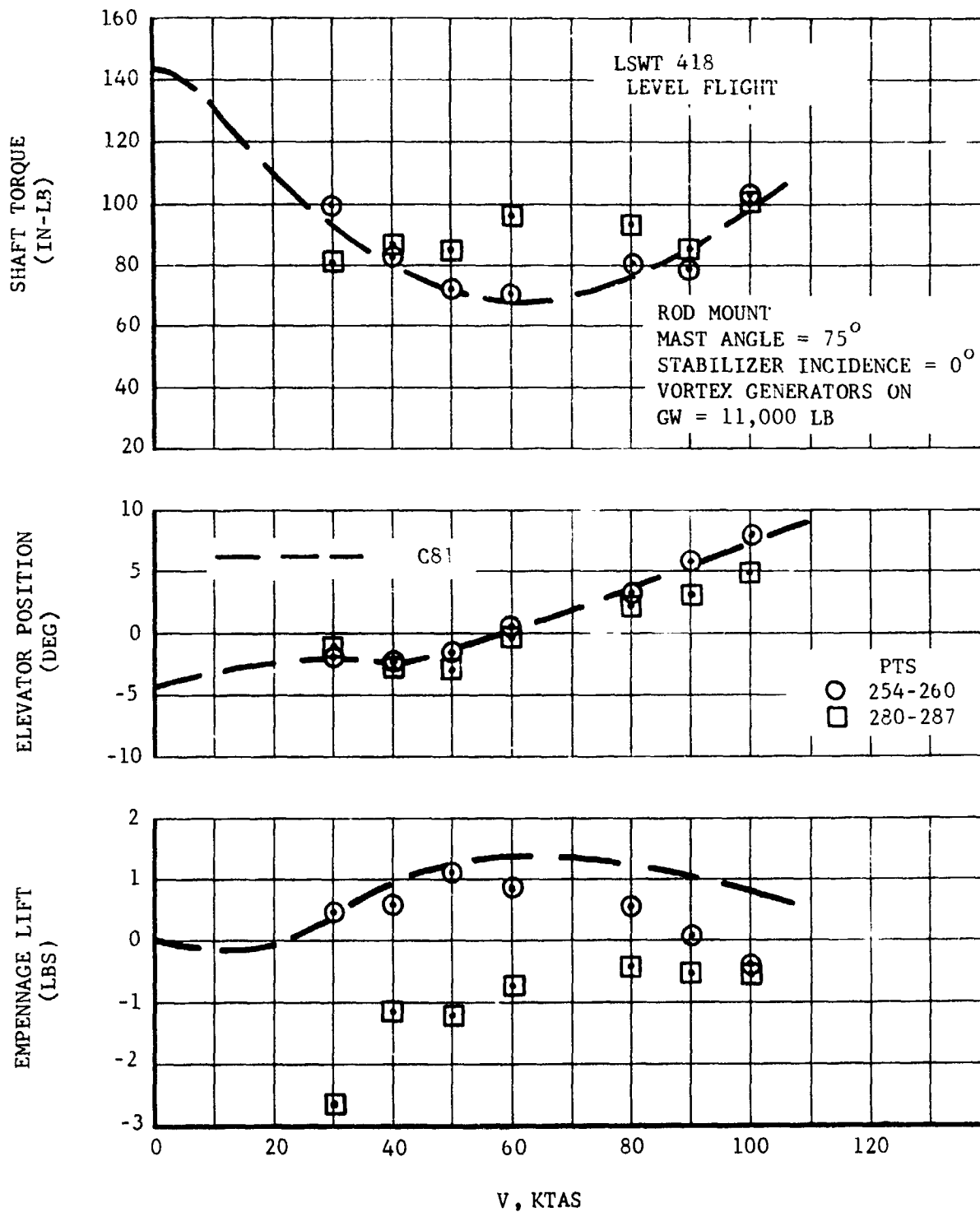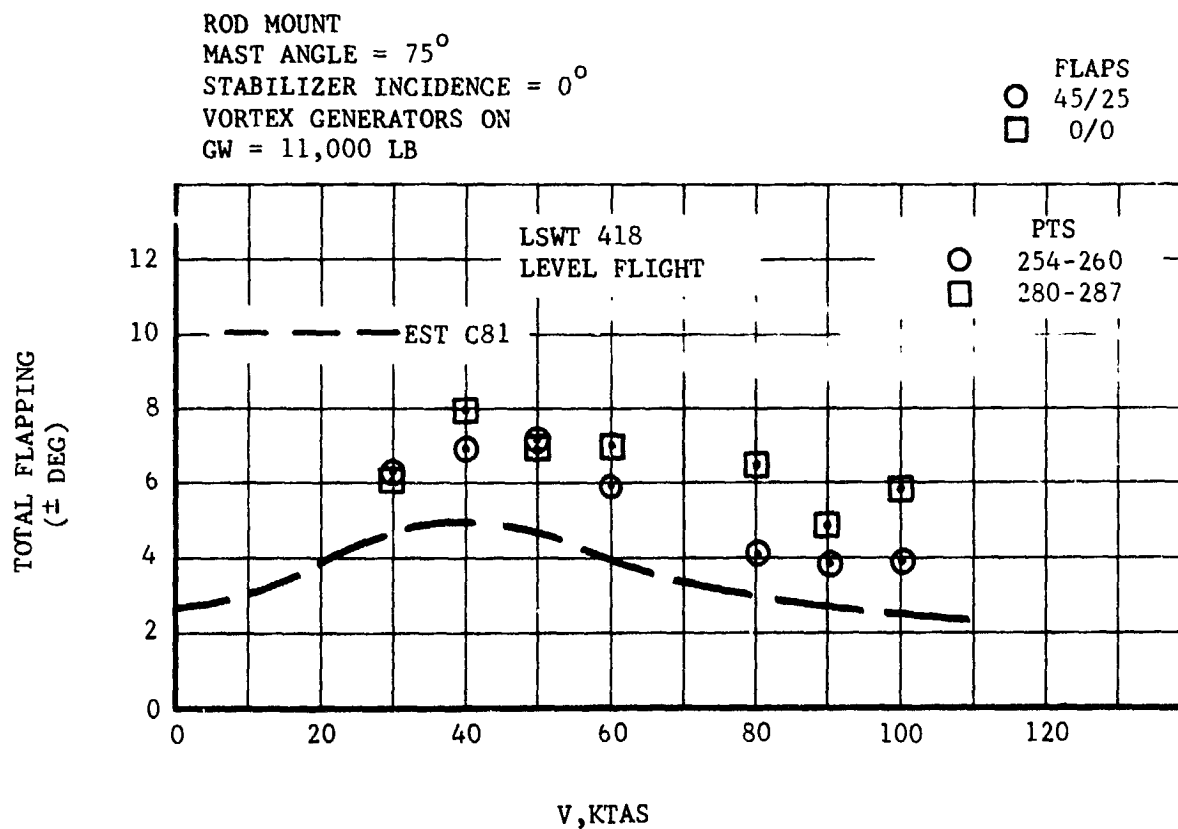
Figure V-22. Concluded
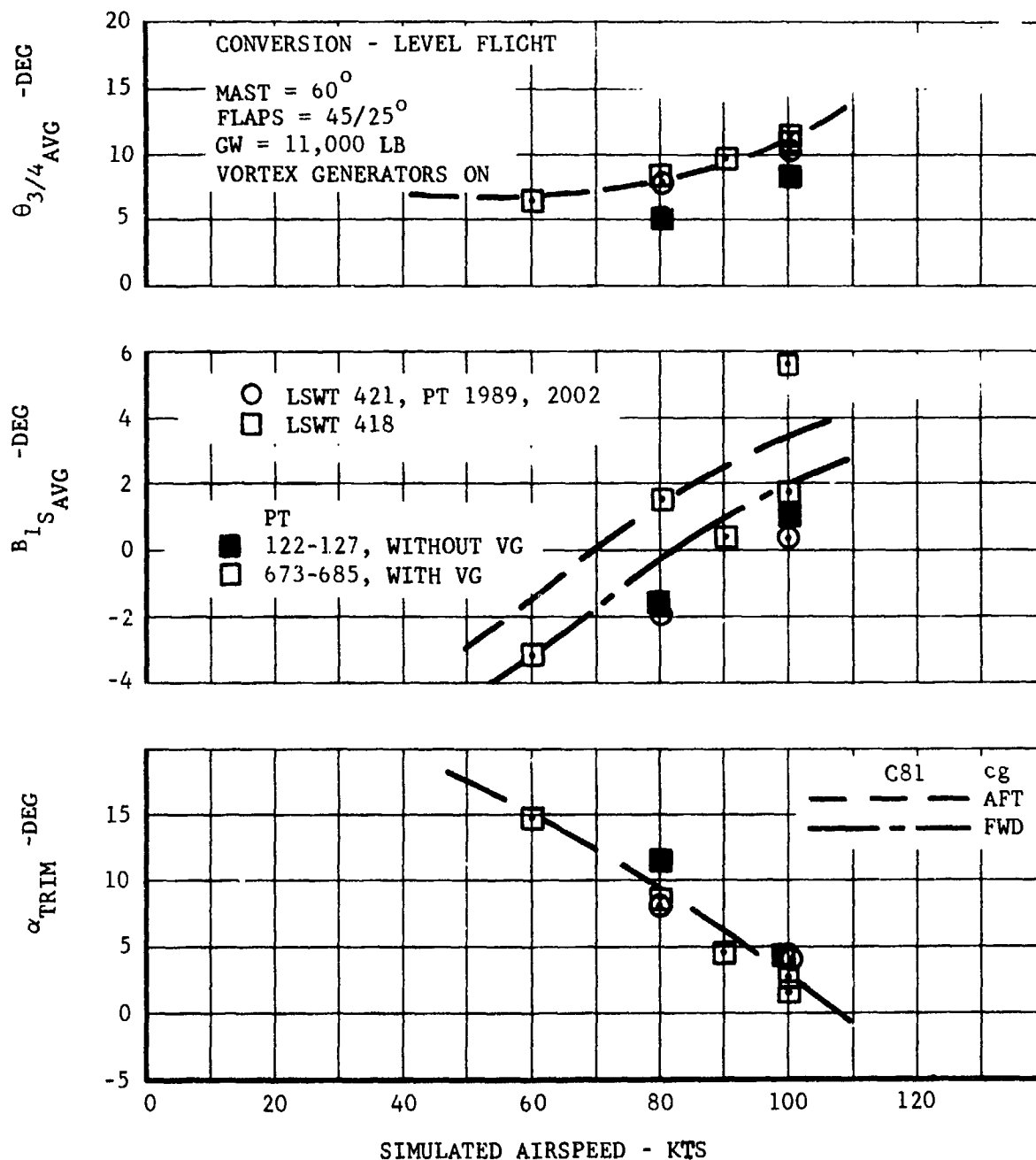
Figure V-23.   Level Flight Trim Parameters,
Mast Angle 60°.

CONVERSION - LEVEL FLIGHT

MAST = 60°
FLAPS = 45/25°
GW = 11,000 LB
VORTEX GENERATORS ON

TOTAL FLAPPING (DEG)

C81   cg
AFT
FWD

○ LSWT 421, PT 1989, 2002
□ LSWT 418

L_H - LB

□ 122-127 - WITHOUT VG
■ 673-685 - WITH VG

Q - IN-LB

SIMULATED AIRSPEED - KTS
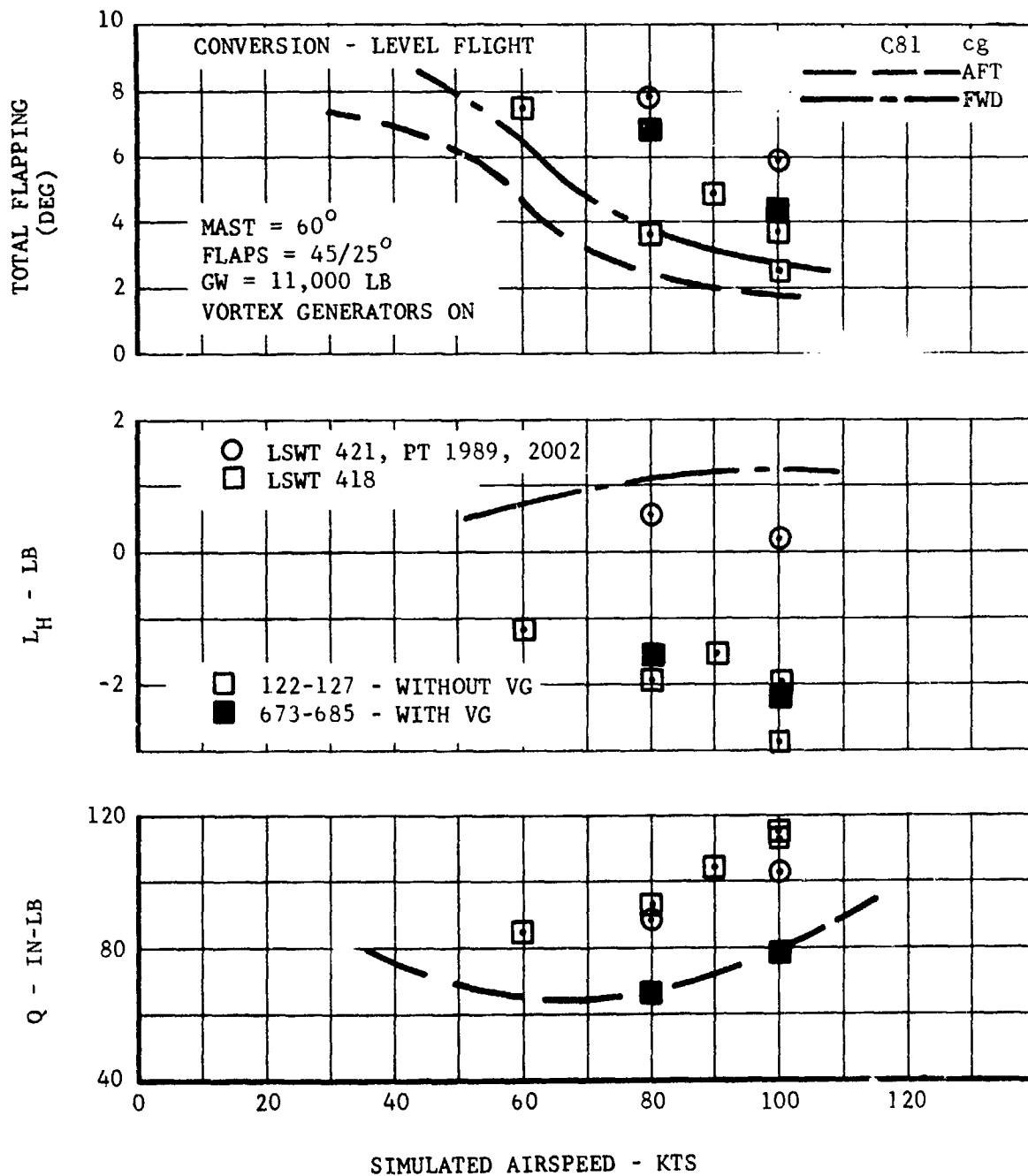
Figure V-23.   Continued
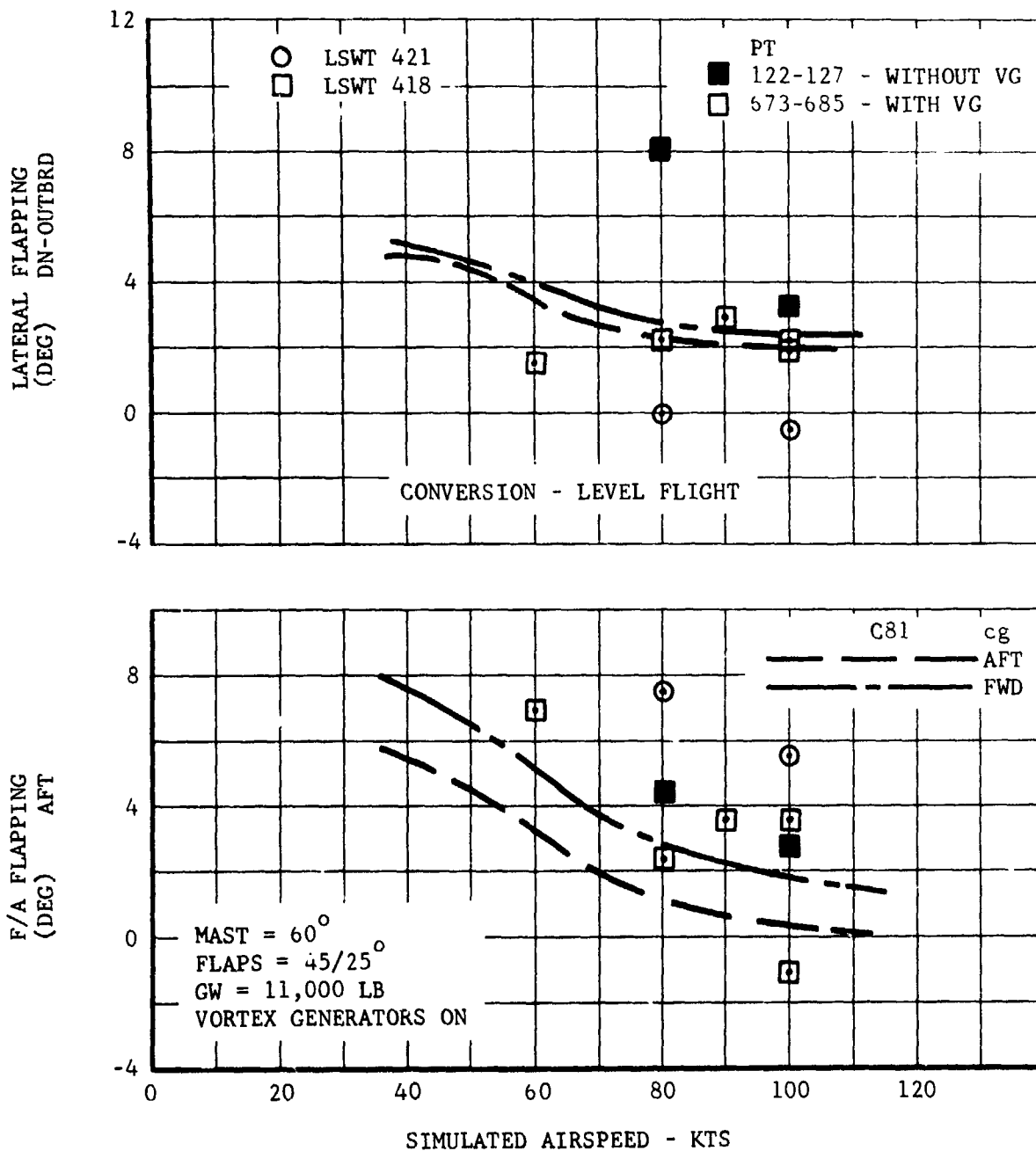
Figure V-23. Concluded
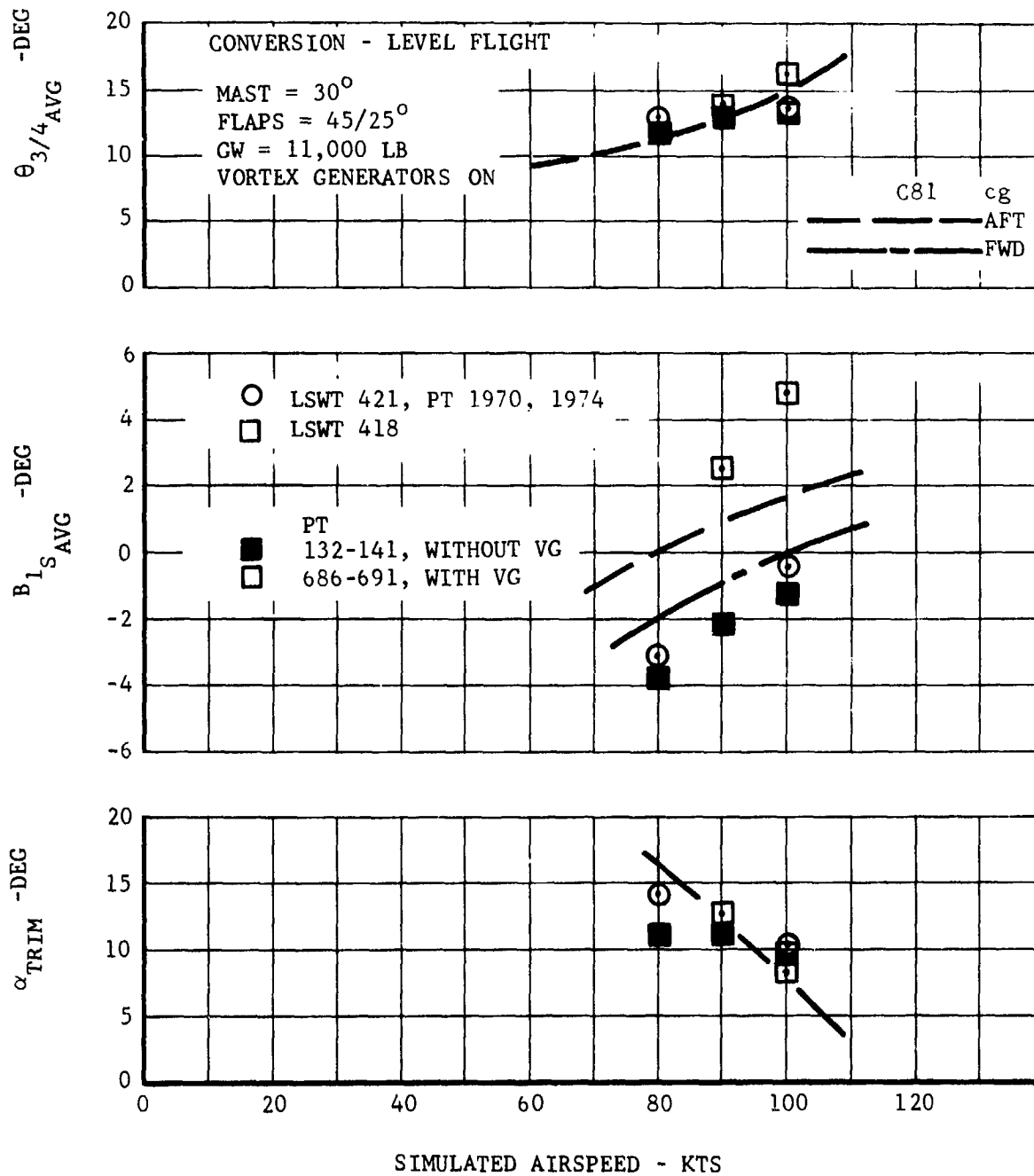
Figure V-24. Level Flight Trim Parameters,
Mast Angle 30°.

CONVERSION - LEVEL FLIGHT

MAST = 30°
FLAPS = $45/25^o$
G.W. = 11000 LBS
VORTEX GENERATORS ON

TOTAL FLAPPING (DEG)

C81   cg
_____ AFT
_____ FWD

PT
■ 132-141, WITHOUT VG
□ 686-691, WITH VG

L_H - LB

O LSWT 421, PT 1970, 1974
□ LSWT 418

Q - IN-LB

SIMULATED AIRSPEED - KTS
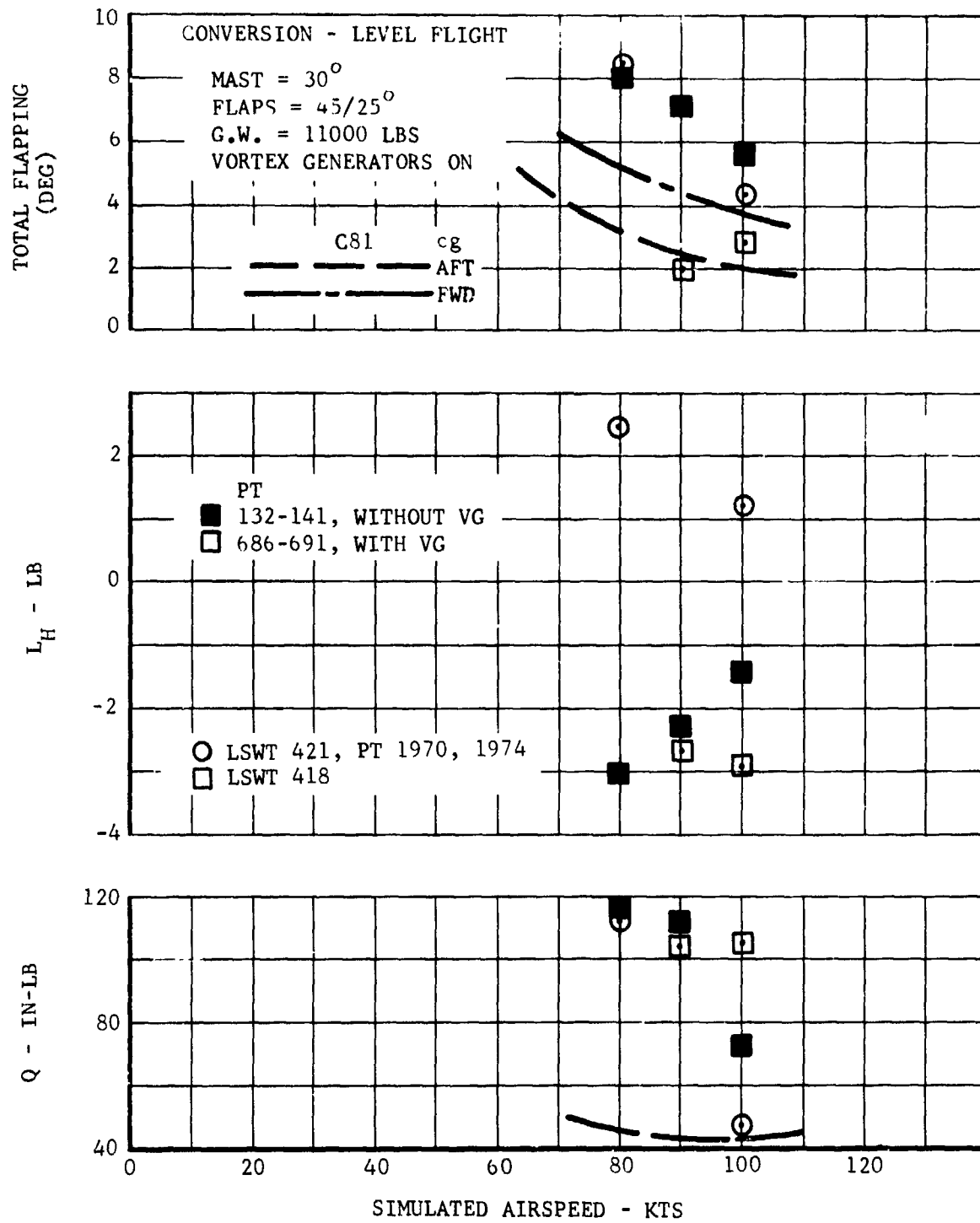
Figure V-24.   Continued
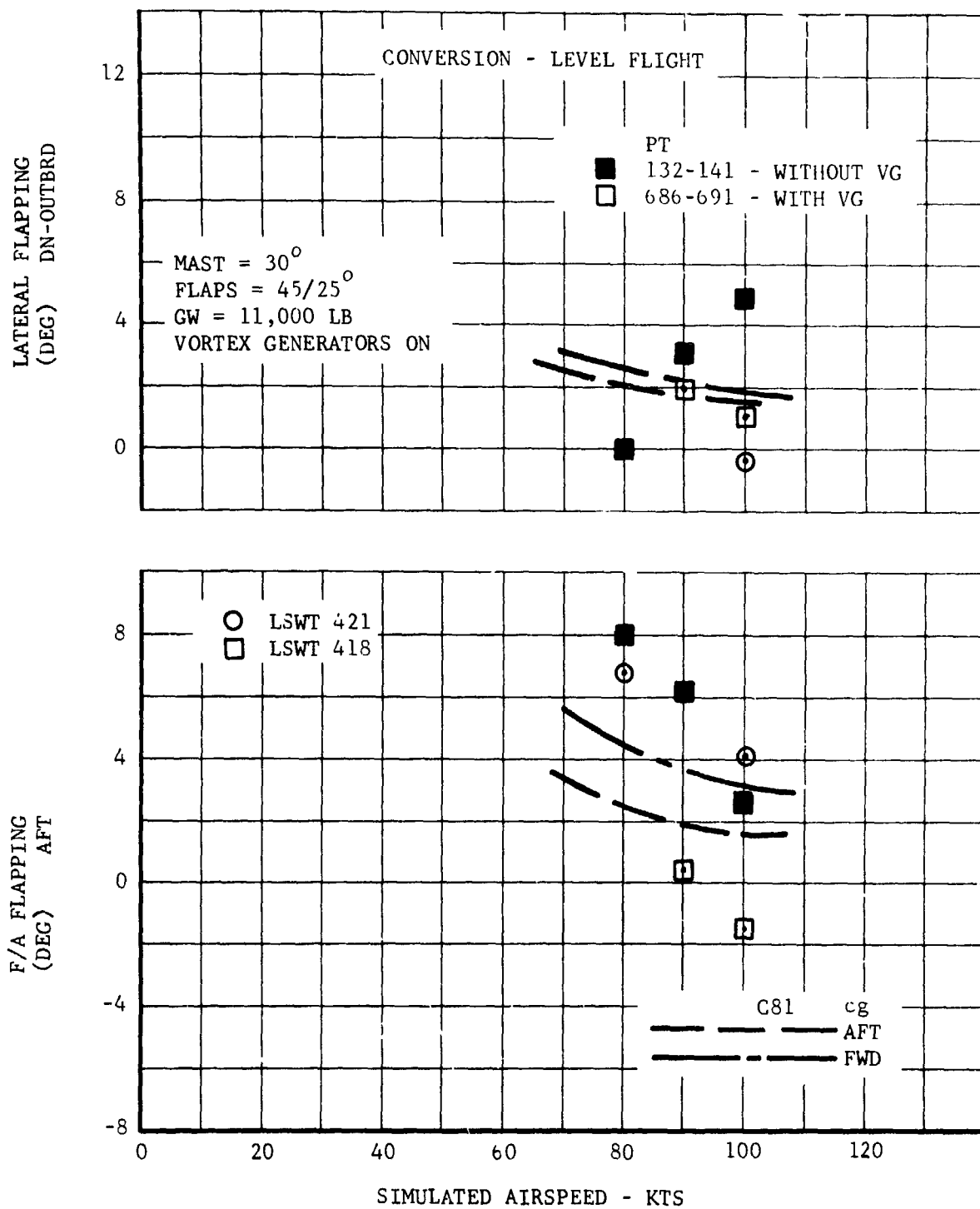
Figure V-24. Concluded
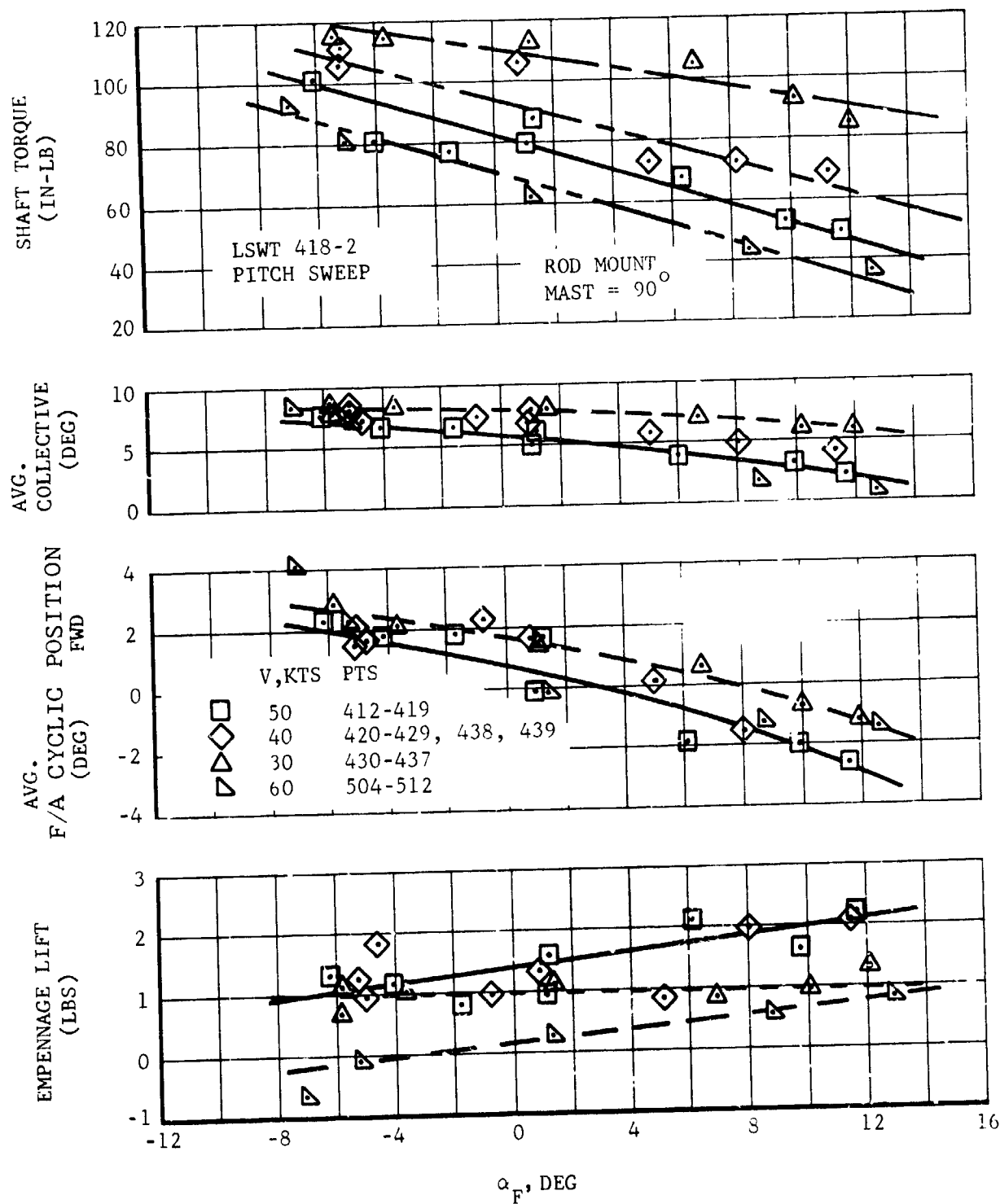
Figure V-25. Variation of Trim Parameters With
Fuselage Angle of Attack, Mast Angle 90°.

301-099-002
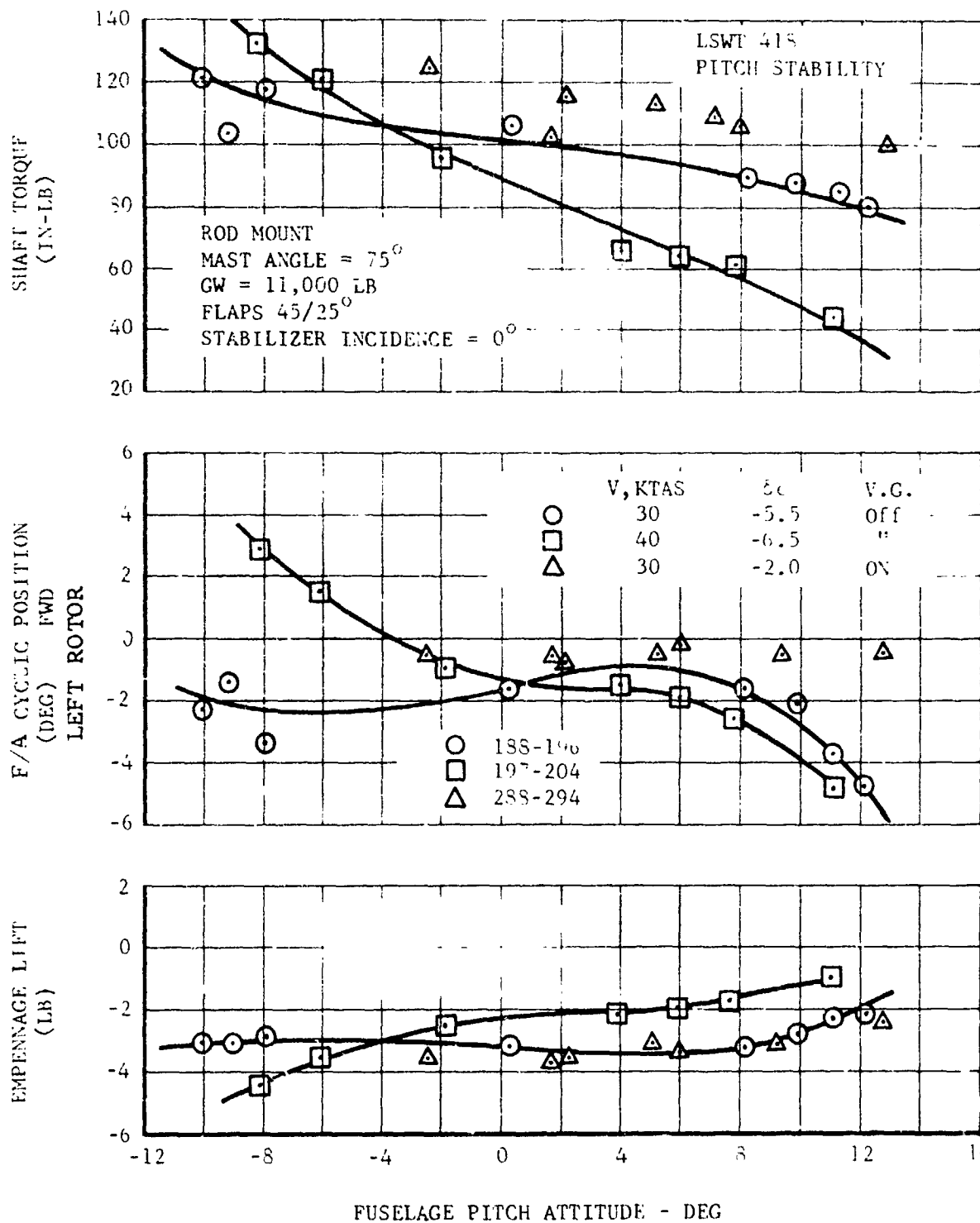
Figure V-26. Variation of Trim Parameters With Fuselage Angle of Attack, Mast Angle 75°.

ROD MOUNT
MAST ANGLE = 90°
FLAPS = +5/25°
STABILIZER INCIDENCE = 0°
VORTEX GENERATORS ON
GW = 11,000 LB
ROLL LOCKED
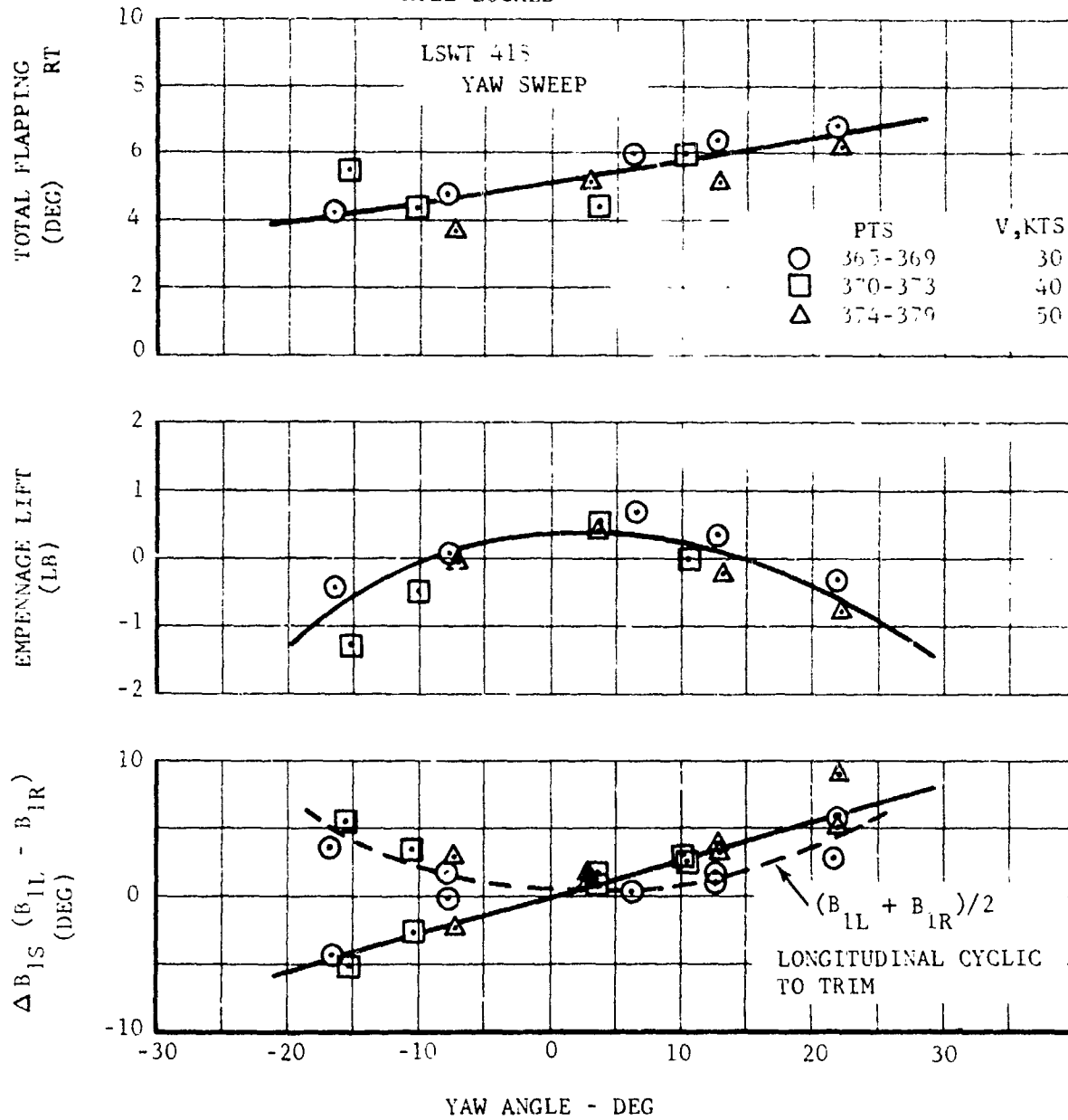
Figure V-27. Variation of Trim Parameters With
Fuselage Yaw Angle, Mast Angle 90°.
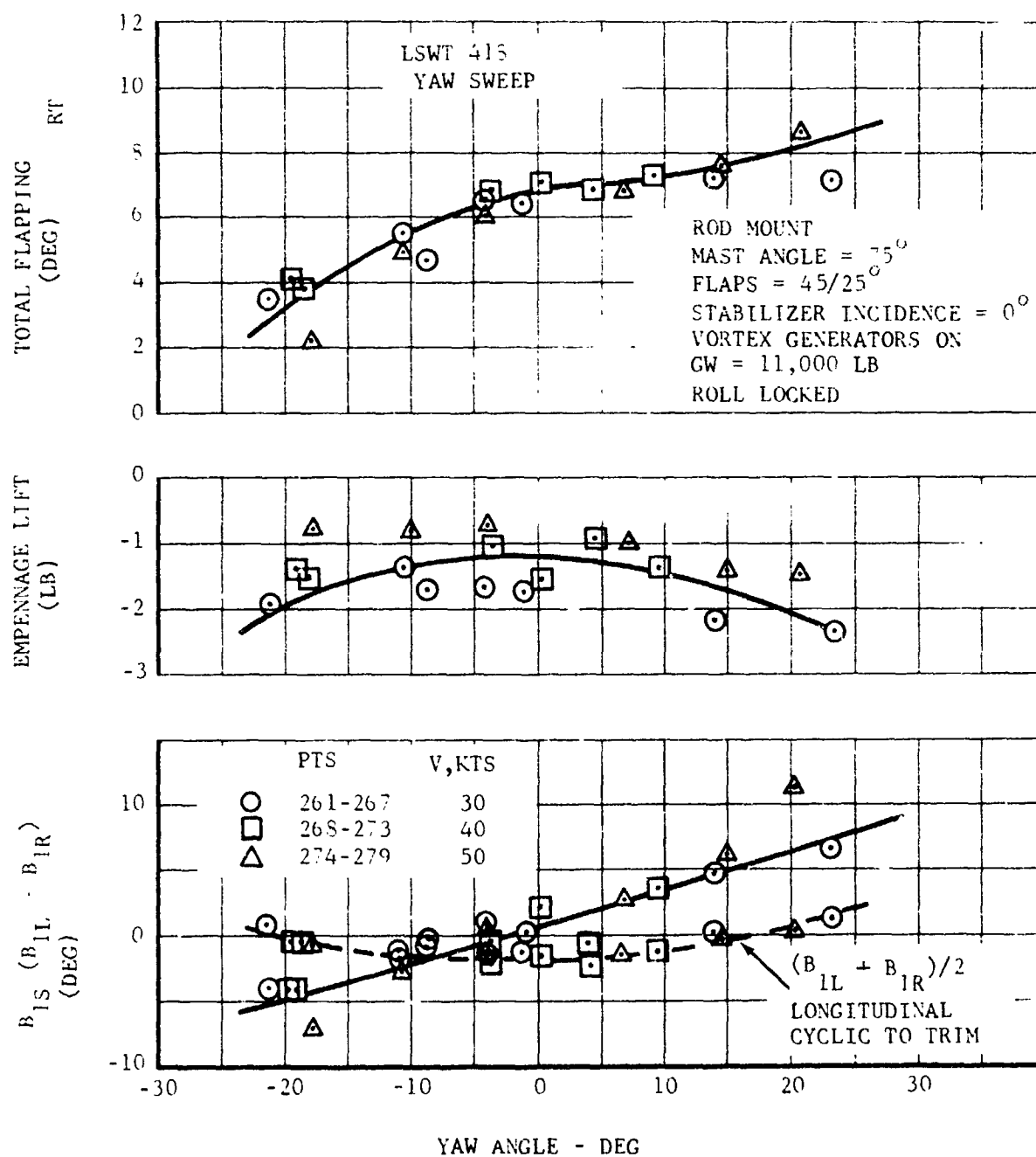
Figure V-28. Variation of Trim Parameters With
Fuselage Yaw Angle, Mast Angle 75°.

BLADE BEAM
31.7%

YOKE CHORD

(OUT)

PITCH LINK

FLAPPING

90°   ψ=90°   90°   90° 90°   ψ=90°   90°   90°

$\alpha_{MAST} = 90°$ $\alpha_{MAST} = 75°$
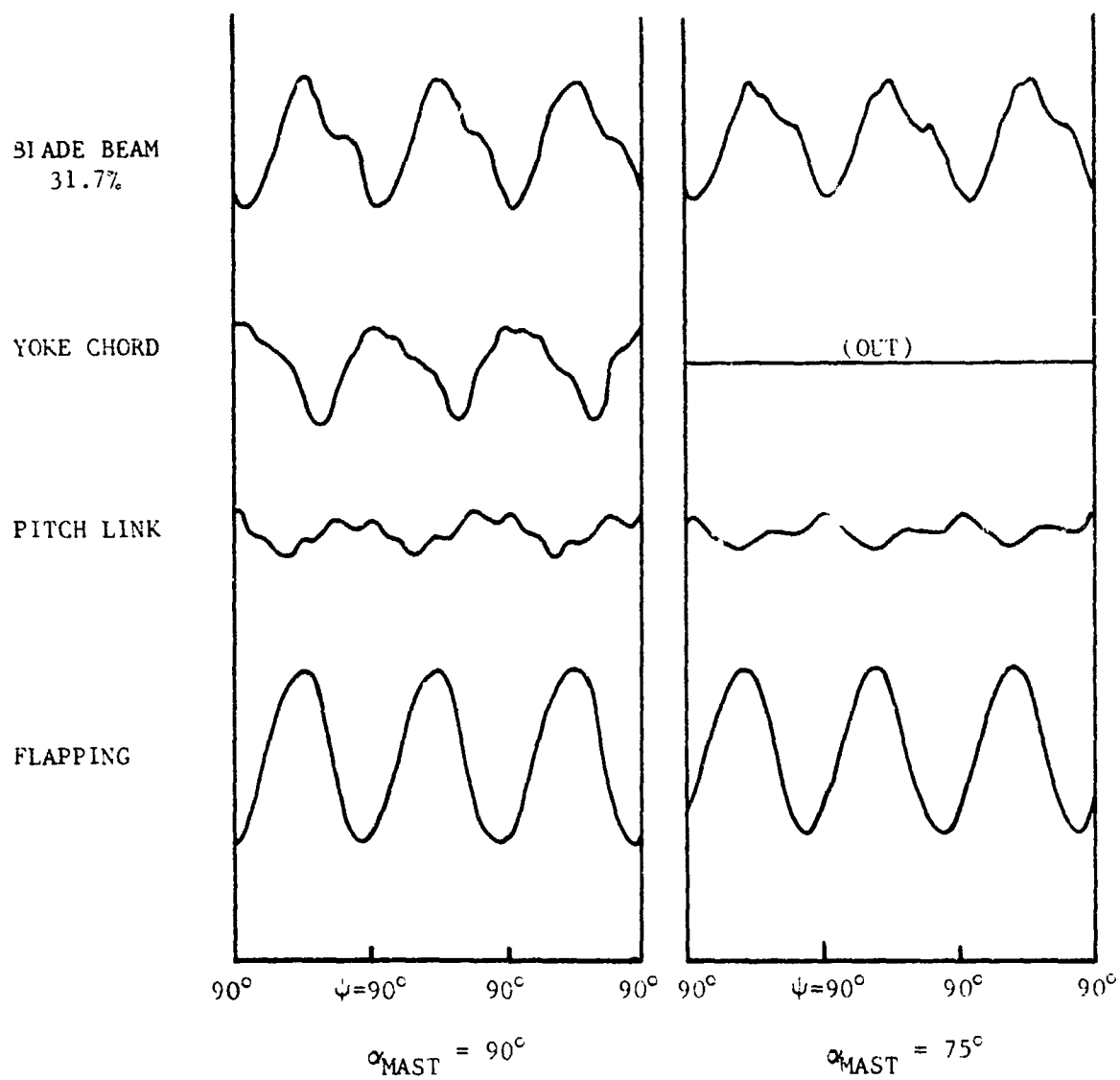
Figure V-29.   Blade and Control System Load Waveforms
at Mast Angles 90° and 75°.
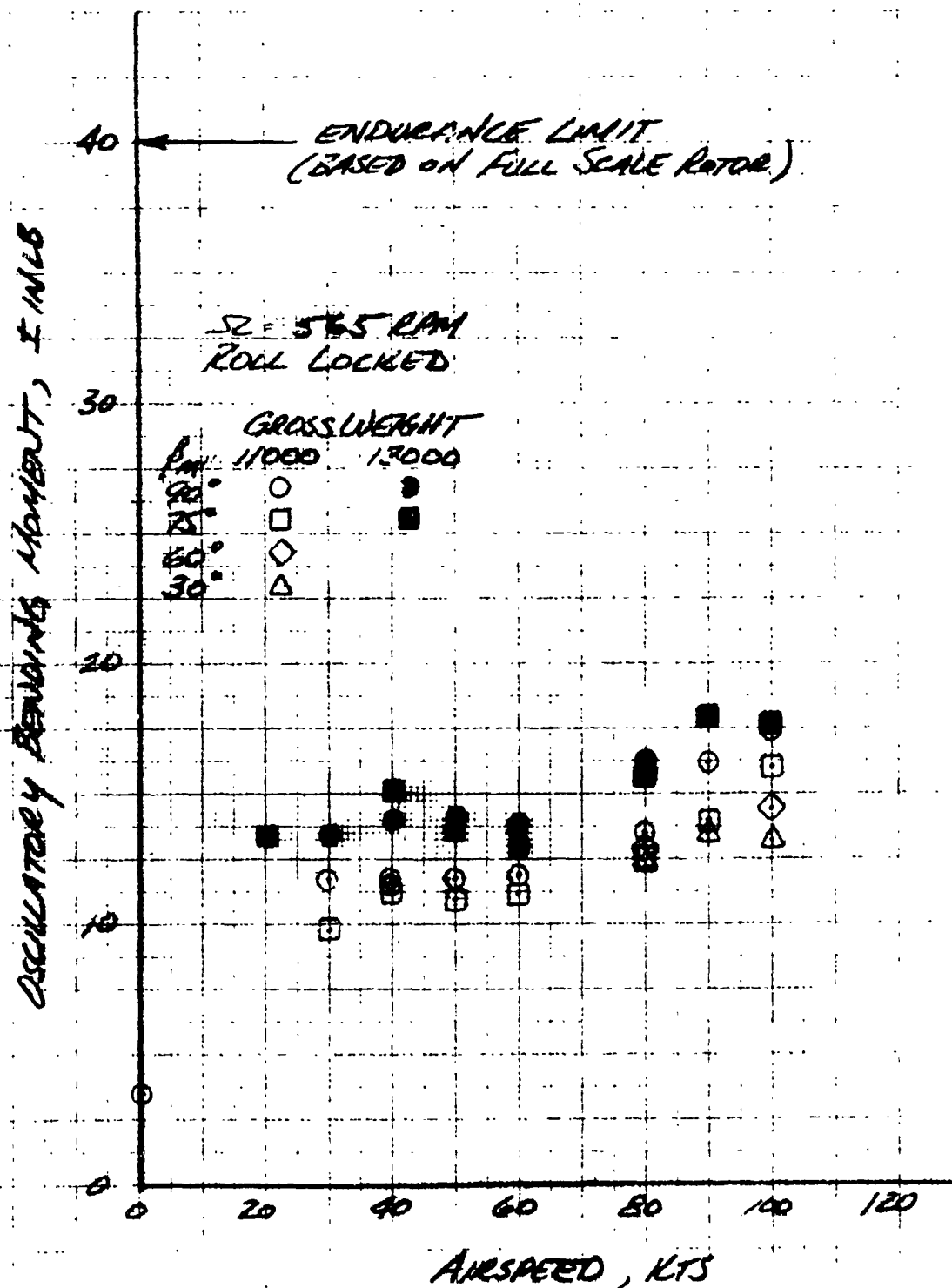
Figure V-30. Blade Station 52.5 Beamwise Oscillatory
Bending Moment versus Airspeed.
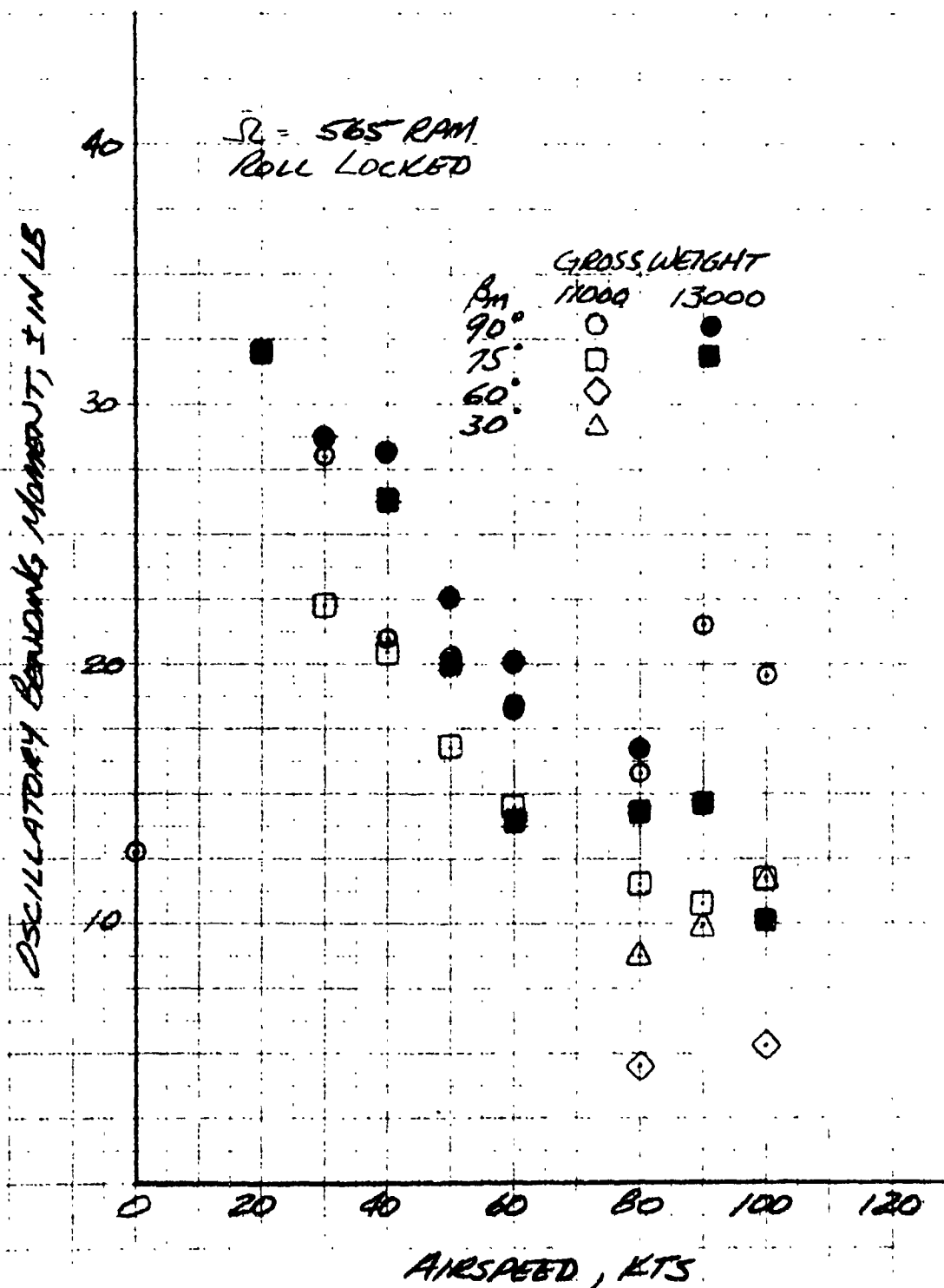
Figure V-31.   Blade Station 52.5 Chordwise Oscillatory
Bending Moment versus Airspeed.
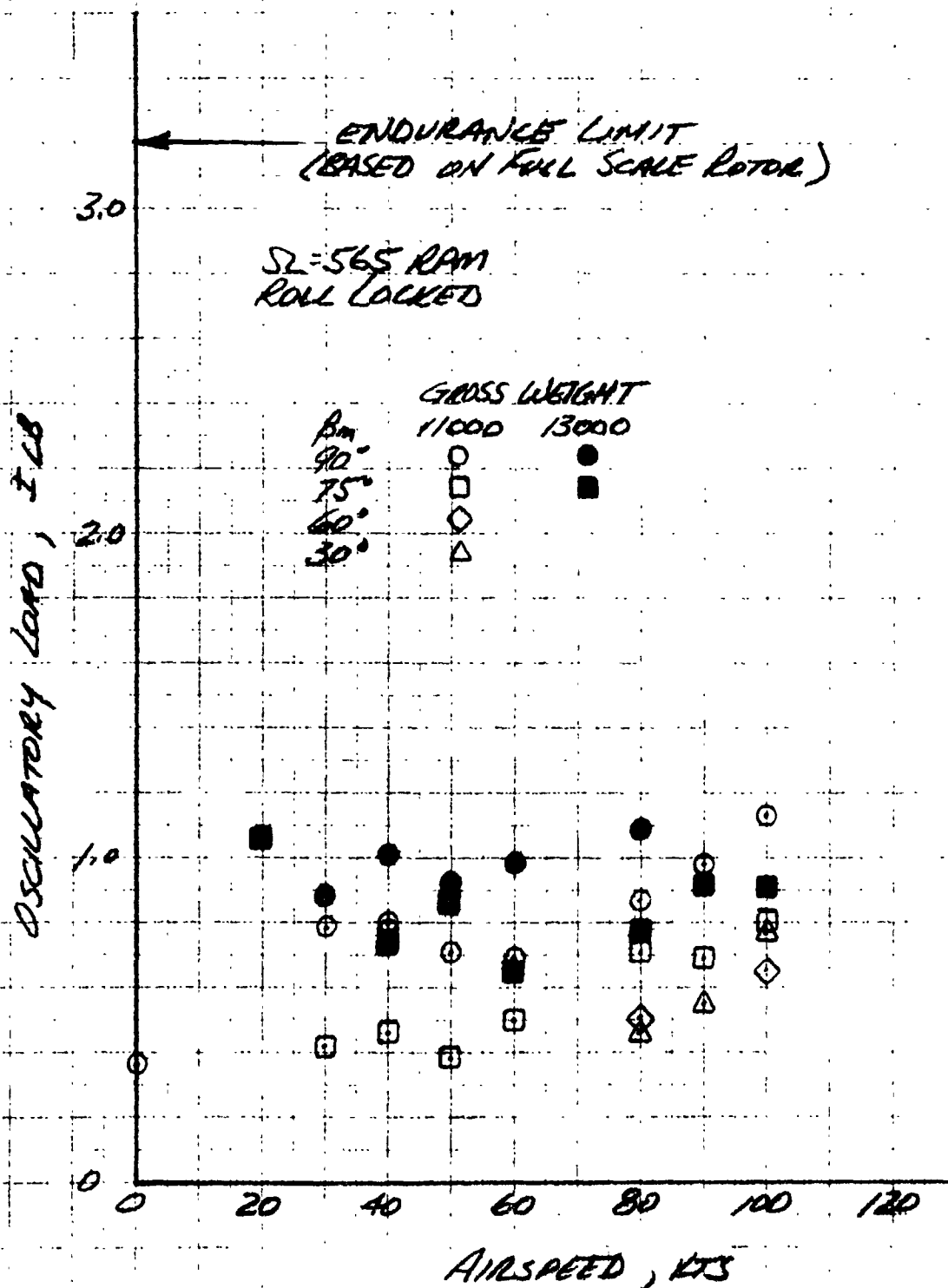
Figure V-32.   Pitch Link Oscillatory Load versus Airspeed.
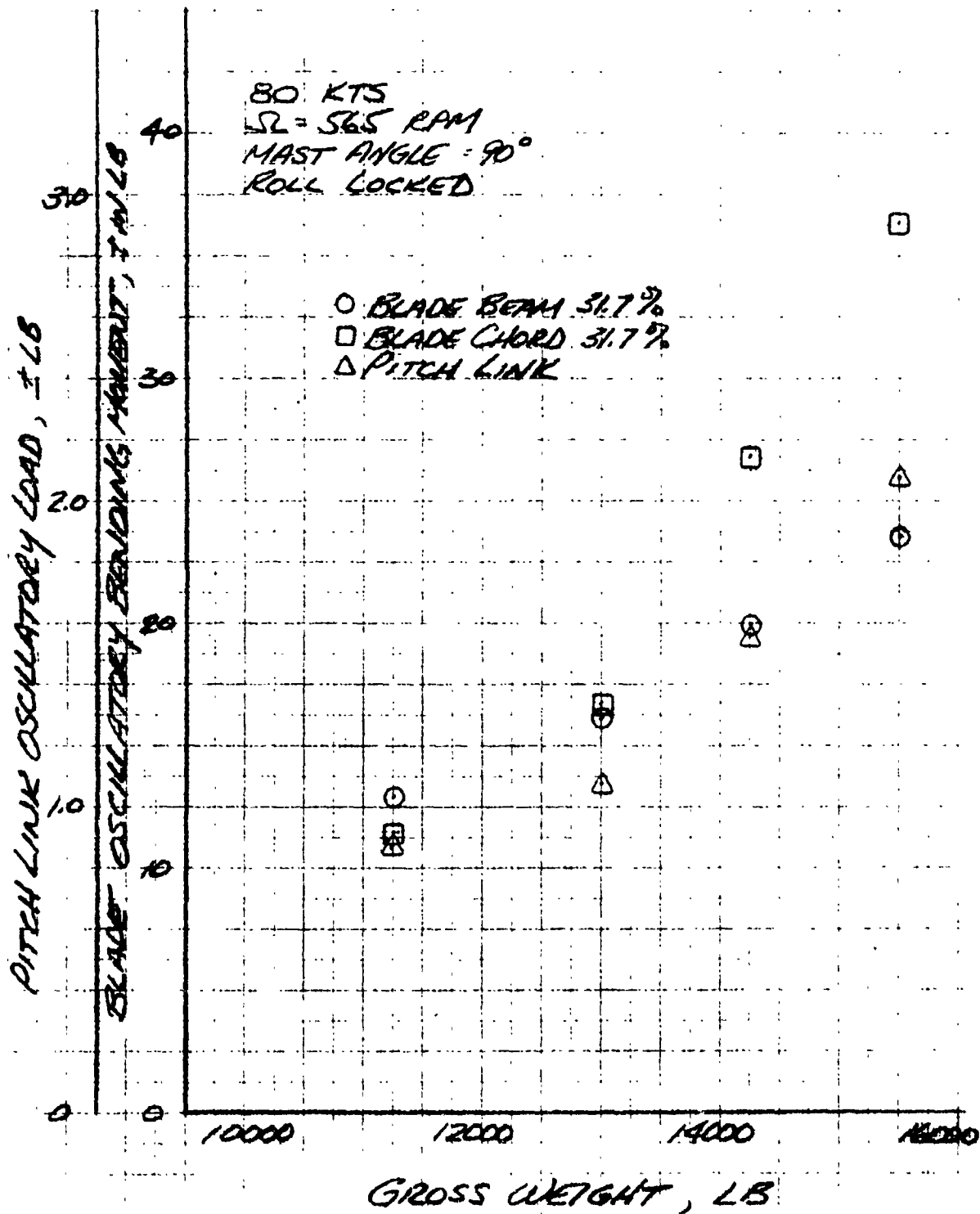
Figure V-33.   Blade and Control System Loads versus
Gross Weight.
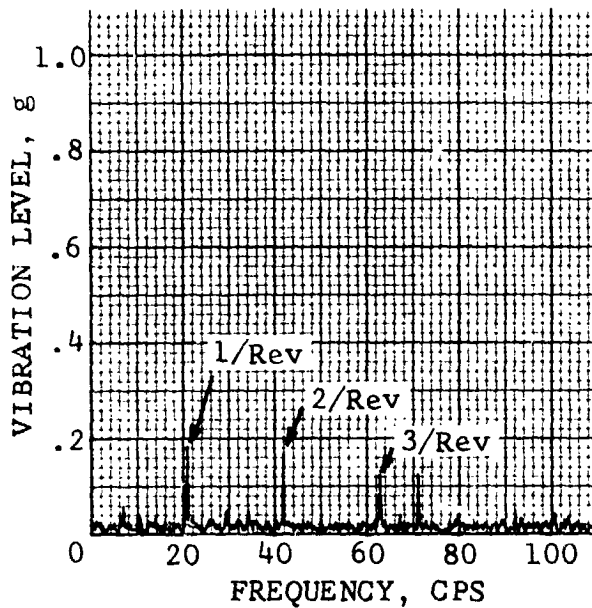
(a) AIRCRAFT CG VERTICAL
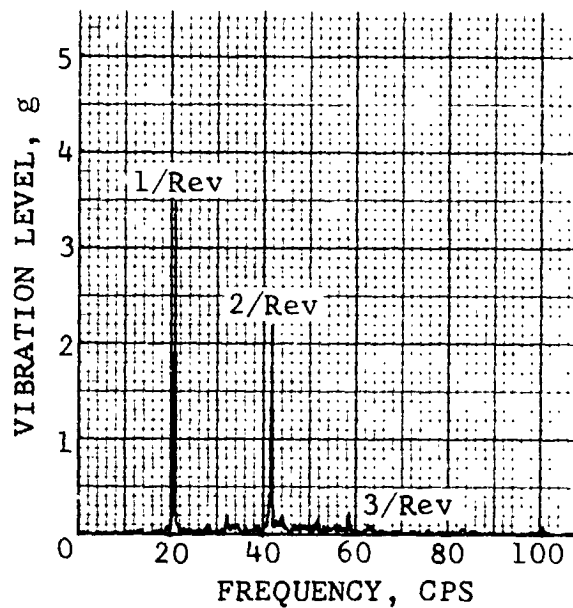
(b) RH VERTICAL STABILIZER F/A
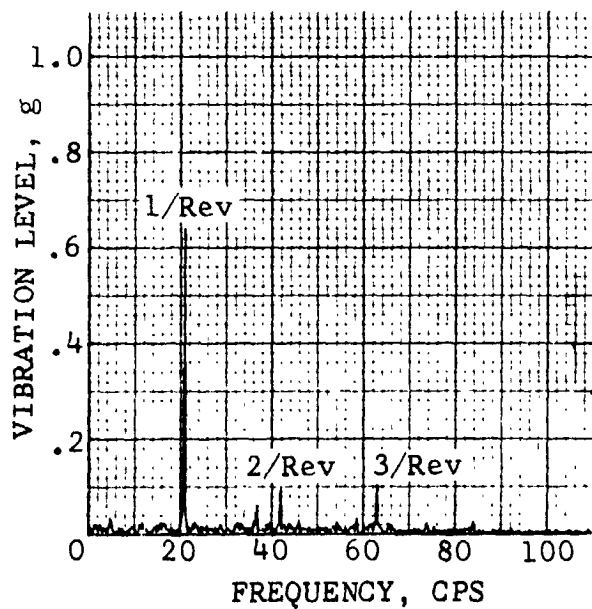
(c) RH POD LATERAL (XMSN CASE)
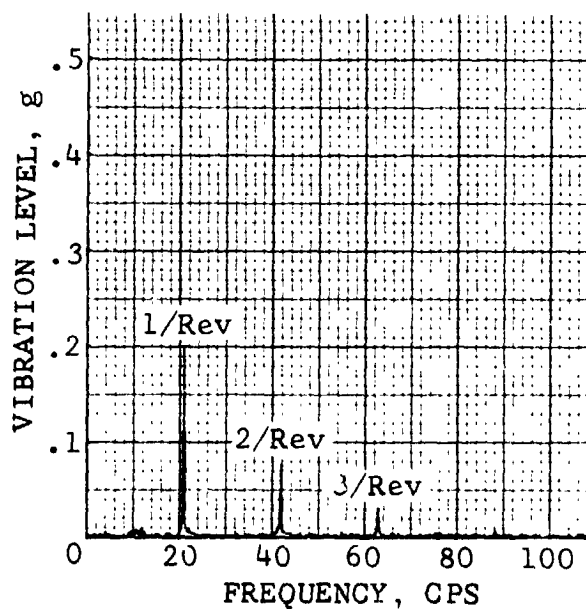
(d) LH POD LONGITUDINAL (XMSN CASE)

Figure V-34. Frequency Spectrum of Airframe Vibration for Several Accelerometer Locations.

GW = 11000 LB
Ω = 565 RPM
MAST ANGLE = 90°
ROLL LOCKED

○ RH VERTICAL STABILIZER F/A
□ RH POD FORWARD F/A
◇ LH POD FORWARD F/A
△ RH POD FORWARD LATERAL
◢ AIRCRAFT CG VERTICAL
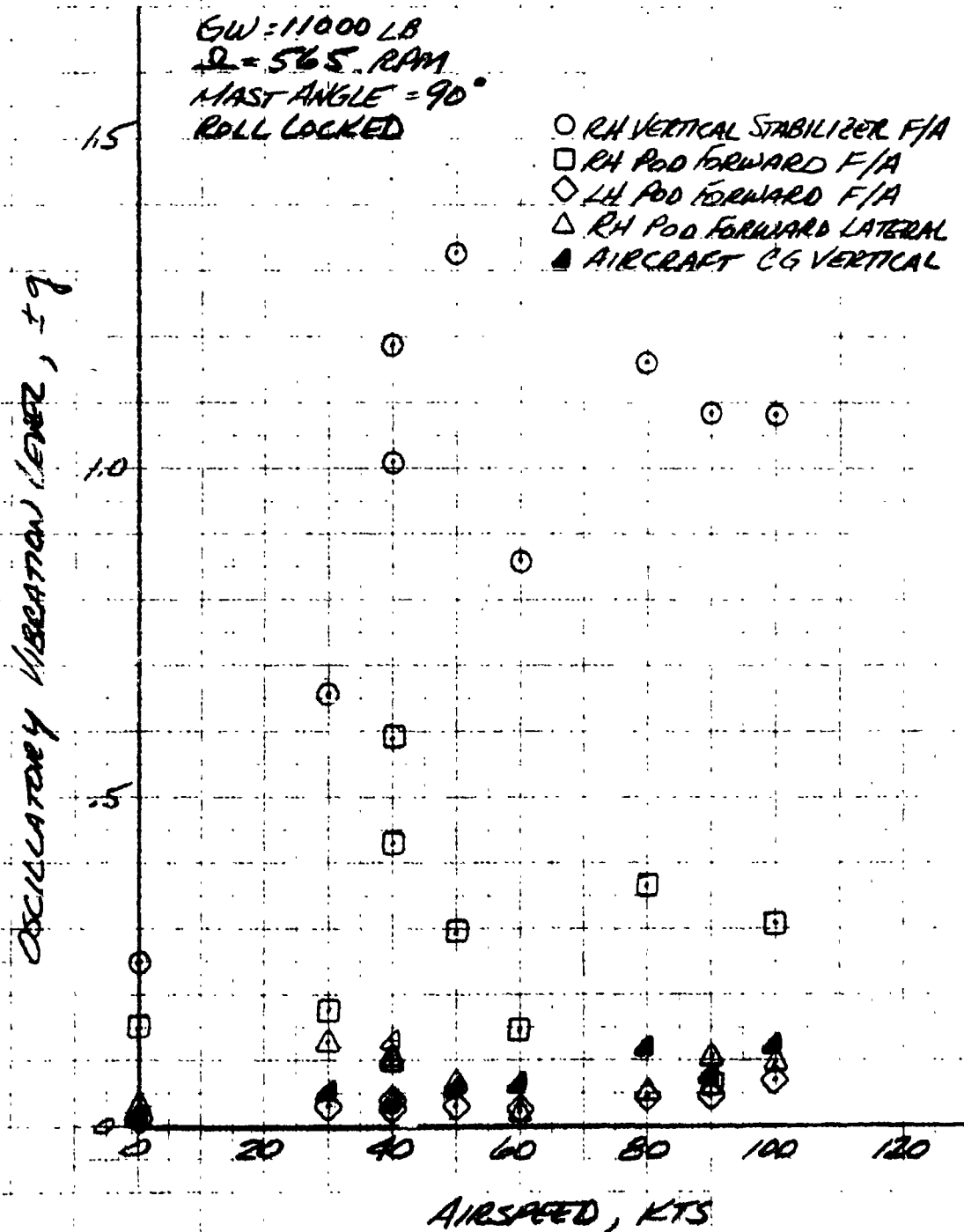
Figure V-35.   Two-Per-Rev Vibration Level versus Airspeed.

GW = 11000 LB
Ω = 565 RPM
MAST ANGLE = 90°
ROLL LOCKED

○ RH VERTICAL STABILIZER F/A
□ RH POD FORWARD F/A
◇ LH POD FORWARD F/A
△ RH POD FORWARD LATERAL
▲ AIRCRAFT CG VERTICAL
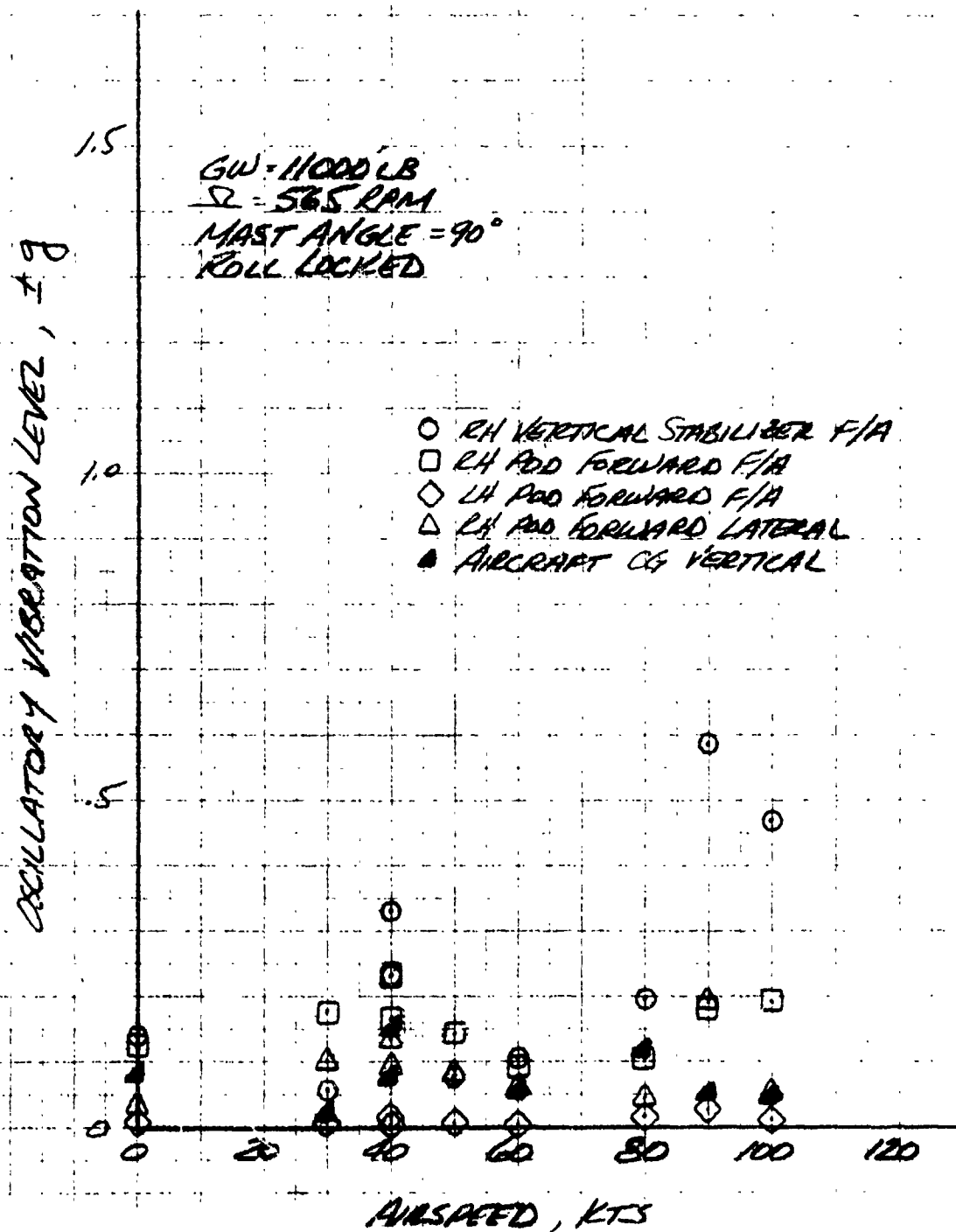
Figure V-36.   Three-Per-Rev Vibration Level
versus Airspeed.
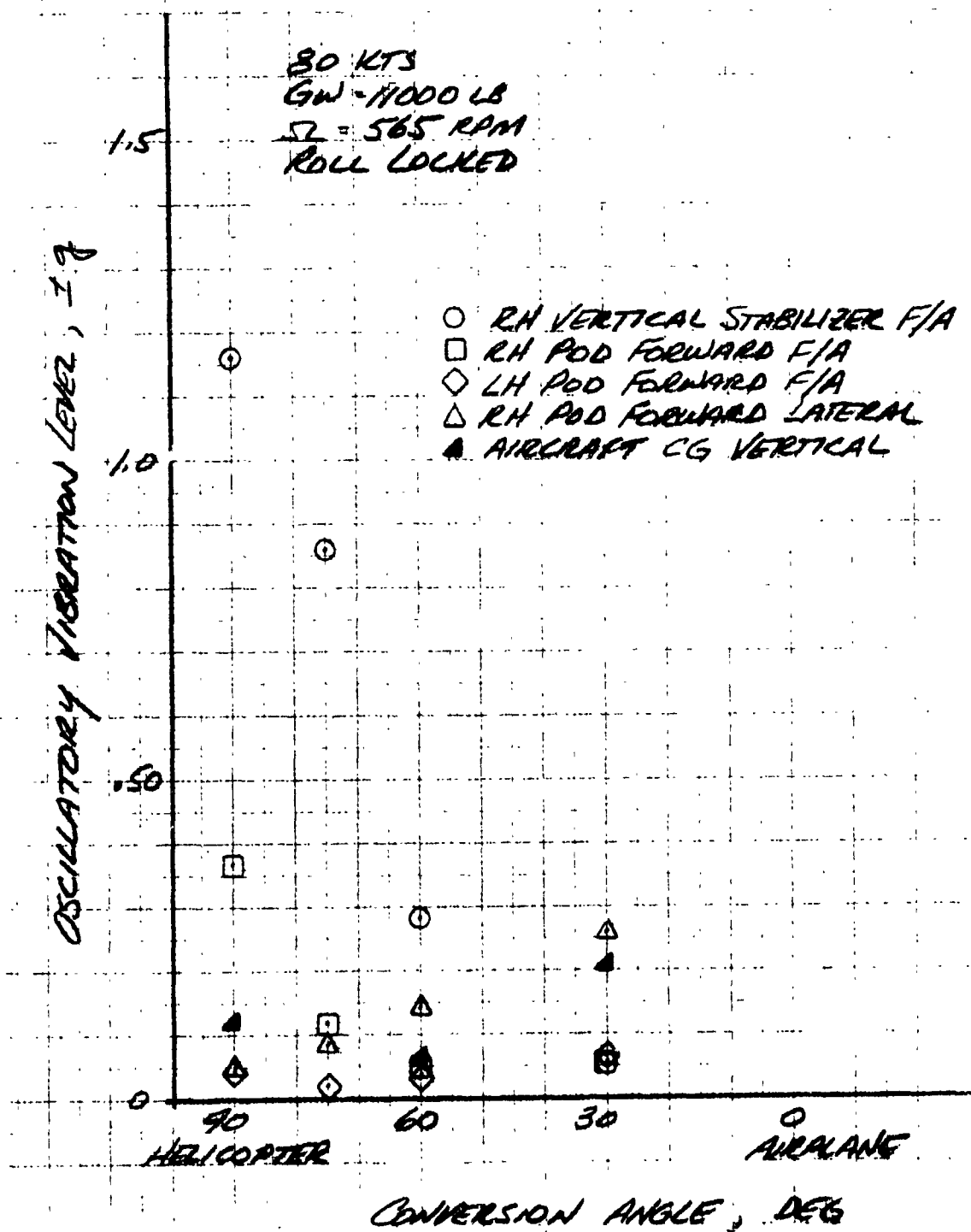
Figure V-37. Two-Per-Rev Vibration Level versus
Conversion Angle.
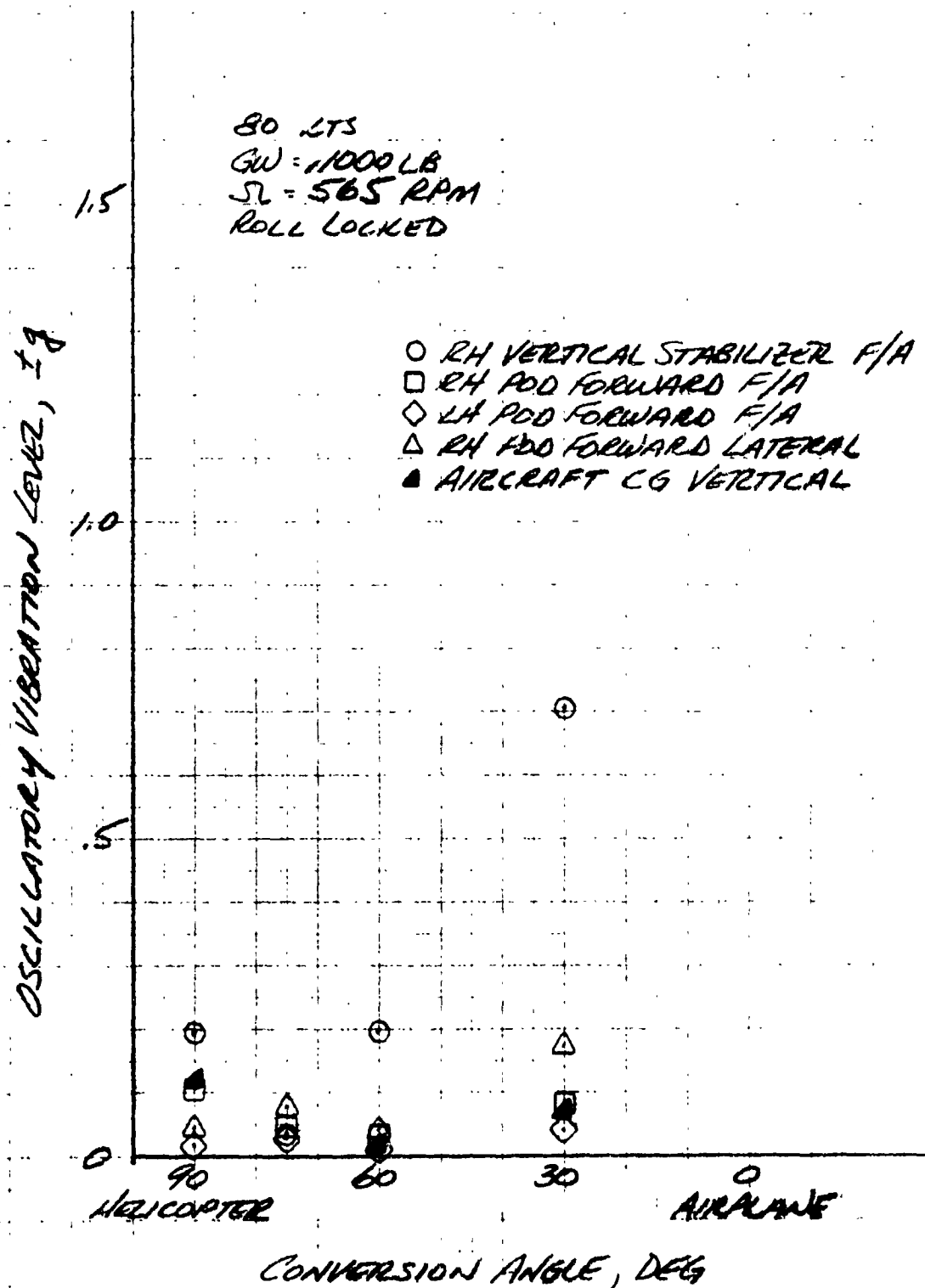
Figure V-38. Three-Per-Rev Vibration Level versus
Conversion Angle.

AIRFRAME LIFT AND PITCHING MOMENT

MAST ANGLE = 90°
FLAPS = 45/25°
VORTEX GENERATORS ON
ROTORS OFF
EMPENNAGE ON (i_H = 0°)

○ LSWT 408, $R_N = 1.2 \times 10^6$
□ LSWT 421, $R_N = .36 \times 10^6$
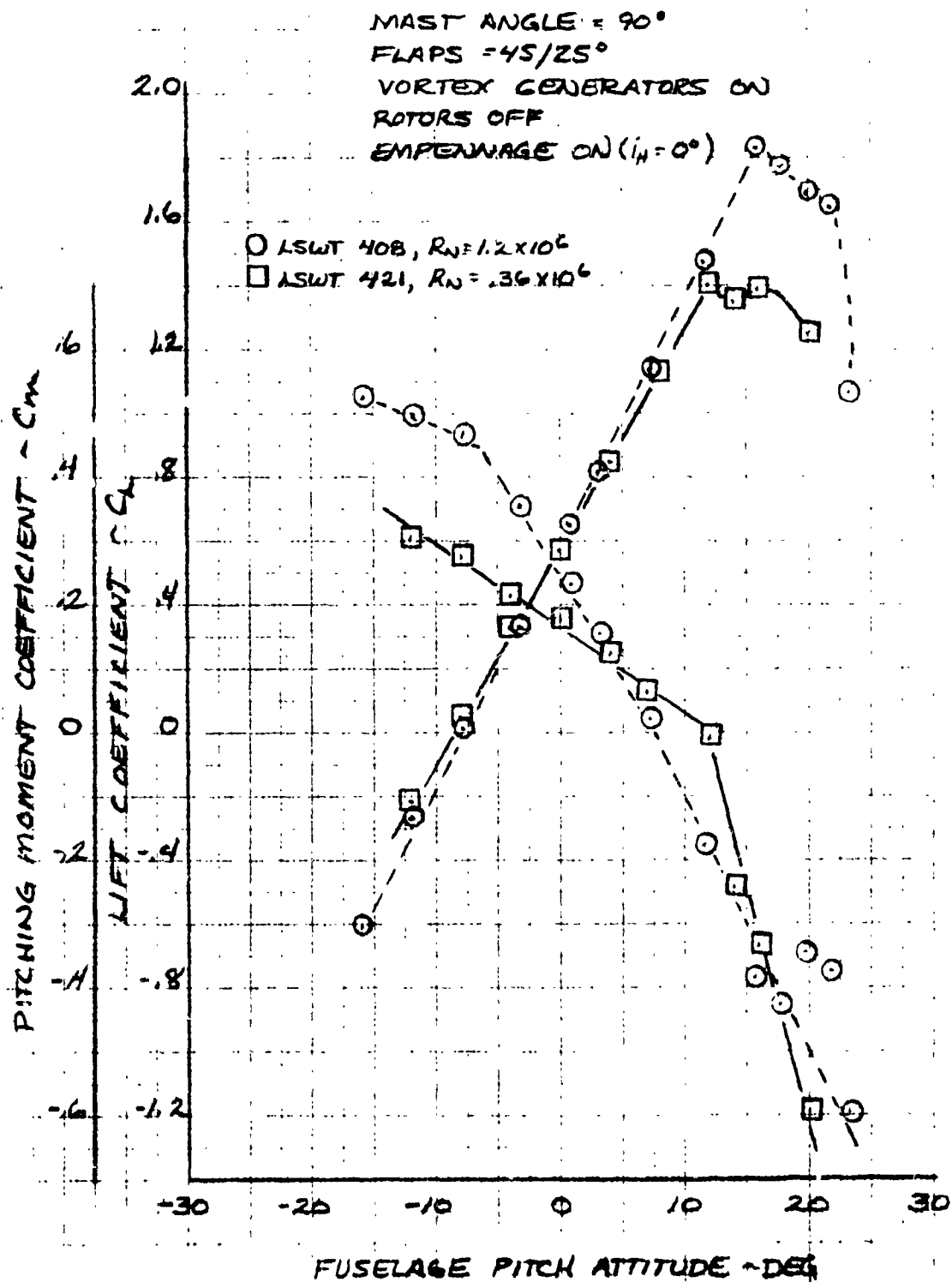
Figure V-39.   Airframe Lift and Pitching Moment versus
Angle of Attack, Rotors Removed.

AIRFRAME YAWING MOMENT

MAST ANGLE = 90°
FLAPS = 45/25°

ROTOR OFF
EMPENNAGE α
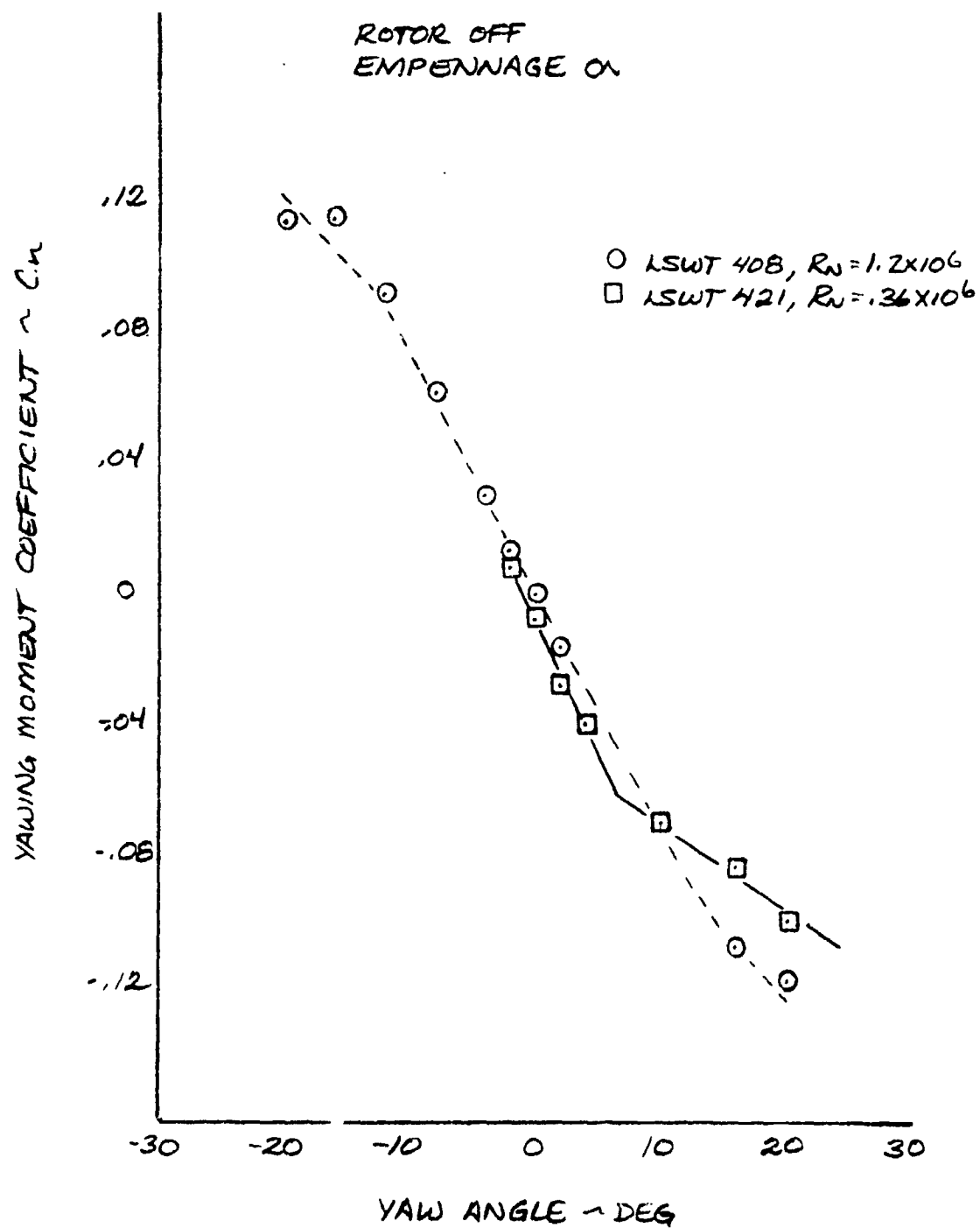
Figure V-40.   Airframe Yaw Moment versus Yaw Angle,
Rotors Removed.
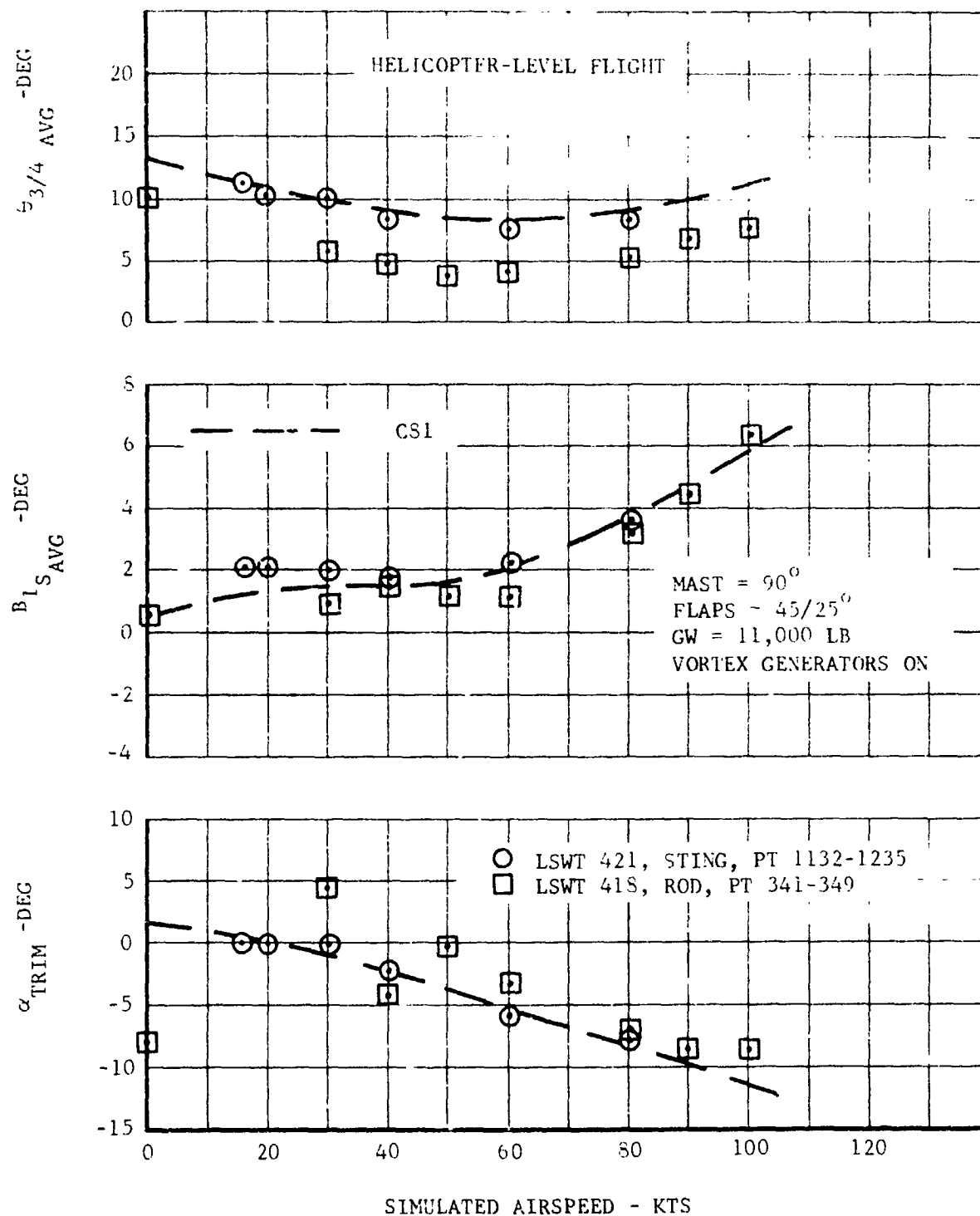
Figure V-41. Comparison of Rod Test with Sting Test
Level Flight Trim Parameters, Mast 90°.
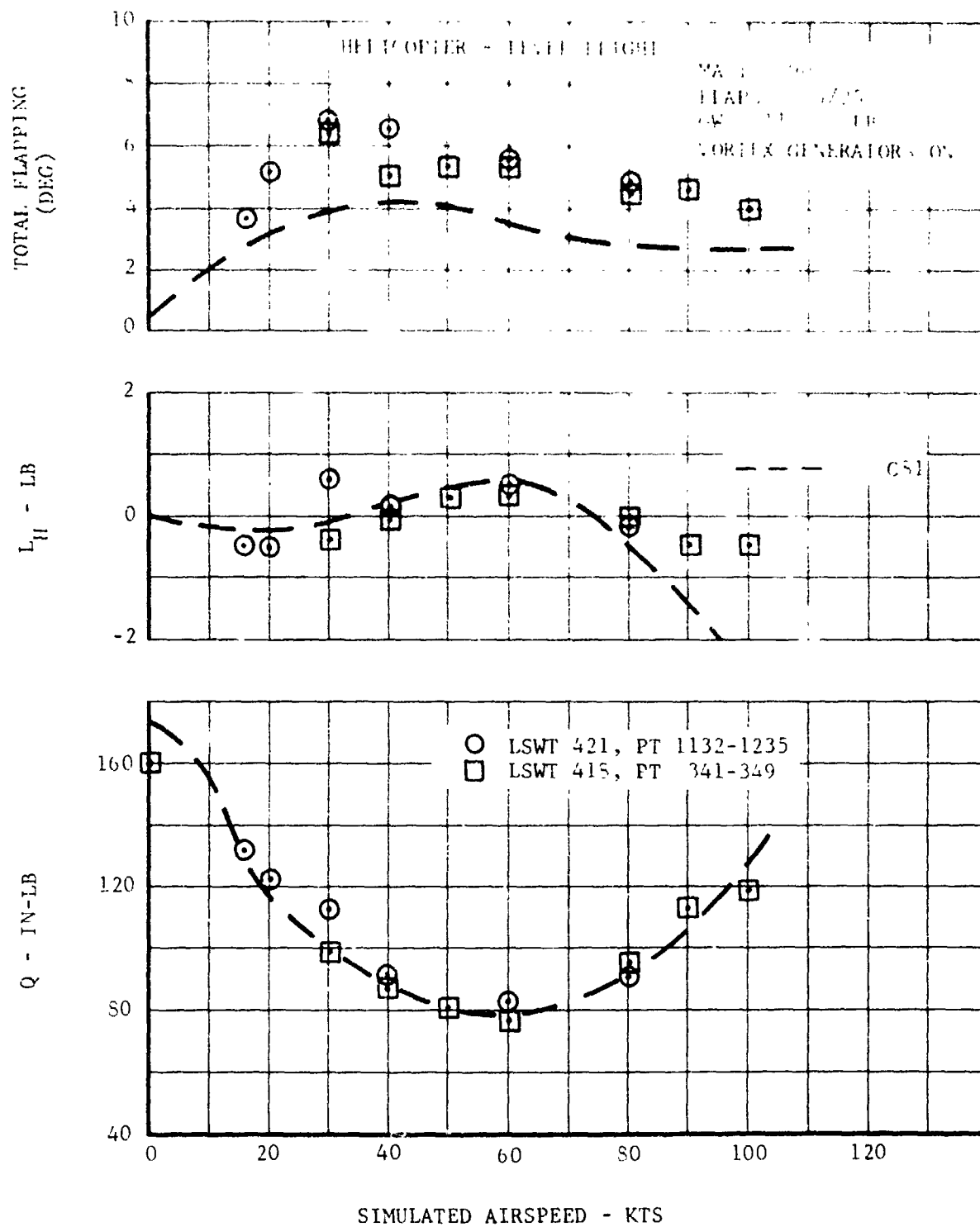
Figure V-41. Concluded
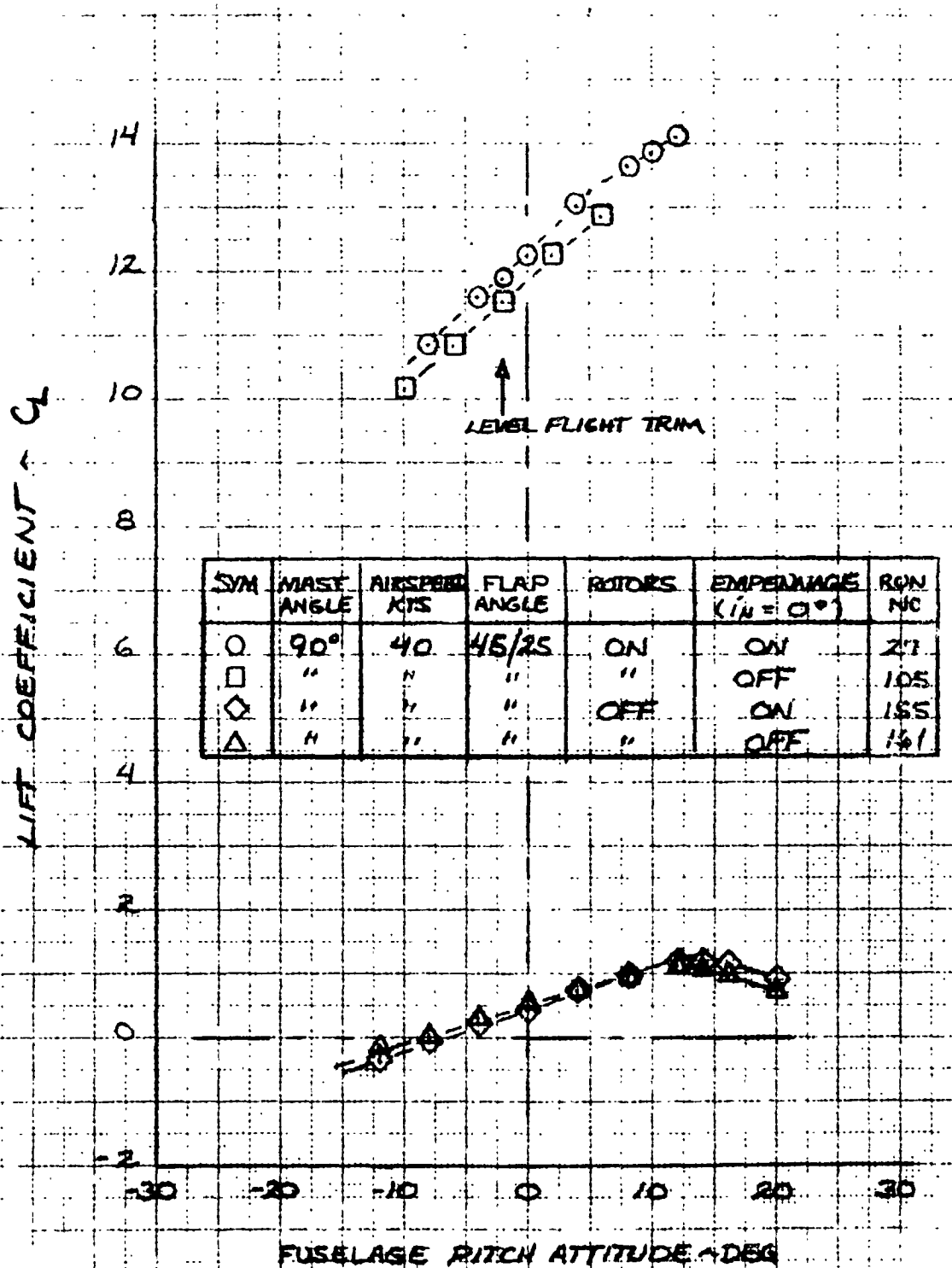
Figure V-42. Lift Coefficient versus Fuselage Angle of Attack, Mast Angle 90°, Airspeed 40 knots.

ROTOR AND AIRFRAME LIFT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE (iₙ = 0°)	RUN NO.
O	90°	60	45/25	ON	ON	29
□	"	"	"	"	OFF	107
◇	"	"	"	OFF	ON	157
△	"	"	"	"	OFF	165

LEVEL FLIGHT TRIM
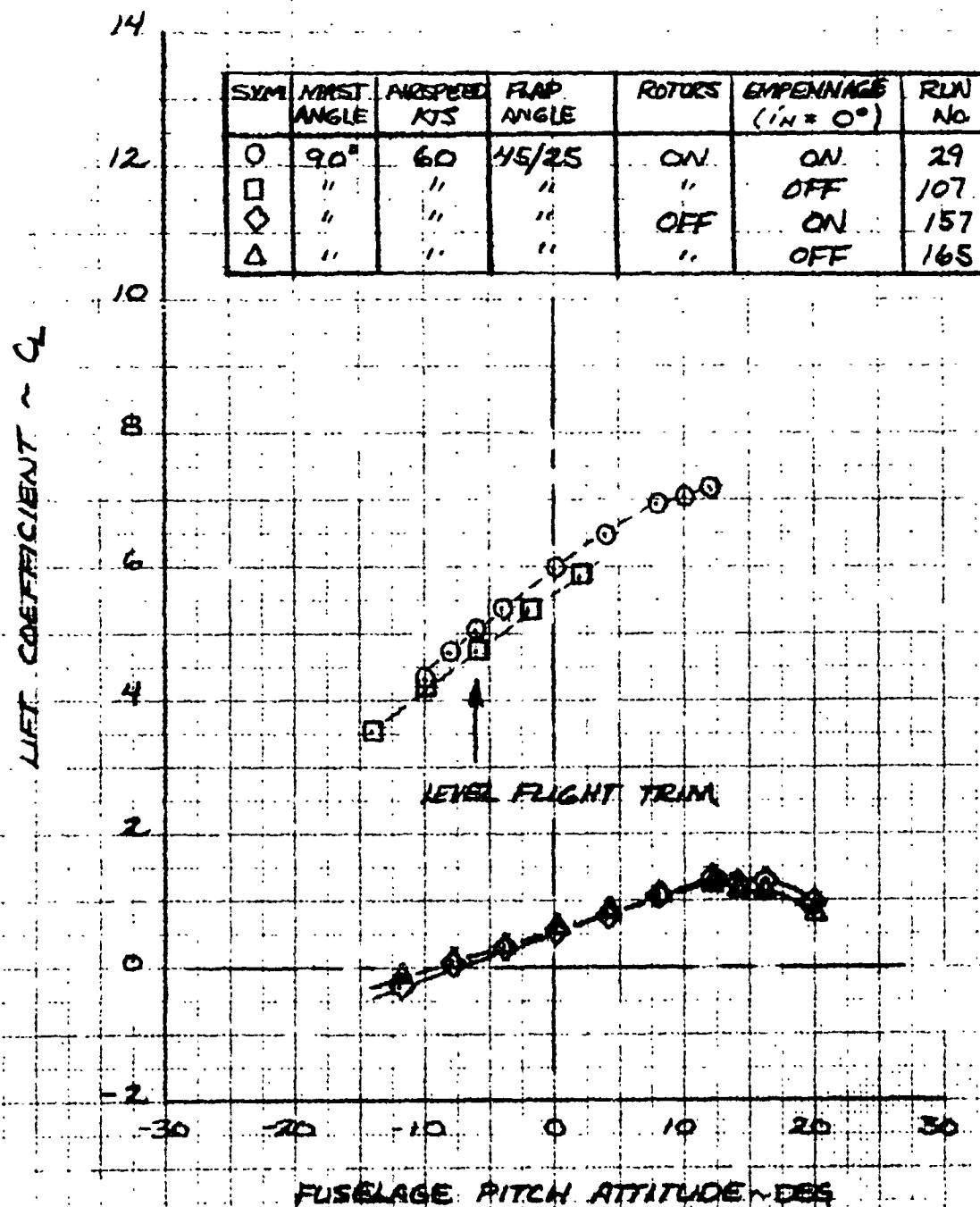
Figure V-43. Lift Coefficient versus Fuselage Angle
of Attack, Mast Angle 90°, Airspeed 60 knots.

ROTOR AND AIRFRAME LIFT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE ($i_N = 0°$)	RUN No
○	90°	80	45/25	ON	ON	32
□	"	"	"	"	OFF	109
◇	"	"	"	OFF	ON	159
△	"	"	"	"	OFF	165

LEVEL FLIGHT TRIM

LIFT COEFFICIENT ~ C_L

FUSELAGE PITCH ATTITUDE ~ DEG
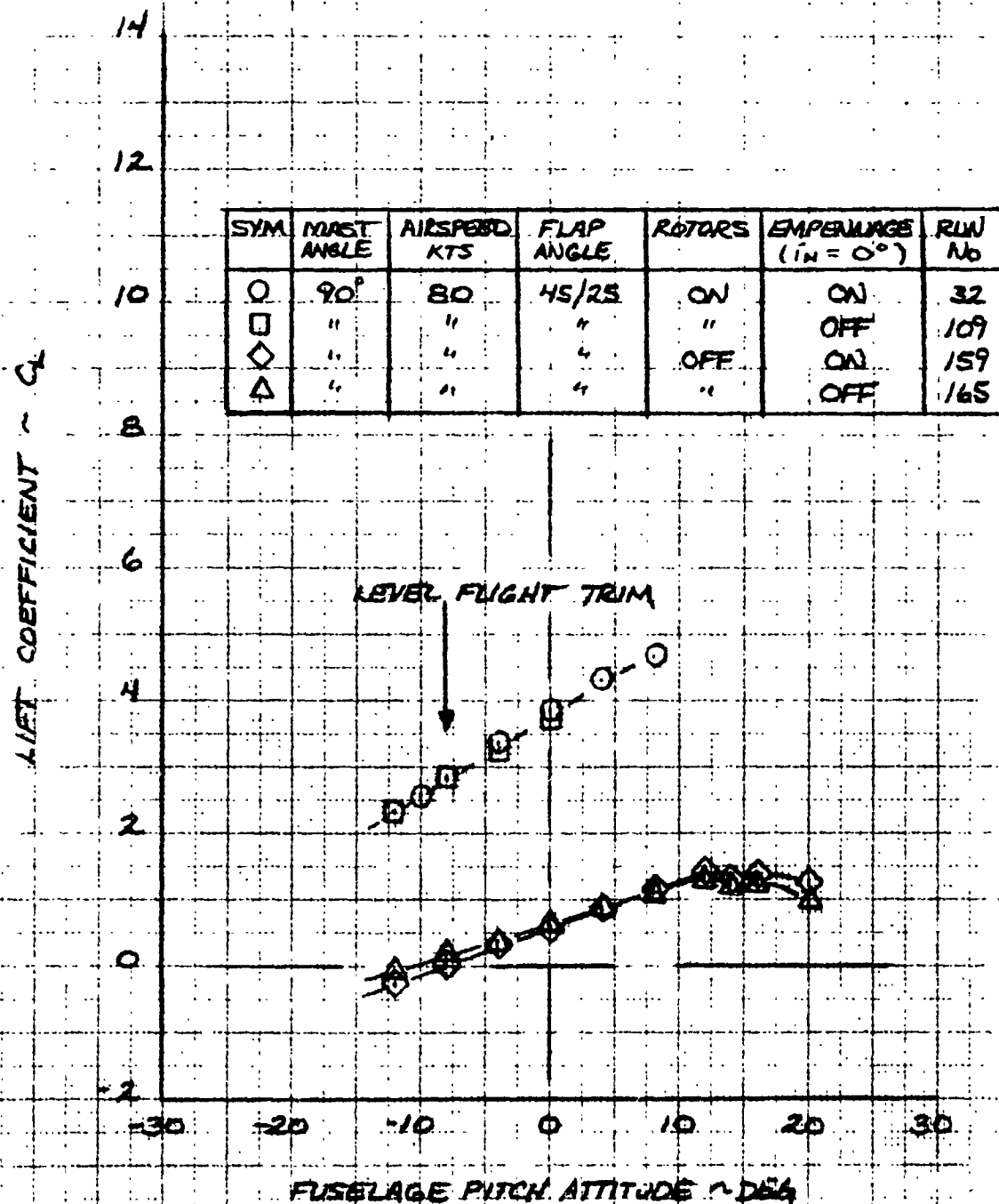
Figure V-44. Lift Coefficient versus Fuselage Angle
of Attack, Mast Angle 90°, Airspeed 80 knots.

ROTOR AND AIRFRAME PITCHING MOMENT
40 KN
(SEE FIG V-42 FOR SYMBOLS.)

LEVEL FLIGHT TRIM
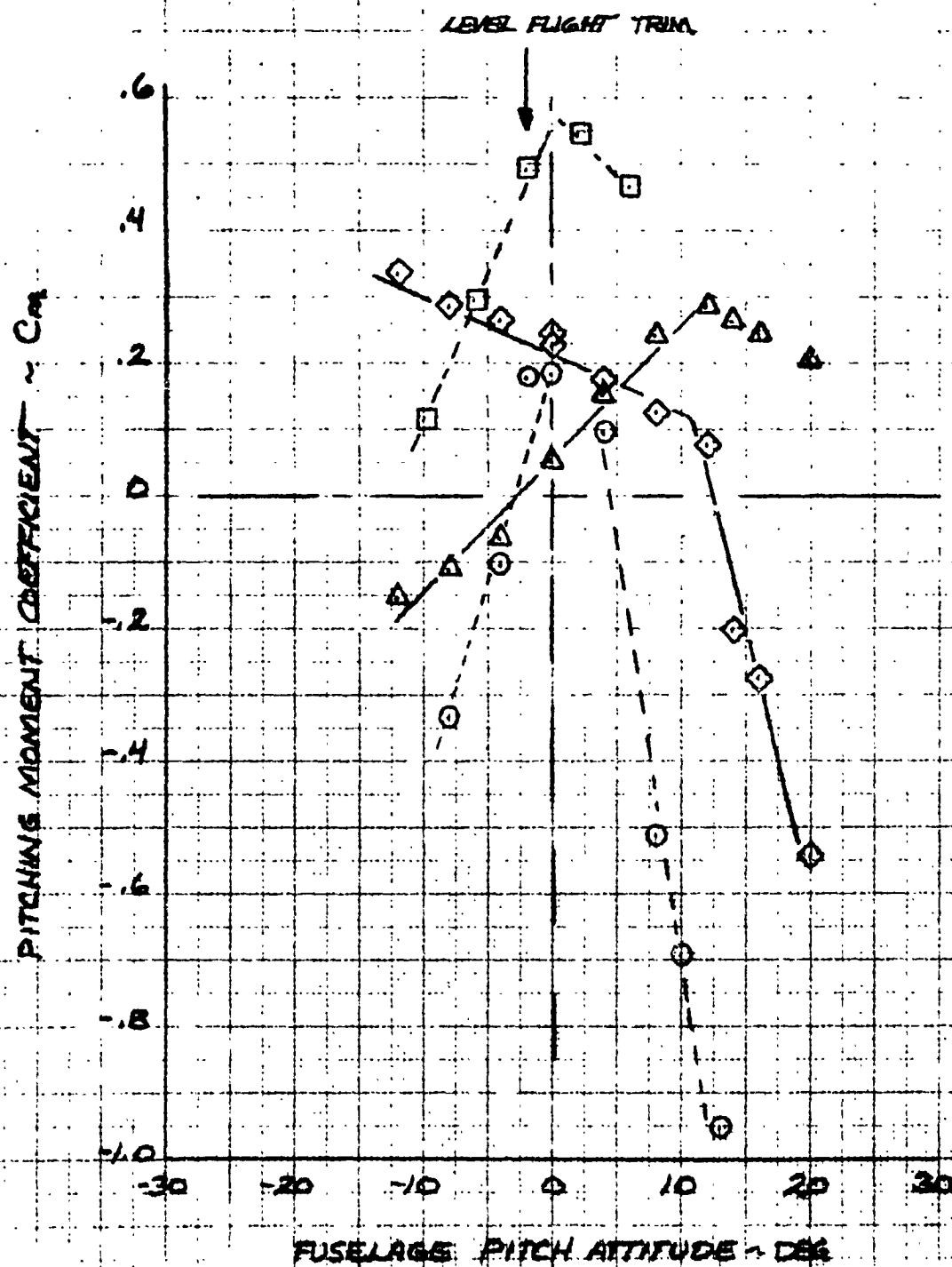
Figure V-45.   Pitching Moment versus Fuselage Angle of
Attack, Mast Angle 90°, Airspeed 40 knots.

ROTOR AND AIRFRAME PITCHING MOMENT
60 KN
(SEE FIG V-43 FOR SYMBOLS)
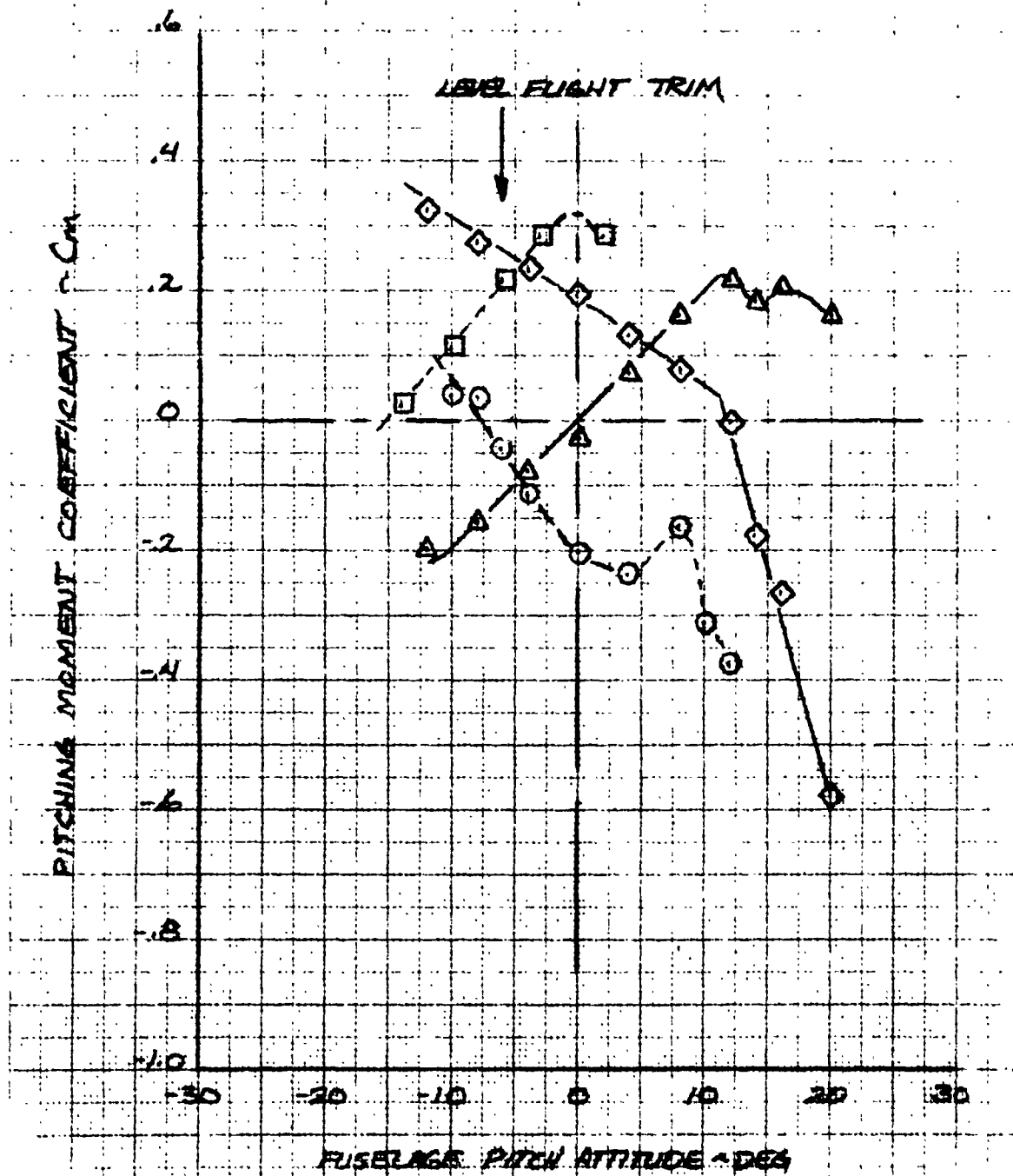
Figure V-46.   Pitching Moment versus Fuselage Angle of
Attack, Mast Angle 90°, Airspeed 60 knots.

ROTOR AND AIRFRAME PITCHING MOMENT
80 KN
(SEE FIG. V -44 FOR SYMBOLS)
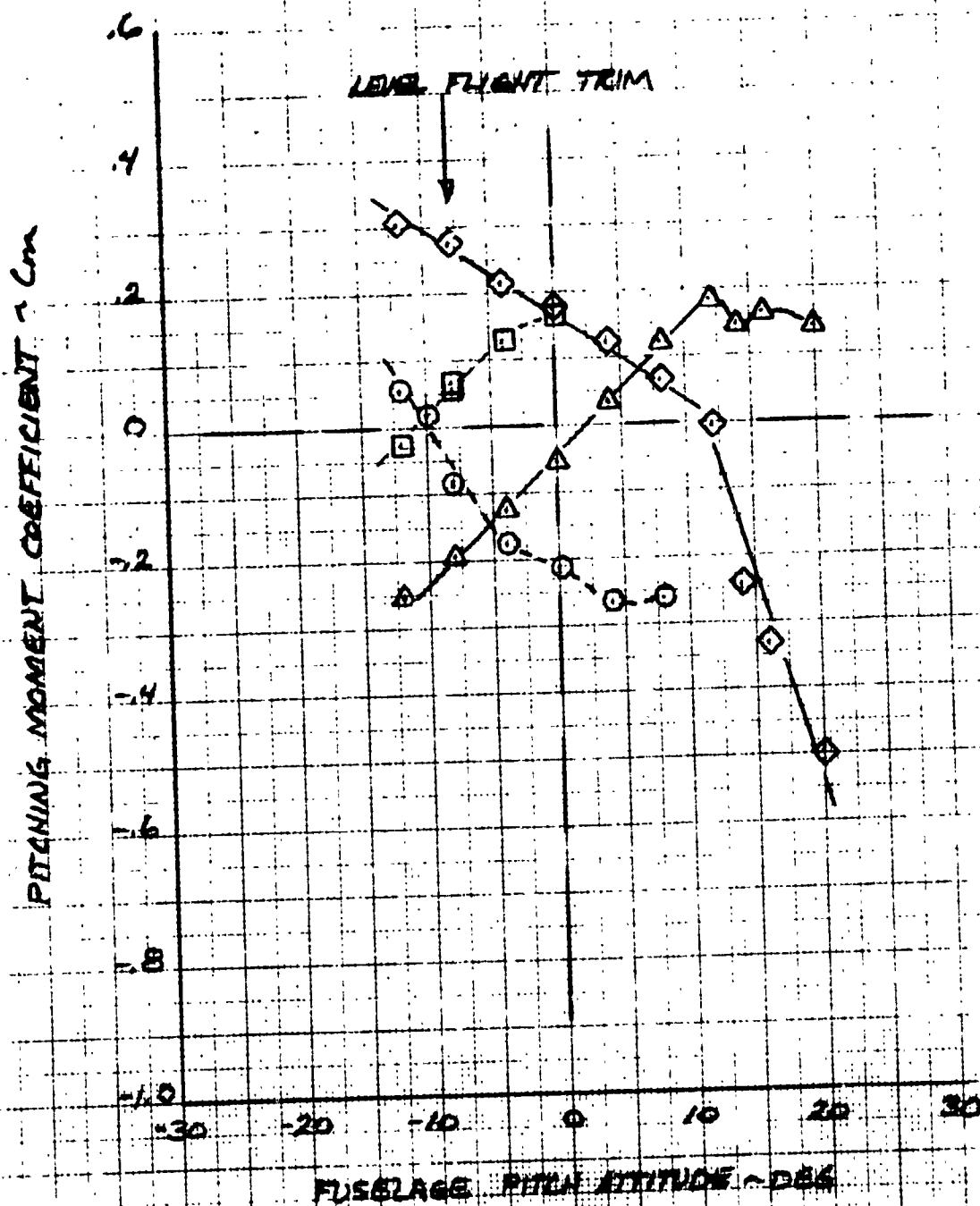
Figure V-47.   Pitching Moment versus Fuselage Angle of
Attack, Mast Angle 90°, Airspeed 80 knots.

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HELICOPTER** COMPANY

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN No
○	90°	40	45/25°	ON	ON	7
□	"	"	"	"	OFF	106
◇	"	"	"	OFF	ON	156
△	"	"	"	"	OFF	162
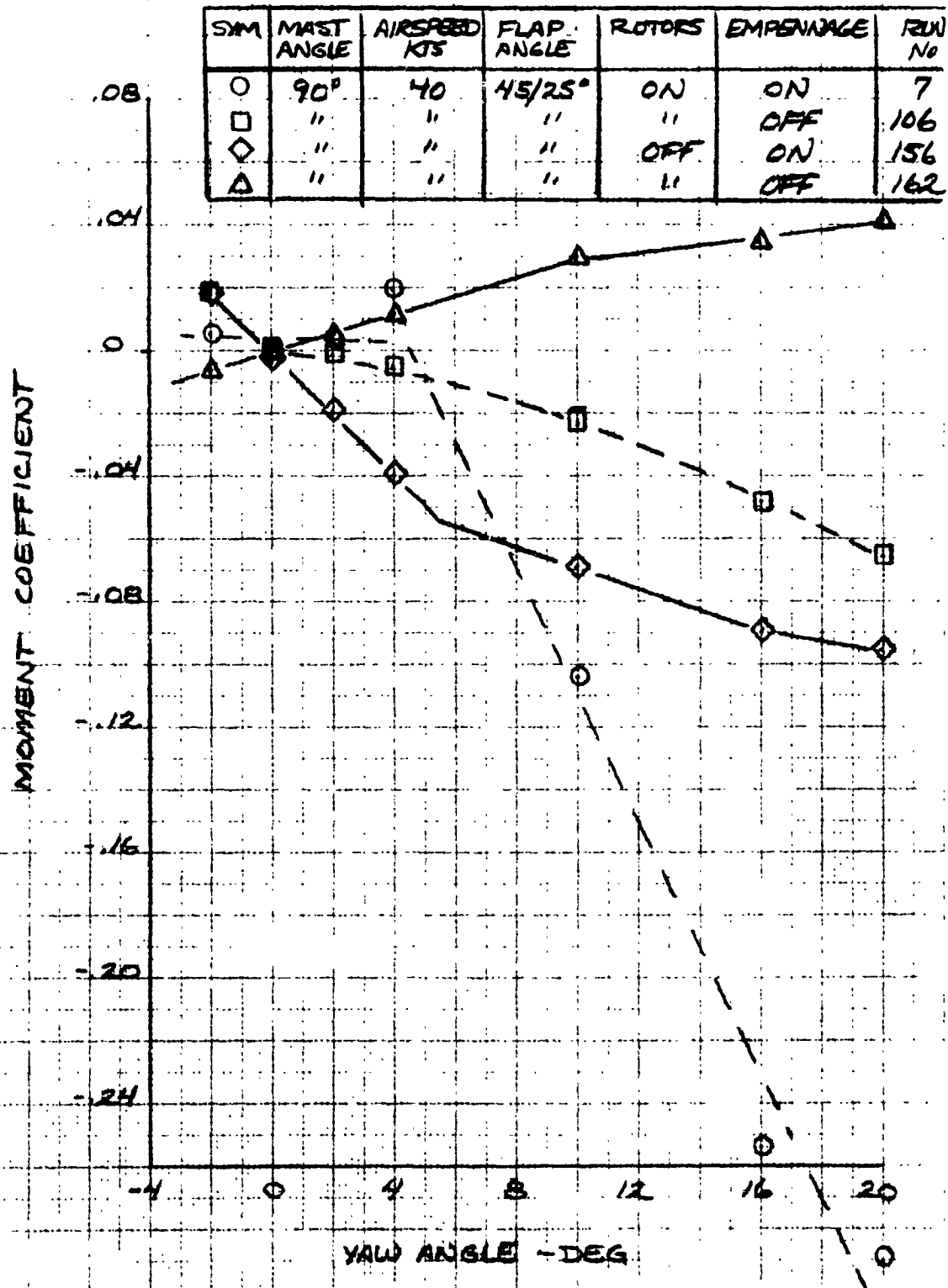
Figure V-48. Yaw Moment Coefficient versus Yaw Angle, Mast Angle 90°, Airspeed 40 knots.

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN No
○	90°	60	45/25°	ON	ON	30
□	"	"	"	"	OFF	108
◇	"	"	"	OFF	ON	158
△	"	"	"	"	OFF	164
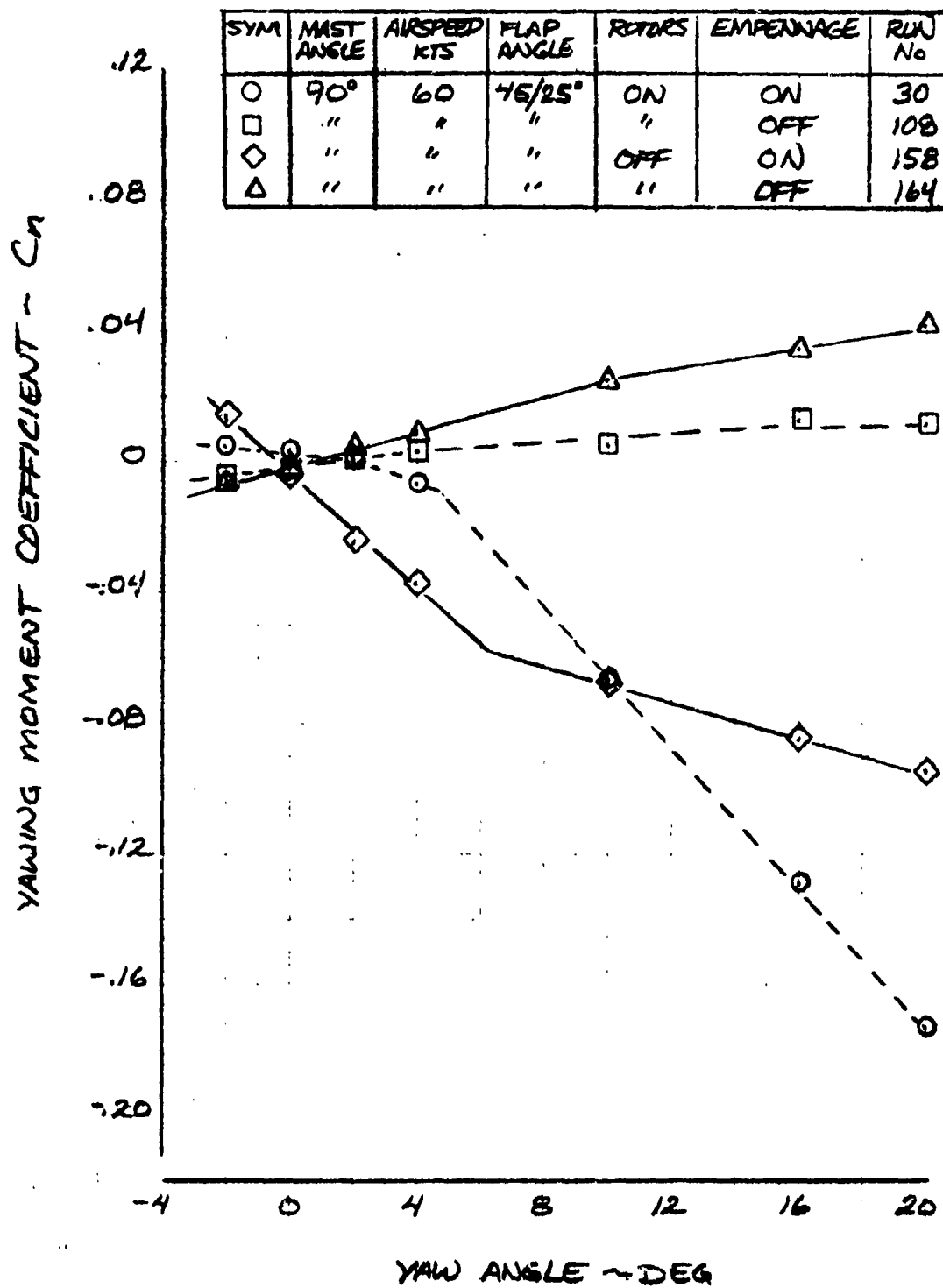
Figure V-49. Yaw Moment Coefficient versus Yaw Angle,
Mast Angle 90°, Airspeed 60 knots.

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN NO
O	90°	80	45/25°	ON	ON	93
□	''	''	''	''	OFF	110
◇	''	''	''	OFF	ON	16°
△	''	''	''	''	OFF	166
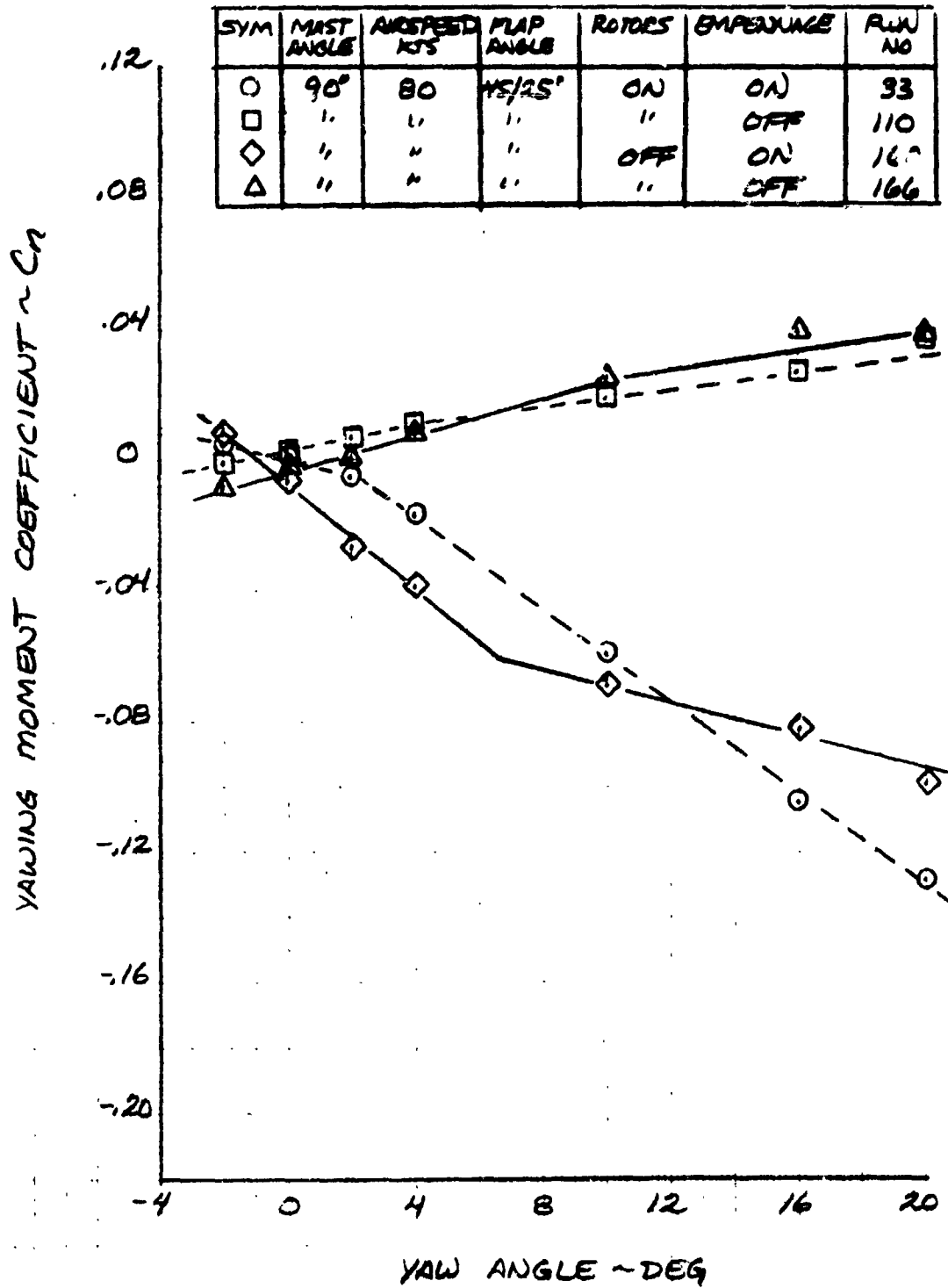
Figure V-50. Yaw Moment Coefficient versus Yaw Angle, Mast Angle 90°, Airspeed 80 knots.

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ROTOR AND AIRFRAME LIFT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE (iH = 0°)	RUN NO
○	75°	40	45/25°	ON	ON	70
□	"	"	"	"	OFF	99
◇	"	"	"	OFF	ON	143
△	"	"	"	"	OFF	167

LEVEL FLIGHT TRIM

LIFT COEFFICIENT ~ C_L

FUSELAGE PITCH ATTITUDE ~ DEG
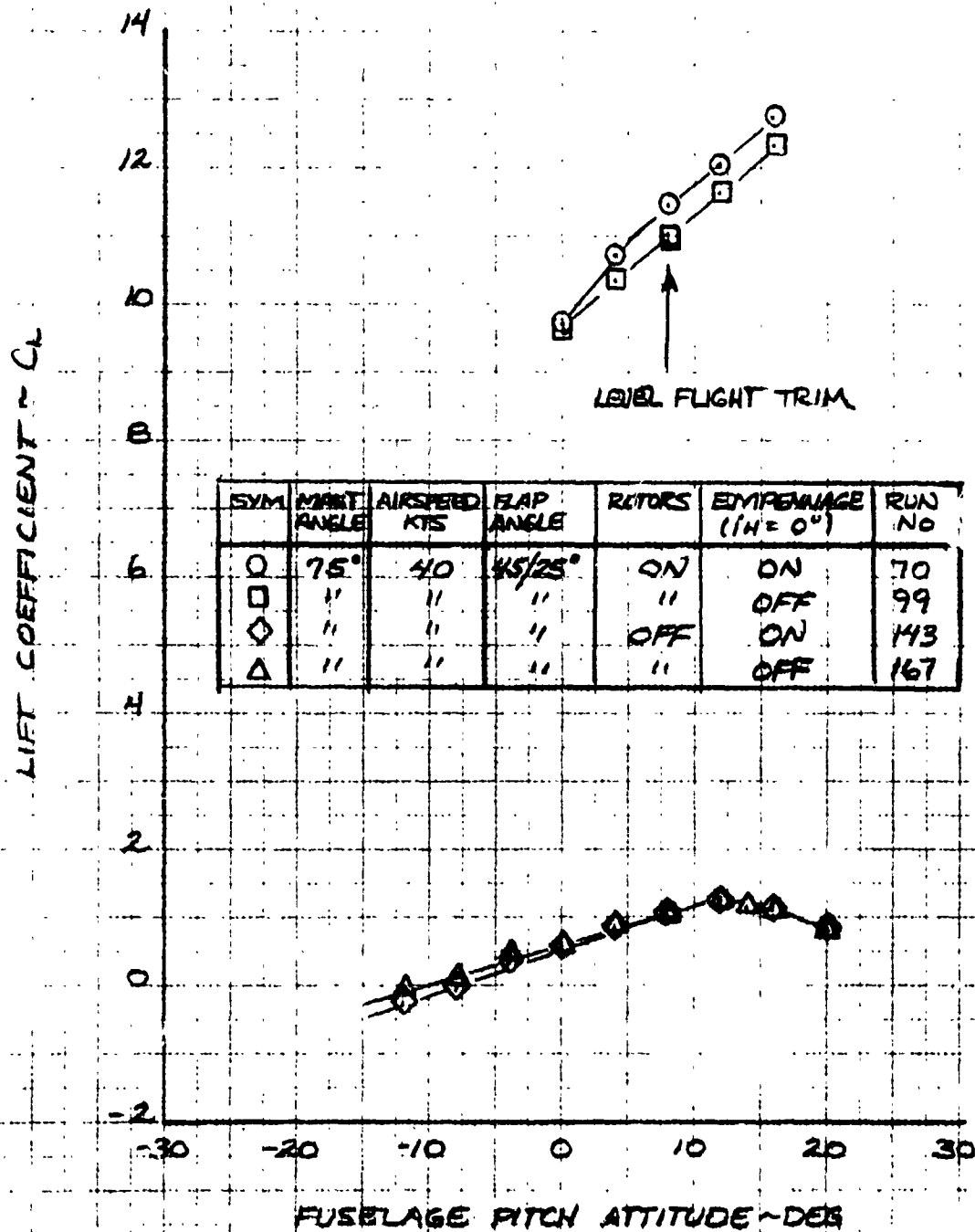
Figure V-51.   Lift Coefficient versus Fuselage Angle of
Attack, Mast Angle 75°, Airspeed 40 knots.

ROTOR AND AIRFRAME LIFT

SYM	MAST ANGLE	AIRSPEED	FLAP ANGLE	ROTORS	EMPENNAGE	RUN No
○	75°	60	45/25	ON	ON	73
□	"	"	"	"	OFF	97
◇	"	"	"	OFF	ON	145
△	"	"	"	"	OFF	169

LEVEL FLIGHT TRIM
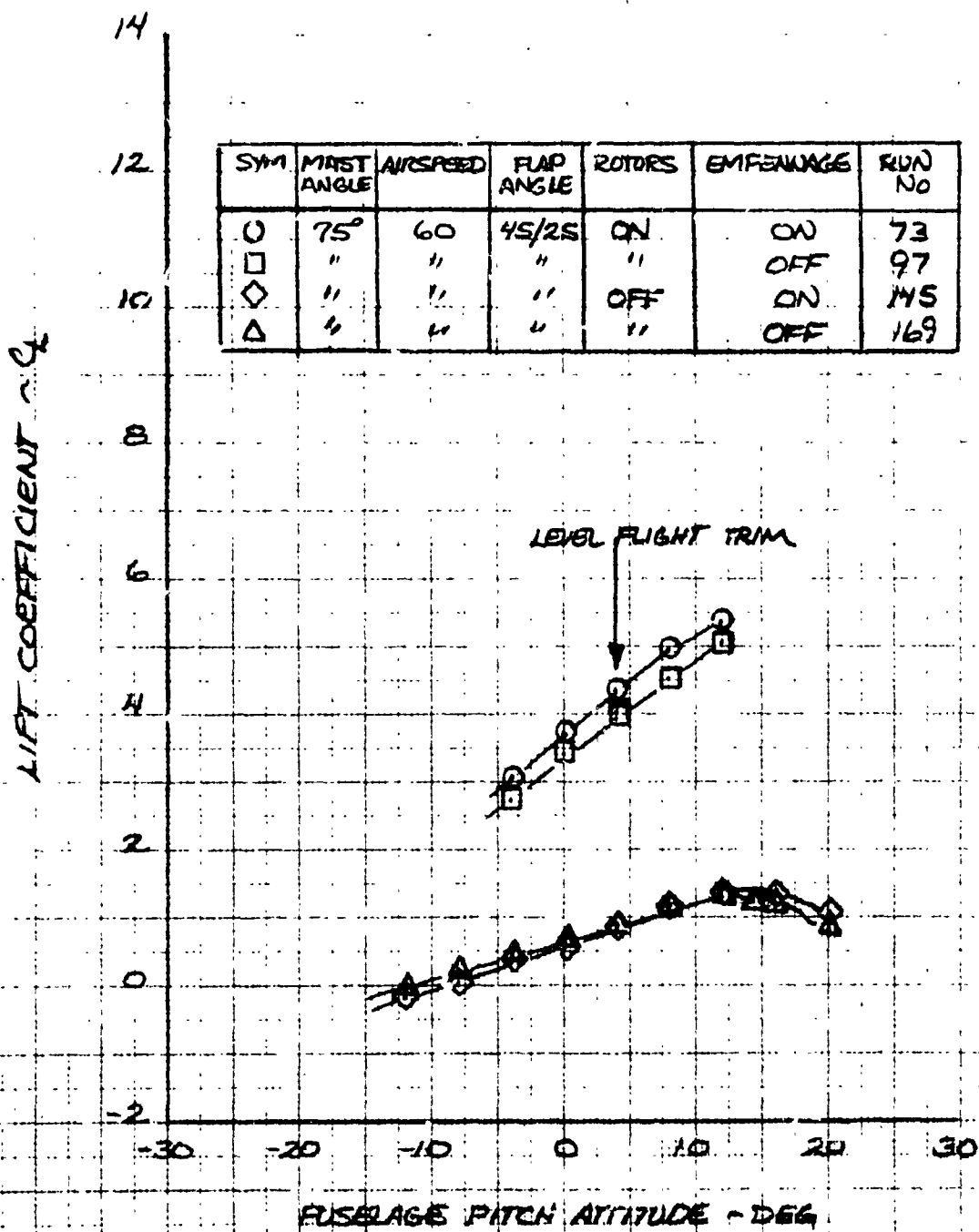
Figure V-52. Lift Coefficient versus Fuselage Angle of
Attack, Mast Angle 75°, Airspeed 60 knots.

ROTOR AND AIRFRAME LIFT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN NO
○	75°	80	45/25°	ON	ON	76
□	"	"	"	"	OFF	95
◇	"	"	"	OFF	ON	147
△	"	"	"	"	OFF	171

LEVEL FLIGHT TRIM
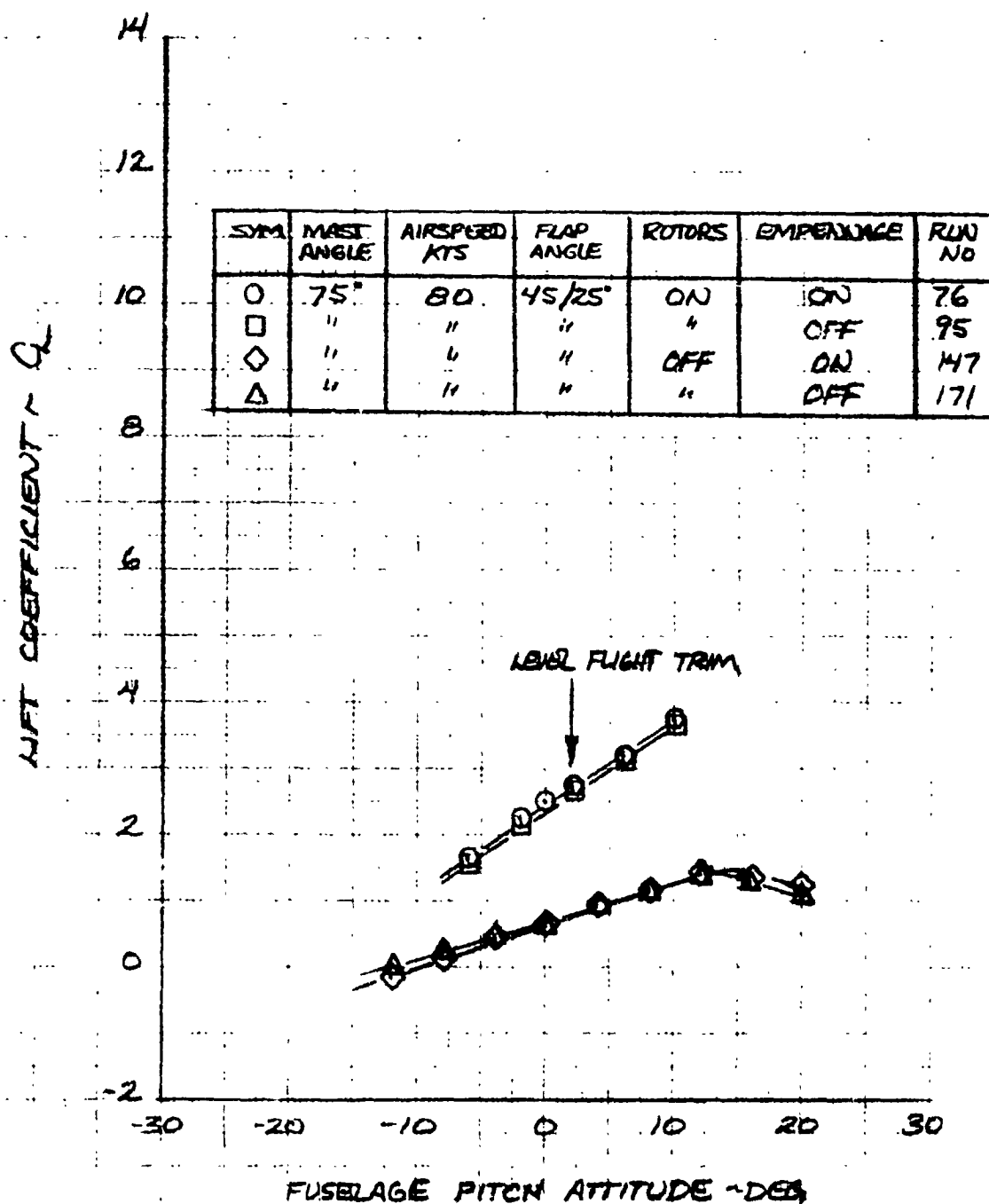
Figure V-53. Lift Coefficient versus Angle of Attack,
Mast Angle 75°, Airspeed 80 knots.

ROTOR AND AIRFRAME PITCHING MOMENT
40 KN

(SEE FIG V-51 FOR SYMBOLS)
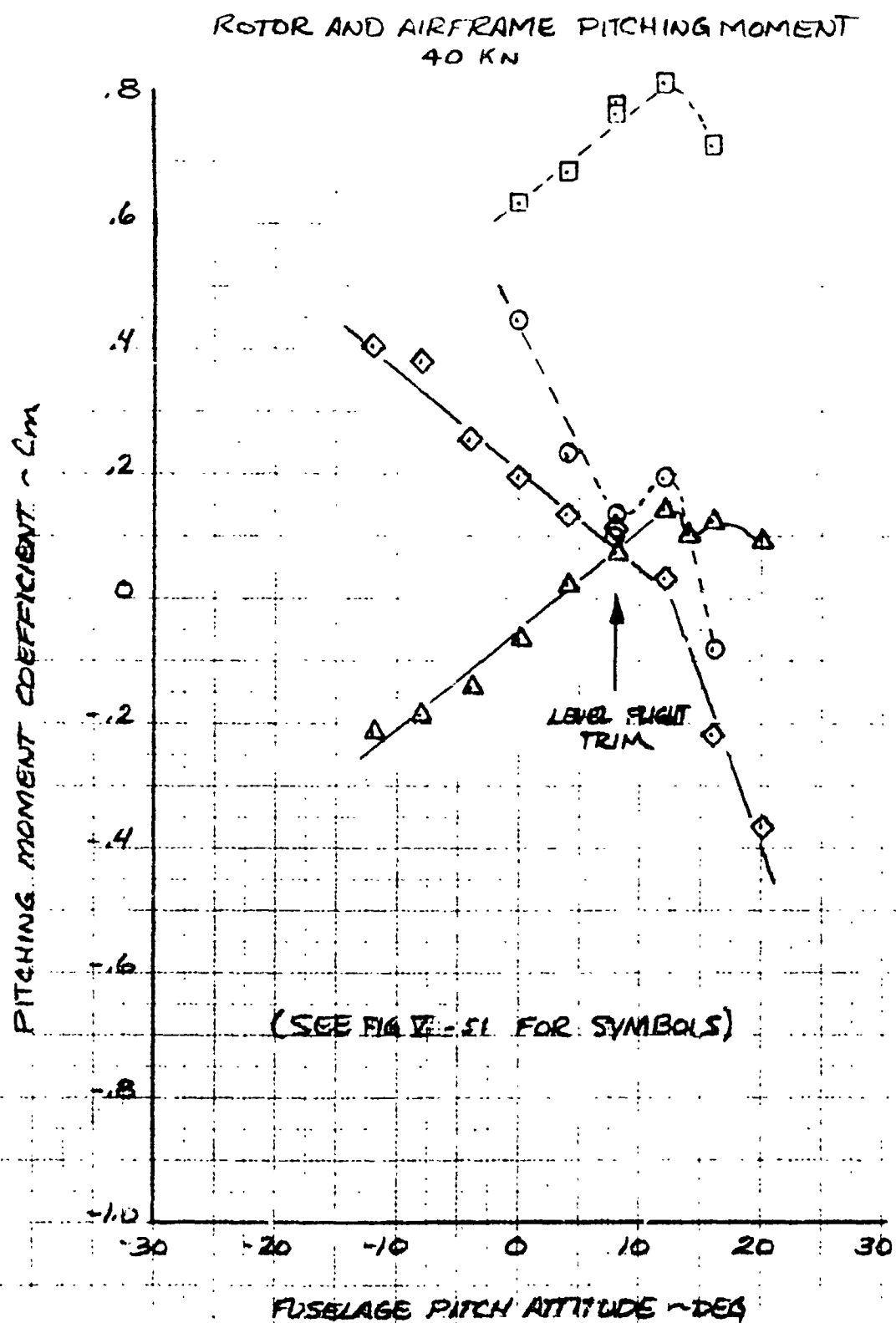
Figure V-54. Pitching Moment versus Fuselage Angle of Attack, Mast Angle 75°, Airspeed 40 knots.

ROTOR AND AIRFRAME PITCHING MOMENT

(SEE FIG V -52 FOR SYMBOLS)
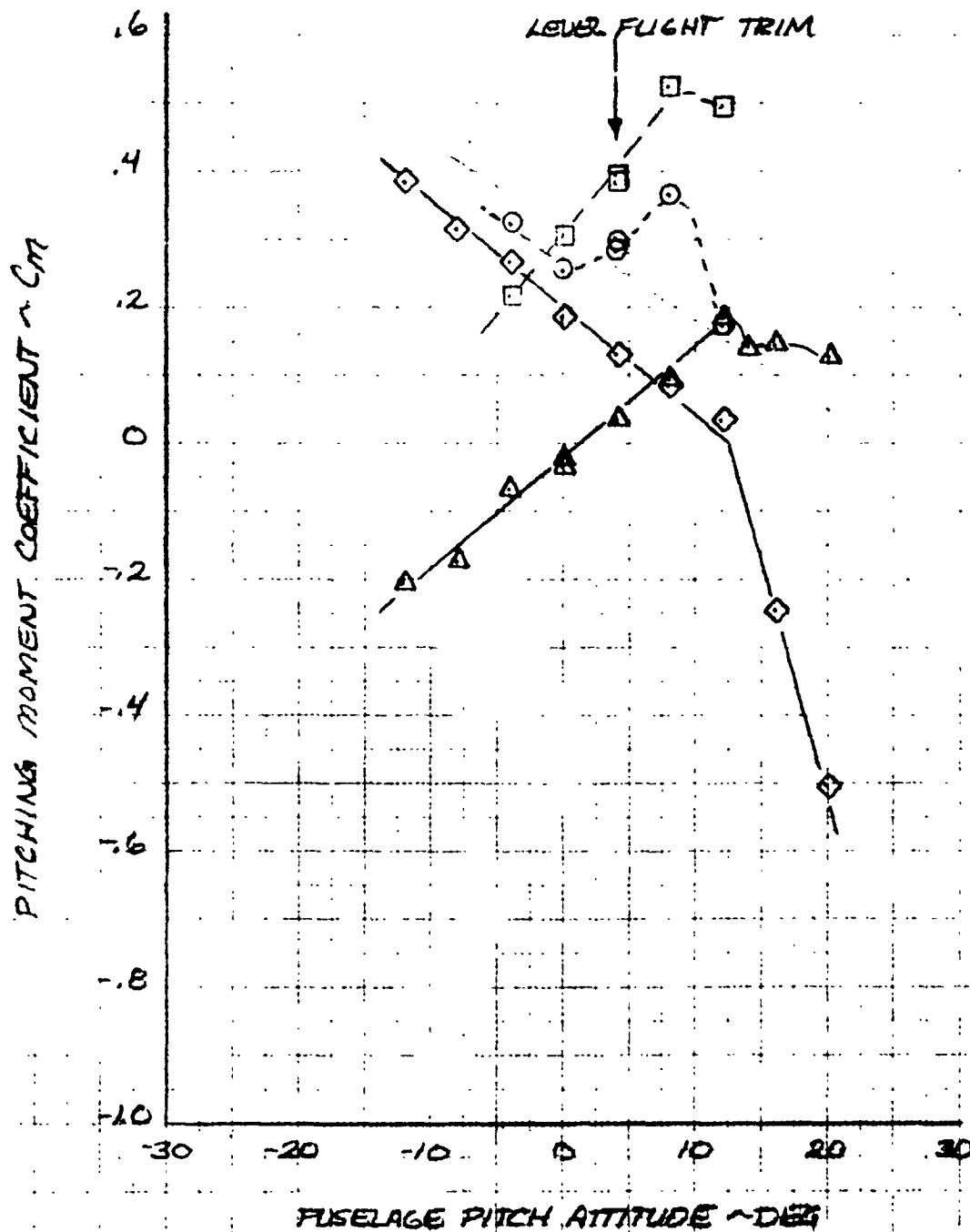
Figure V-55. Pitching Moment versus Fuselage Angle of
Attack, Mast Angle 75°, Airspeed 60 knots.

ROTOR AND AIRFRAME PITCHING MOMENT

(SEE FIG Ⅴ-53 FOR SYMBOLS)
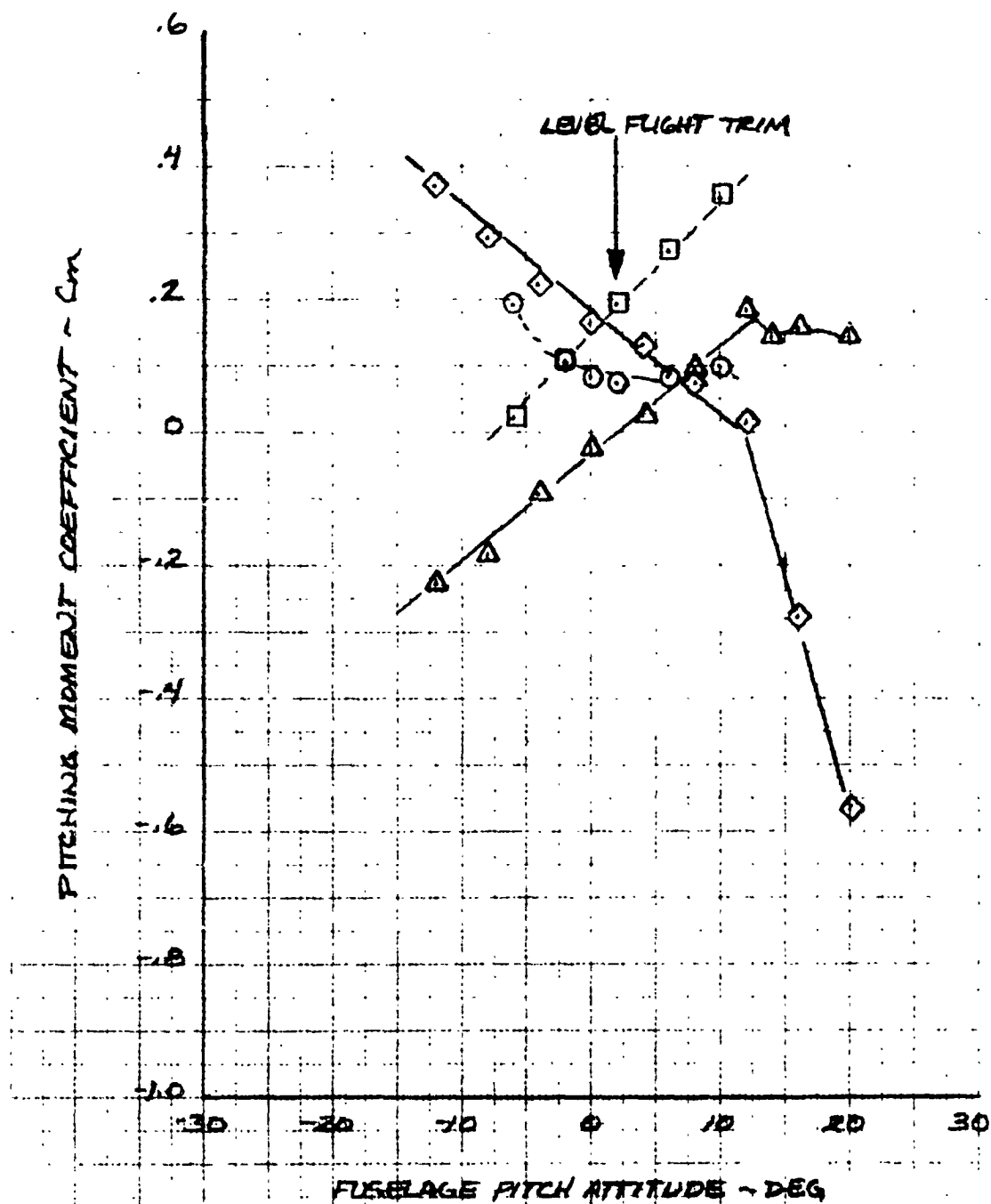
Figure V-56. Pitching Moment versus Fuselage Angle of Attack, Mast Angle 75°, Airspeed 80 knots.

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN No
○	75°	40	45/25°	ON	ON	71
□	"	"	"	"	OFF	100
◇	"	"	"	OFF	ON	144
△	"	"	"	"	OFF	168
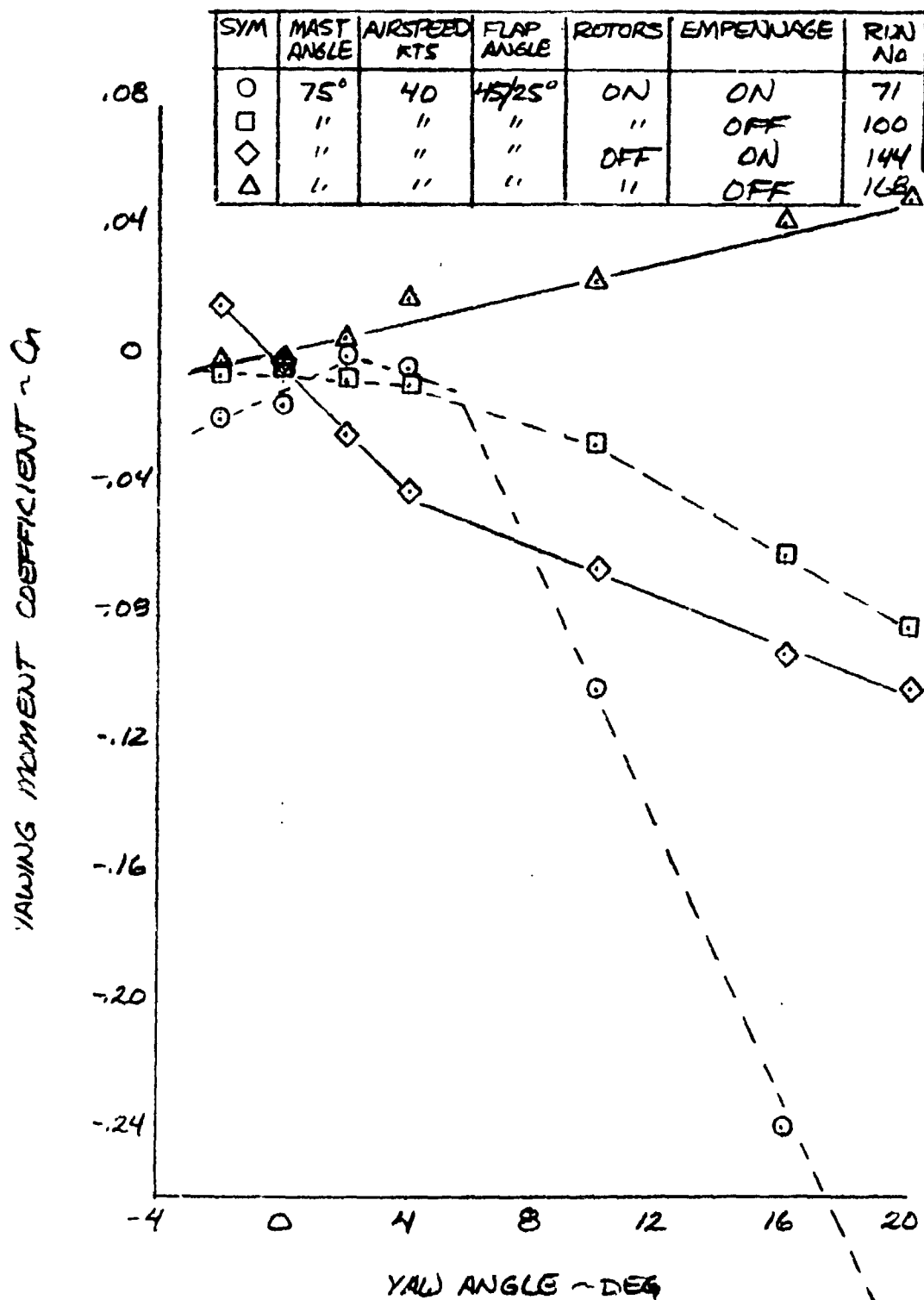
Figure V-57. Yawing Moment Coefficient versus Yaw Angle, Mast Angle 75°, Airspeed 40 knots.

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN No
○	75°	60	45/25°	ON	ON	74
□	,,	,,	,,	,,	OFF	98
◇	,,	,,	,,	OFF	ON	146
△	,,	,,	,,	,,	OFF	170
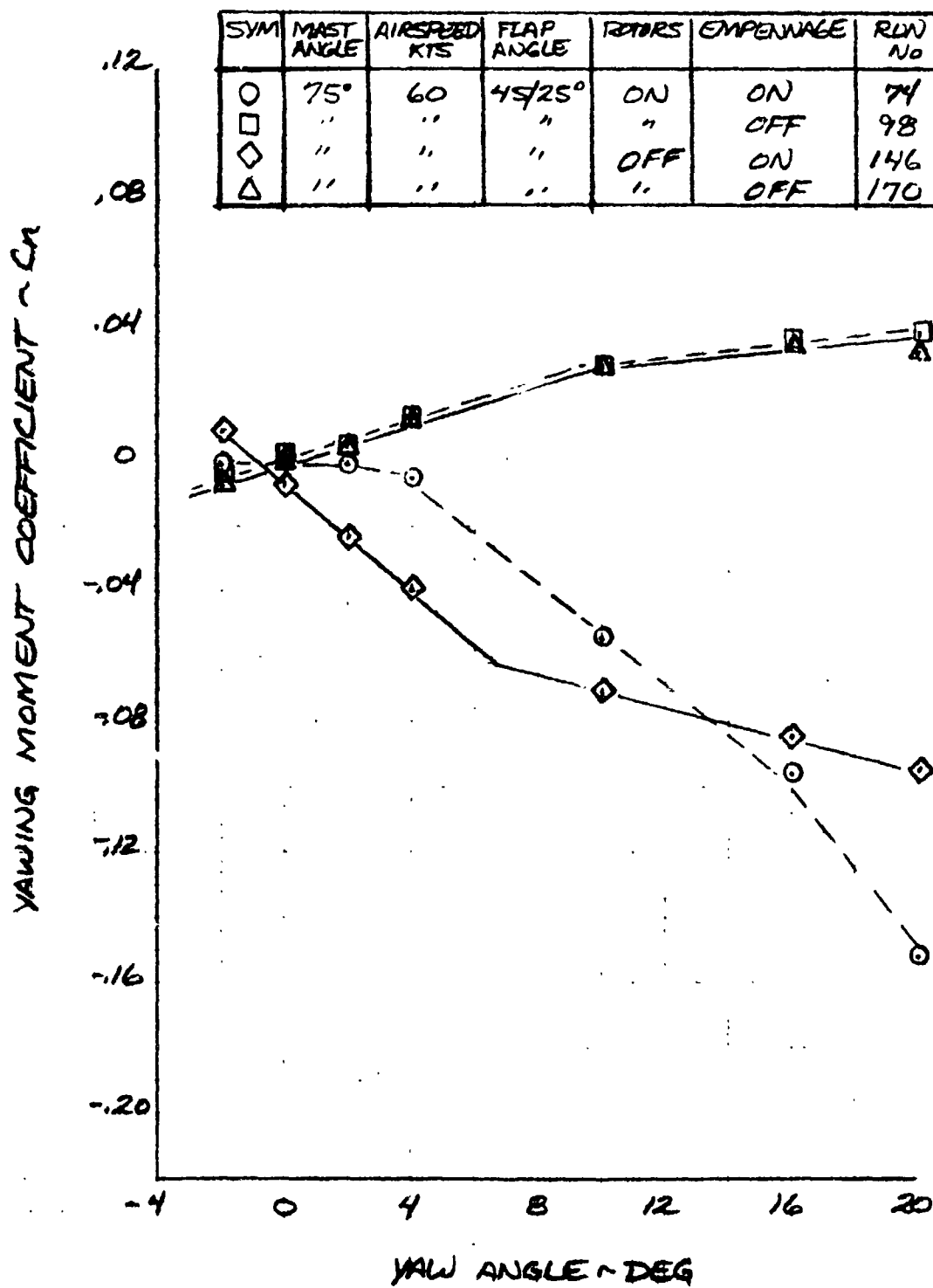
Figure V-58.   Yawing Moment versus Yaw Angle, Mast Angle 75°
Airspeed 60 knots.

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN NO
○	75°	80	45/25°	ON	ON	77
□	"	"	"	"	OFF	96
◇	"	"	"	OFF	ON	148
△	"	"	"	"	OFF	172
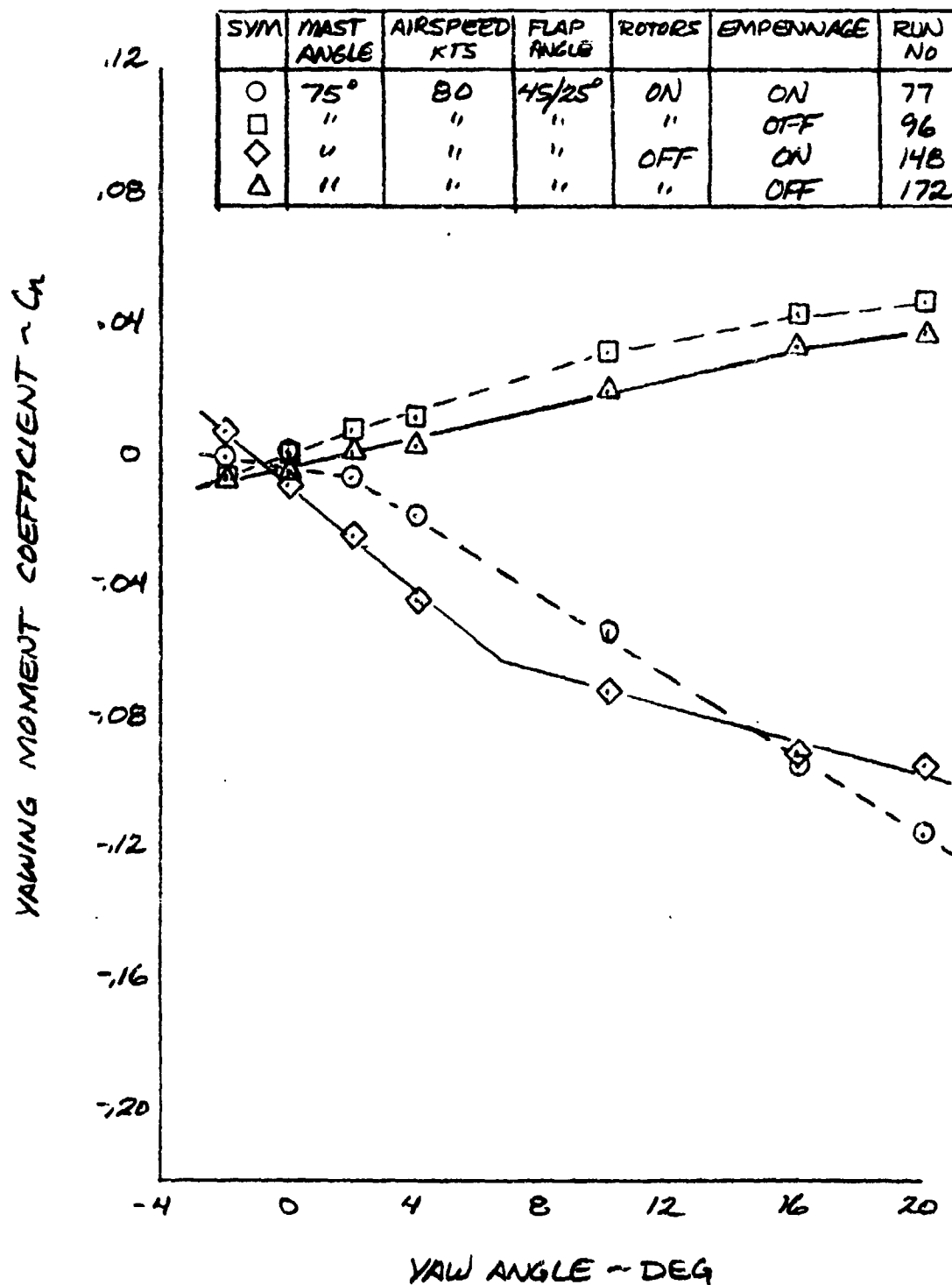
Figure V-59.   Yawing Moment Coefficient versus Yaw Angle,
Mast Angle 75°, Airspeed 80 knots.

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ROTOR AND AIRFRAME LIFT
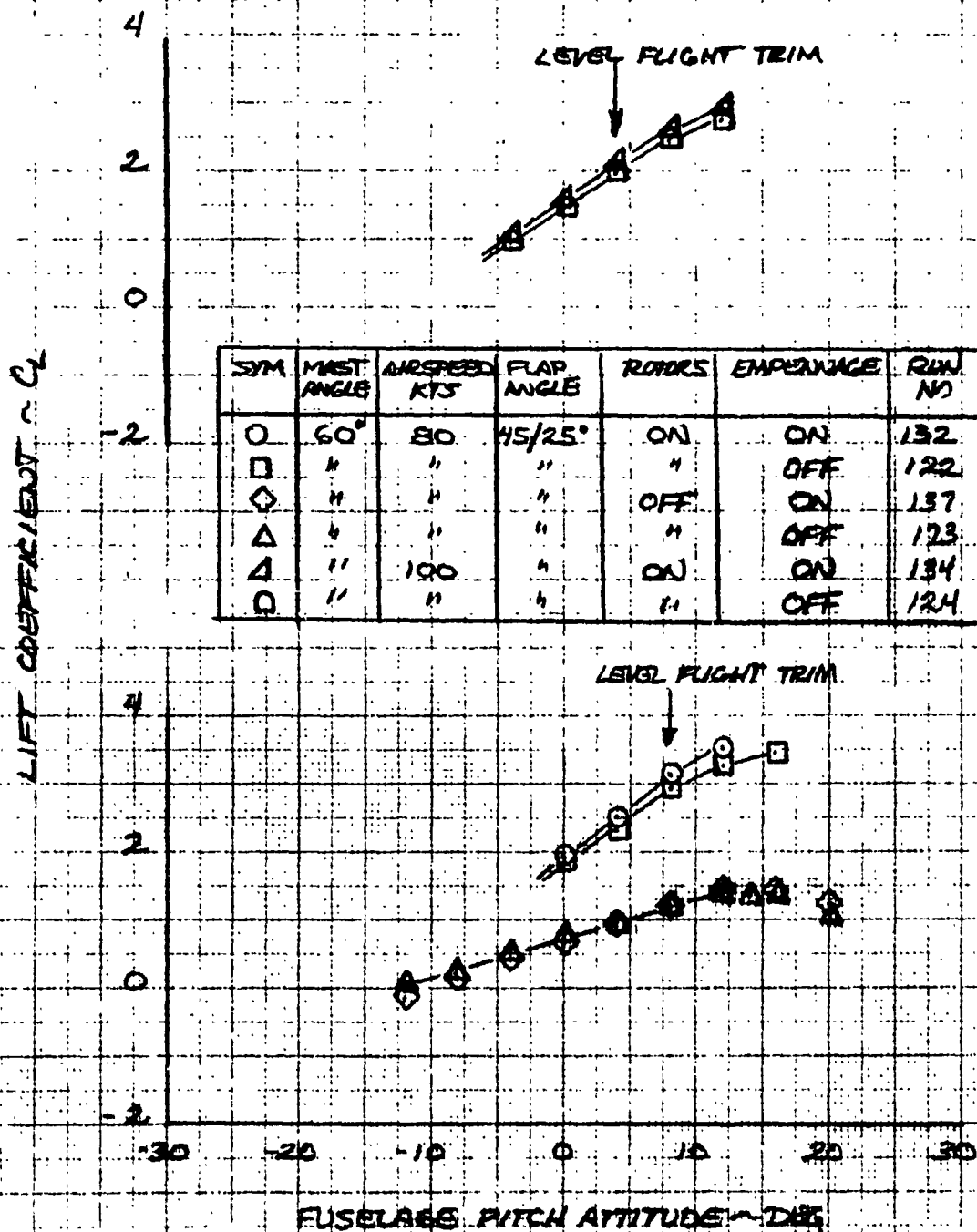
Figure V-60. Lift Coefficient versus Fuselage Angle of
Attack, Mast Angle 60°.

ROTOR AND AIRFRAME LIFT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE (iₕ = 0°)	RUN NO
○	30°	80	45/25°	ON	ON	129
□	"	"	"	"	OFF	126
◇	"	"	"	OFF	ON	141
△	"	"	"	"	OFF	175
◮	"	100	"	ON	ON	130
⬡	"	"	"	"	OFF	127
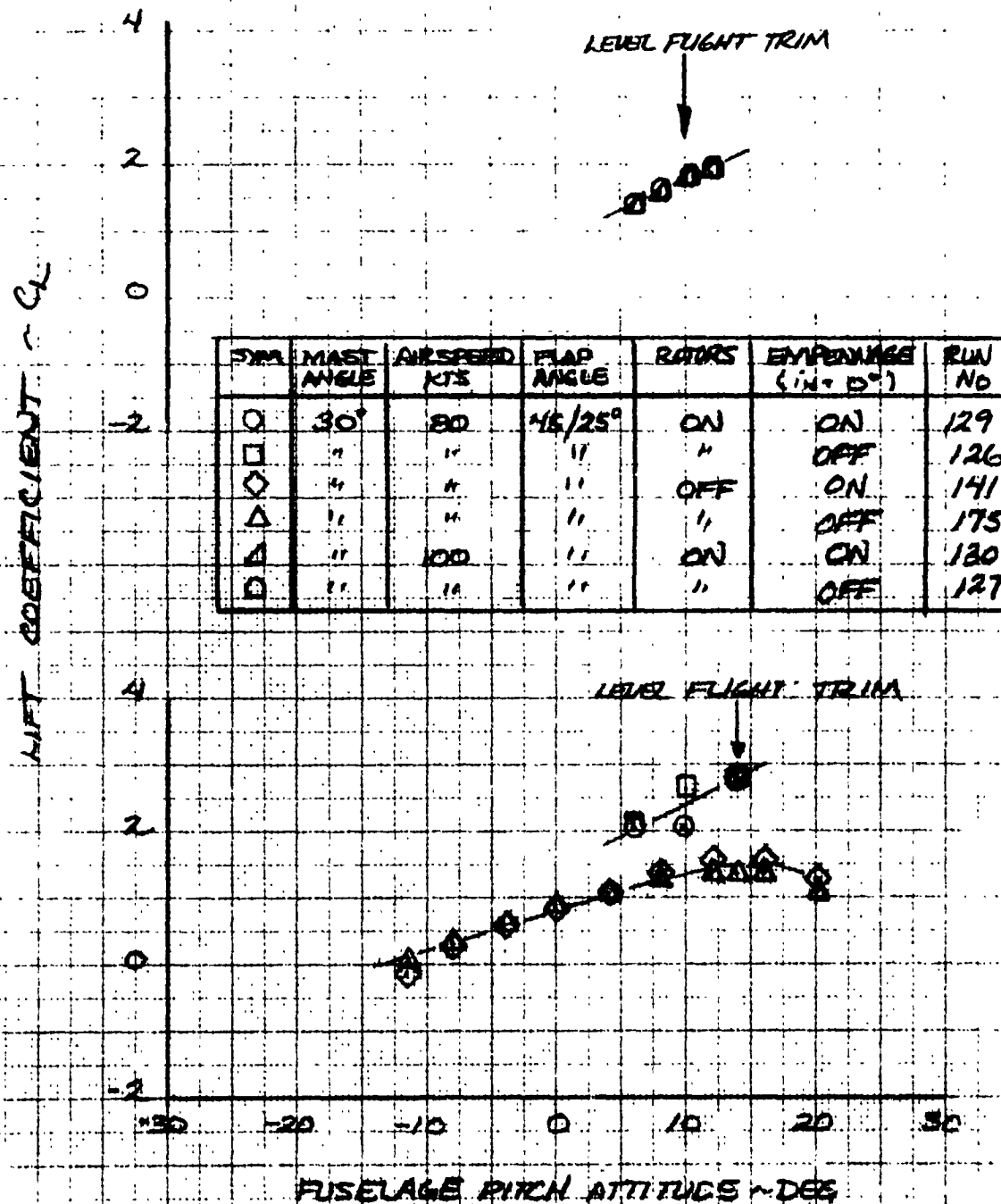
Figure V-61. Lift Coefficient versus Fuselage Angle of
Attack, Mast Angle 30°.

ROTOR AND AIRFRAME PITCHING MOMENT

(SEE FIG V-60 FOR SYMBOLS)
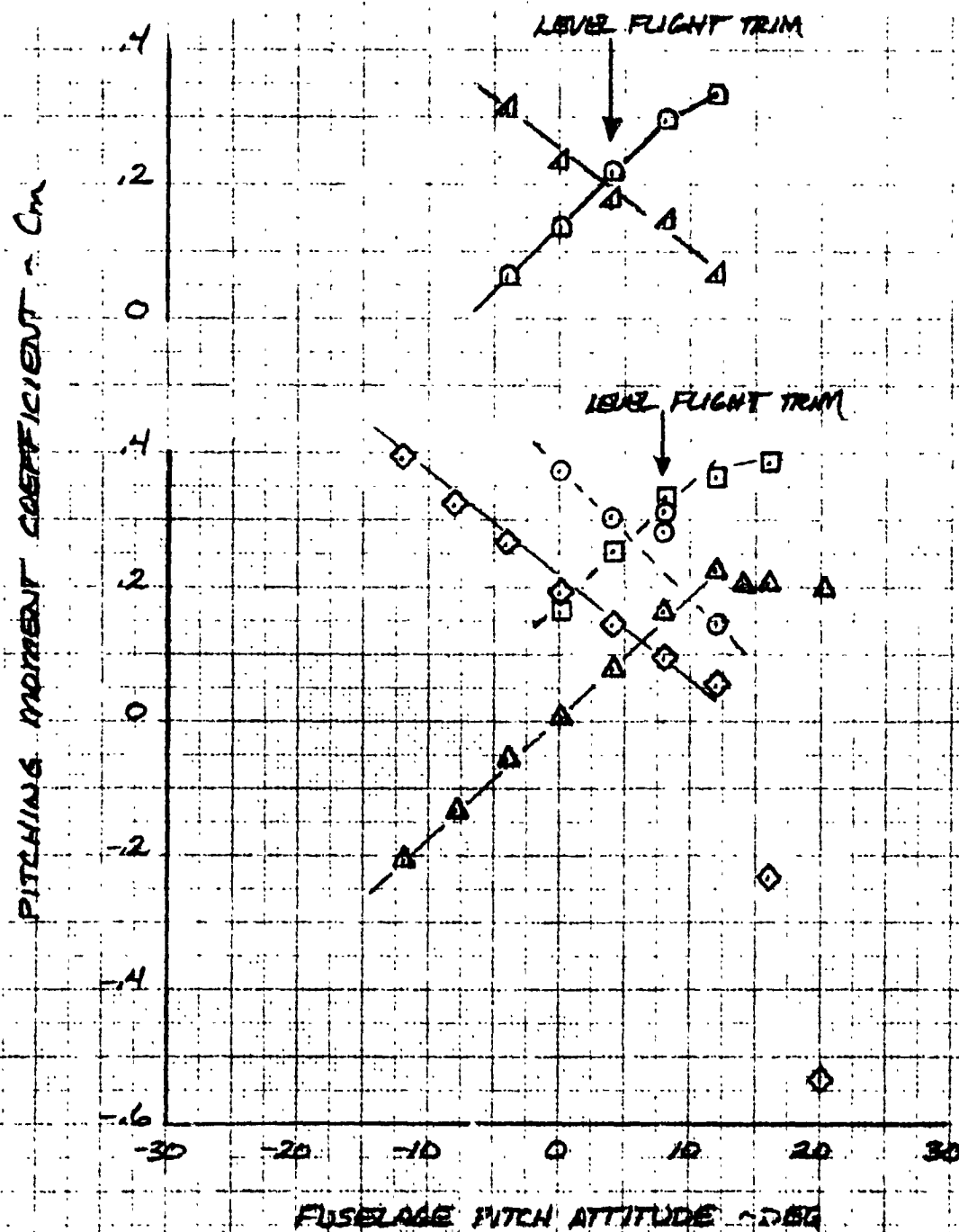
Figure V-62.   Pitching Moment versus Fuselage Angle
of Attack, Mast Angle 60°.

ROTOR AND AIRFRAME PITCHING MOMENT.

(SEE FIG V-61 FOR SYMBOLS)
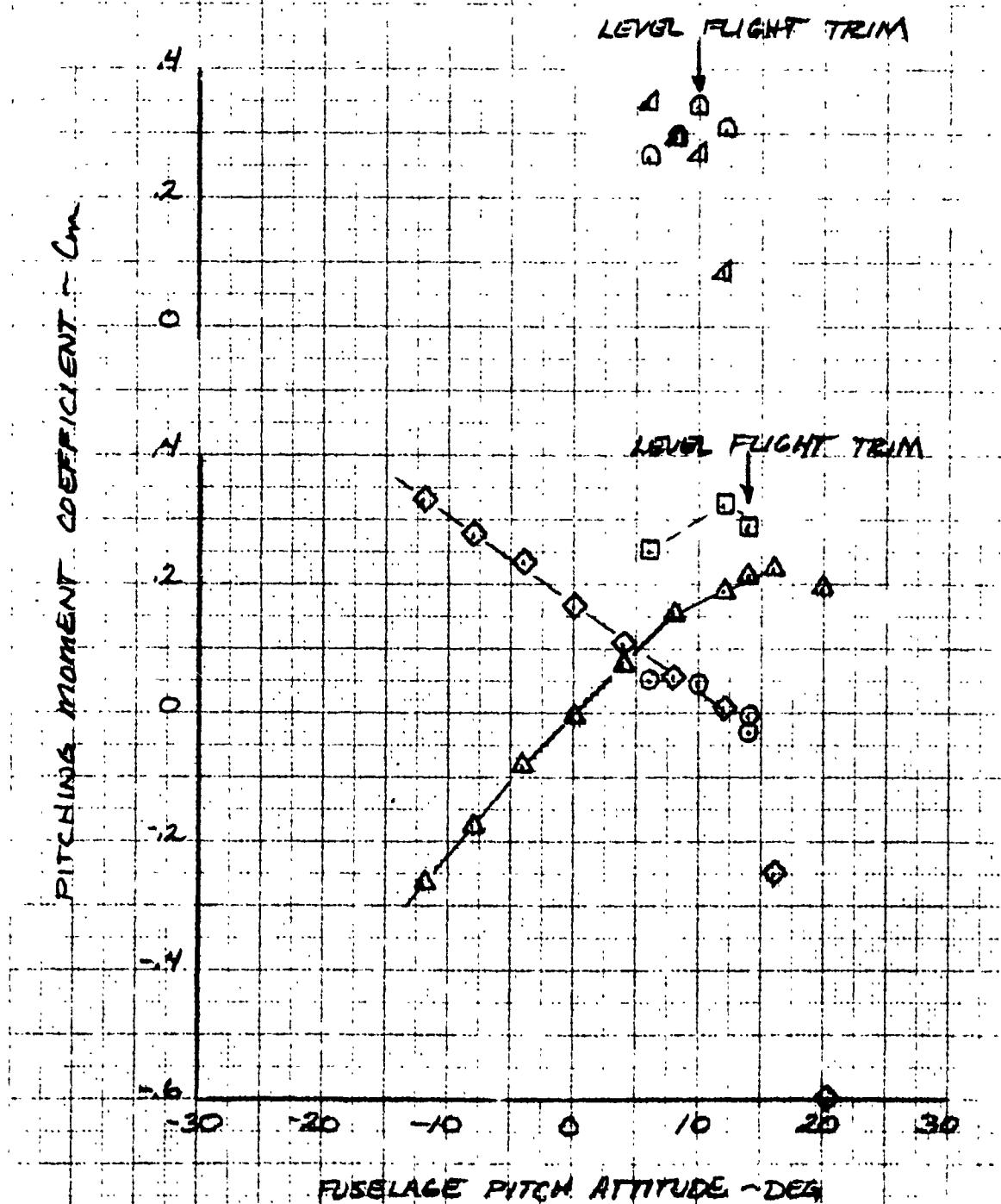
Figure V-63.   Pitching Moment versus Fuselage Angle
of Attack, Mast Angle 30°.

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN No
○	60°	80	45/25°	ON	ON	133
□	"	"	"	"	OFF	123
◇	"	"	"	OFF	ON	138
△	"	"	"	"	OFF	174
◿	"	100	"	ON	ON	135
◠	"	"	"	"	OFF	125
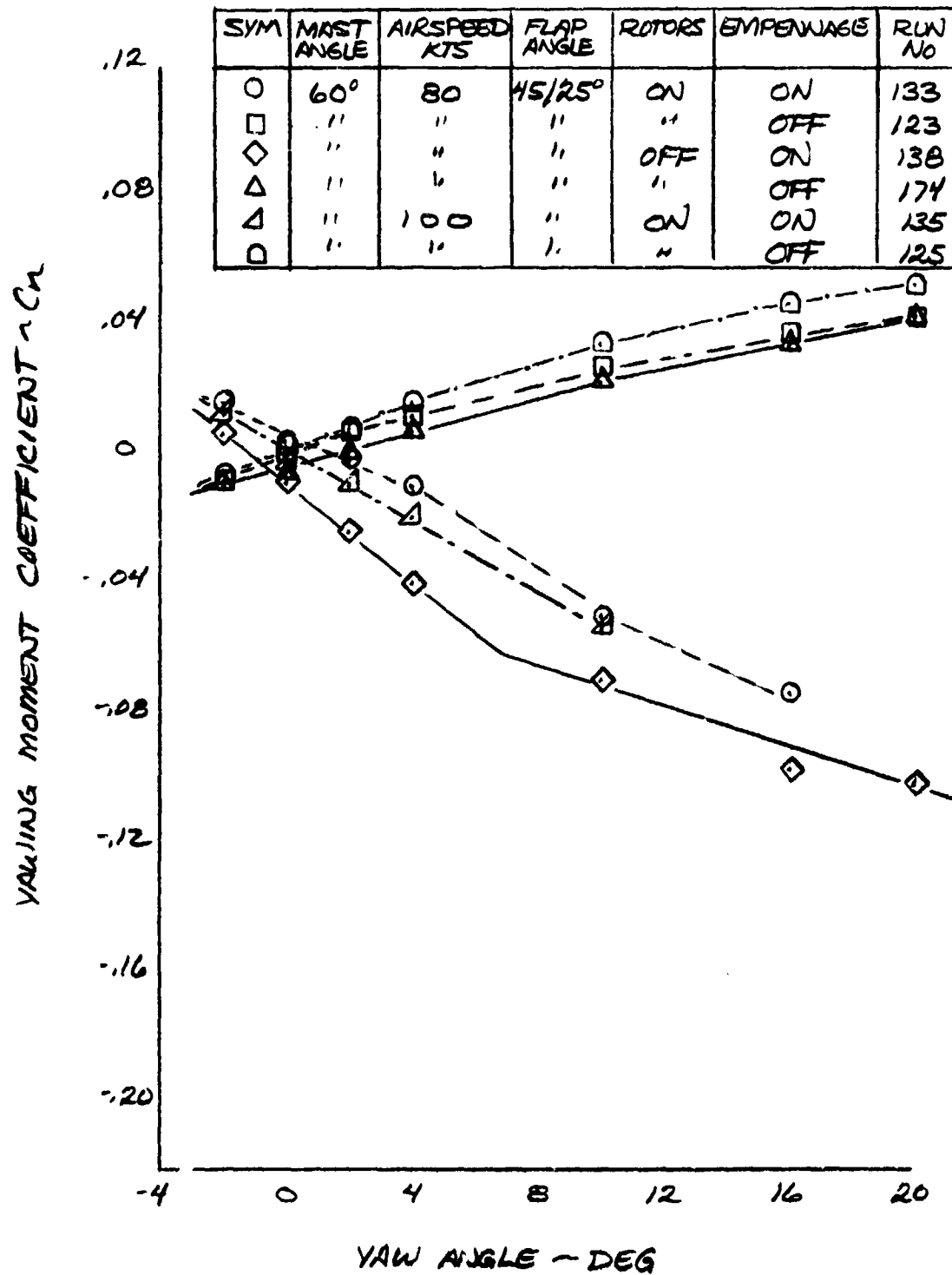
Figure V-64.   Yawing Moment Coefficient versus Yaw
Angle, Mast Angle 60°.

ROTOR AND AIRFRAME YAWING MOMENT

SYM	MAST ANGLE	AIRSPEED KTS	FLAP ANGLE	ROTORS	EMPENNAGE	RUN NO
◇	30°	80	45/25"	OFF	ON	142
△	"	"	"	"	OFF	126
◁	"	100	"	ON	ON	131
▢	"	"	"	"	OFF	128
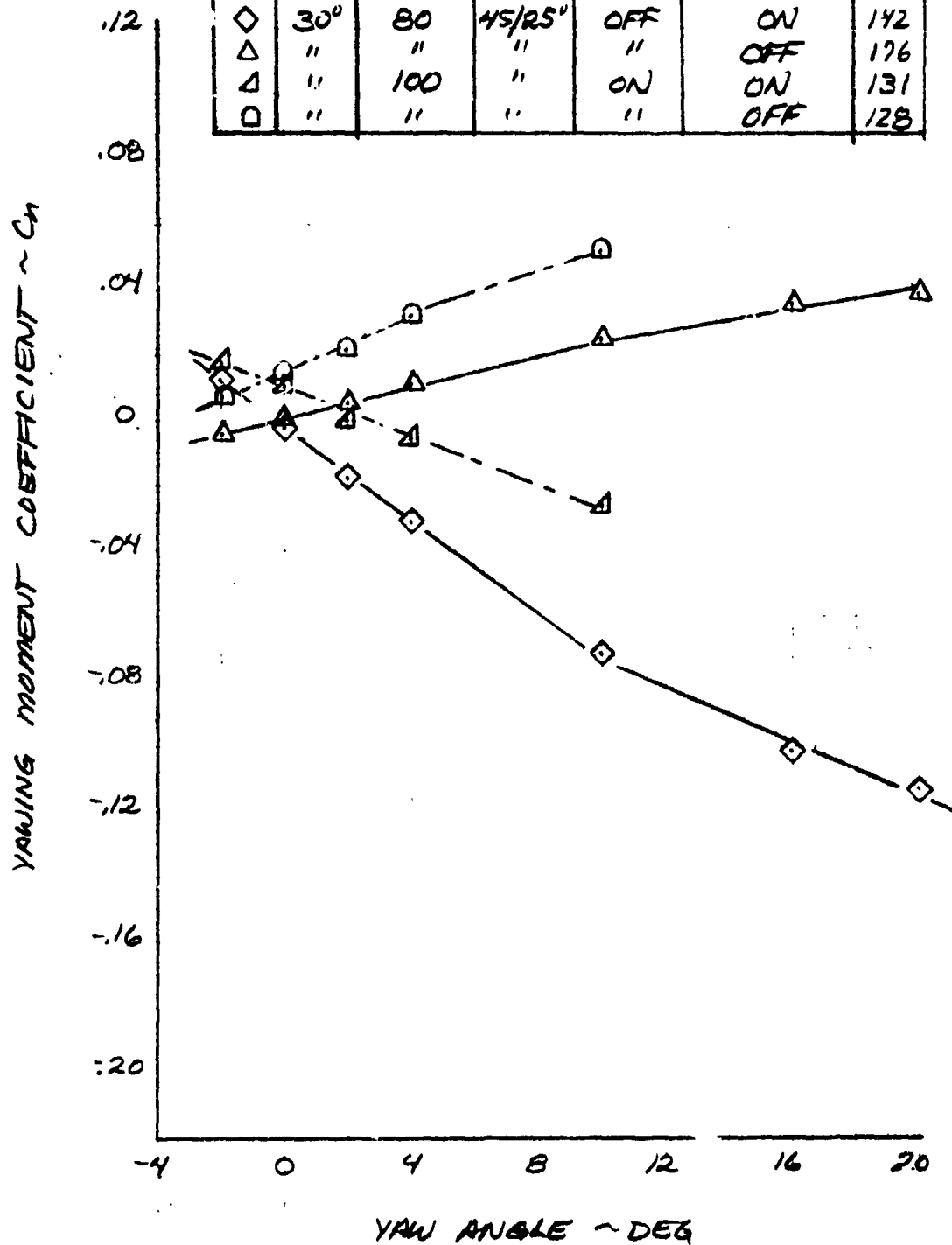
Figure V-65. Yawing Moment versus Yaw Angle, Mast Angle 30°.

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VI. ANALYSIS OF RESULTS

The test data have been analyzed to determine qualitatively and quantitatively the effects of the rotor wake on the empennage. The results of this analysis will be used to update the Model 301 flight simulation mathematical model.

An analysis of the discrepancy with regard to predicted and measured flapping was also made and is discussed below.

A. Analysis of Rotor Wake Tuft Grid Patterns and Smoke Observations

A tuft grid measuring 3 by 4 feet and located 6 inches aft of the horizontal stabilizer trailing edge was used to visually observe the rotor wake in the vicinity of the empennage. A camera located downstream of the grid was used to record the tuft patterns. The tuft grid installation was shown in Figure IV-14.

Figures VI-1 and VI-2 have been sketched from photographs of the tuft grid patterns at several airspeeds and attitude conditions. The photographs are presented in Appendix C. The camera was laterally displaced from the center of the grid creating a skewed image of the tufts in the photographs. The sketches in Figures VI-1 and VI-2 have been drawn without the distortion of the patterns as shown in the photographs. In the sketches the flow streamlines represent the direction of flow only and do not indicate magnitude.

The flow at the horizontal stabilizer for 16 and 20 knots, Figure VI-1(a) and (b), was not well defined as the tufts fluctuated considerably. The photographs and sketches at these speeds represent one instant in the fluctuations; hence care must be taken in interpreting them. The net effect of the rotor wake at these speeds appears to be a downwash on the horizontal stabilizer (this was also indicated during smoke studies of the rotor wake). However, the strain gage balance data indicates a net upwash effect on the stabilizer at these speeds.

At speeds above 20 knots the flow became more distinct. The flow patterns at 30 and 40 knots, presented in Figure VI-1(c) and (d), shows the rotors vortices rolled up directly above the horizontal stabilizer. At these speeds the rotor wake has the characteristic of the wake of a low aspect ratio wing with the net effect of an upload at the horizontal stabilizer. The upload is probably due more to vortex induced lift than to a physical upwash at the horizontal

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A. (Continued)

stabilizer. Analysis of balance data also shows that the
effective dynamic pressure at the horizontal stabilizer
is up to twice the freestream dynamic pressure.

The tuft grid was not used at speeds higher than 40 knots.
Smoke patterns at higher speeds show rotor vortices move
downward and outward as speed increases. At 50 knots the
vortices were slightly outboard and below the horizontal
stabilizer - vertical fin junctions. A net upwash over the
span of the horizontal stabilizer, induced by the vortices,
was clearly visible. The upwash was evident at speeds up
to 100 knots.

Flow patterns at an airspeed of 40 knots and yaw angles of
0 and 10 degrees are compared in Figure VI-2. At 10 degrees
right yaw the right rotor vortex core is clear of the empen-
nage while the left rotor vortex core is nearly centered
on the horizontal stabilizer. This shift in the position
of the vortices with respect to the empennage has two effects:

(1) The strength of the vortex lift on the horizontal
stabilizer is reduced, reducing the upload and pro-
ducing a nose up pitching moment, and

(2) for yaw angles between 0 and 12 degrees, directional
stability is reduced by the right fin being immersed
in the right rotor vortex. At yaw angles greater than
12 degrees the vortex from the left rotor increases
the effectiveness of the fins and the directional
stability is increased over that of the basic airframe.

Smoke studies showed that the rotor induces a strong upwash
on the wing inboard panel even at speeds as low as 20 knots.
This is clearly evident in the balance pitching moment data
presented in Section V, where wing stall occurs at several
degrees lower angle of attack rotors-on than for rotors-
off.

B. Determination of Net Effect of Rotor Wake at Empennage

1. Downwash at Horizontal Stabilizer - Rotors Off

The downwash at the horizontal stabilizer was deter-
mined using pitching moment data from tail-off, tail-
on and incidence ($i_N = 0°$ and $i_H = -4°$) runs to obtain
the wake angle. Rotors-off, the wake is the wing down-
wash ($\epsilon_{W/H}$). The method of obtaining the wing downwash
is as follows:

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1. (Continued)

Knowing,

$$C_{m_H} = C_{m_{TAIL-ON}} - C_{m_{TAIL-OFF}} \tag{1}$$

$$= -a_H \, \eta_{H_{WB}} \, \overline{V}_H \, \alpha_H \tag{2}$$

$$C_{m_{i_H}} = (C_{m_{i_H} = -4^\circ} - C_{m_{i_H} = 0^\circ})/-4^\circ \tag{3}$$

$$= -a_H \, \eta_{H_{WB}} \, \overline{V}_H \tag{4}$$

then, $\alpha_H = C_{m_H}/C_{m_{i_H}}$ \hfill (5)

$$= \alpha_F - \epsilon_{W/H} + i_H^{\;0^\circ} + \tau_e^{\;0^\circ} \delta_e \tag{6}$$

giving $\quad \epsilon_{W/H} = \alpha_F - \alpha_H$ \hfill (7)

The horizontal stabilizer lift curve slope was estimated
using the method of USAF Datcom[18] corrected to the
model's Reynolds number. The horizontal tail volume
(V_H) was obtained from model geometry. Knowing these
two parameters the dynamic pressure ratio at the hori-
zontal stabilizer was determined as follows:

$$a_H = .054/degree$$

$$\overline{V}_H = \frac{S_H \ell_H}{S_W C_W} = \frac{7.01(4.33)}{7.25(1.05)} = 1.14$$

$$\eta_{H_{WB}} = C_{m_{i_H}}/a_H \overline{V}_H \tag{8}$$

The horizontal stabilizer characteristics (C_{m_H}, $C_{m_{i_H}}$, α_H

and $\eta_{H_{WB}}$) and wing downwash ($\epsilon_{W/H}$) for rotors-off,

mast angles 90, 75, 60, and 30 degrees are shown in

1. (Continued)

Figures VI-3 through VI-6. The large amount of scatter in $\eta_{H_{WB}}$ is due to the very low test dynamic pressures.

Force model data at higher dynamic pressures do not show as much scatter. Also, the force model data show higher values of $\eta_{H_{WB}}$, indicating the horizontal stabilizer

lift curve slope for the aeroelastic model to be esti-mated too high. The values shown for $\eta_{H_{WB}}$ should only be

used in conjunction with the lift curve slope noted in the figures.

The rotors-off downwash is used in the current flight simulation mathematical model in computing the total wake. The wing downwash is assumed not to change due to the induced flow of the rotor other than by how the average wing angle of attack changes due to the induced flow of the rotor.

Rotors-off runs where made at the same equivalent air-speeds as for the rotors-on tests. This was to elimi-nate any differences due to Reynolds number effects.

2. <u>Downwash at Horizontal Stabilizer - Rotors On</u>

The horizontal stabilizer characteristics, rotors-on, are computed similarly to rotors-off except the total wake (ϵ_T) includes the elevator settings for trim. Elevator effectiveness (τ_e) was obtained from an ele-vator sweep enabling the total wake angle (ϵ_T) due to the wing and rotor to be determined as follows:

$$C_{m_{H_{H_{ROTORS-ON}}}} = -a_H \ \eta_{H_T} \ \overline{V}_H \ \alpha_H \tag{9}$$

$$C_{m_{i_{H_{ROTORS-ON}}}} = -a_H \ \eta_{H_T} \ \overline{V}_H \tag{10}$$

$$C_{m_{\delta_{e_{ROTORS-ON}}}} = -a_H \ \eta_{H_T} \ \overline{V}_H \ \tau_e \tag{11}$$

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VI-4

2. (Continued)

therefore, $\tau_e = C_{m_{\delta_e}}/C_{m_{i_H}}$ (12)

$\alpha_H = C_{m_{i_H}}/C_{m_{i_H}}$ (13)

and $\alpha_H = \alpha_F - \epsilon_T + i_H^{\,0^\circ} + \tau_e \delta_e$ (14)

giving $\epsilon_T = \alpha_F - \alpha_H + \tau_e \delta_e$ (15)

$\eta_{H_T} = -C_{m_{i_H}}_{\text{ROTOR-ON}}/a_H \overline{V}_H$ (16)

These values for level flight mast angles 90, 75, 60,
and 30 degrees are shown in Figures VI-7 through VI-10.
A comparison with the rotors-off values for 80 knots is
also shown. The total wake (ϵ_T) indicates an upwash

occurring for all mast angles and pitch attitudes tested
except for mast angle 30 degrees. Most noticeable is
the large increase in dynamic pressure ratio at the
tail (η_{H_T}) at low speeds for mast angles of 90 and 75

degrees. (This term was omitted for mast angles of
60 and 30 degrees since incidence sweeps were not made
at these mast angles.)

The slope of the total wake with angle of attack
($\partial \epsilon_T/\delta \alpha_F$) is an indication of the level of aircraft

static stability. The steeper the slope the less stable
the aircraft. This is obtained from the following
equation.

$$C_{m_\alpha} = C_{m_{\alpha_{\text{TAIL-OFF}}}} + C_{m_{\alpha_H}} \qquad (17)$$

Since static stability (C_{m_α}) is proportional to the

horizontal stabilizer $C_{m_{\alpha_H}}$, and

$$C_{m_{\alpha_H}} = -a_H \eta_{H_T} \overline{V}_H \left(1 - \frac{\partial \epsilon_T}{\partial \alpha_F}\right) \qquad (18)$$

2. (Continued)

Therefore, the greater $\partial_{\epsilon_T}/\partial_{\alpha_F}$ the smaller the contribution of the horizontal stabilizer to static stability. As shown in Figure VI-7 for 40 knots this slope is greater than 1.0 at negative pitch attitudes and almost 0.0 at positive attitudes, i.e., the model is unstable at negative attitudes and stable at positive attitudes. At 60 and 80 knots the slope of the wake about trim pitch attitude is less than for rotors-off showing the model to be more stable rotors-on than for rotors-off.

The total wake angle (ϵ_T) at the horizontal stabilizer is made up of the wing downwash ($\epsilon_{W/H}$) and rotor wake ($\epsilon_{R/H}$). The rotor wake angle was computed using the following equations:

$$\epsilon_T = \epsilon_{W/H} + \epsilon_{R/H} \tag{19}$$

therefore,

$$\epsilon_{R/H} = \epsilon_{W/H} - \epsilon_T \tag{20}$$

The magnitude of the rotor upwash contribution was determined as follows:

$$K_{R/H} V_{iR} = V_\infty \tan \epsilon_{R/H} \tag{21}$$

Where the term V_{iR} is the rotor induced velocity at the rotor disk. The $K_{R/H}$ term is that used in the current math model to include the rotor wake at the horizontal. This parameter is given in Figures VI-11 through VI-15 for the various mast angles, airspeeds, and flight conditions tested.

It should be noted that the term $K_{R/H} V_{iR}$ is merely a convenient way to represent the rotor wake effects on the horizontal stabilizer and does not represent the actual upwash or downwash from the rotor. This is illustrated by the fact that by taking the vector sum of the freestream velocity and the rotor induced velocity factor ($K_{R/H} V_{iR}$) does not give the total velocity indicated by the measured horizontal stabilizer efficiency (Figures VI-7 and -8). Furthermore the variation of $K_{R/H} V_{iR}$ with airspeed shown in Figures VI-11 through -15 implies the rotor induced velocity increases with airspeed which is opposite to the momentum theory of rotor induced velocity. These differences are believed to be the result of vortex induced lift. However, combining the η_H and $K_{R/H} V_{iR}$ in the math model does give the correct empennage lift.

2. (Continued)

The rotor wake at the horizontal stabilizer measured in
this test is compared to that used in the Phase I flight
simulation in Figure VI-16. It should be noted that in
the Phase I simulation η_T was assumed equal to 1.0 and
the downwash/upwash was assumed invariant with angle of
attack.

3. Effect of Yaw on Downwash at Horizontal Stabilizer

As the aircraft yaws the net rotor wake effect on the
horizontal changes from an upwash to downwash causing
a pitch-up with yaw. The rotor wake velocity is shown
in Figure VI-17 through VI-19 to illustrate the change
in the magnitude of the rotor wake with yaw angle.
This effect was evaluated during the rod test and found
to require only a small amount of longitudinal cyclic
to maintain pitch attitude with yaw.

4. Effect of Rotor Wake on Vertical Stabilizer Character-
istics

Rotor wake effects on the vertical stabilizer were eval-
uated in terms of a parameter defined as the rotor side-
wash factor (K_β). This is the ratio of the vertical
stabilizer yawing moment rotors-on to rotors-off and is
determined as follows:

Knowing,
$$C_{n_V} = C_{n_{TAIL\text{-}ON}} - C_{n_{TAIL\text{-}OFF}} \tag{22}$$

$$C_{n_{V_{ROTORS\text{-}OFF}}} = a_V \, \overline{V} \, \eta_{WB} \, (1 - \partial\sigma/\partial\beta)_{OFF}\,\beta \tag{23}$$

$$C_{n_{V_{ROTORS\text{-}ON}}} = a_V \, \overline{V}_V \, \eta_T \, (1 - \partial\sigma/\partial\beta)_{ON}\,\beta \tag{24}$$

therefore,
$$K_\beta = \frac{C_{n_{V_{ROTORS\text{-}ON}}}}{C_{n_V} \, ROTORS\text{-}OFF} \tag{25}$$

$$\frac{\eta_T \, (1 - \partial\sigma/\partial\beta)_{ON}}{\eta_{WB} \, (1 - \partial\sigma/\partial\beta)_{OFF}} \tag{26}$$

4. (Continued)

This term is shown in Figure VI-20 through VI-22 for
the various airspeeds and mast angles tested. The
rotor wake is destabilizing for yaw angles of ±4 degrees
at low speeds for mast angles 90 and 75 degrees. Above
twelve degrees yaw the rotors-on yawing moment exceeds
that of the rotors-off. This effect also exists at mast
angles of 60 and 30 degrees, although to a lesser degree.
The directional stability is decreased for up to about
four degrees of yaw and then increases as yaw angle
increases. However, the yawing moment coefficient
rotors-on did not exceed that of rotors-off.

5. Empennage Lift Characteristics

Empennage lift, rotors-on, is compared with lift rotors-
off in Figures VI-23 through VI-26. Empennage lift
is in general higher than that for rotors-off, reflec-
ting; the net upwash effect of the rotors on the
horizontal stabilizer. The decreasing difference
between the rotors on and off lift with increasing speed
reflects the fact that the rotor wake effects become
less significant at the higher speeds.

C. Analysis of Discrepancy Between Measured and Calculated
   Lateral Flapping

At mast angles 90 and 75 degrees the measured lateral flap-
ping was found to be approximately 40 percent greater than
that predicted prior to the test. Figure VI-27 compares the
measured and predicted flapping. Several possible reasons
for this discrepancy were theorized including aerodynamic
interference between the wing and the rotor. (This was
resolved by removing the wing fairings and flying the model
at the same trim condition. Flapping was not changed.)
However, smoke studies of the rotor induced flow indicated
that the longitudinal distribution of induced velocity was
considerably different than that used in the theory. It
was therefore concluded that the theory used to predict
lateral flapping was deficient with regard to the math
model of the rotor induced velocity.

1. Induced Velocity Representation Used in Pretest Predic-
   tions

Program C81 was used to predict blade flapping prior to
the test. In hover C81 uses a triangular distribution
of induced velocity such that

$$V_i = \frac{4}{3} \times \overline{V}$$

1. (Continued)

   where $\overline{V}_i$ is the local spanwise induced velocity, X is the nondimensional blade radius, r/R, and $\overline{V}$ is the average induced velocity. For forward flight the triangular distribution is modified to increase the induced velocity at the rear of the disc ($\psi = 0°$) and reduce it at the front of the disc ($\psi = 180°$) as suggested in Reference 19. The downwash distribution is then assumed equal to

   $$V_i = \frac{4}{3} \, X \, (1 + K \cos \psi) \, \overline{V}$$

   where K has the following values

μ	K
$0 < \mu < .1067$	$11.25 \, \mu$
$.1067 < \mu < .5733$	$1.36 - 1.5 \, \mu$
$\mu \geq .5733$	0.5

   These values for K were derived for low disc loading, low twist rotors[20] and have provided reasonable correlation with such rotors. By modifying the value of K arbitrarily to reflect the higher disc loading, high twist, tilt rotor, good correlation with the test data can be achieved. Figure VI-28 compares the original longitudinal distribution of induced velocity with one obtained by modifying the value of K to achieve correlation. Also shown is the approximate shape of the longitudinal distribution observed during smoke studies of the rotor wake. The very high value of induced velocity at the trailing edge of the disc is probably due to roll up of the rotor wake.

2. **Investigation of More Advanced Methods of Predicting Lateral Flapping**

   The significant difference between the computed and observed distribution of induced velocity indicted a better method of predicting the distribution is needed. Therefore, an investigation into more advanced means of predicting the distribution was made.

   A recent study of Harris, Reference 21, indicates the problem of predicting lateral flapping accurately is not confined to high disk loading tilt rotors. In Reference 21 a comparison is made between the lateral flapping

2. (Continued)

predicted by several induced velocity theories with that
measured with a low disc loading model rotor. The trend
of the measured flapping with advance ratio was similar
to that measured in this test and the maximum lateral
flapping occurred at nearly the same advance ratio
(μ = .08 compared to .091). Harris compared downwash
theories ranging from uniform downwash to prescribed
helical wakes. In no case did the predicted flapping
exceed 70 percent of the measured flapping. Several
theories, specifically that of Castles and DeLeeuw,
Reference 22, and Heyson and Katzoff, Reference 23, pre-
dicted the correct trend with advance ratio but were off
in magnitude by as much as 50 percent.

A comparison between lateral flapping predicted using
various induced velocity theories, including those
investigated by Harris is given in Table VI-I for mast
angle 90 degrees, airspeed 40 knots (the speed for
maximum lateral flapping). Castles and DeLeeuw's and
Heyson and Katzoff's theories predict 57 percent and
67 percent of the measured lateral flapping respectively.
Both are based on a prescribed wake geometry with Castles
theory being for a uniform disc loading and Heyson's
accounting for the nonuniformity of the loading. The
distribution of induced velocity predicted using Heyson
and Katzoff's method is very close to that observed in
the test but the magnitude is too low. Also shown in Table
VI-I is the flapping predicted by Bell computer program
BRAM, Reference 24, which has a free trailing tip vortex.
This accounts to some extent for the roll up of the wake
but neglects the nonuniformity of the blade loading.
BRAM comes closest to predicting the measured lateral
flapping.

3. Prediction of Lateral Flapping For Design Purposes

Although none of the more advanced induced velocity
theories satisfactorily predicts the lateral flapping, a
method is available for design purposes. As noted
earlier, by arbitrarily modifying the factor K used in
the present C81 induced velocity representation to
distribute the induced velocity longitudinally, good
correlation with the measured flapping can be achieved.
Figures VI-29 through VI-32 compare the flapping pre-
dicted using the modified value of K with the measured
values. Correlation is good at all mast angles.

While this approach is not sophisticated it does achieve
the basic objective of predicting the lateral flapping
and is felt to be acceptable for design purposes. This
method has the advantage of using a minimum of computer
time whereas a free wake analysis such as BRAM requires
a large computer run time.

TABLE VI-I. COMPARISON OF LATERAL FLAPPING CALCULATED USING
SEVERAL THEORIES WITH MEASURED LATERAL FLAPPING

Flight Condition $\alpha_m = 90°$, GW = 11,000 LB, 40 Knots

Measured b_1 Flapping = 6 Degrees[1]

Theory	b_1 Flapping	% Of Measured
Pretest (Drees)	4.1°	68%
Post Test (Drees)[2]	6.0°	100%
Castles and DeLeeuw	3.3°	57%
Heyson and Katzoff	4.0°	67%
BRAM $\left(\begin{array}{c}\text{Free}\\\text{Vortex}\end{array}\right)$	5.0°	83%

1. Based on line faired through test data.

2. Longitudinal distribution factor adjusted to achieve
correlation.

VIEW LOOKING FORWARD FROM
AFT OF EMPENNAGE

(a) AIRSPEED = 16 KNOTS
$\alpha = \Psi = 0^{o}$
RPM = 1260

LEFT ROTOR   EMPENNAGE   RIGHT ROTOR

(b) AIRSPEED = 20 KNOTS
$\alpha = \Psi = 0^{o}$
RPM = 1260

FLOW STREAMLINES   TUFT GRID

(c) AIRSPEED = 30 KNOTS
$\alpha = \Psi = 0^{o}$
RPM = 1260

(d) AIRSPEED = 40 KNOTS
$\alpha = -2^{o}, \Psi = 0^{o}$
RPM = 1260
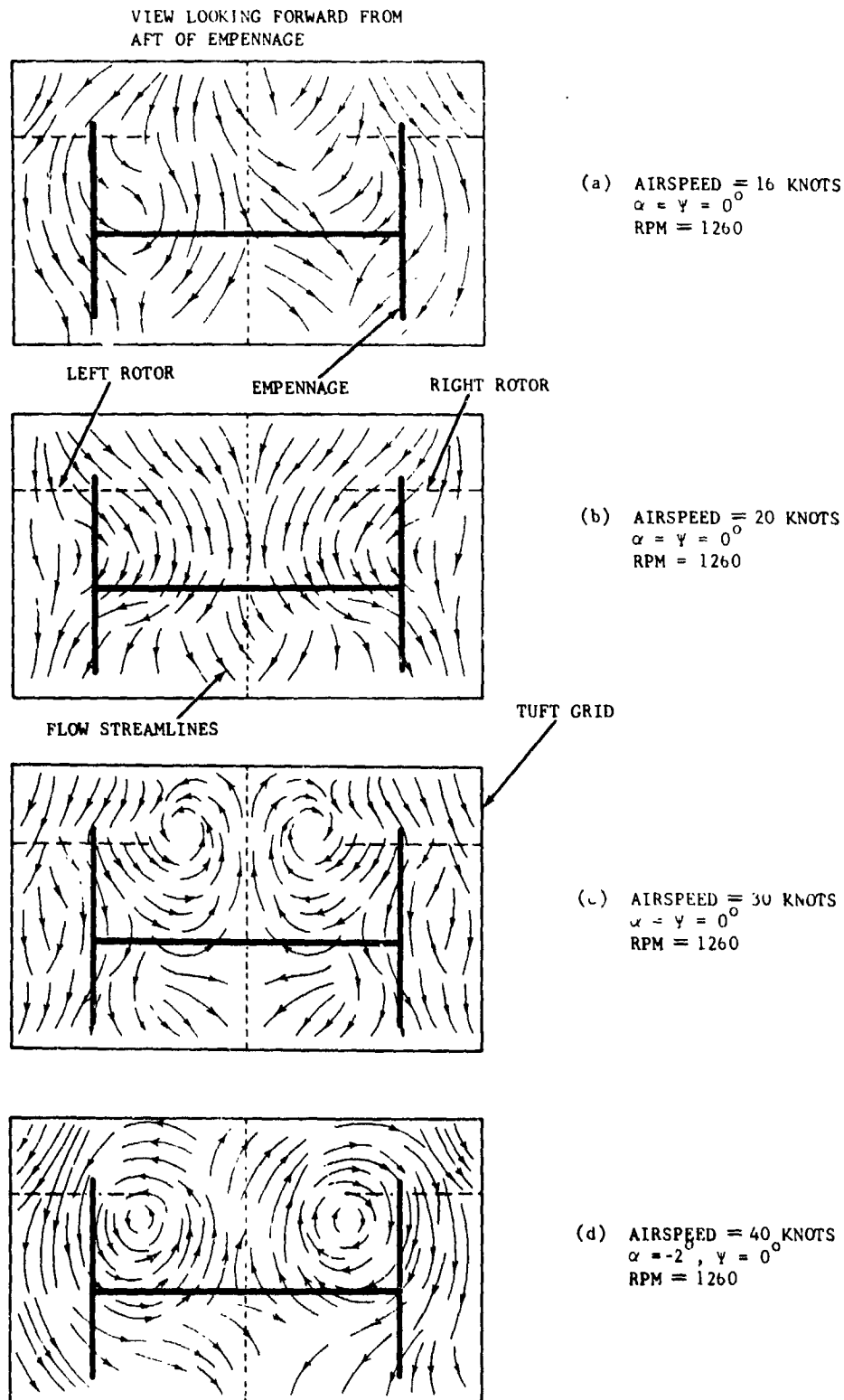
Figure VI-1.   Flow Patterns in the Vicinity of the Empennage
in Level Flight, Mast Angle 90°.

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VIEW LOOKING FORWARD
FROM AFT OF EMPENNAGE

TUFT GRID

(a) Ψ = 0°, α = -2°

EMPENNAGE

FLOW STREAMLINES

LEFT ROTOR

RIGHT ROTOR

(b) Ψ = 10°, α = -2°

DYNAMIC PRESSURE LOSS
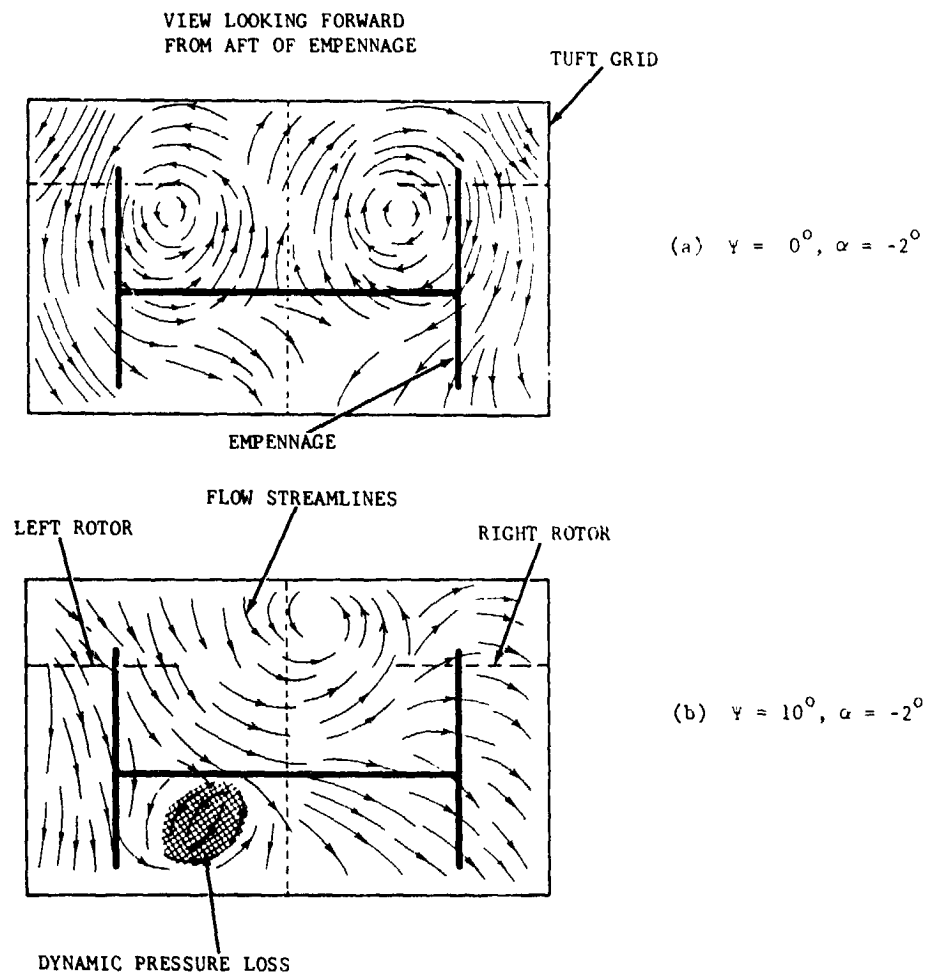
Figure VI-2. Effect of Yaw Angle on Flow Pattern,
Mast Angle 90°, Airspeed 40 Knots.

HORIZONTAL STABILIZER CHARACTERISTICS

○ 40 KTS
□ 60 KTS
◇ 80 KTS

MAST = 90°
FLAPS = 45/25°
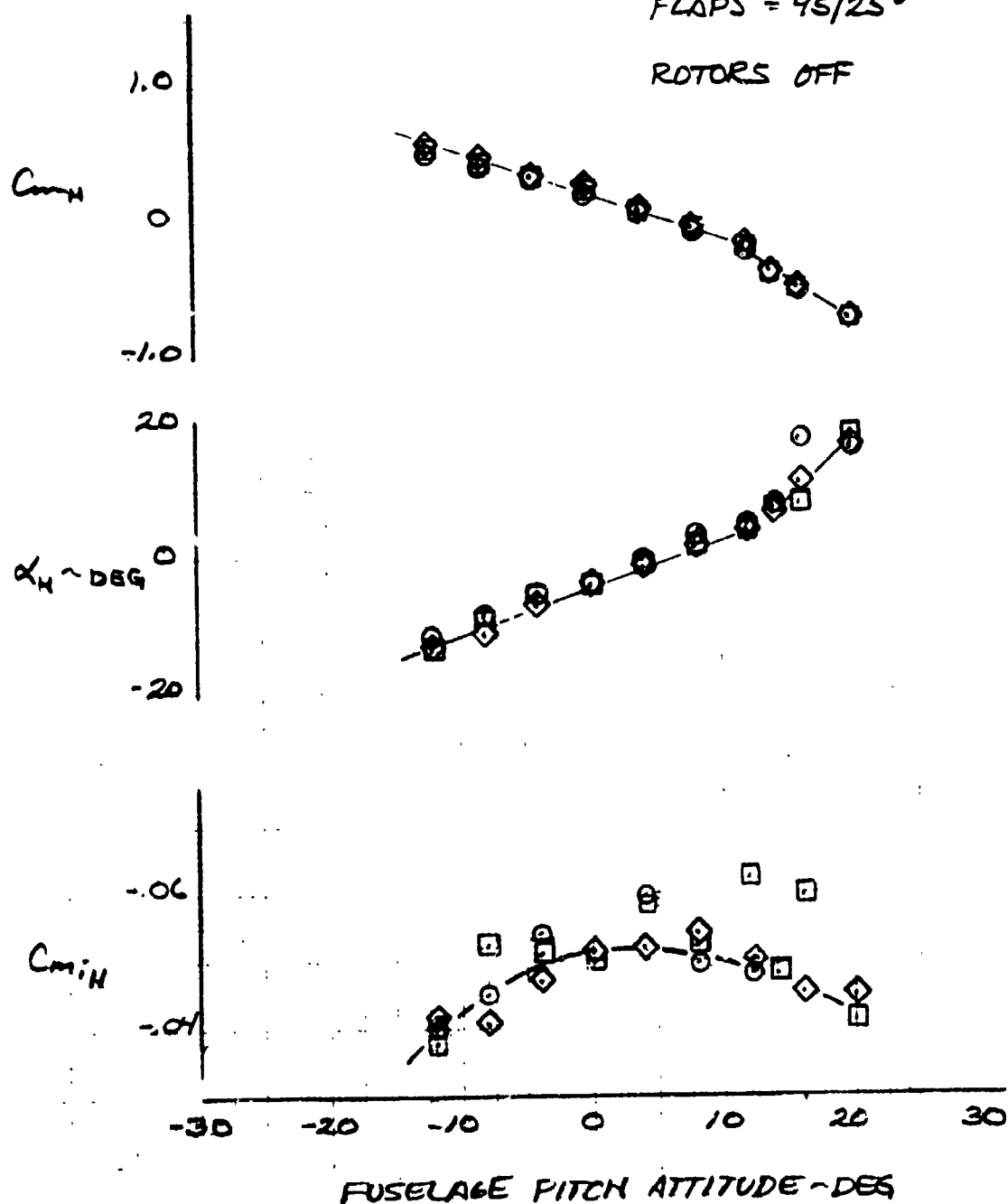
ROTORS OFF

Figure VI-3. Horizontal Stabilizer Aerodynamic
Characteristics, Mast Angle 90°,
Rotors off.

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HORIZONTAL STABILIZER DOWNWASH

O 40 KTS
□ 60 KTS   MAST = 90°
△ 80 KTS   FLAPS = 45/25°
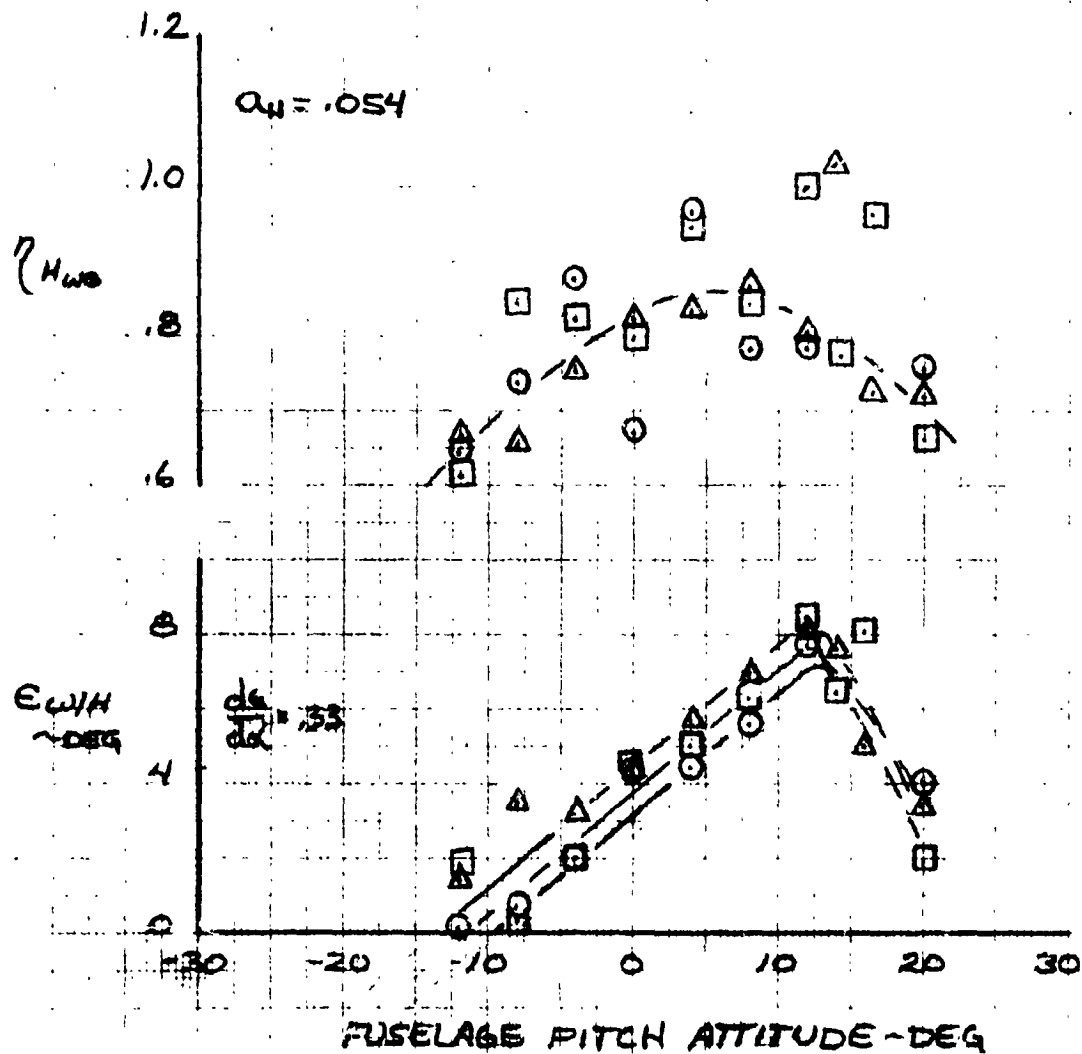
ROTORS OFF

Figure VI-3. Concluded

HORIZONTAL STABILIZER CHARACTERISTICS

○ 40 KTS
□ 60 KTS
△ 80 KTS

MAST 75°
FLAPS = 45/25°
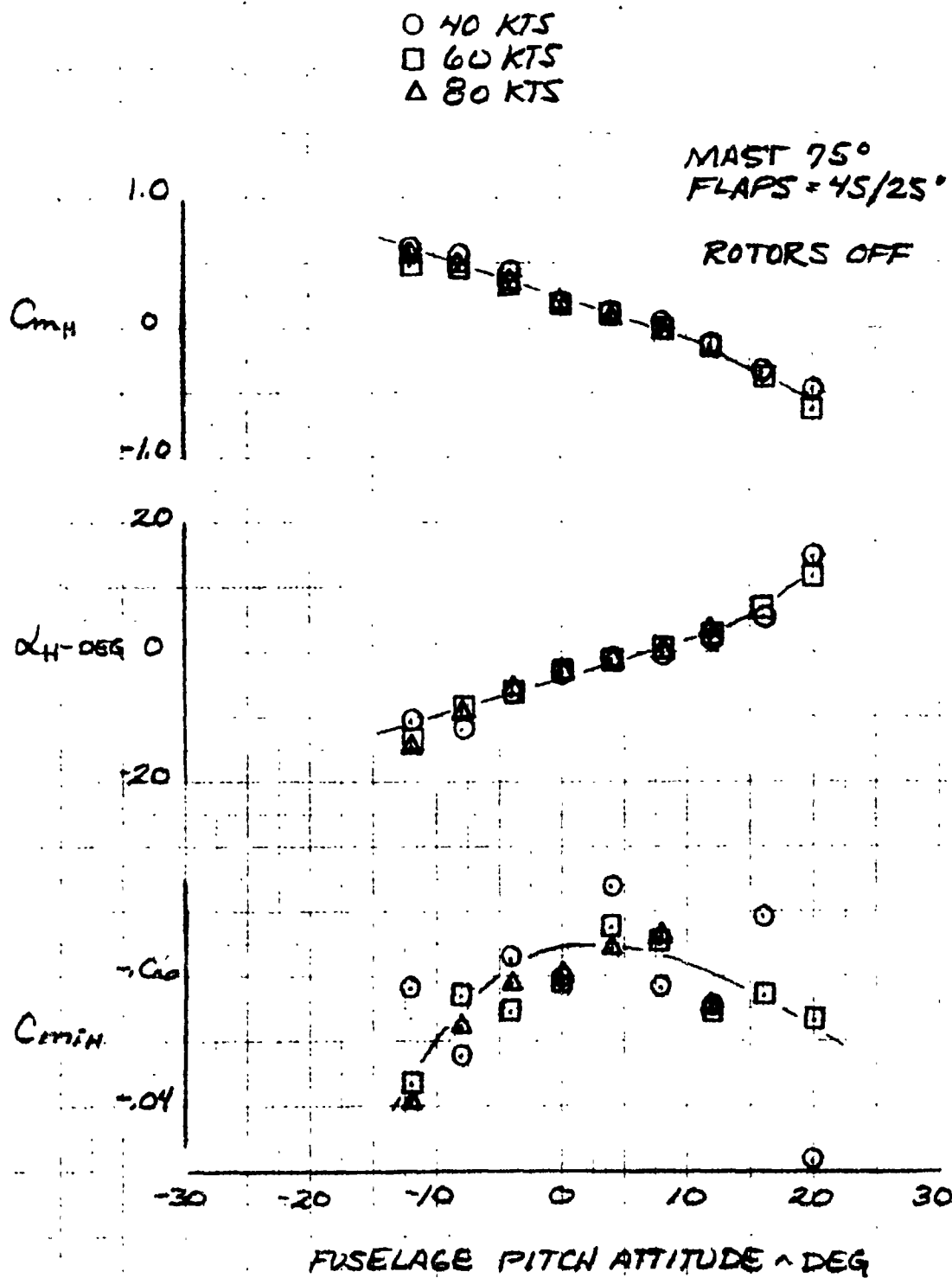
ROTORS OFF

Figure VI-4. Horizontal Stabilizer Aerodynamic
Characteristics, Mast Angle 75°,
Rotors off.

HORIZONTAL STABILIZER DOWNWASH

○ 40 KTS
□ 60 KTS
△ 80 KTS

MAST = 75°
FLAPS = 45/25°
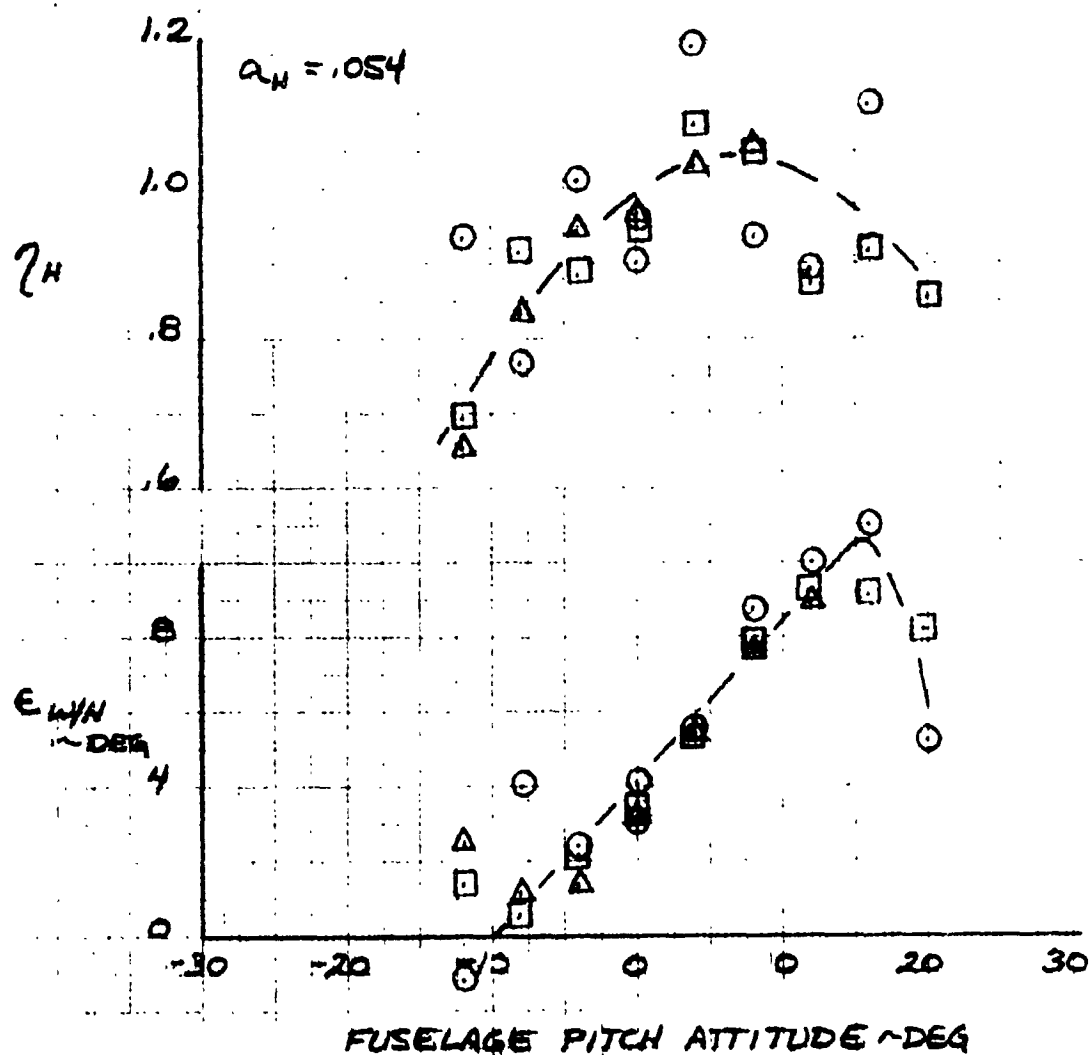
ROTORS OFF

Figure VI-4.   Concluded

HORIZONTAL STABILIZER CHARACTERISTICS

MAST = 60°
FLAPS = 45/25°
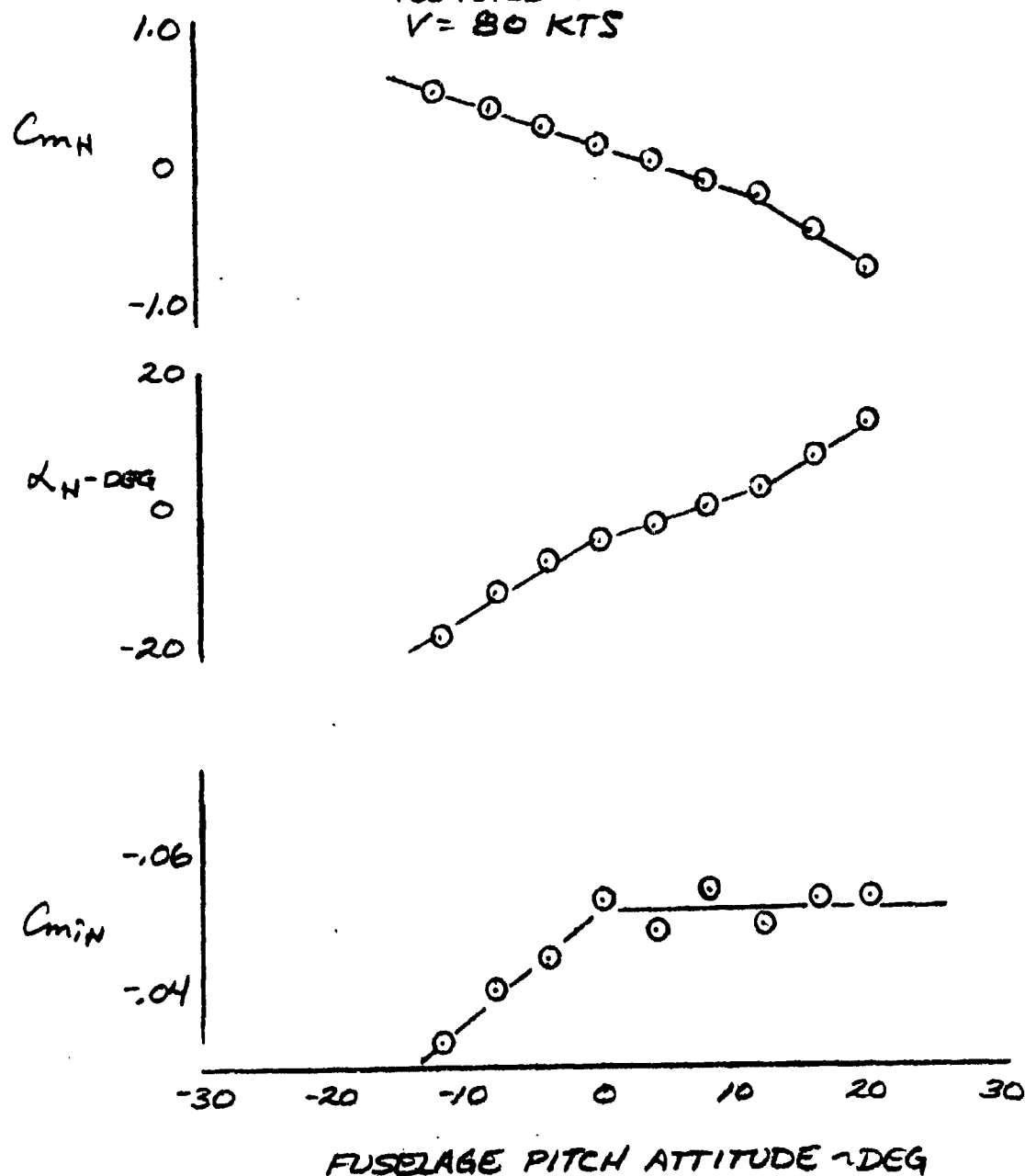
ROTORS OFF
V = 80 KTS

Figure VI-5. Horizontal Stabilizer Aerodynamic
Characteristics, Mast Angle 60°,
Rotors off.

HORIZONTAL STABILIZER DOWNWASH

MAST = 60°
FLAPS = 45/25°
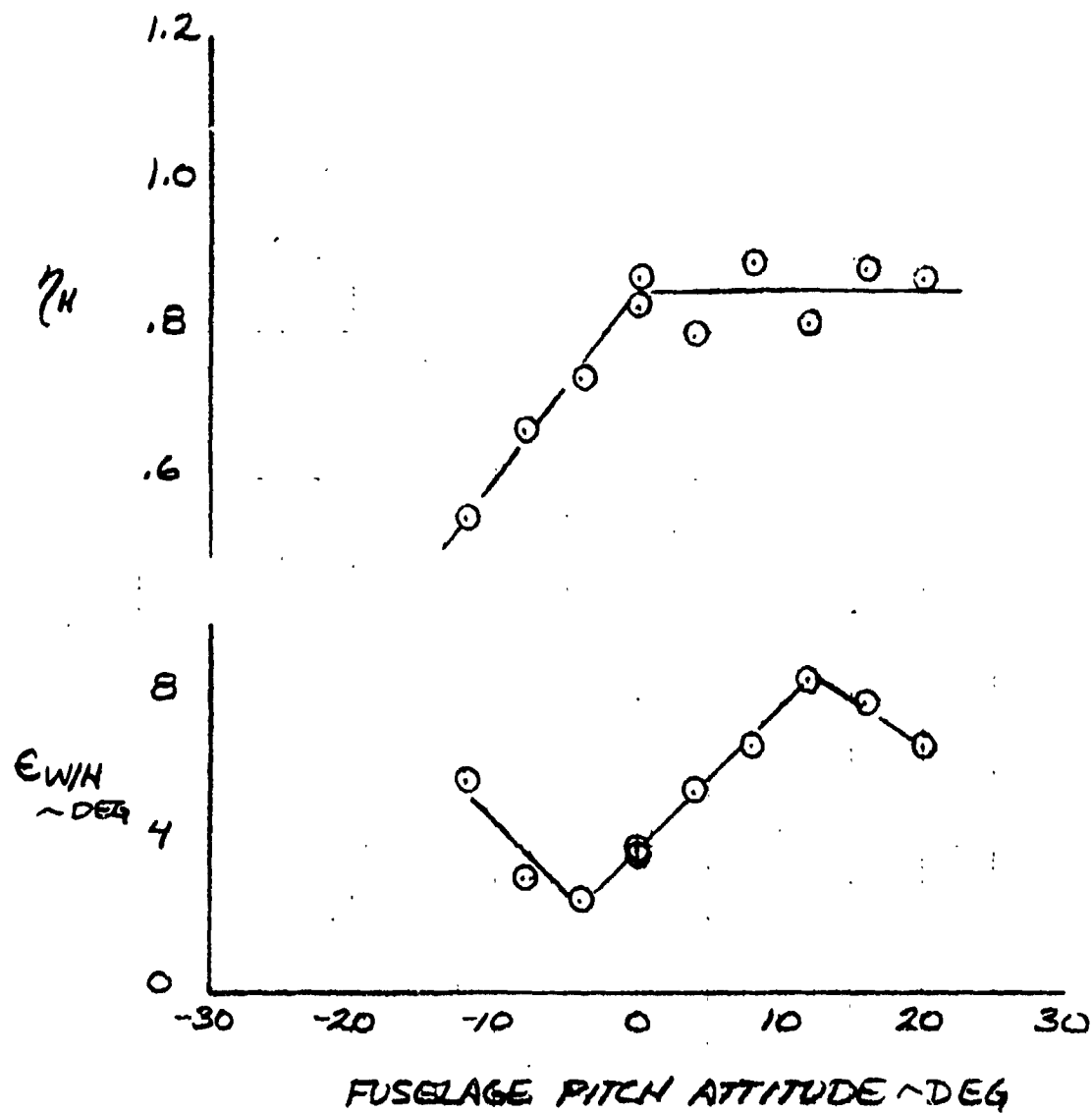
ROTORS OFF
V = 80 KTS

Figure VI-5.   Concluded

HORIZONTAL STABILIZER CHARACTERISTICS

MAST = 30°
FLAPS = 45/25°
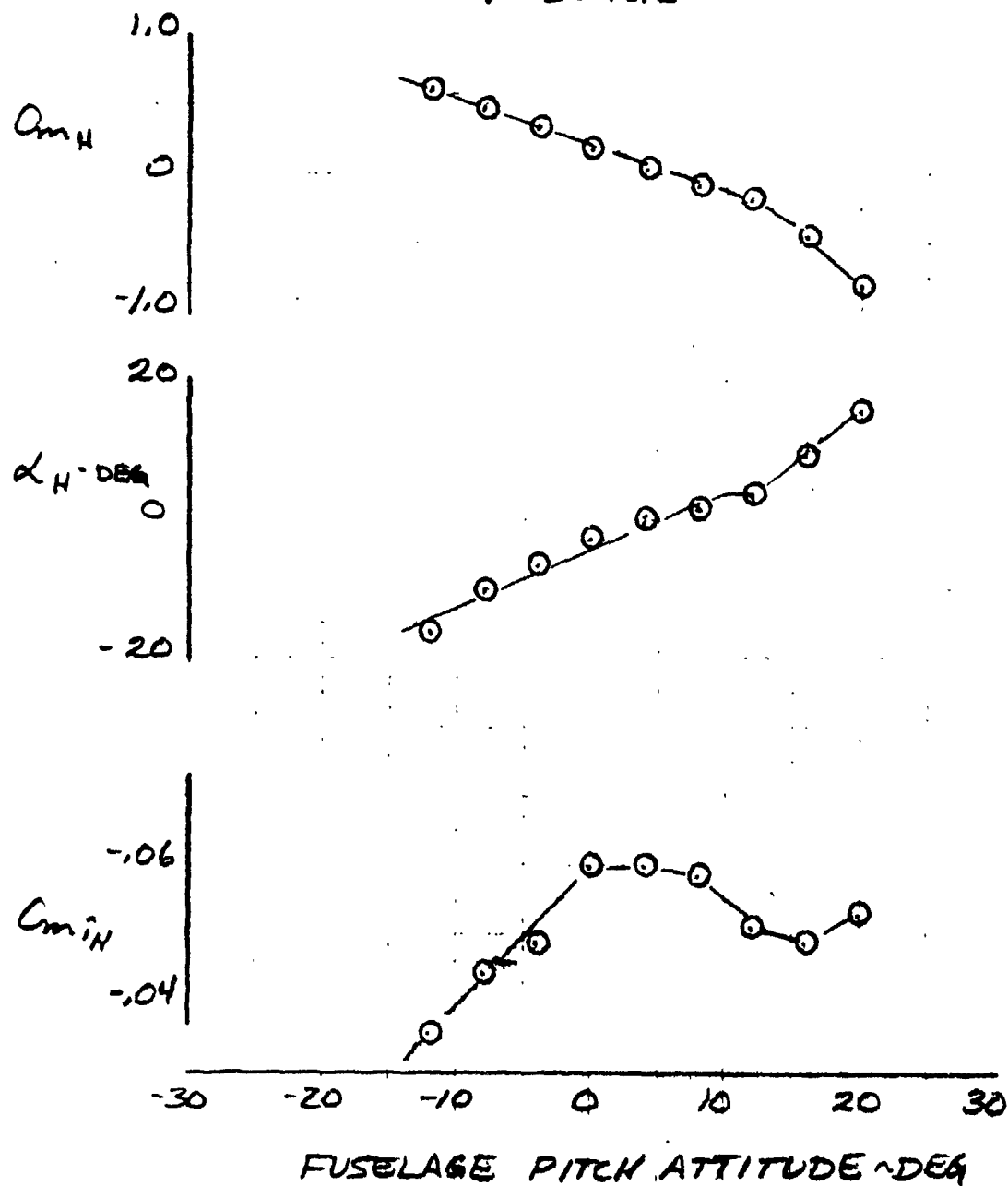
ROTORS OFF
V = 80 KTS

Figure VI-6. Horizontal Stabilizer Aerodynamic
Characteristics, Mast Angle 30°,
Rotors off.

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HORIZONTAL STABILIZER DOWNWASH

MAST = 30°
FLAPS = 45/25°

ROTORS OFF
V = 80 KTS

FUSELAGE PITCH ATTITUDE ~DEG
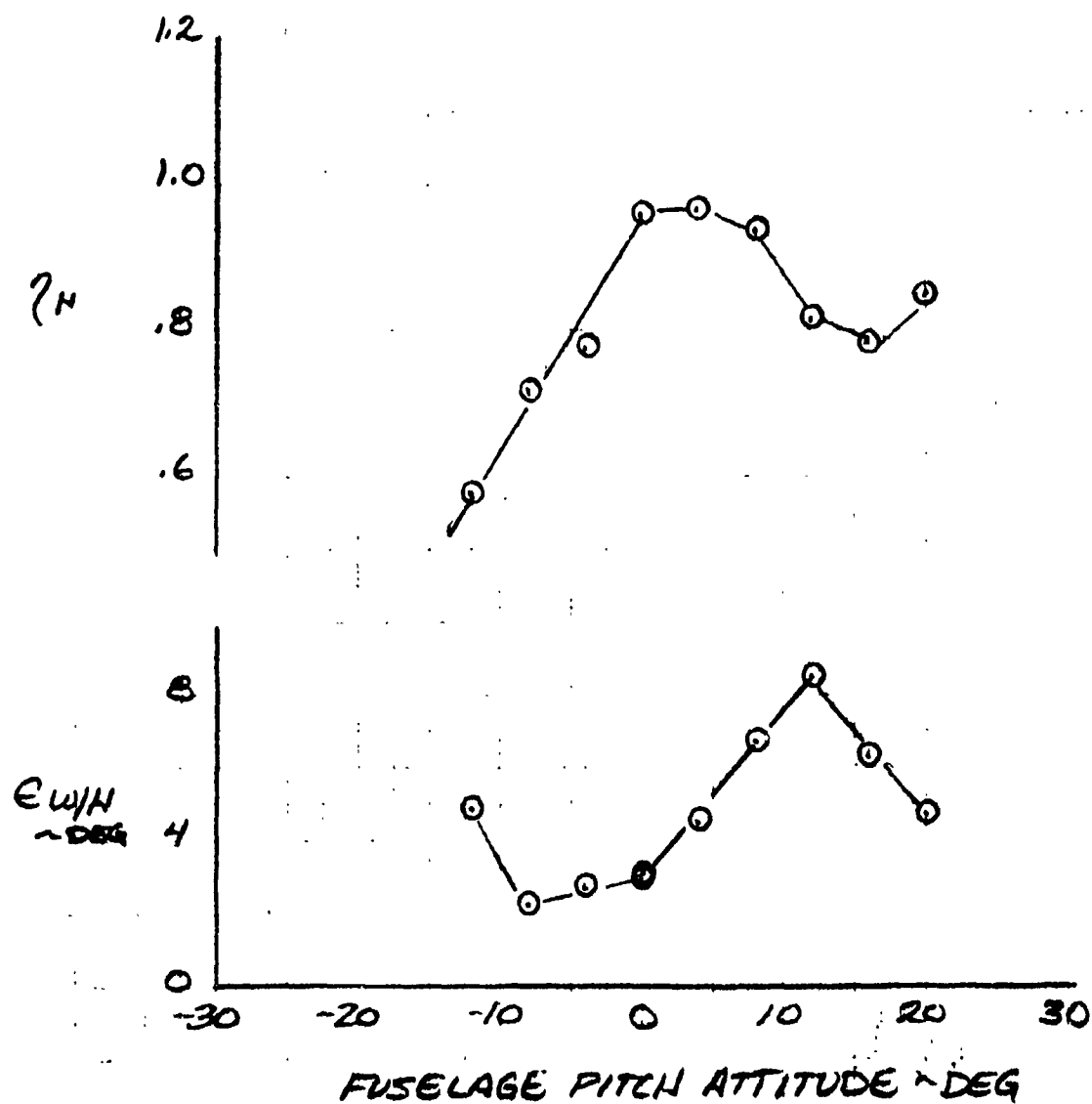
Figure VI-6. Concluded

HORIZONTAL STABILIZER CHARACTERISTICS

O 40 KTS $\frac{Se}{3.0°}$ MAST = 90°
□ 60 KTS 4.0° FLAPS = 45/25
△ 80 KTS 7.0°

- - - - ROTORS OFF
———— ROTORS ON

FUSELAGE PITCH ATTITUDE ~ DEG
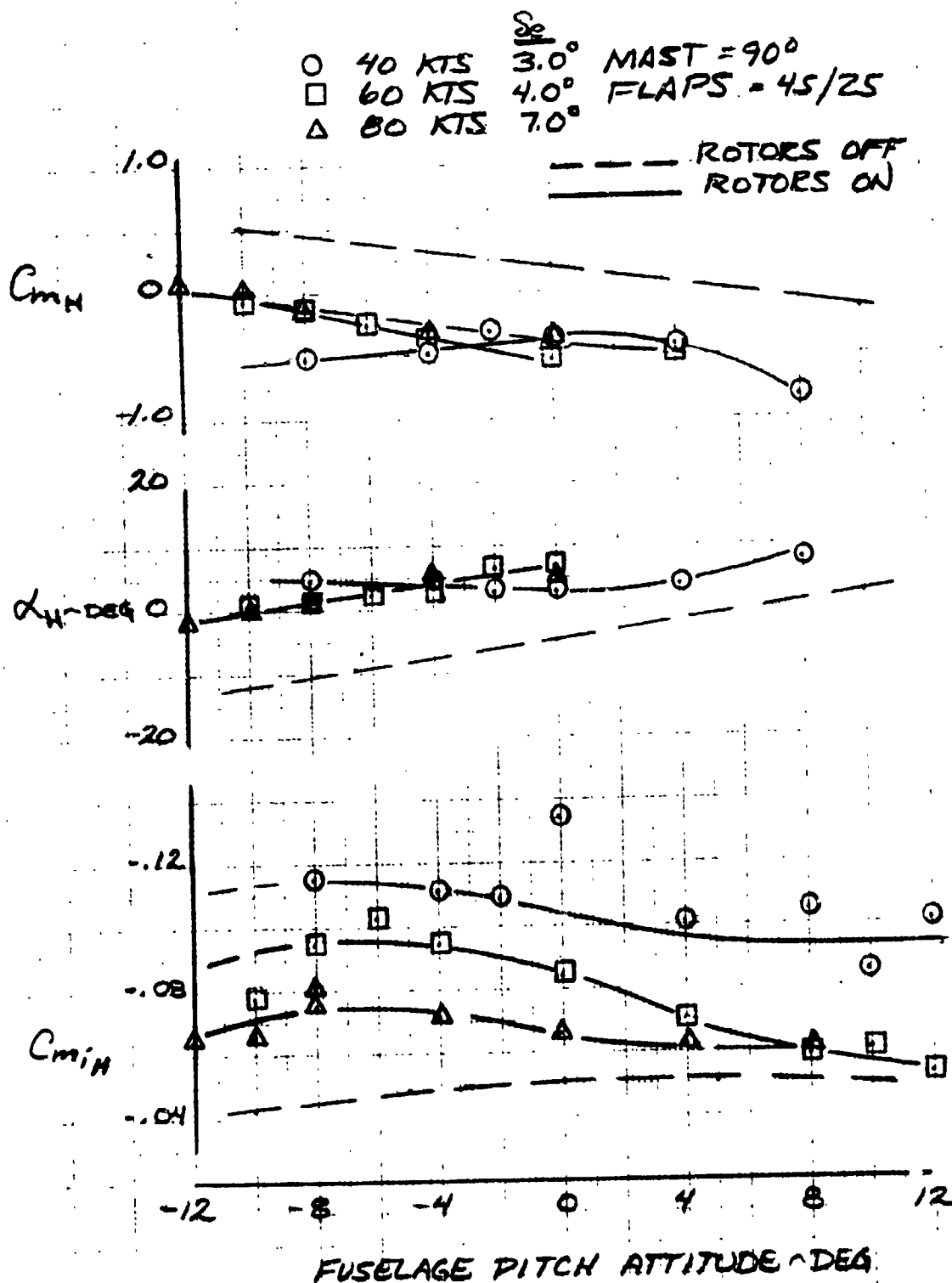
Figure VI-7. Horizontal Stabilizer Aerodynamic
Characteristics, Mast Angle 90°,
Rotors on.

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HORIZONTAL STABILIZER DOWNWASH

○ 40 KTS
□ 60 KTS
△ 80 KTS

MAST = 90°
FLAPS = 45/25°

ROTORS ON

$\alpha_H = .054$

η_{H_T}

ROTORS OFF

ϵ_{DWH}
ROTORS OFF
(80 KTS)

ϵ_T ~ DEG

FUSELAGE PITCH ATTITUDE ~ DEG.

Figure VI-7. Concluded

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HORIZONTAL STABILIZER CHARACTERISTICS

$$\frac{S_e}{}$$

O	40 KTS	-3.6
□	60 KTS	-3.4
△	80 KTS	2.2

MAST = 75°
FLAPS = 45/25°

— — — ROTORS OFF
——— ROTORS ON

FUSELAGE PITCH ATTITUDE ~DEG
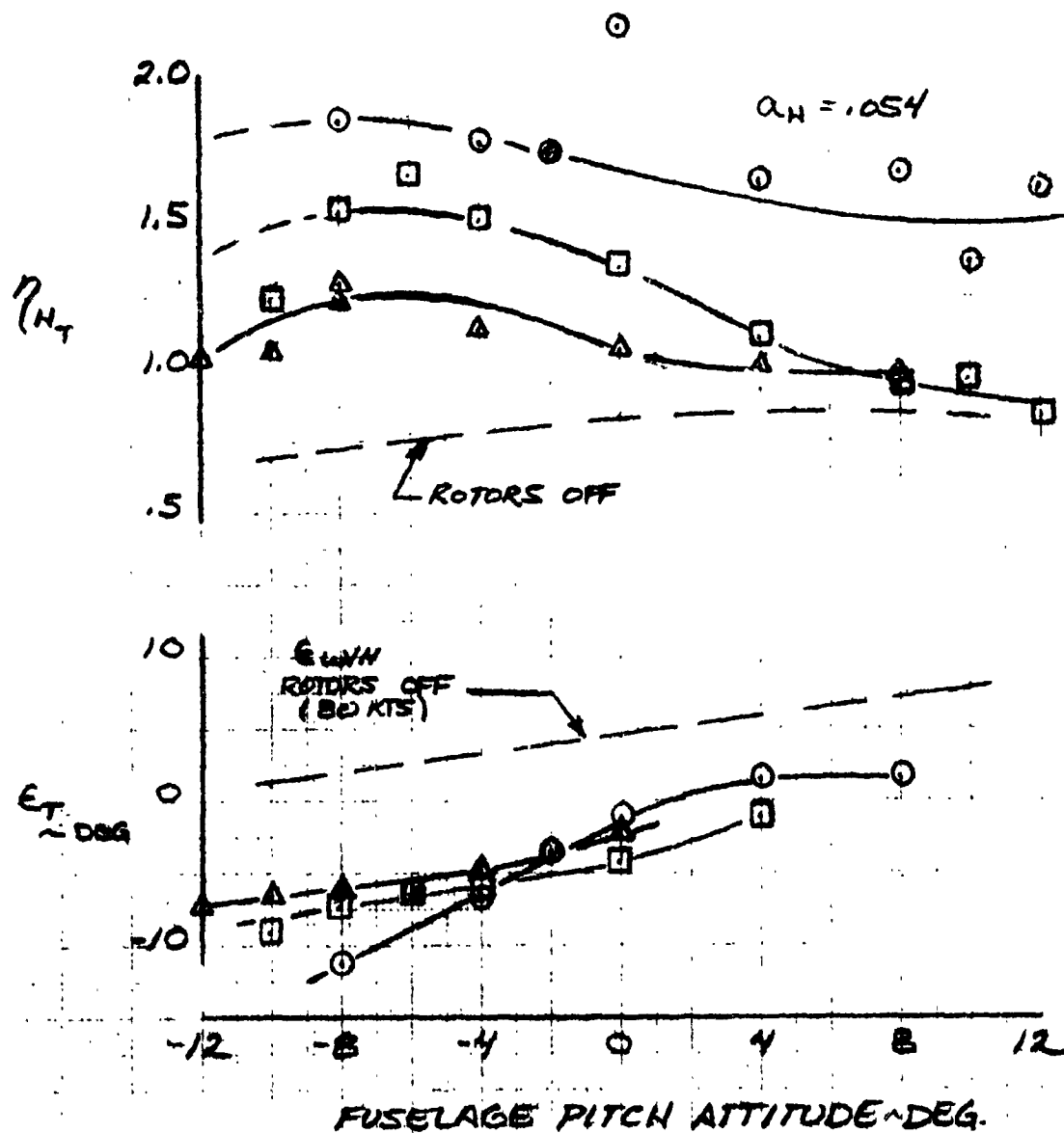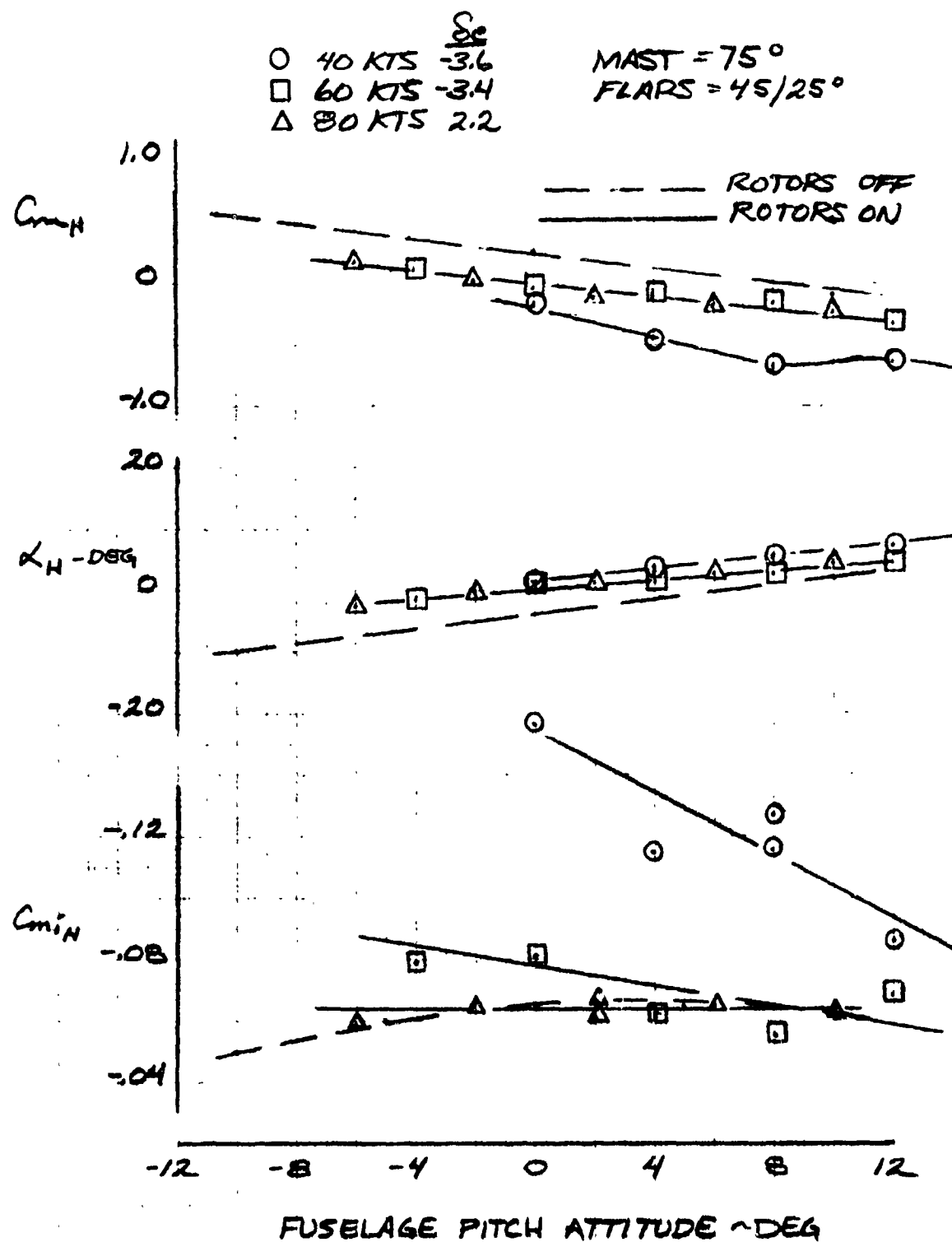
Figure VI-8. Horizontal Stabilizer Aerodynamic
Characteristics, Mast Angle 75°,
Rotors on.

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HORIZONTAL STABILIZER DOWNWASH

○ 40 KTS
□ 60 KTS
△ 80 KTS

MAST = 75°
FLAPS = 45/25°

— — — — ROTORS OFF
—————— ROTORS ON
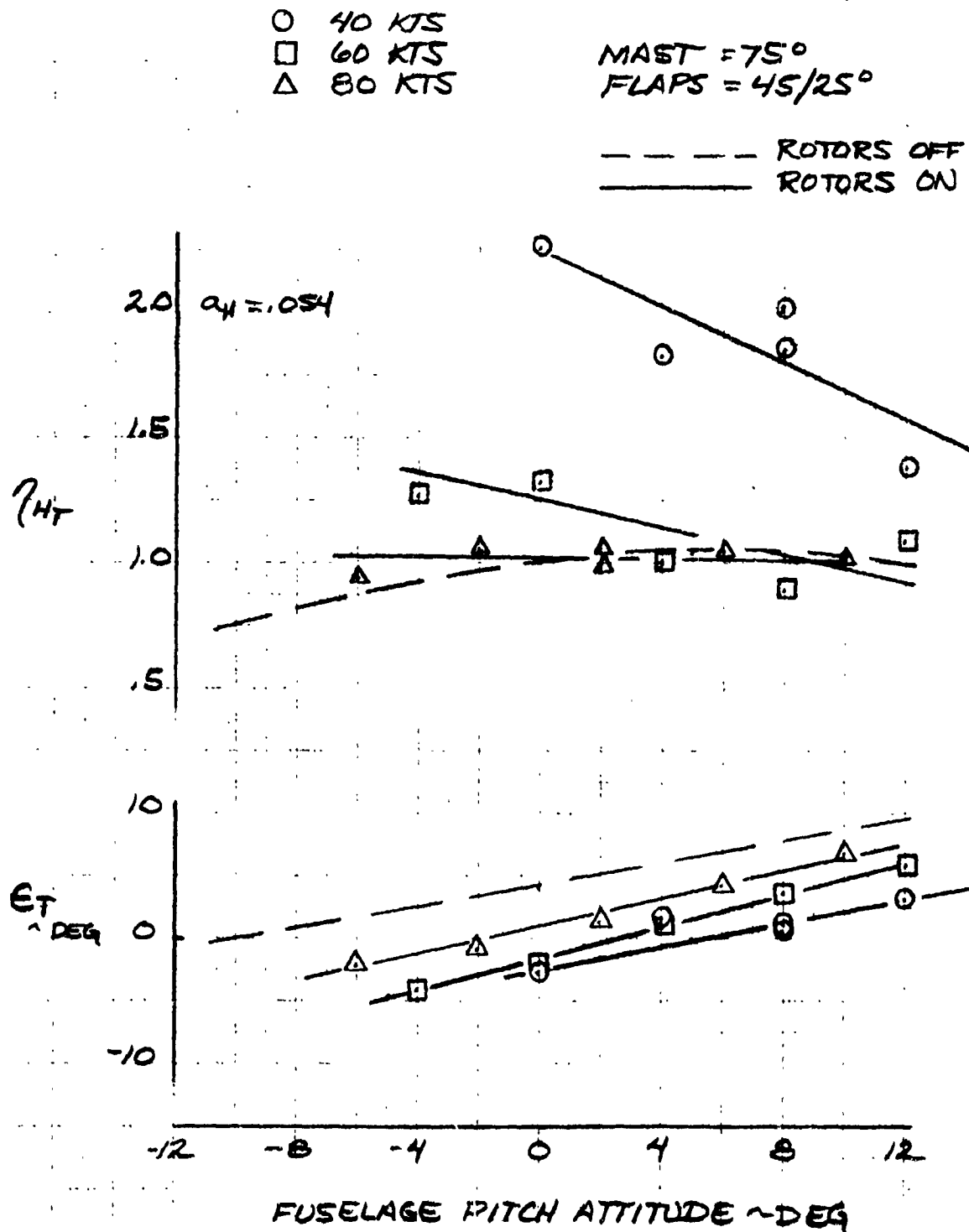
Figure VI-8. Concluded

HORIZONTAL STABILIZER CHARACTERISTICS

& DOWNWASH

$\frac{\delta e}{}$
O 80 KTS -4°
□ 100 KTS .7°

MAST = 60°
FLAPS = 45/25°

— — — ROTORS OFF
——— ROTORS ON

$a_N \eta_H \bar{V}_H = .050$

FUSELAGE PITCH ALTITUDE ~DEG
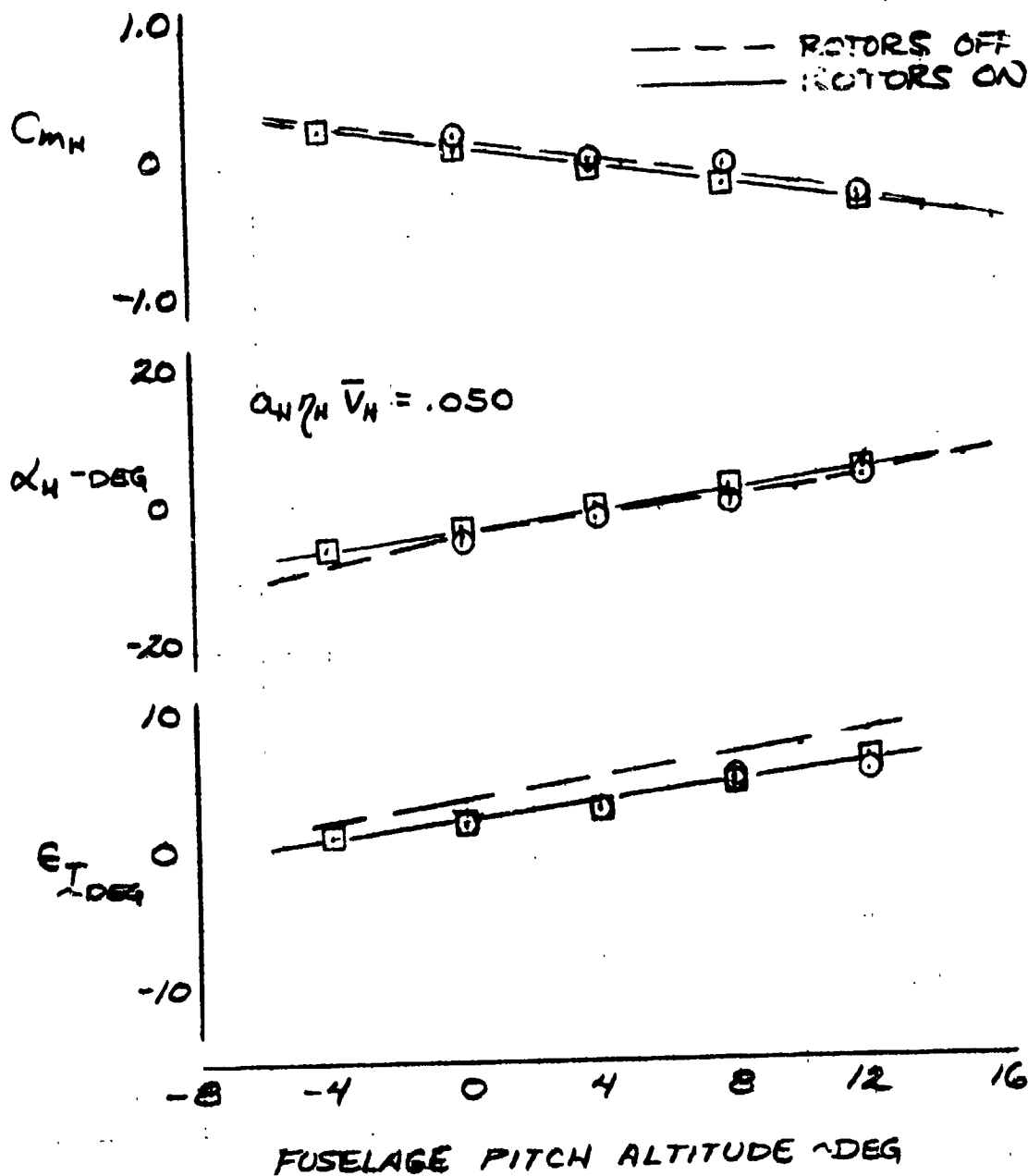
Figure VI-9. Horizontal Stabilizer Aerodynamic Characteristics, Mast Angle 60°, Rotors on.

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HORIZONTAL STABILIZER CHARACTERISTICS
& DOWNWASH

O 80 KTS $\frac{\delta_c}{-9°}$
□ 100 KTS $-4°$

MAST = 30°
FLAPS = 45/25°

— — — ROTORS OFF
——— ROTORS ON

$a_H \eta_N \bar{V}_H = .050$

FUSELAGE PITCH ATTITUDE ~ DEG
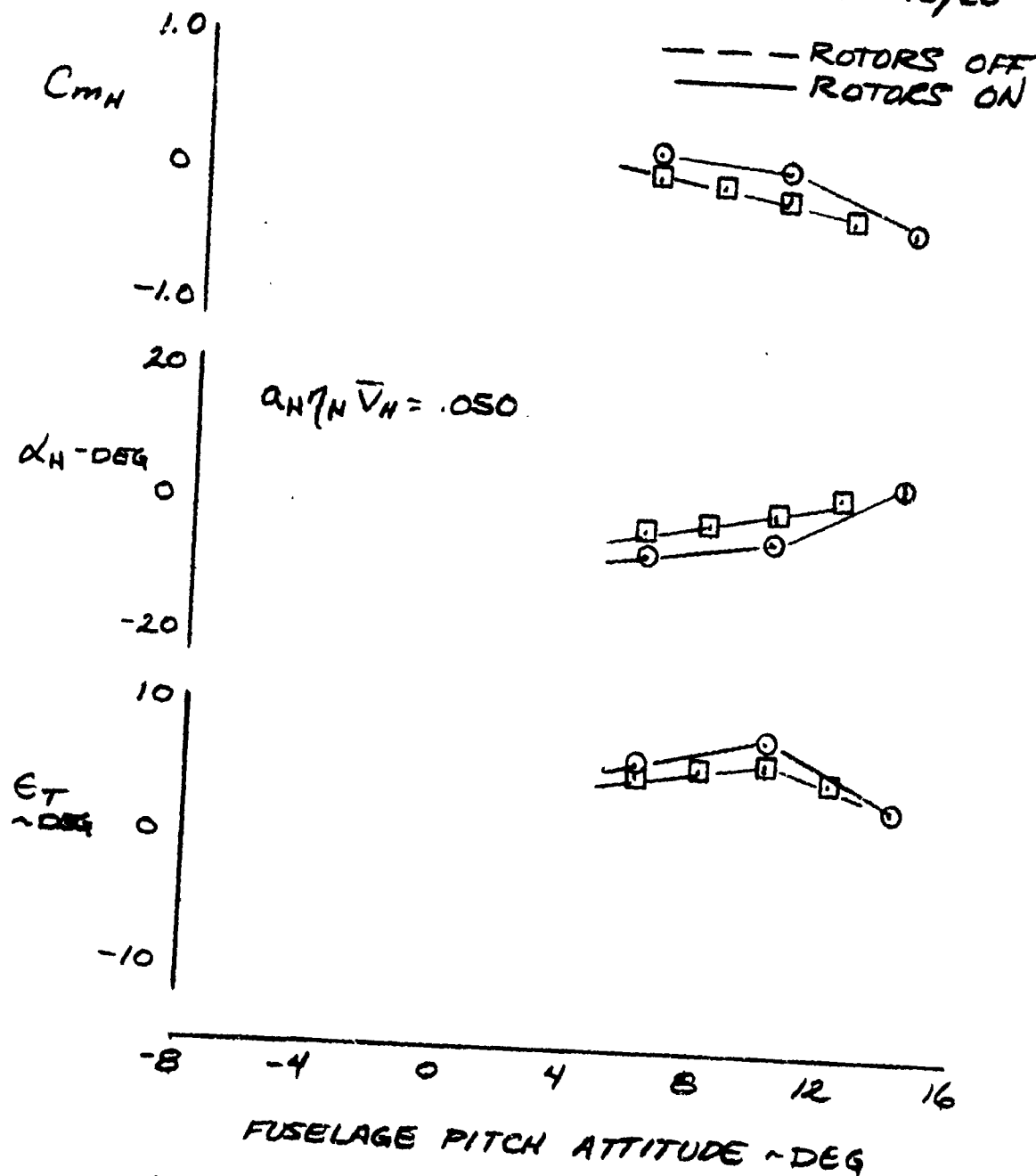
Figure VI-10. Horizontal Stabilizer Aerodynamic Characteristics, Mast Angle 30°, Rotors on.

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ROTOR WAKE AT HORIZONTAL TAIL

⬡ 20 KTS   MAST = 90°
○ 40 KTS   FLAPS = 45/25°
□ 60 KTS
△ 80 KTS

LEVEL FLIGHT PITCH SWEEP

● TRIM PITCH ATTITUDE
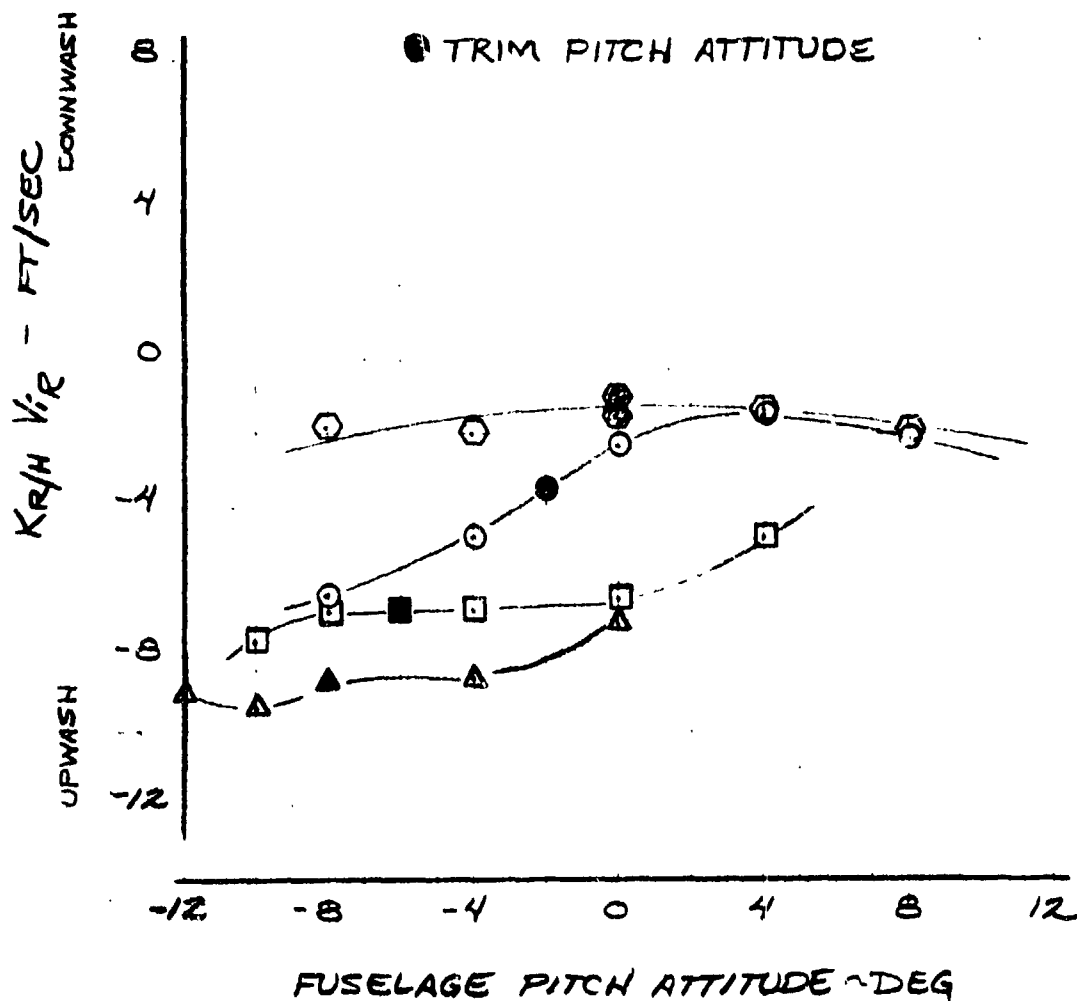
Figure VI-11. Rotor Upwash at the Horizontal Stabilizer versus Angle of Attack for Several Airspeeds, Mast Angle 90°.

ROTOR WAKE AT HORIZONTAL TAIL

O 40 KTS
□ 60 KTS
△ 80 KTS

MAST = 90°
FLAPS = 45/25°

POWER
● CLIMB $(\eta_N = 1.0)$
O LEVEL
◑ DESCENT $(\eta_N = 1.0)$
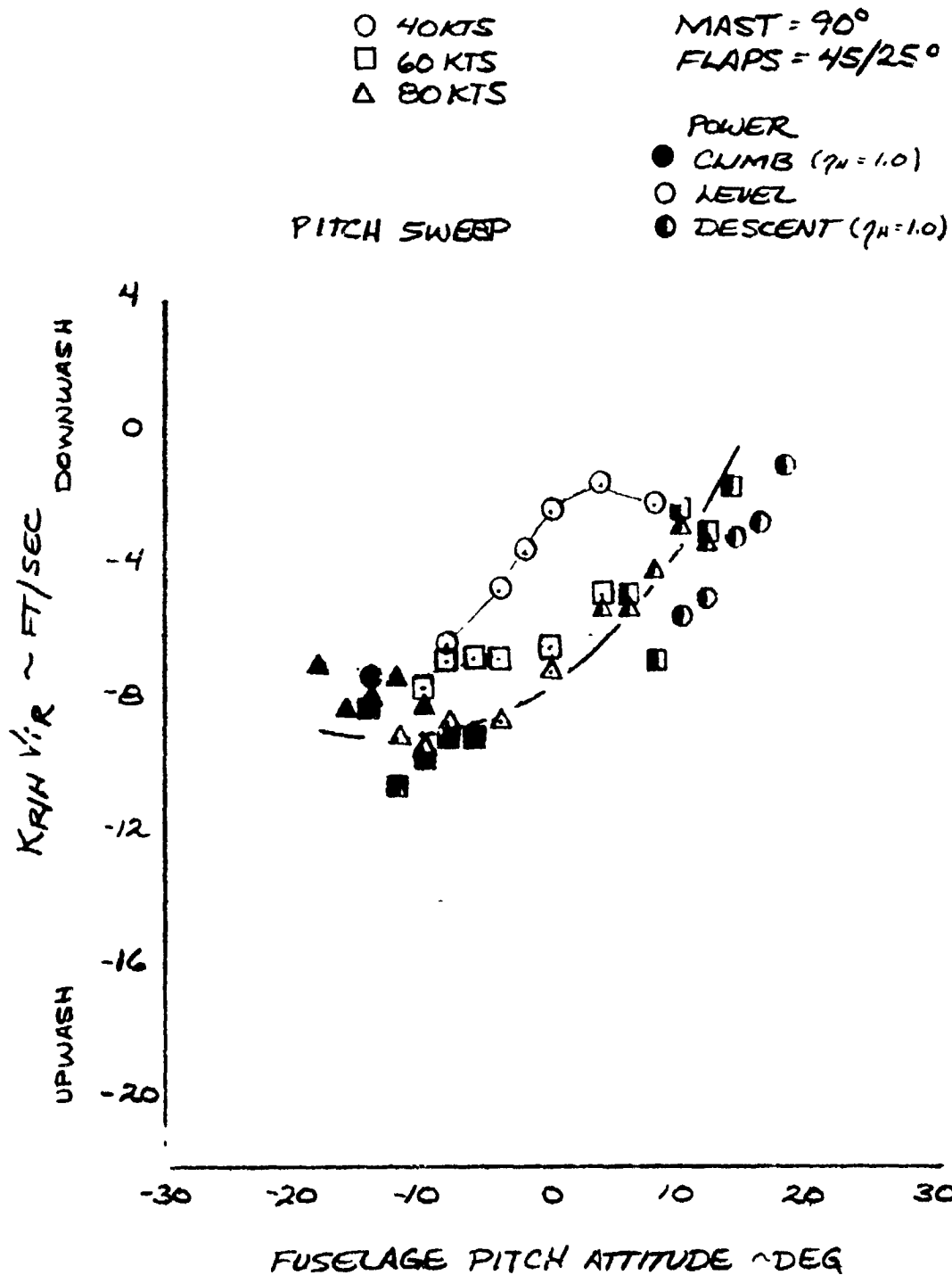
PITCH SWEEP

Figure VI-12. Rotor Upwash at Horizontal Stabilizer in
Climb and Descent Compared to Level Flight,
Mast Angle 90°.

ROTOR WAKE AT HORIZONTAL TAIL

○ 30 KTS
○ 40 KTS   MAST = 75°
□ 60 KTS   FLAPS = 45/25°
△ 80 KTS

LEVEL FLIGHT PITCH SWEEP

● TRIM PITCH ATTITUDE
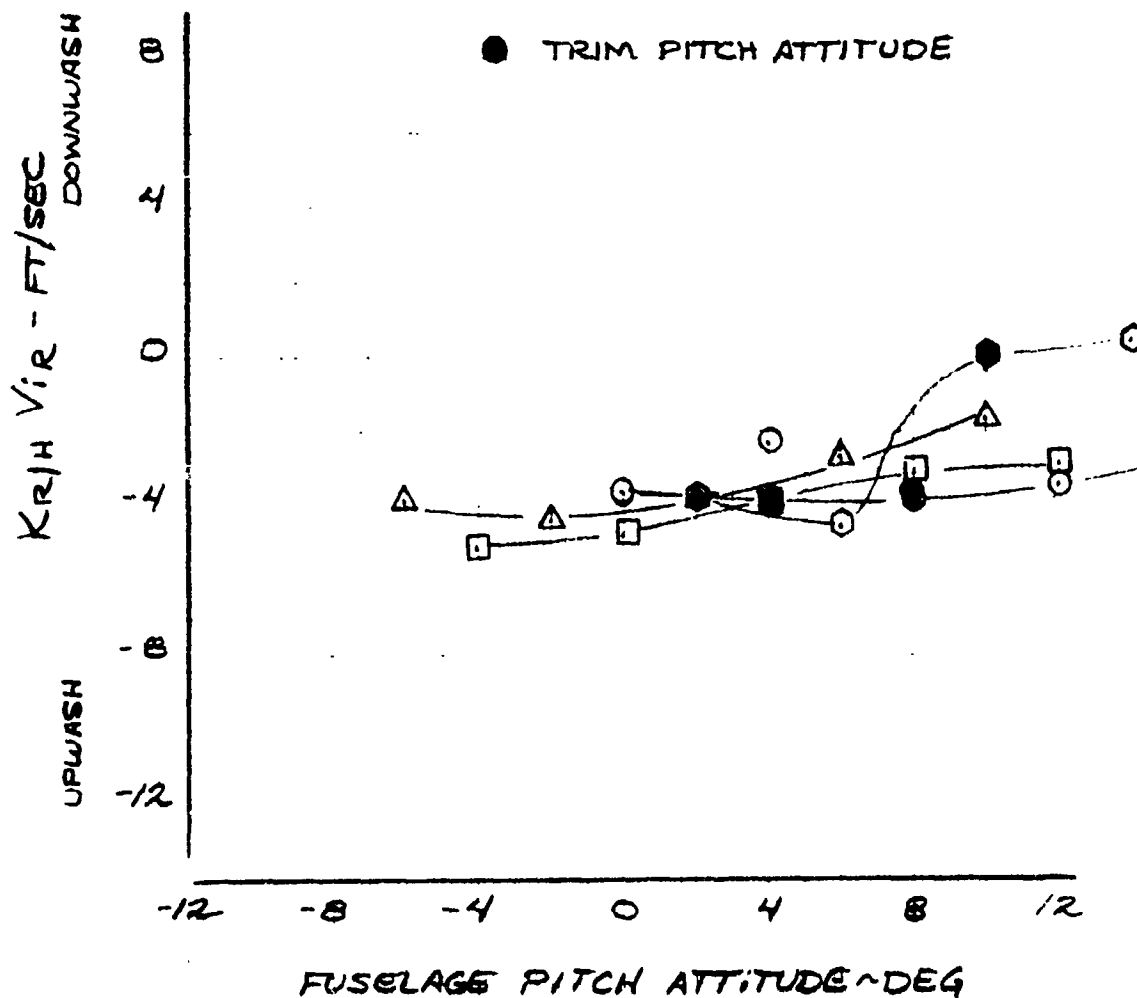
Figure VI-13. Rotor Upwash at Horizontal Stabilizer
versus Fuselage Angle of Attack,
Mast Angle 75°.

ROTOR WAKE AT HORIZONTAL TAIL

O 80 KTS   MAST = 60°
□ 100 KTS   FLAPS = 45/25°

LEVEL FLIGHT PITCH SWEEP
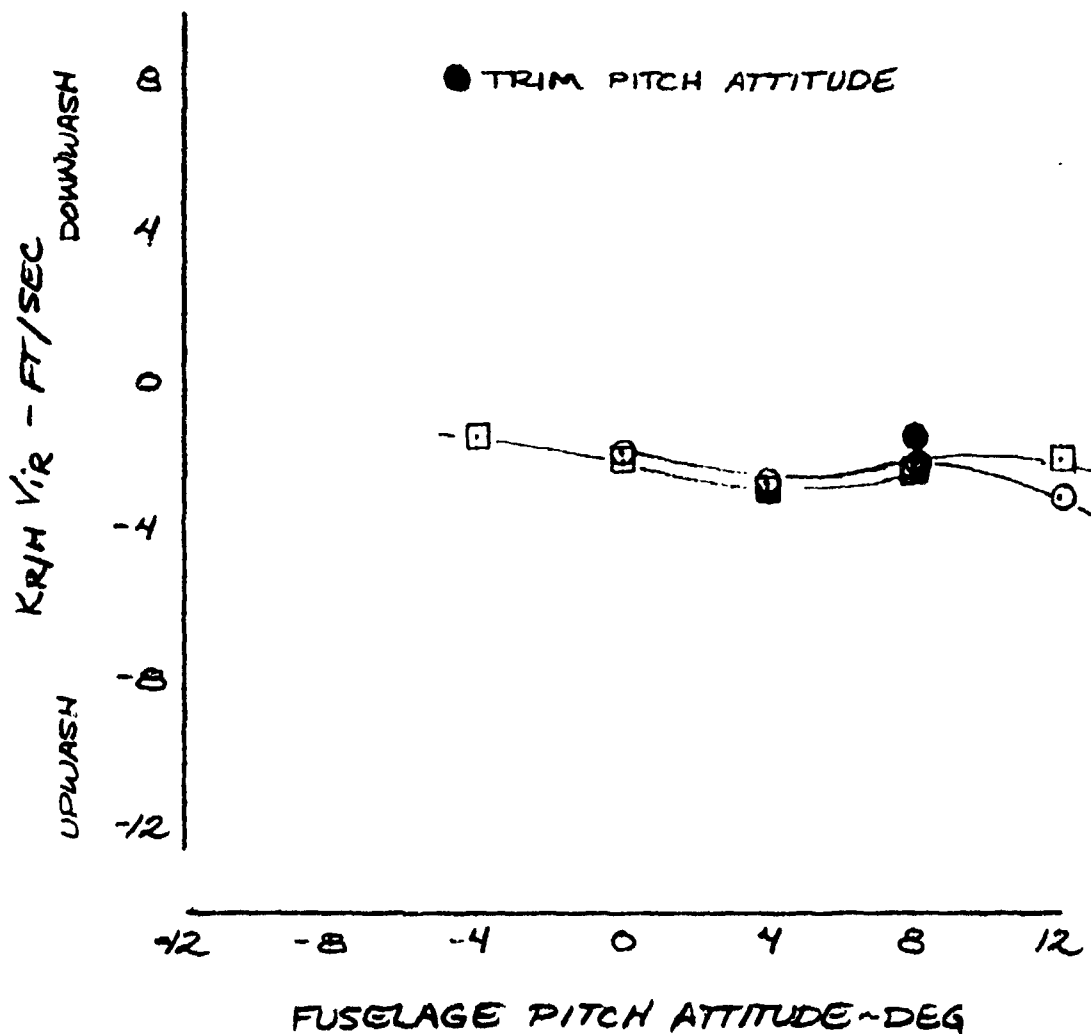
Figure VI-14. Rotor Upwash at Horizontal Stabilizer versus Fuselage Angle of Attack, Mast Angle 60°.

ROTOR WAKE AT HORIZONTAL TAIL

O 80 KTS   MAST = 30°
□ 100 KTS   FLAPS = 45/25°

LEVEL FLIGHT PITCH SWEEP

● TRIM PITCH ATTITUDE
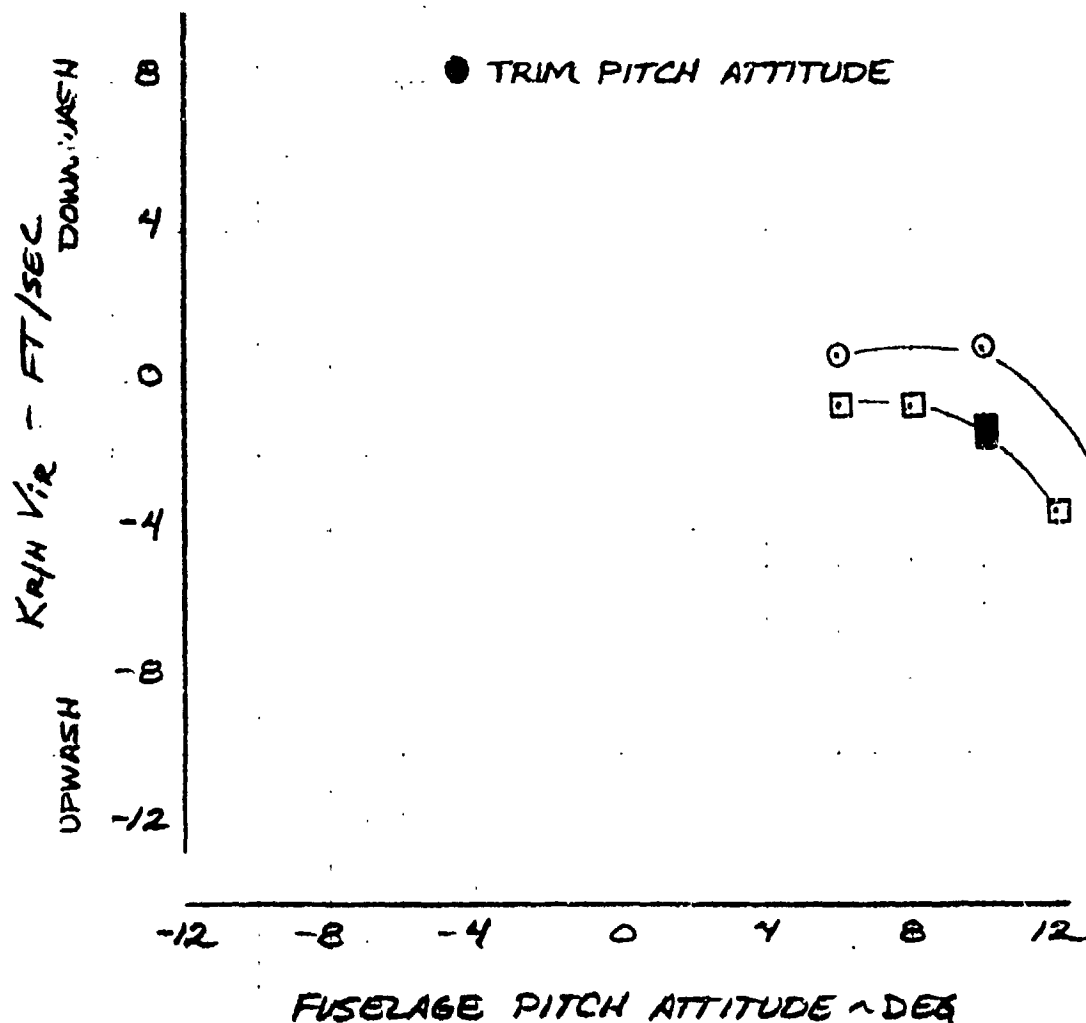
Figure VI-15. Rotor Upwash at Horizontal Stabilizer
versus Fuselage Angle of Attack,
Mast Angle 30°.

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ROTOR WAKE AT HORIZONTAL TAIL

	MAST ANGLE	USED IN PHASE I
○	90°	— — — —
□	75°	— · — · —
◇	60°	— — — —
△	30°	— x —
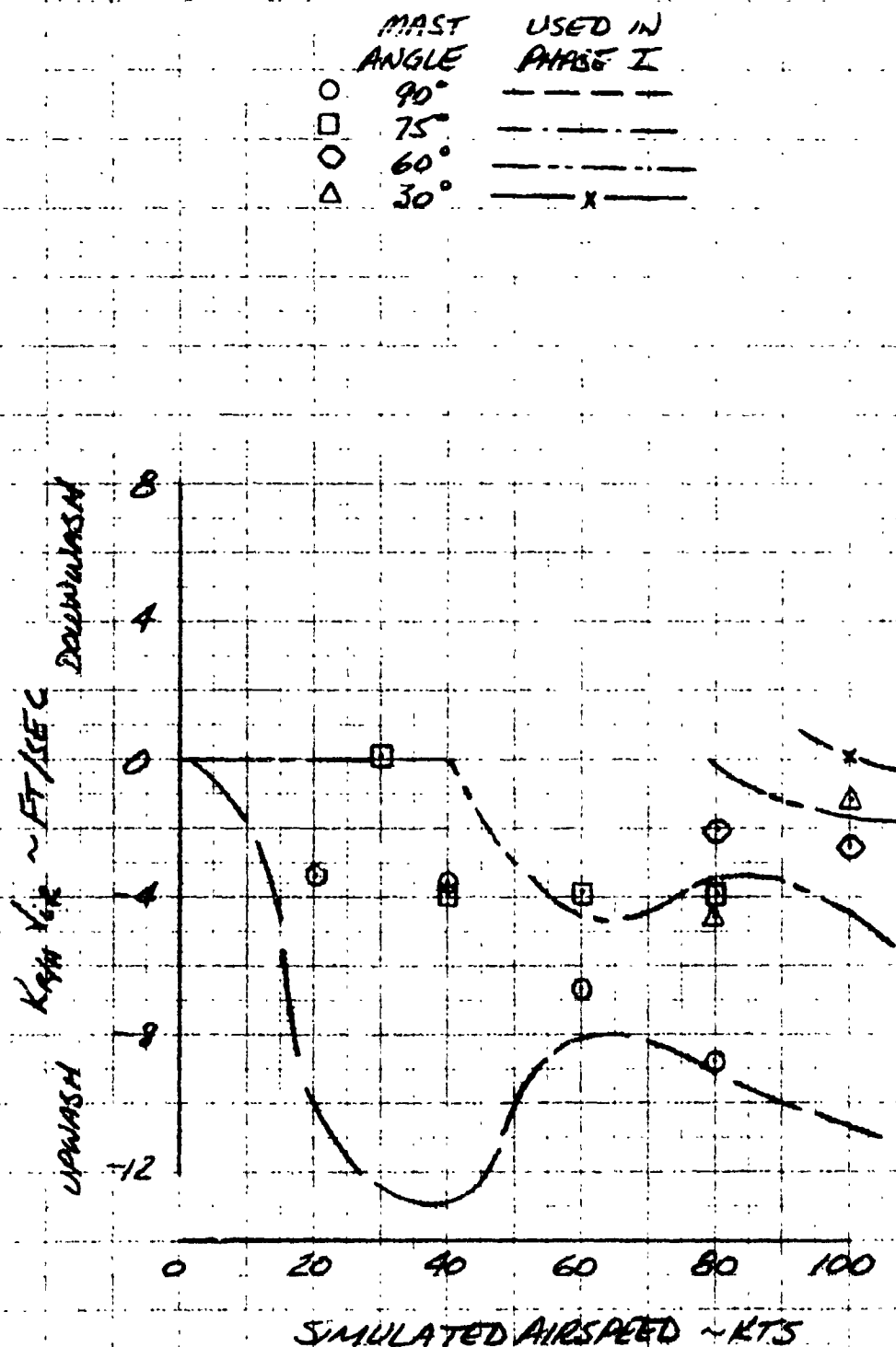
Figure VI-16. Comparison of Measured Upwash at
Horizontal Stabilizer with Upwash Used
in Phase I Study.

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ROTOR WAKE AT HORIZONTAL TAIL

		η_{HT}	
○	40 KTS	1.75	MAST = 90°
□	60 KTS	1.54	FLAPS = 45/25°
△	80 KTS	1.22	

YAW SWEEP LEVEL FLIGHT
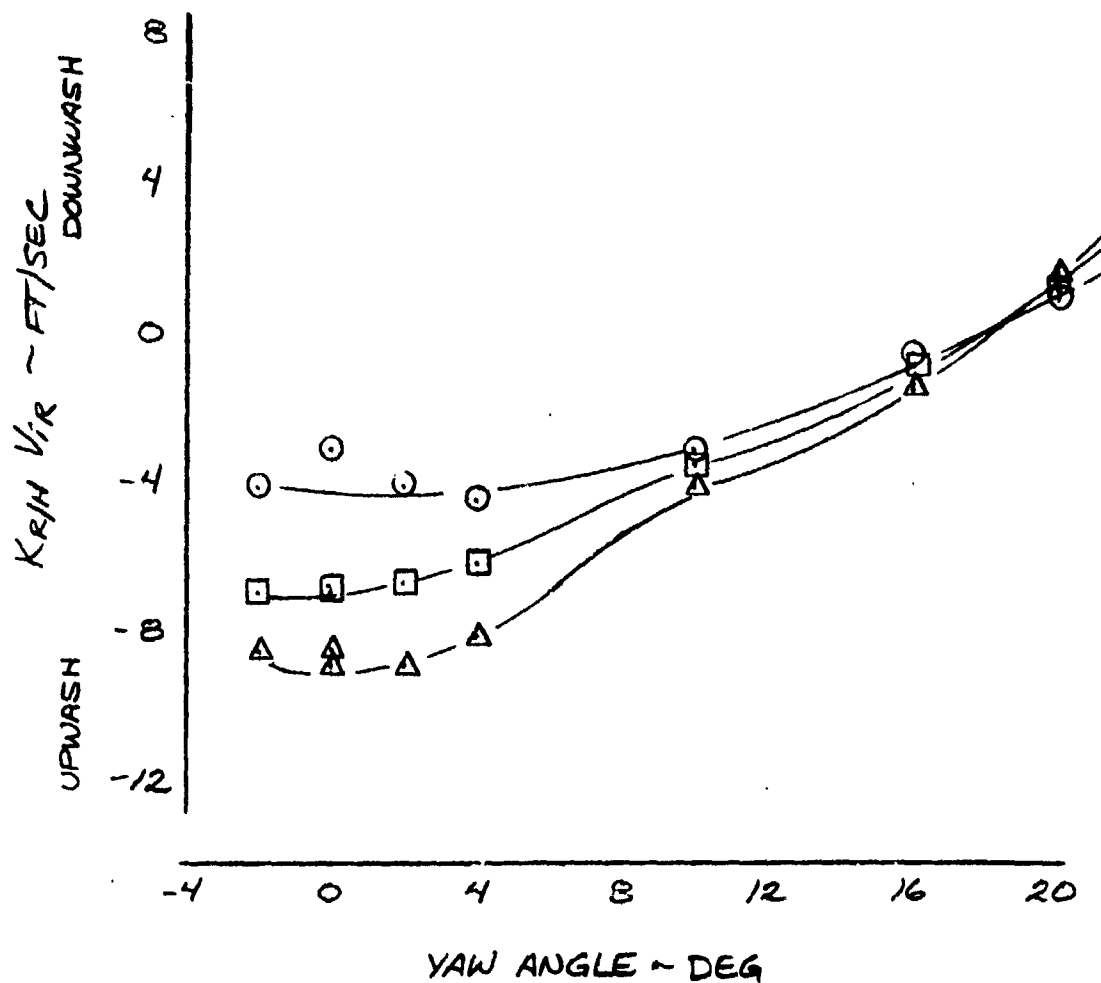
Figure VI-17. Effect of Yaw Angle on Rotor Wake Upwash at Horizontal Stabilizer, Mast Angle 90°.

ROTOR WAKE AT HORIZONTAL TAIL

		η_{HT}	
O	40 KTS	1.85	MAST = 75°
□	60 KTS	1.10	FLAPS = 45/25°
△	80 KTS	1.07	
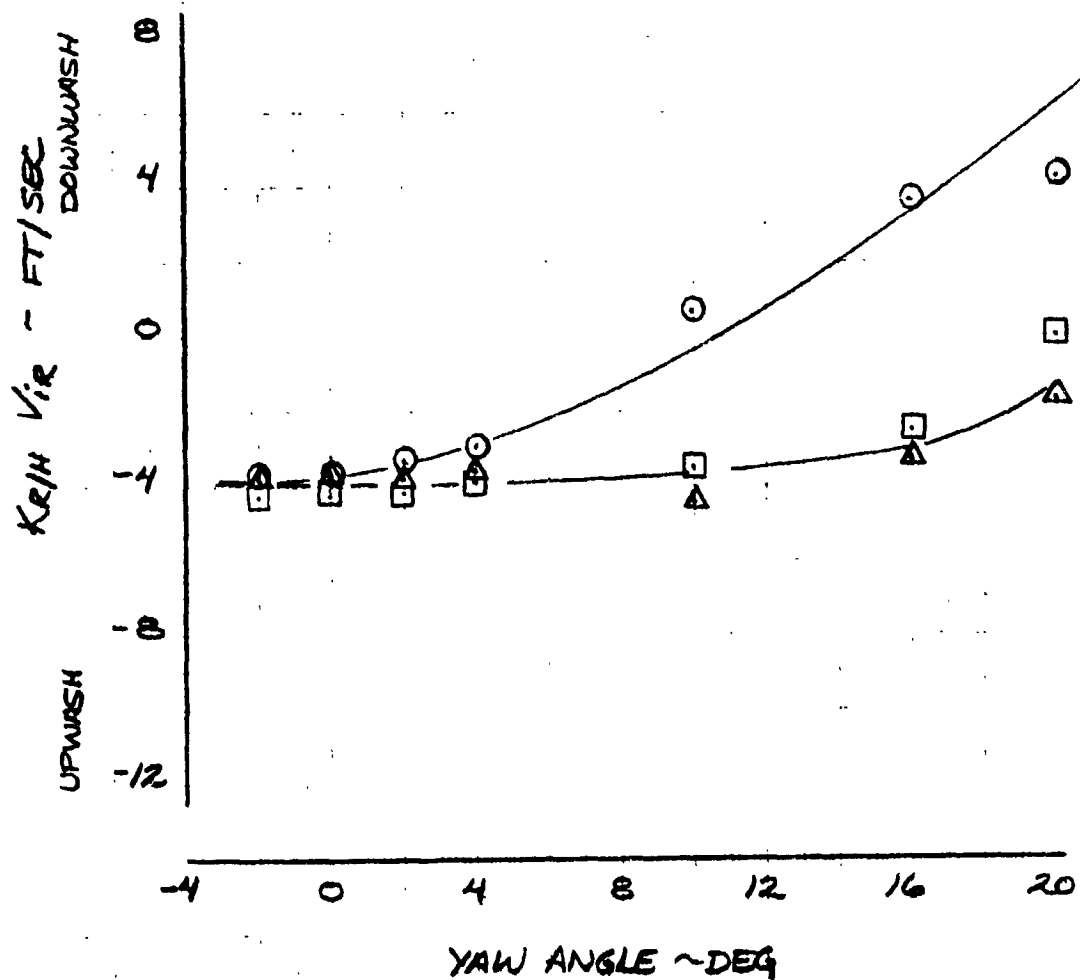
Figure VI-18. Effect of Yaw Angle on Rotor Wake Upwash at Horizontal Stabilizer, Mast Angle 75°.

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ROTOR WAKE AT HORIZONTAL TAIL

MAST 60° η_{HT}
  ○ 80 KTS .81 FLAPS = 45/25°
  □ 100 KTS .81

MAST 30°
  ○ 100 KTS .81

YAW SWEEP LEVEL FLIGHT
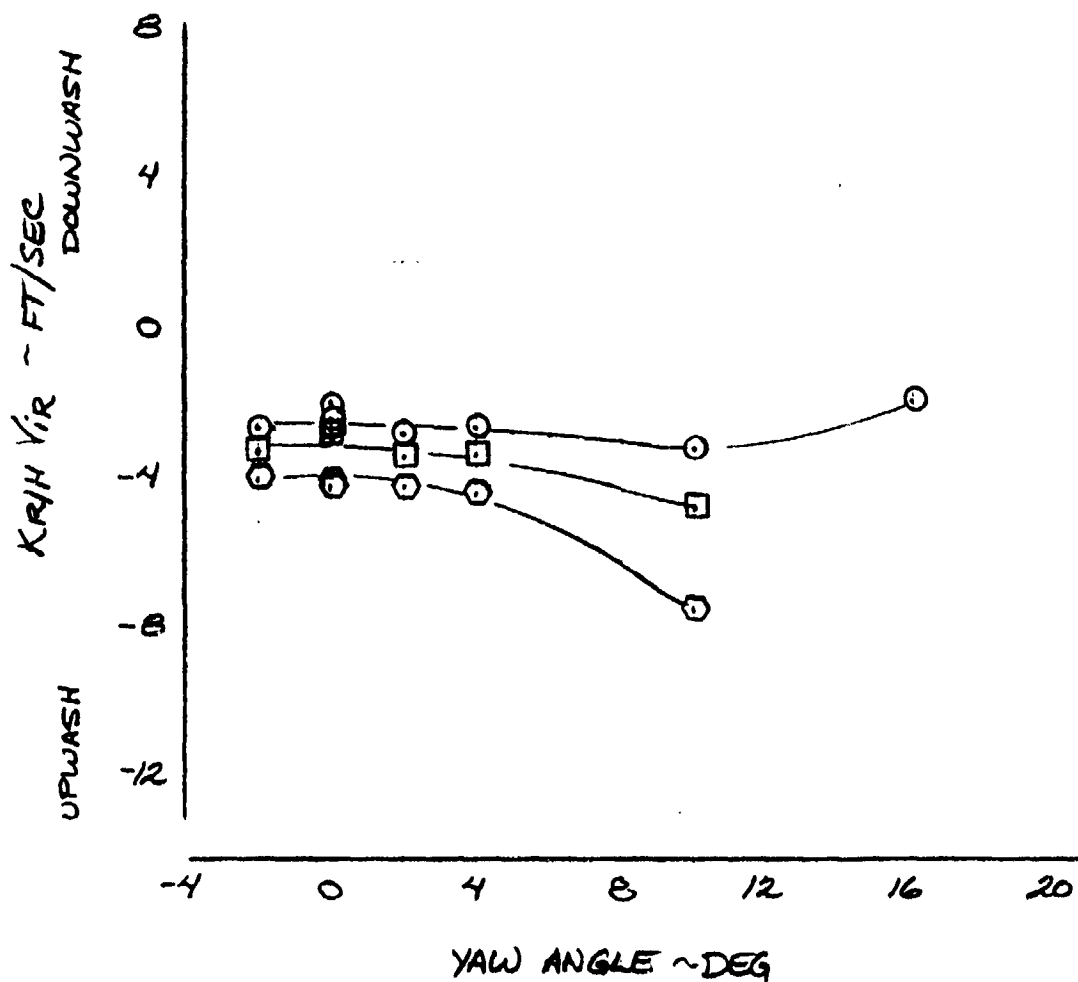
Figure VI-19. Effect of Yaw Angle on Rotor Wake Upwash
at Horizontal Stabilizer, Mast Angles 60°
and 30°.

ROTOR EFFECT ON DIRECTIONAL STABILITY

O 40 KTS   MAST = 90°
□ 60 KTS   FLAPS = 45/25°
△ 80 KTS

$$K_\beta = \frac{\text{FIN YAWING MOMENT, ROTORS-ON}}{\text{FIN YAWING MOMENT, ROTORS-OFF}}$$
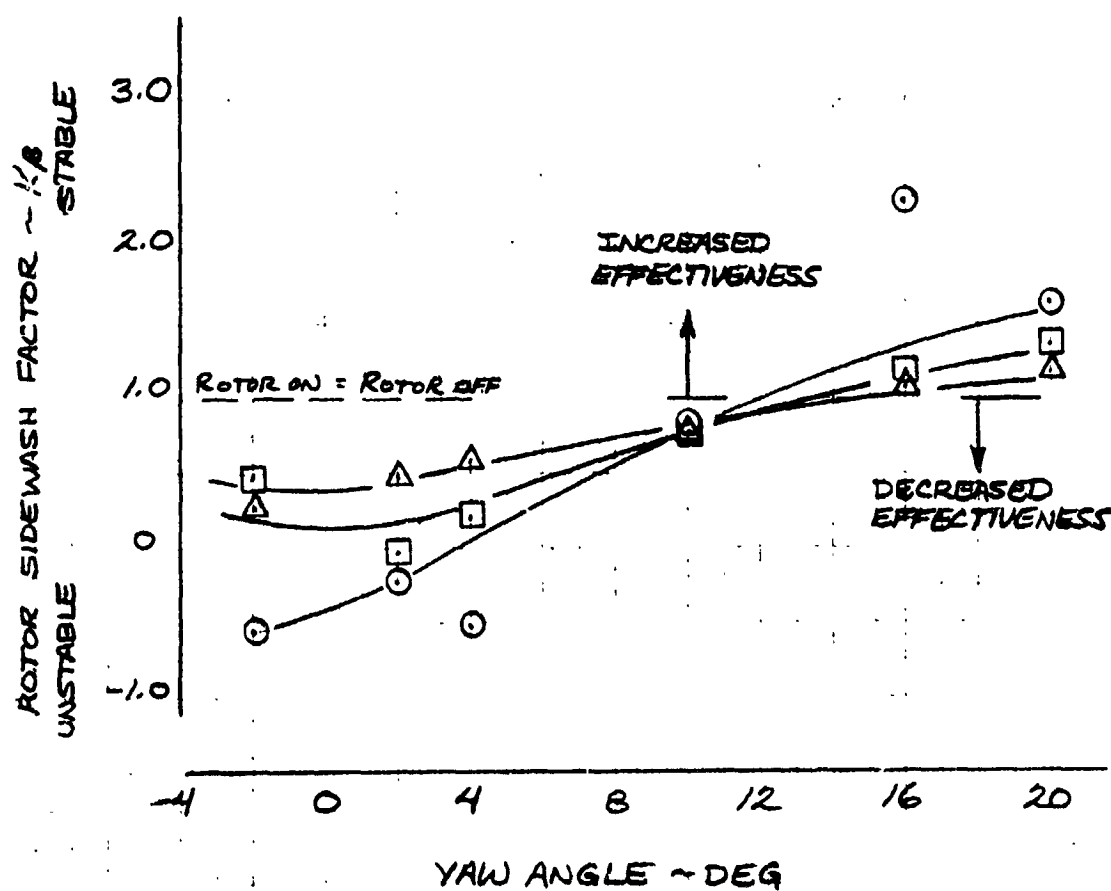
Figure VI-20. Effect of Rotor Wake on Directional
Stability, Mast Angle 90°.

ROTOR EFFECT ON DIRECTIONAL STABILITY

○ 40 KTS MAST = 75°
□ 60 KTS FLAPS = 45/25°
△ 80 KTS

$$K_\beta = \frac{\text{FIN YAWING MOMENT, ROTORS-ON}}{\text{FIN YAWING MOMENT, ROTORS-OFF}}$$
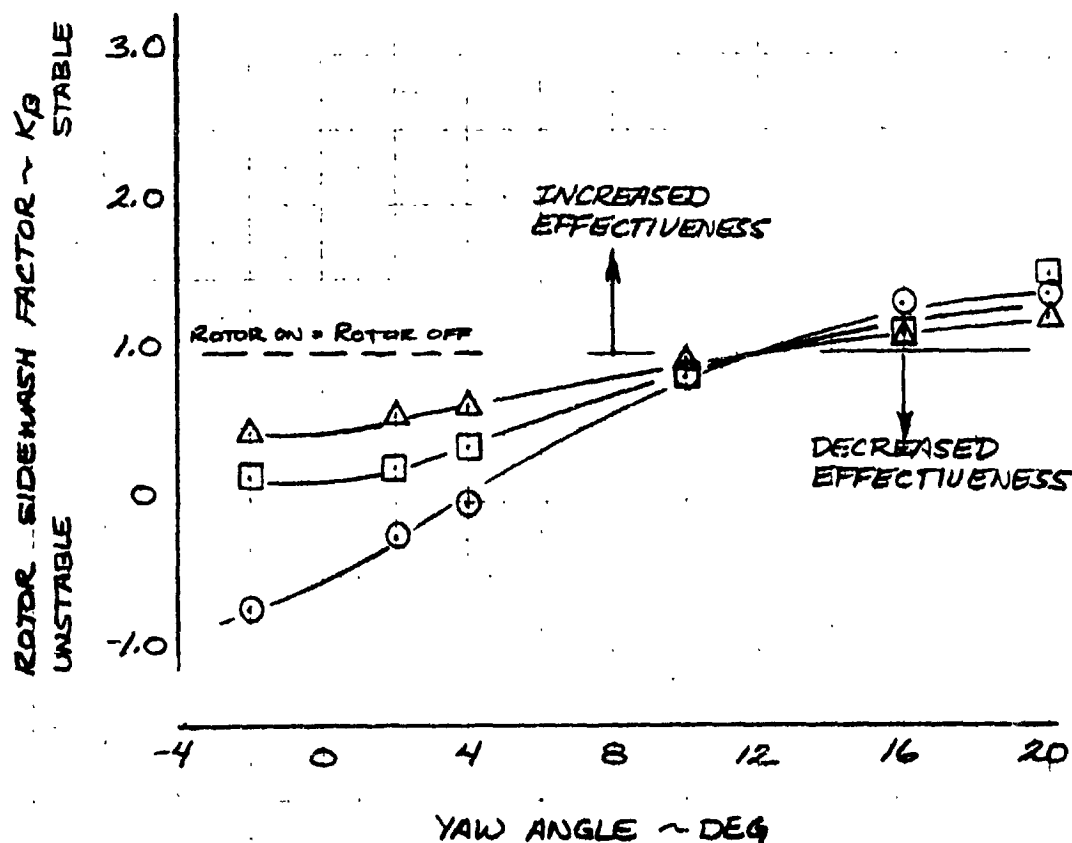
Figure VI-21. Effect of Rotor Wake on Directional Stability, Mast Angle 75°.

ROTOR EFFECT ON DIRECTIONAL STABILITY

MAST 60°
 ○ 80 KTS   FLAPS = 45/25°

MAST 30°
 □ 100 KTS

$$K_\beta = \frac{\text{FIN YAWING MOMENT, ROTORS-ON}}{\text{FIN YAWING MOMENT, ROTORS-OFF}}$$
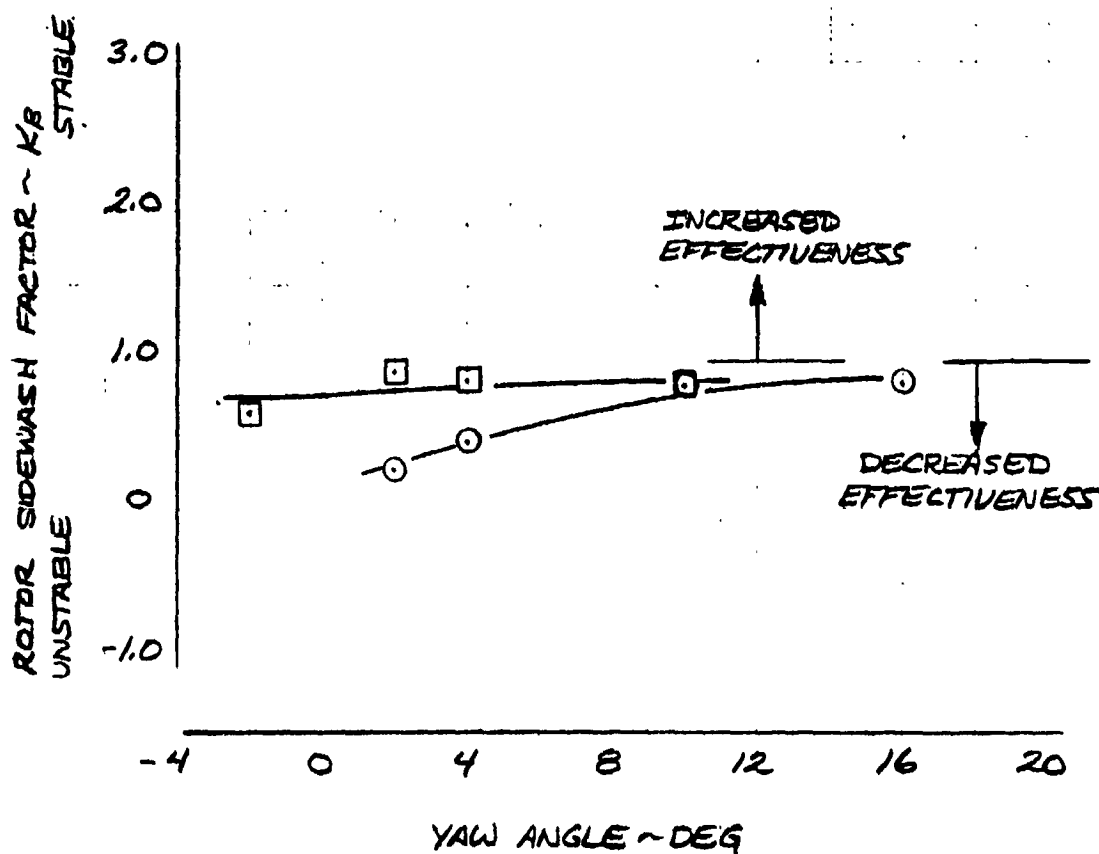
Figure VI-22. Effect of Rotor Wake on Directional
Stability, Mast Angles 60° and 30°.

EMPENNAGE LIFT

MAST = 90°
FLAPS = 45/25°
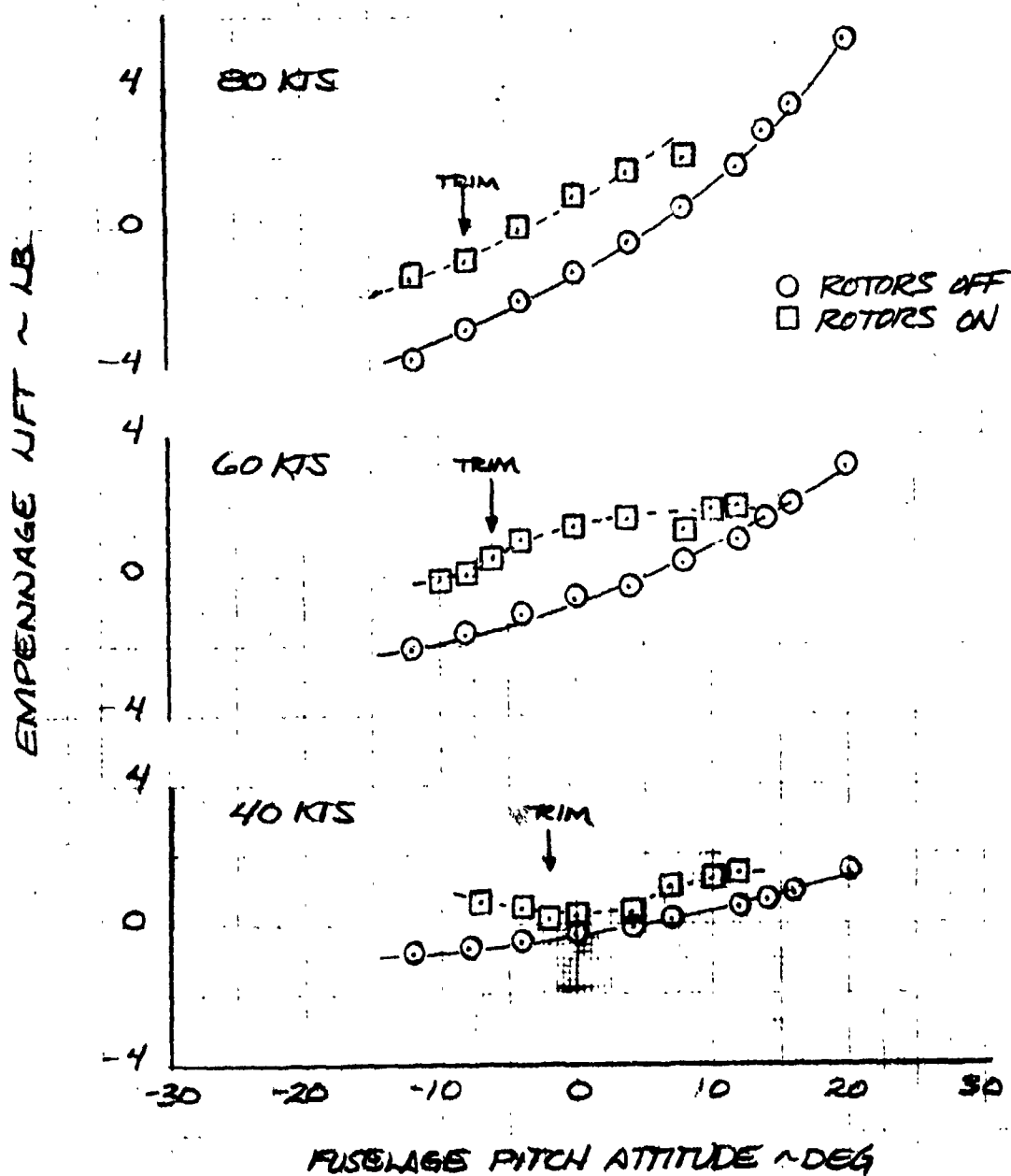
Figure VI-23. Comparison of Empennage Lift Rotors On
and Rotors Off. Mast Angle 90°.

EMPENNAGE LIFT

MAST = 75°
FLAPS = 45/25°
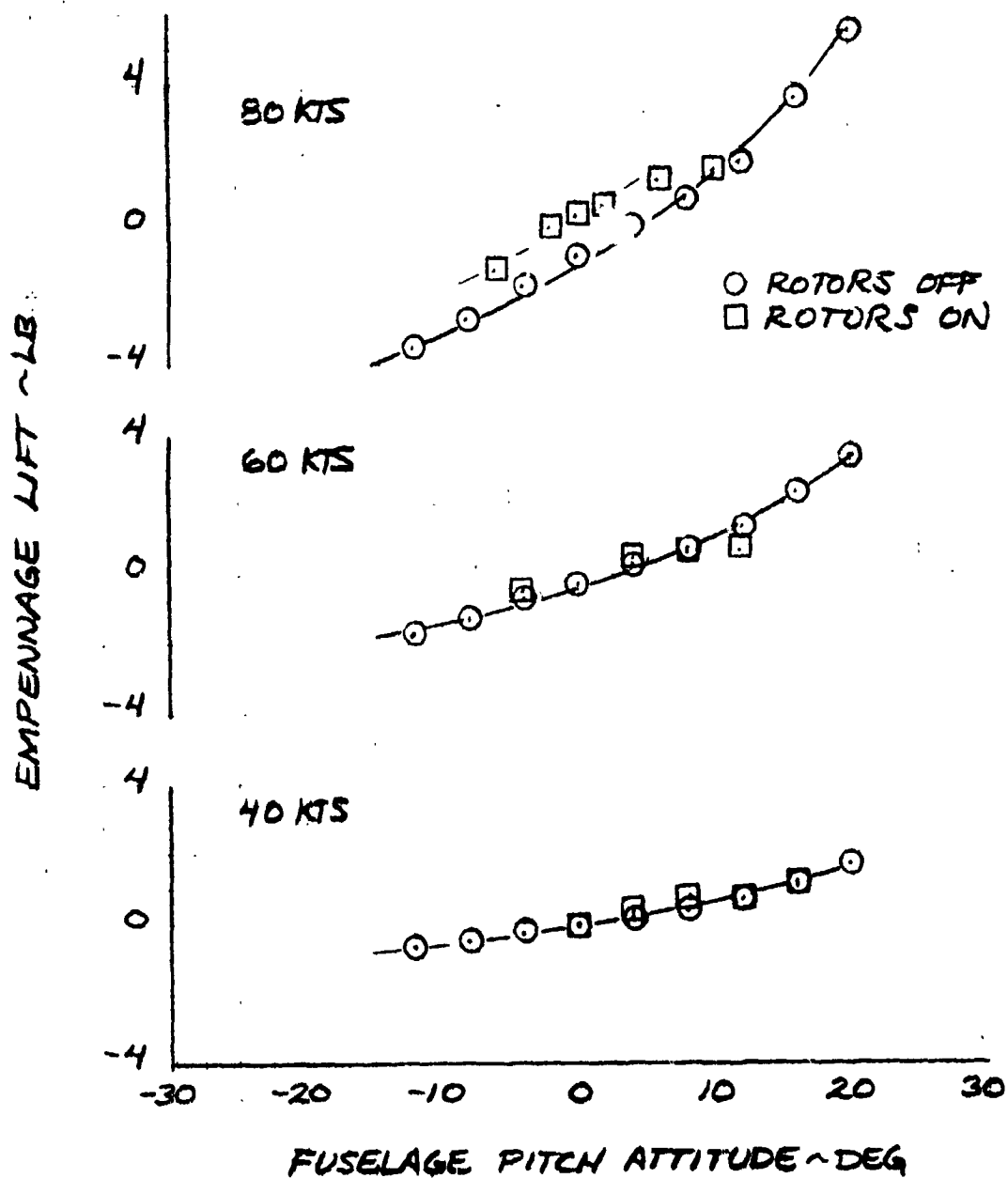
Figure VI-24. Comparison of Rotor Lift Rotors On
and Rotors Off, Mast Angle 75°.

EMPENNAGE LIFT

MAST 60°
FLAPS = 45/25°

○ ROTORS OFF
□ ROTORS ON
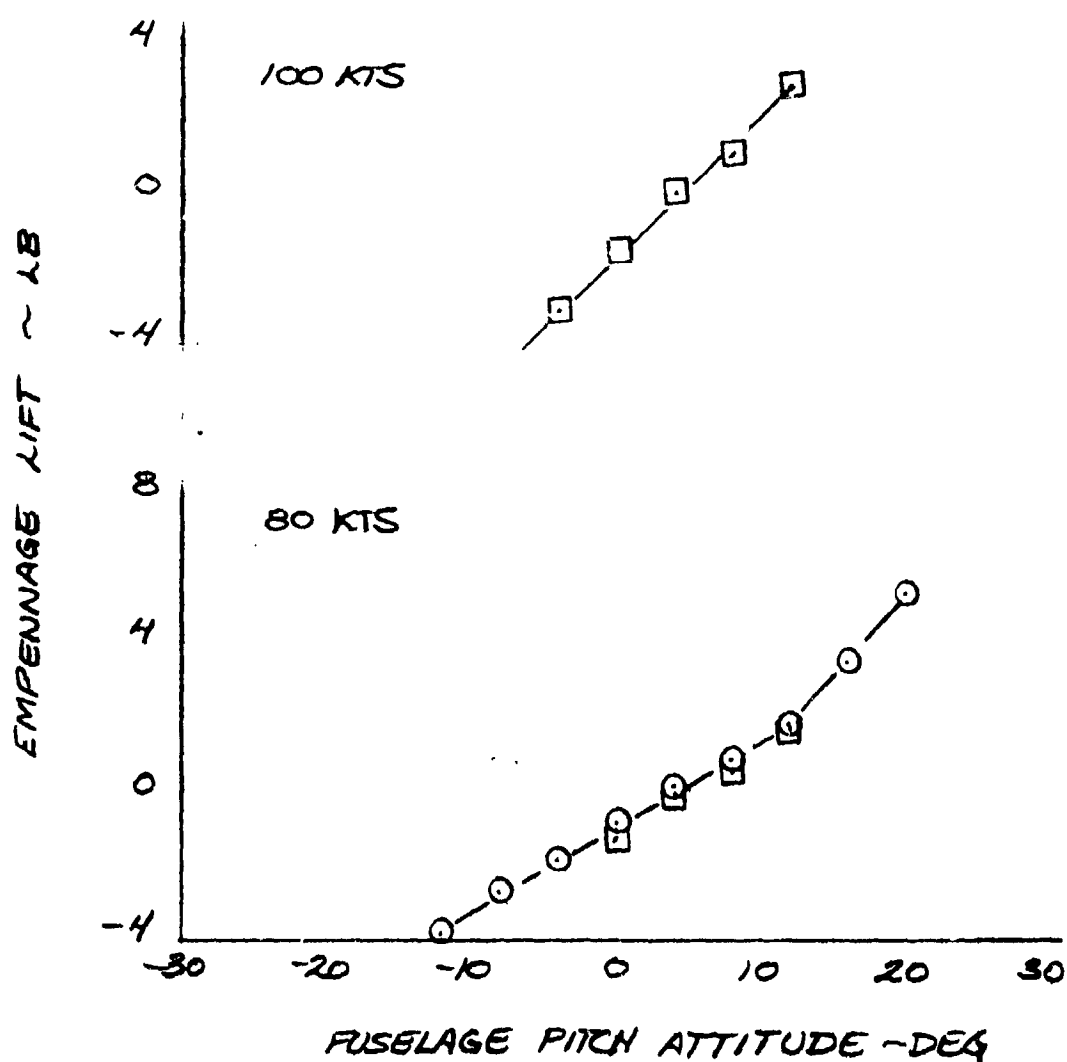
Figure VI-25. Comparison of Empennage Lift Rotors On
and Rotors Off, Mast Angle 60°.

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EMPENNAGE LIFT

MAST 30°
FLAPS = 45/25°

O ROTORS OFF
□ ROTORS ON
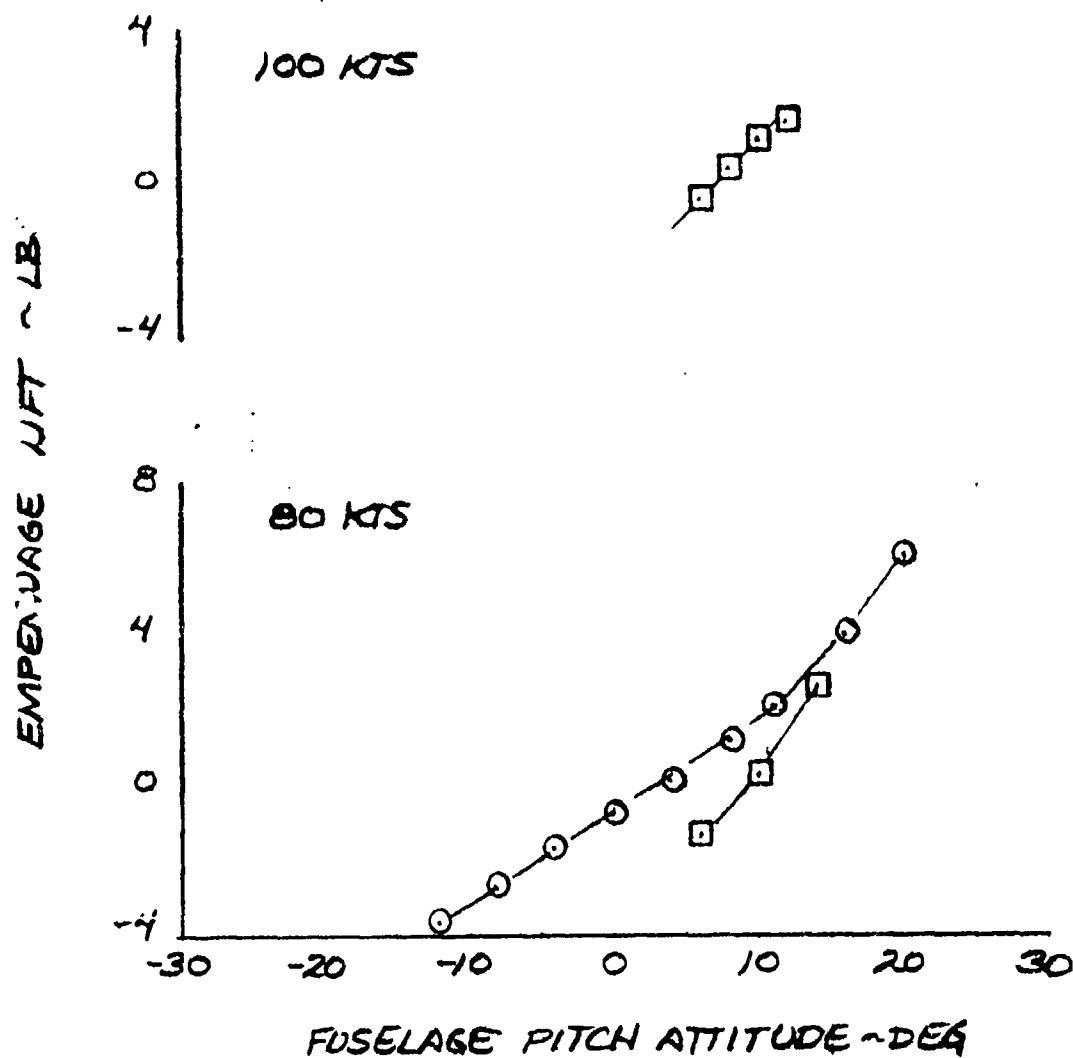
Figure VI-26. Comparison of Empennage Lift Rotors On
and Rotors Off, Mast Angle 30°.
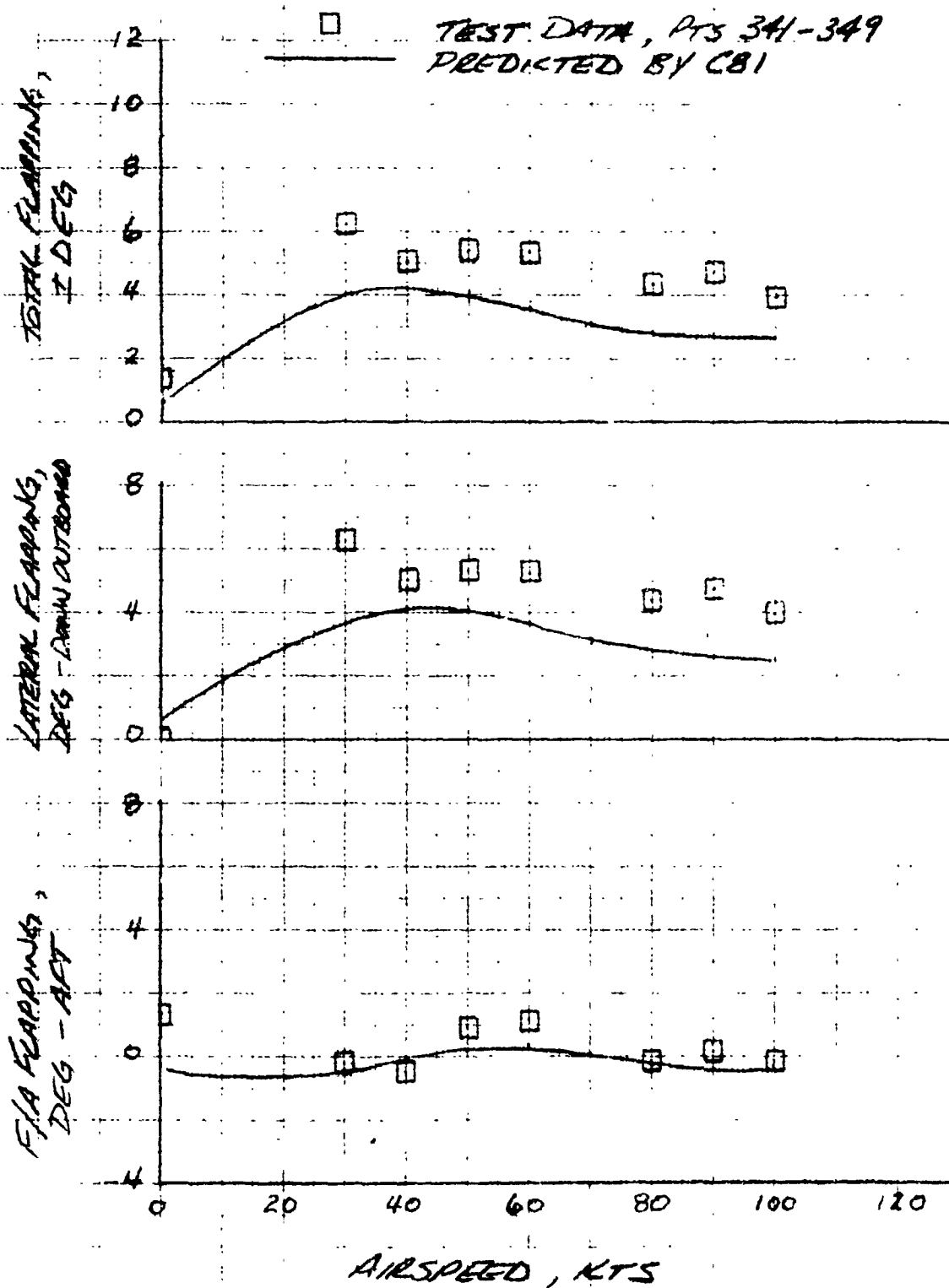
Figure VI-27. Comparison of Pretest Predicted Flapping
With Measured Flapping, Mast Angle 90°.

AIRSPEED 40 KNOTS
GROSS WEIGHT - 11,000 LB

--- --- C81 WITH K = 1.20 (PRETEST)
----- C81 WITH K = 1.62 (REQ'D TO CORRELATE)
——— OBSERVED DISTRIBUTION (APPROXIMATE)
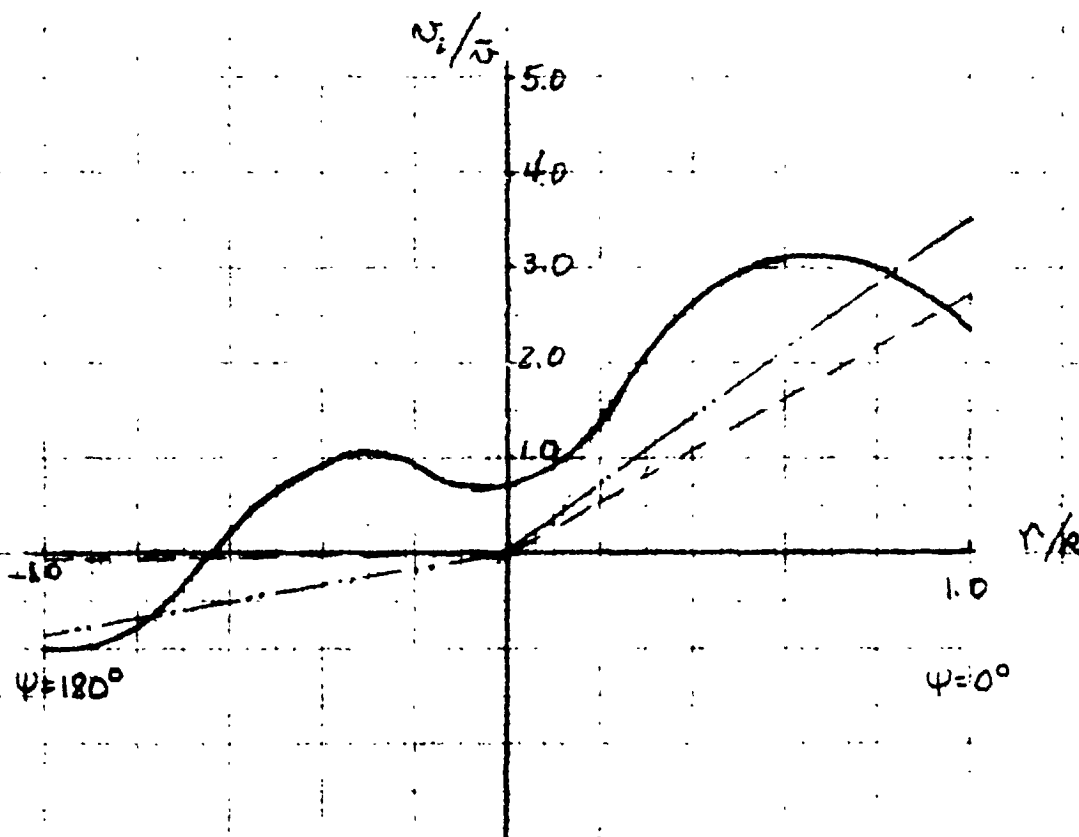
Figure VI-28. Comparison of Computer Program and Observed Longitudinal Distribution of Rotor Induced Velocity.

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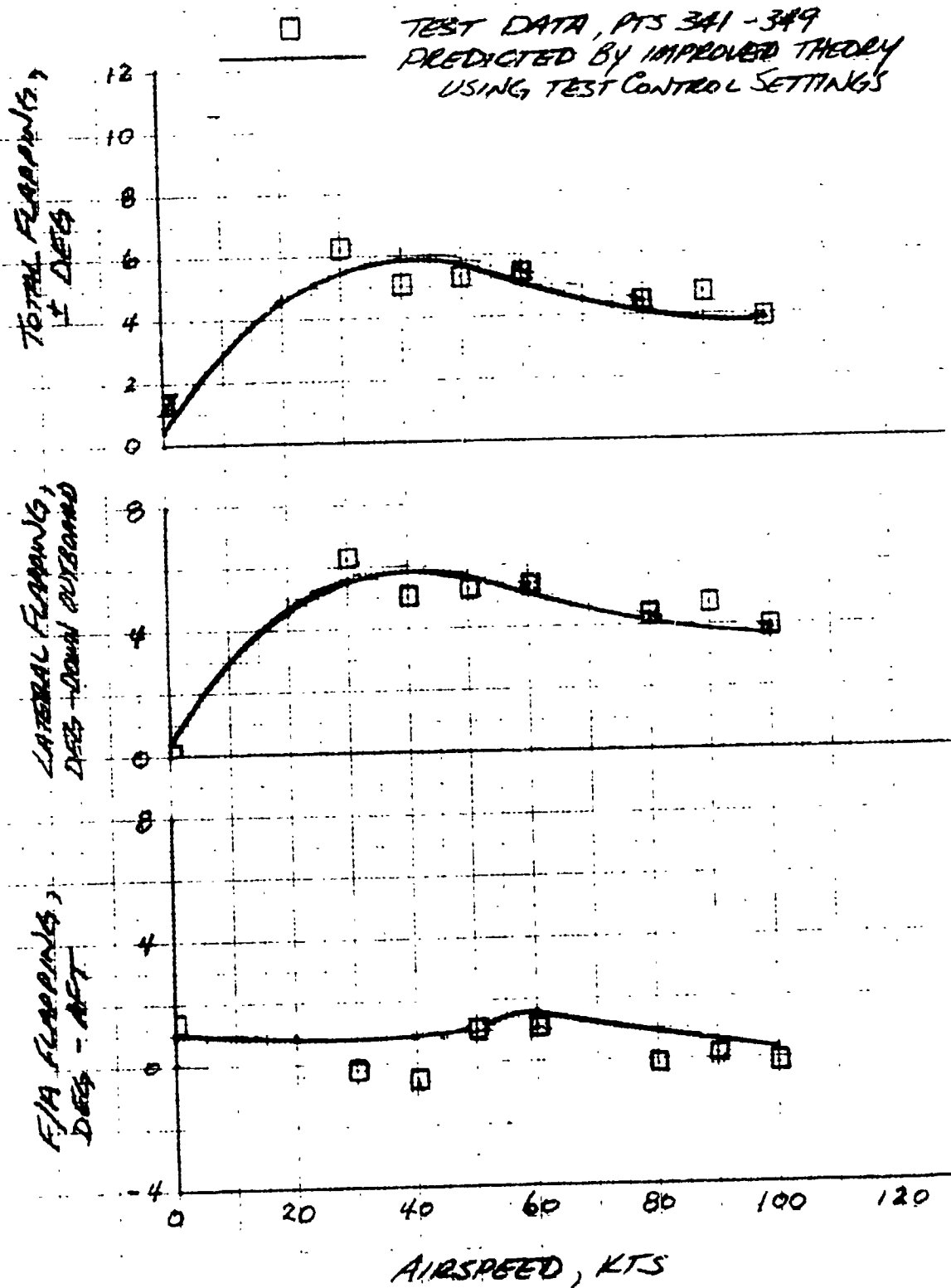
Figure VI-29.   Comparison of Post Test Predicted Flapping
With Measured Flapping, Mast Angle 90°.

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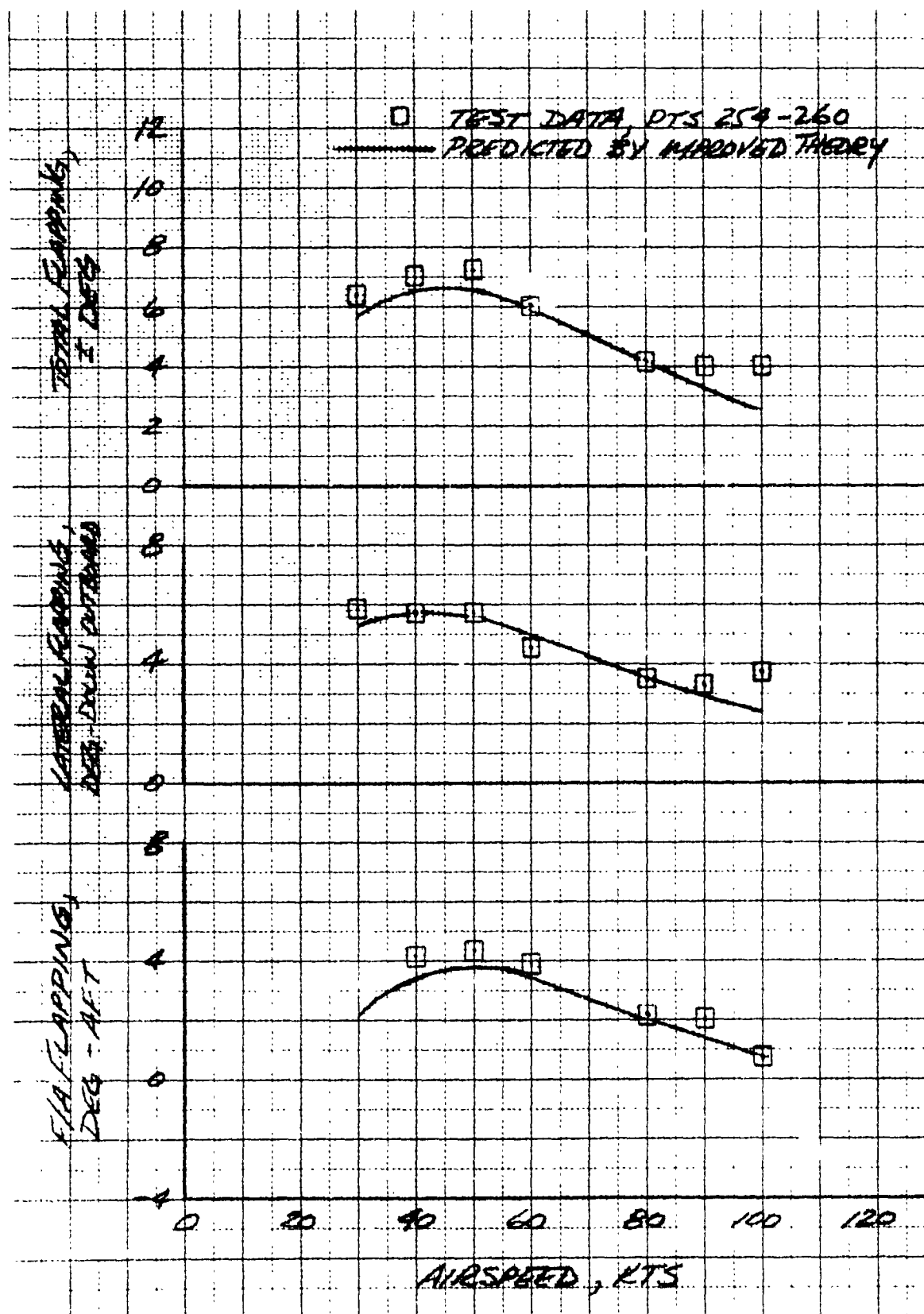
Figure VI-30. Correlation of Post Test Predicted Flapping with Measured Flapping, Mast Angle 75°.
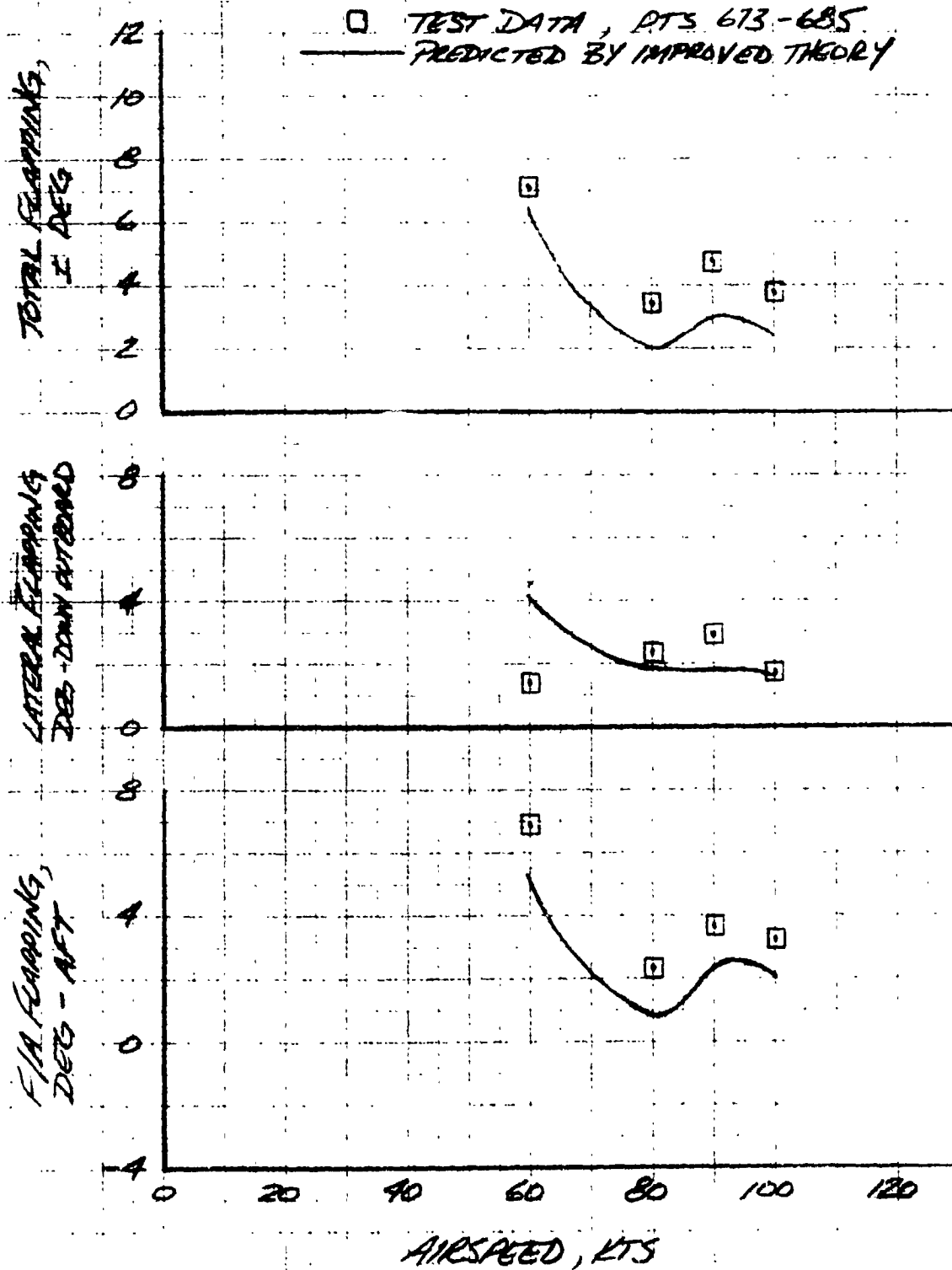
Figure VI-31. Comparison of Post Test Predicted
Flapping with Measured Flapping,
Mast Angle 60°.

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Use or disclosure of data on this page is subject to the restriction on the title page.
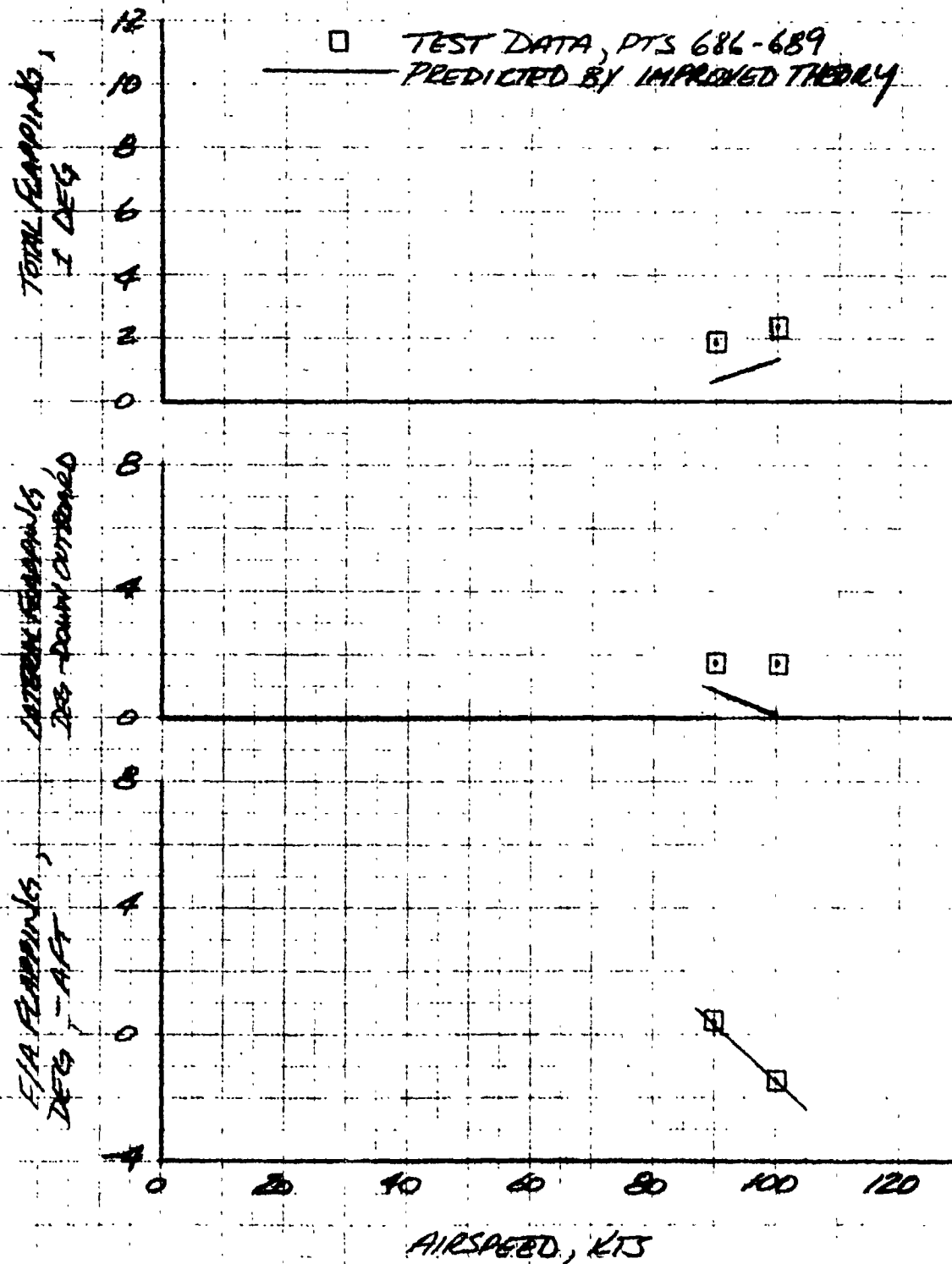
Figure VI-32. Comparison of Post Test Predicted Flapping with Measured Flapping, Mast Angle 30°.

VII. CONCLUSIONS

Observation of model behavior and analysis of measured data re-
sulted in the following conclusions:

A. Hover

1. The hover test confirmed the presence of a static roll
   instability in an IGE hover. The magnitude of the
   instability was comparable to previous estimates and the
   instability vanishes at airspeeds above approximately
   20 knots.

2. The wing download in hover was greater than that estimated
   from previous test results. The measured download was
   11.4 percent compared to the 7 percent estimated for the
   full-scale aircraft. Analysis indicates this discrepancy
   may be the result of the model's very low Reynolds
   number.

3. Attempts to fly the model in a controlled hover on an
   essentially free-flight mount were not successful.
   Failure is attributed to the model's response being
   faster than full-scale by a factor of 2.24, and because
   the model pilots lacked many of the cues normally avail-
   able in hover.

B. Low Speed Helicopter and Conversion Mode

1. Flow visualization techniques confirmed that the wakes
   from the two rotors do not merge and analysis of balance
   data indicate that the rotor wake has the effect of an
   upwash on the horizontal stabilizer.

2. At mast angles of 90 and 75 degrees, a shallow longitu-
   dinal stick gradient occurs between hover and 60 knots
   and a slight stick reversal occurs between 20 and 40
   knots. The shallow gradient and reversal are caused by
   the rotor wake influence on the horizontal stabilizer.

3. The interaction between the rotor wake and the horizontal
   stabilizer causes a nose-up pitching moment when the air-
   craft is yawed. The longitudinal cyclic required to
   compensate for the pitching moment is easily within a
   pilot's ability. With the SCAS on, the altitude-hold
   loop will make the required cyclic input.

4. The rotor wake acts on the vertical fins in such a manner
   as to reduce directional stability for sideslip angles

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4. (Continued)

less than about 12 degrees and to increase it for larger
sideslip angles.

5. In autorotation, at the speeds tested (80 and 90 knots),
the model was very stable but had a rate of descent in
excess of 4000 fpm, compared to the 2200 fpm predicted
for the Model 301. Analysis indicates that the model's
low Reynolds number is responsible for its higher rates
of descent and that the rate of descent predicted for
the Model 301 is considered a reasonable estimate.

6. On the rod, the model was difficult to fly at speeds
below 30 knots but was controllable. Above 30 knots,
the model was relatively easy to fly. Controllability
was adequate for rearward and sideward flight at speeds
up to 35 knots.

7. Lateral flapping in helicopter mode was approximately
40 percent higher than predicted, but was within flapping
limits for all conditions tested. Analysis shows that
the theory used for the pretest predictions employed a
representation of the longitudinal distribution of induced
velocity which was not representative of the observed in-
duced velocity distribution. By modifying the distribution
good correlation with the measured flapping is achieved.

C. Dynamic Stability, Loads, and Vibration

1. There was no evidence of rotor or rotor-pylon-wing
instability during the tests.

2. Scaled rotor and control system loads were significantly
lower than those predicted for the full-scale aircraft.

3. Airframe vibration levels were higher than predicted but
within design limits. The magnitude of the two-per-rev
vibration indicates care will have to be taken to avoid
resonance of airframe modes with two-per-rev.

D. Correlation of Theory With Measured Data

1. When Reynolds number effects are properly accounted for,
theory adequately predicts model trim parameters and
stability characteristics. (Extrapolation of model
flight characteristics directly to full scale is not
recommended because of Reynolds number effects.)

2. Theory predicts the measured lateral flapping when the
rotor longitudinal distribution of induced velocity is
correctly represented.

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APPENDIX A

Run Schedule Summary

Run Schedule Summary
Model C300-12B Aeroelastic Model – Hover Test (α_M = 90°)

RUN No.			FLAP ANGLE (DEG)	Model Weight (LBS)	h/D	WING Panel	Comments
1			0/0	100	.5 to 1.0	ON	Hover Performance
2			"	90	"	"	"
3			"	75	"	"	"
4			"	50	"	"	"
5			"	75	1.0	"	"
6			45/25	79.5	"	"	"
7			75/75	78.5	"	"	"
8			75/45	79.5	"	"	"
9			60/45	"	"	"	"
10			75/45	90	.5 to 1.0	"	"
11			"	100	.5	"	"
12			"	75	.5 to 1.0	"	"
13			"	50	"	"	"
14			"	75	1.0	"	"
15-26			"	66	"	"	Roll Stability, φ=±8°
28-37			"	"	.75	"	"
38			"	"	.50	"	Power-off calibration
39			"	"	"	"	Roll Stability, φ=±8°
40			–	47	"	OFF	Hover Performance
41-42			–	66	"	"	Roll Stability, φ=±8°
43			–	"	.75	"	Hover Performance
44			–	"	"	"	Roll Stability, φ=±8°
45			–	"	1.0	"	Hover Performance
46			–	"	"	"	"
47			–	80	"	"	Roll Stability, φ=±8°
48			–	"	"	"	
50			–	50	"	"	Twang, zero RPM
51			–	"	"	"	", Hover
52			–	"	"	"	", zero RPM
53			–	70	"	"	", Hover
54(a)			–	–	"	"	Roll calibration
55-56			–	66	"	"	Hover Performance
57			–	"	"	"	Roll Stability, φ=+6,-4°
58			–	"	.75	"	Hover Performance
59			–	"	"	"	Roll Stability, φ=±6°
60			–	"	"	"	Roll Calibration
61			–	66	"	"	Hover Performance
64			–	"	"	"	Roll Stability, φ=±6°
65			–	"	.50	"	Hover Performance
66			–	"	"	"	Roll Calibration
67			–	66	"	"	Roll Stability, φ=±6°
68			–	50	.5 to 1.0	"	Hover Performance
69			–	60	"	"	"
70			–	70	"	"	"
71			–	80	"	"	"
72			75/45	66	.5	ON	Roll Calibration, +6°
73			"	–	"	"	Roll Stability, φ=±6°
74			"	66	"	"	Hover Performance
75			"	–	.75	"	Roll Calibration
76			"	–	"	"	Roll Stability, +6°
77			"	66	"	"	Hover Perf, Roll Stab, φ=±6°
78			"	"	1.0	"	Hover Perf, Roll Stab, φ=±6°

LSWT 418
Run Schedule Summary
Model C300-A2B Aeroelastic Model – Rod Test*

RUN No	POINT No	MAST ANGLE (DEG)	AIRSPEED (KTS-FPS)	INCIDENCE iH (DEG)	Model WEIGHT (LBS)	VORTEX Generator	Comments
1	5	90	30	0	88.5	OFF	Trim – level Fit
	6-10	"	"	"	"	"	Stability – "
2	11	"	40	"	"	"	Trim – "
	12-15	"	"	"	"	"	Stability – "
	16	"	60	"	"	"	Trim – "
3	17-23	"	"	"	"	"	Stability – "
	24-25	"	80	"	"	"	Trim – "
	26-29	"	"	"	"	"	Stability – "
	30	"	"	"	"	"	Trim – "
	31,32	"	"	"	"	"	Stability – "
4	33	"	90	"	"	"	Trim – "
	34-37	"	"	"	"	"	Stability – "
	38	"	100	"	"	"	Trim – "
	39-42	"	"	"	"	"	Stability –
5	43	"	16	"	"	"	Trim –
	44-47	"	"	"	"	"	Stability –
	48	"	20	"	"	"	Trim –
	49-52	"	"	"	"	"	Stability –
	53	"	40	"	"	"	Trim –
6	56	"	16	"	"	"	" –
	57-60	"	"	"	"	"	Stability –
7	61	"	20	"	"	"	Trim –
	62-65	"	"	"	"	"	Stability –
	66	"	60	"	"	"	Trim –
	67-72	"	80	"	"	"	Trim-Se –
	73	"	90	"	"	"	Trim –
	74-75	"	100	"	"	"	" –
8	80	"	30	"	"	"	Trim –
	81	"	40	"	"	"	" –
	82-84	"	60	"	"	"	Trim-Se –
	85	"	80	"	"	"	Trim –
	86	"	70	"	"	"	" –
	87-89	"	100	"	"	"	Trim-B1 –
9	90	"	40	+10	"	"	Trim –
	91	"	60	"	"	"	" –
10	92	75	60	0	"	"	Trim –
	93-96	"	"	"	"	"	Stability –
	97	"	80	"	"	"	Trim –
	98-101	"	"	"	"	"	Stability –
	103	"	100	"	"	"	Trim –
	104-107	"	"	"	"	"	Stability –
11	108	"	40	-10	"	"	Trim –
	109-113	"	"	"	"	"	Stability –
	114	"	60	"	"	"	Trim –
	115-118	"	"	"	"	"	Stability – Climb
	119	"	"	"	"	"	Trim –
12	120	"	40	+10	"	"	Trim – Level Fit
	121	"	60	"	"	"	" – "
13	122	60	80	0	"	"	" –
	123-126	"	100	"	"	"	Stability – "
	127	"	"	"	"	"	Trim –

* δF: 15/25 unless noted , Ω = 1260 RPM

LSWT 418
Run Schedule Summary
Model C300 Aeroelastic Model - Rod Test

RUN No	POINT No	MAST ANGLE (DEG)	AIRSPEED (KTS-FS)	INCIDENCE iH (DEG)	Model Weight (LBS)	VORTEX Generators	Comments
14	132	30	80	0	88.5	OFF	Trim — Level Flt
	133	"	90	"	"	"	" — "
	134	"	100	"	"	"	" — "
	135-136	"	"	"	"	"	Stability — "
15	147	90	30	"	104.0	"	Trim — "
	148	"	40	"	"	"	" — "
	149	"	50	"	"	"	" — "
	151	"	60	"	"	"	" — "
	152	"	80	"	"	"	" — "
	153	"	100	"	"	"	" — "
16	155-156	"	60	"	114.0	"	" — "
	157	"	80	"	"	"	" — "
	158	"	"	"	124.0	"	" — "
17	159-160	75	"	"	124.0	"	" — "
	161-162	"	"	"	129.0	"	" — "
18	163-167	90	60	"	88.5	"	Trim — Descent
	168-169	"	80	"	"	"	" — "
19	171-172	"	60	"	"	"	" — δr=25° — "
	173-175	"	80	"	"	"	" — "
20	176	75	60	"	"	"	Trim — Climb
	177-179	"	80	"	"	"	" — "
21	180-181	"	30	"	"	"	Trim — Level Flt.
	182-186	"	"	"	"	"	Stability — "
	187	"	40	"	"	"	Trim s — "
22	188-196	"	30	"	"	"	Pitch Sweep
23	197-204	"	60	"	"	"	" "
24	208	"	Ho.er	"	"	"	Trim — Level Flt.
	209	"	30	"	104.0	"	" — "
	210	"	40	"	"	"	" — "
	211	"	50	"	"	"	" — "
	212	"	60	"	"	"	" — "
	214	"	80	"	"	"	" — "
	215	"	90	"	"	"	" — "
	216	"	100	"	"	"	" — "
	217-219	"	80	"	89.5	"	" — Climb
25	220-221	"	30	04.0	"	"	" — Level Flt.
26	224	"	30	"	"	ON	" — "
	225	"	40	"	"	"	" — "
	226	"	50	"	"	"	" — "
	227-233	"	Hover	"	"	"	Trim - B₁ — "
27	234-240	"	30	"	89.5	"	Trim - θ — "
	241-244	"	40	"	"	"	Trim - B₁ — "
28	249-253	"	"	"	"	"	Trim - B₁ — "
	254	"	30	"	"	"	Trim — Level Flt.
	255	"	40	"	"	"	
	256	"	50	"	"	"	
	257	"	60	"	"	"	
	258	"	80	"	"	"	
	259	"	90	"	"	"	
	260	"	100	"	"	"	

LSWT 418
Run Schedule Summary
Model C30 A2B Aeroelastic Model · Rod Test

Run No	Point No	Mast Angle (DEG)	Airspeed (KTS)	Incidence in (DEG)	Model Weight (LBS)	Vortex Generators	Comments
29	261-267	75	30	0	89.5	ON	Yaw Sweep
	268-273	"	40	"	"	"	" "
	274-279	"	50	"	"	"	" "
30	280	"	30	"	"	"	Trim -δF = 0° - level Flt.
	281	"	40	"	"	"	" " = "
	282	"	50	"	"	"	" " = "
	283	"	60	"	"	"	" " = "
	284	"	80	"	"	"	" " = "
	285-286	"	90	"	"	"	" " = "
	287	"	100	"	"	"	" " = "
31	288-294	"	30	"	"	"	Pitch Sweep = 0°
32	295-306	90	80	"	"	"	Trim, δF = 0°, 1100 RPM, level
33	307-312	"	60	"	"	"	" " " "
34,35	313-317	"	80	"	"	"	Trim - Descent
36	318-320	95	"	"	"	"	" - "
	321-323	"	"	"	"	"	" , 1100 RPM - "
	324	"	"	"	"	"	" , 900 - "
	325	"	"	"	"	"	" , 950 - "
37	326	"	70	"	"	"	" , 1100 - "
	327	"	"	"	"	"	" " - "
38	328	"	60	"	"	"	" " - "
	329-330	"	50	"	"	"	" " - "
	331-332	"	90	"	"	"	" " - "
39	333-336	"	"	"	"	"	" , 1100 - "
40	337	90	80	"	1040	"	" " - "
	338	"	"	"	"	"	" , 1100 - "
	339	"	"	"	"	"	" , 950 - "
	340	"	"	"	"	"	" , 900 - "
41	341	"	Hover	"	89.5	"	Trim - Level Flt
	342	"	30	"	"	"	" " - "
	343-344	"	40	"	"	"	" " - "
	345	"	50	"	"	"	" " - "
	346	"	60	"	"	"	" " - "
	347	"	80	"	"	"	" " - "
	348	"	90	"	"	"	" " - "
	349	"	100	"	"	"	" " - "
42	350-351	"	30	-4	"	"	" " - "
	352	"	40	"	"	"	" " - "
	353	"	50	"	"	"	" " - "
	354	"	60	"	"	"	" " - "
	355	"	80	"	"	"	" " - "
	356	"	90	"	"	"	" " - "
	357	"	100	"	"	"	" " - "
43	361	"	70	+4	"	"	" " - "
	362	"	50	"	"	"	" " - "
	363	"	60	"	"	"	" " - "
	364	"	80	"	"	"	" " - "
44	365-369	"	30	0	"	"	Yaw Sweep
	370-373	"	40	"	"	"	" "
	374-379	"	50	"	"	"	" "

Note: Run 1-44 Roll Locked out, Roll Free Run 45 oN

LSWT 418
Run Schedule Summary
Model C300 A2B Aeroelastic Model - Rod Test

Run No	Point No	Mast Angle (DEG)	Airspeed (KTS-F.S.)	Incidence iH (DEG)	Model Weight (LBS)	Vortex Generators	Comments
45	383	90	50	0	89.5	ON	Trim -Lael Fu. Roll Sweep
	384-397	"	60	"	"	"	" "
46	398-403	"	80	"	"	"	" "
48	406-419	"	50	"	90.5	"	Pitch Sweep
49	420-429	"	40	"	"	"	" "
	430-437	"	30	"	"	"	" "
	438-429	"	40	"	"	"	" "
50	442-453	"	40	"	"	"	Pitch Sweep, δf=75°
51	455-471	"	30	"	"	"	" " , "
52	473-486	"	50	"	"	"	" " , "
	489-501	"	60	"	"	"	" " , "
53	504-512	"	60	"	"	"	" "
54	517	"	30	"	"	—	Trim, WING-OFF, LEVEL FG
	519	"	40	"	"	—	" "
55	523-528	"	16	"	"	—	" "
	530-536	"	20	"	"	—	" "
	537-543	"	30	"	"	—	" "
	546-550	"	40	"	"	—	" "
	552-556	"	50	"	"	—	" "
56	558	"	16	"	"	—	" "
	559	"	20	"	"	—	" "
	560	"	30	"	"	—	" "
	561	"	40	"	"	—	" "
57	562-569	"	60	"	"	—	" "
	571-572	"	80	"	"	—	" "
	573	"	"	"	105.5	—	" "
	574	"	"	"	103.5	—	" "
58	577-578	"	16	"	90.5	ON	Trim - Loueti/o
	579-580	"	20	"	"	"	" "
	581-582	"	30	"	"	"	" "
59	583-585	"	40	"	"	"	" "
60	586-595	"	"	"	"	"	" "
61	596	"	20	"	"	"	" "
	597	"	40	"	"	"	" "
62	598	"	20	"	"	"	" "
	599-600	"	40	"	"	"	" "
	601-602	"	50	"	"	"	" "
63	603-604	"	20	"	"	"	Trim, δf=75° - Loueti/o
	605-607	"	30	"	"	"	" "
	608-615	"	40	"	"	"	" "
	616-617	"	50	"	"	"	" "
64	618-619	"	16	"	"	"	" , δf=0° -
	620-624	"	20	"	"	"	" "
	626-631	"	30	"	"	"	" "
65	633-636	"	40	"	"	"	" "
	637-641	"	50	"	"	"	" "
66	646	"	80	"	"	"	" "
	647	"	"	"	105.5	"	" "
	648	"	"	"	120.5	"	" "
	649	"	"	"	117.0	"	" "

BELL HELICOPTER COMPANY

LSWT 418
Run Schedule, Summary
Model C300 A2B Aeroclastic Model - Rod Test

Run No	Point No	Mast Angle (DEG)	Airspeed (KTS-FS)	Incidence in (DEG)	Model Weight (LBS)	Vortex Generators	Comments
67	650	90	30	0	90.5	ON	Trim, Roll free, level FLT
	651	"	40	"	"	"	" = "
	652	"	50	"	"	"	" = "
	653	"	60	"	"	"	" = "
	654	"	80	"	"	"	" = "
	657	"	90	"	"	"	" = "
	658	"	100	"	"	"	" = "
	659	"	90	"	"	"	" - Descent
68	663	"	Hover	"	75	"	Trim, δr: 35
	664	"	"	"	"	"	" = 45
	665	"	"	"	"	"	" = 60
	666	"	"	"	"	"	" = 75
	667	"	"	"	"	"	" = 45
69	668	"	16	"	60	"	Trim - Level FLT
	669	"	20	"	"	"	" = "
	670	"	30	"	"	"	" = "
70-74	673	60	60	"	"	"	" = "
	675-680	"	80	"	"	"	" = "
	681-682	"	90	"	"	"	" = "
	683-685	"	100	"	"	"	" = "
75	686-689	30	90	"	"	"	" = "
	690-692	"	100	"	"	"	" = "
77	695	90	16	"	60	"	Trim - Sideward Flt
78	696	"	Hover	"	"	"	" = "
	697	"	16	"	"	"	" = "
	698	"	20	"	"	"	" = "
	699	"	25	"	"	"	" = "
	700	"	30	"	"	"	" = "
79	702	"	Hover	"	90	"	" = "
	703	"	16	"	"	"	" = "
	704	"	20	"	"	"	" = "
	705	"	25	"	"	"	" = "
	706	"	30	"	"	"	" = "
	707-711	"	30	"	63	"	" = "
80	713	"	16	"	60	"	" - Rearward Flt
	714-717	"	20	"	"	"	" = "
	718	"	25	"	"	"	" = "
	719-721	"	30	"	"	"	" = "
81	722	"	16	"	90	"	" = "
	723-724	"	20	"	"	"	" = "
	725-728	"	25	"	"	"	" = "
	731-732	"	35	"	"	"	" = "
82	733	"	20	"	"	"	" - Tail cg, Rearward
	734-736	"	25	"	"	"	" = "
	737	"	30	"	"	"	" = "
	738	"	35	"	"	"	" = "
83	740-741	"	Hover	"	"	"	Trim, IGE, h/D=.75
	742-743	"	16	"	"	"	"
	744	"	20	"	"	"	"
	745-751	"	25	"	"	"	"
	752-757	"	30	"	"	"	"

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A-7

LSWT 418
Run Schedule Summary
Model C30D A2B Aeroelastic Model - Rod Test

Run No	Point No	Mast Angle (DEG)	Arrested (INS FS)	Trailing In (DEG)	Model Weight (LBS)	Vortex Generator	Comments
83	759	90	16	0	90	ON	Trim, IGE, h/D=.75
	760-761	"	20	"	"	"	" , " , "
	762-763	"	25	"	"	"	" , " , "
	764-765	"	30	"	"	"	" , " , "
84	770-772	"	16	"	"	"	" , " , h/D=.6
	773-775	"	20	"	"	"	" , " , "
	776-781	"	25	"	"	"	" , " , "
	782-788	"	30	"	"	"	" , " , "
	789-790	"	20	"	"	"	" , " , "
	791-793	"	40	"	"	"	" , " , "
85	794-796	"	40	"	"	"	" , " , h/D=.75
	797-799	"	60	"	"	"	" , " , "
86	800-802	"	60	"	"	"	" , " , h/D=.6
	805	75	40	"	"	"	" , " , h/D=.75
	806	"	"	"	"	"	" , " , h/D=.6
	807-809	"	"	"	"	"	" , " , h/D=.75
	810-812	"	"	"	"	"	" , " , h/D=.6
	813-817	"	"	"	"	"	" , " , h/D=.75
	818-820	"	60	"	"	"	" , " , h/D=.75
87	821-823	"	16	"	"	"	" , " , h/D=.6
	824-826	"	"	"	"	"	" , " , =.75
	827-829	"	20	"	"	"	" , " , =.6
	830-832	"	"	"	"	"	" , " , =.75
	833-838	"	25	"	"	"	" , " , =.6
	839-845	"	"	"	"	"	" , " , =.75
	846-852	"	30	"	"	"	" , " , =.6
	853-858	"	"	"	"	"	" , " , =.75
	859-860	"	80	"	"	"	" , " , "
89	864	90	30	"	"	"	" , " , h/D=.75
	865	"	40	"	"	"	" , " , "
	866	"	50	"	"	"	" , " , "
	867-868	"	30	"	"	"	" , " , "
90,91	869-875	"	30,40	"	"	"	smoke studies
	End of Rod Test						

BELL HELICOPTER COMPANY

LSWT 421
Run Schedule Summary
Model C300-A2B Aeroclastic Model - Sting Test*

Run No	Point No	Mast Angle (DEG)	Airspeed (KTS-F.S.)	Incidence iH (DEG)	αTRIM (DEG)	Rotors	Comments
1	1003 - 1004	90	40	0	-2	ON	Elevator Sweep
2	1005 - 1009	"	"	"	"	"	"
3	1010 - 1015	"	"	"	"	"	"
4	1017 - 1025	"	"	"	"	"	Pitch Sweep - Level
5	1026 - 1034	"	"	"	"	"	Yaw Sweep α = -2
6	1035 - 1043	"	"	"	"	"	" ⊧ 0
7	1047 - 1055	"	"	"	"	"	" ⊧ -2
8	1056 - 1064	"	"	"	"	"	" ⊧ 0
9	1065 - 1073	"	"	"	"	"	" ⊧ -4
10	1074 - 1081	"	"	"	—	"	Pitch Sweep - Descent
11	1082 - 1088	"	"	"	—	"	"
12	1089 - 1095	"	"	"	—	"	"
13	1096 - 1103	"	"	"	—	"	" - Climb
14	1104 - 1110	"	"	"	—	"	"
15	1111 - 1114	"	"	"	—	"	"
16	1116 - 1119	"	16 - 40	"	0 to -2	"	Smoke Tubes - Side
17	1120 - 1123	"	"	"	"	"	" - Top
18	1126 - 1131	"	30	"	0	"	Elevator Sweep
19	1132 - 1137	"	"	"	"	"	Pitch Sweep - Level
20	1138 - 1146	"	"	"	"	"	Yaw Sweep - "
21	1147 - 1154	"	20	"	"	"	Elevator Sweep
22	1155 - 1161	"	"	"	"	"	Pitch Sweep - Level
23	1162 - 1170	"	"	"	"	"	Yaw Sweep - "
24	1171 - 1177	"	16	"	"	"	Pitch Sweep - "
25	1178 - 1185	"	"	"	"	"	Yaw Sweep - "
26	1188 - 1194	"	40	"	-2	"	Pitch Sweep - "
27	1195 - 1202	"	"	"	"	"	Yaw Sweep - "
28	1203 - 1208	"	60	"	-6	"	Elevator Sweep - "
29	1209 - 1212	"	"	"	"	"	Pitch Sweep - "
30	1218 - 1225	"	"	"	"	"	Yaw Sweep - "
31	1228 - 1234	"	80	"	-8	"	Elevator Sweep - "
32	1235 - 1243	"	"	"	"	"	Pitch Sweep - "
33	1244 - 1254	"	"	"	"	"	Yaw Sweep - "
34	1255 - 1264	"	"	"	—	"	Pitch Sweep - Descent
35	1265 - 1267	"	"	"	—	"	" - Autorot
36	1268 - 1273	"	"	"	—	"	" - Climb
37	1276 - 1283	"	40	-4	-2	"	Pitch Sweep - Level
38	1284 - 1292	"	60	"	-6	"	"
39	1293 - 1300	"	80	"	-8	"	"
40	1301 - 1306	"	20	"	0	"	"
41	1307 - 1312	"	30	"	0	"	"
42	1319 - 1321	"	16 - 40	0	0 to -2	"	Tuft grid - Level
43	1333 - 1341	"	40	"	—	"	Pitch Sweep - Descent
44	1342 - 1319	"	"	"	—	"	" - Climb
45	1350 - 1356	"	30	"	—	"	Pitch Sweep - Descent
46	1357 - 1365	"	"	"	—	"	Yaw Sweep - "
47	1366 - 1372	"	"	"	—	"	Pitch Sweep - Climb
48	1373 - 1380	"	"	"	—	"	Yaw Sweep - "
49	1382 - 1388	"	25	"	—	"	Pitch Sweep - Descent
50	1389 - 1396	"	"	"	—	"	Yaw Sweep - "
51	1397 - 1403	"	"	"	—	"	Pitch Sweep - Climb

* SF = 45/25, Model Wt = 90 LBS, Ω = 1160 RPM, Vortex Generators ON unless noted

LSWT 421
Run Schedule Summary
Model C300-A2B Aeroelastic Model - Sting Test

Run No	Point No.	Mast Angle (DEG)	Airspeed (KTS-F.S)	Incidence iₙ (DEG)	α_TRIM (DEG)	Rotors	Comments
52	1405-1411	90	60	0	—	ON	Pitch Sweep - Descent
53	1412-1420	"	"	"	—	"	Yaw Sweep - "
54	1421-1427	"	"	"	—	"	Pitch Sweep - Climb
55	1428-1435	"	"	"	—	"	Yaw Sweep - "
56	1436-1444	"	80	"	—	"	" " - Descent
57	1445-1454	"	"	"	—	"	" " - Climb
58	1456-1461	"	"	"	—	"	Pitch Sweep, α=0 Descent
59	1462-1470	"	"	"	—	"	" " "
60	1473-1478	75	16	"	12	"	Elevator Sweep - Level
61	1479-1484	"	"	"	"	"	Pitch Sweep - "
62	1485-1492	"	"	"	"	"	Yaw Sweep - "
63	1493-1498	"	20	"	"	"	Elevator Sweep - "
64	1499-1503	"	"	"	"	"	Pitch Sweep - "
65	1504-1511	"	"	"	"	"	Yaw Sweep - "
66	1514-1520	"	30	"	10	"	Elevator Sweep "
67	1521-1526	"	"	"	"	"	Pitch Sweep - "
68	1527-1534	"	"	"	"	"	Yaw Sweep - "
69	1535-1540	"	40	"	8	"	Elevator Sweep - "
70	1541-1546	"	"	"	"	"	Pitch Sweep - "
71	1547-1554	"	"	"	"	"	Yaw Sweep - "
72	1557-1562	"	60	"	4	"	Elevator Sweep - "
73	1563-1569	"	"	"	"	"	Pitch Sweep - "
74	1570-1577	"	"	"	"	"	Yaw Sweep - "
75	1579-1584	"	80	"	2	"	Elevator Sweep - "
76	1585-1591	"	"	"	"	"	Pitch Sweep - "
77	1592-1601	"	"	"	"	"	Yaw Sweep - "
78	1602-1608	"	20	-4	12	"	Pitch Sweep - "
79	1609-1614	"	30	"	10	"	" " - "
80	1615-1620	"	40	"	8	"	" " - "
81	1621-1626	"	60	"	4	"	" " - "
82	1627-1632	"	80	"	2	"	" " - "
83	1633-1639	"	40	0	—	"	Pitch Sweep - Descent
84	1640-1648	"	"	"	—	"	Yaw Sweep - "
85	1652-1657	"	"	"	—	"	Pitch Sweep - Climb
86	1658-1665	"	"	"	—	"	Yaw Sweep - "
87	1666-1671	"	60	"	—	"	Pitch Sweep - Descent
88	1672-1678	"	"	"	—	"	Yaw Sweep - "
89	1679-1685	"	"	"	—	"	Pitch Sweep - Climb
90	1686-1693	"	"	"	—	"	Yaw Sweep - "
91	1694-1699	"	80	"	—	"	Pitch Sweep - Descent
92	1700-1707	"	"	"	—	"	Yaw Sweep - "
93	1708-1713	"	"	"	—	"	Pitch Sweep - Climb
94	1714-1721	"	"	"	—	"	Yaw Sweep - "
Run 1-94 tail on, Run 95-129 tail off							
95	1727-1732	75	80	—	2	ON	Pitch Sweep - Level
96	1733-1740	"	"	—	"	"	Yaw Sweep - "
97	1741-1746	"	60	—	4	"	Pitch Sweep - "
98	1747-1754	"	"	—	"	"	Yaw Sweep - "
99	1755-1760	"	40	—	8	"	Pitch Sweep - "
100	1761-1768	"	"	—	"	"	Yaw Sweep - "
101	1769-1774	"	30	—	10	"	Pitch Sweep - "
102	1775-1783	"	"	—	"	"	Yaw Sweep - "

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LSWT 421
Run Schedule Summary
Model C300-A2B Aeroelastic Model - Sting Test

RUN No	POINT No	MAST Angle (DEG)	AIRSPEED (KTS·FS)	Incidence iH (DEG)	αTRIM (DEG)	Rotors	Comments
103	1785-1790	90	30	—	0	ON	Pitch Sweep - Level
104	1791-1799	"	"	—	"	"	Yaw Sweep - "
105	1800-1805	"	40	—	-2	"	Pitch Sweep - "
106	1806-1814	"	"	—	"	"	Yaw Sweep - "
107	1813-1823	"	60	—	-6	"	Pitch Sweep - "
108	1824-1832	"	"	—	"	"	Yaw Sweep - "
109	1833-1837	"	80	—	-8	"	Pitch Sweep - "
110	1938-1946	"	"	—	"	"	Yaw Sweep - "
111	1847-1852	"	80	—	—	"	Pitch Sweep - Descent
112	1853-1858	"	"	—	—	"	" - Climb
113	1859-1861	"	60	—	—	"	" - Descent
114	1867-1872	"	20	—	0	"	" - Level
115	1873-1880	"	"	—	"	"	Yaw Sweep - "
116	1881-1886	"	"	—	—	"	Pitch Sweep - Descent
117	1887-1892	"	30	—	—	"	" - Climb
118	1893-1898	"	"	—	—	"	" - Descent
119	1900-1905	"	40	—	—	"	" - Climb
120	1906-1911	"	"	—	—	"	" - "
121	1912-1917	"	60	—	—	"	Pitch Sweep - Level
122	1920-1925	60	80	—	8	"	Yaw Sweep - "
123	1926-1933	"	"	—	4	"	Pitch Sweep - "
124	1934-1940	"	100	—	"	"	Yaw Sweep - "
125	1941-1948	"	"	—	"	"	Pitch Sweep - "
126	1951-1954	30	80	—	14	"	" - "
127	1955-1959	"	100	—	10	"	Yaw Sweep - "
128	1960-1967	"	"	—	"	"	"
RUN	129-136	TAIL-ON					
129	1970-1973	30	80	0	14	ON	Pitch Sweep - Level
130	1974-1978	"	100	"	10	"	" - "
131	1979-1985	"	"	"	"	"	Yaw Sweep - "
132	1986-1993	60	80	"	8	"	Pitch Sweep - "
133	1994-2001	"	"	"	"	"	Yaw Sweep
134	2002-2007	"	100	"	4	"	Pitch Sweep - "
135	2008-2013	"	"	"	"	"	Yaw Sweep - "
136	2014-2018	"	"	"	"	"	Elevator Sweep - "
END OF	ROTOR-ON TESTING						
137	2021-2030	60	80	0	—	OFF	Pitch Sweep
138	2031-2039	"	"	"	—	"	Yaw Sweep
139	2040-2019	"	"	-4	—	"	Pitch Sweep
140	2051-2061	30	"	"	—	"	" "
141	2063-2072	"	"	0	—	"	Yaw Sweep
142	2073-2080	"	"	"	—	"	Pitch Sweep
143	2083-2092	75	40	0	—	"	Yaw Sweep
144	2093-2100	"	"	"	—	"	Pitch Sweep
145	2101-2110	"	60	"	—	"	Yaw Sweep
146	2111-2118	"	"	"	—	"	Pitch Sweep
147	2120-2129	"	80	"	—	"	Yaw Sweep
148	2130-2119	"	"	-4	—	"	Pitch Sweep
149	2139-2140	"	40	"	—	"	" "
150	2149-2158	"	60	"	—	"	
151	2159-2168	"	80				

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LSWT 421
Run Schedule Summary
Model C300-A2B Aeroelastic Model - Sting Test

Run No	Point No	Mast Angle (DEG)	Airspeed (KTS-FS)	Incidence iN (DEG)	α_TRIM (DEG)	Rotors	Comments
152	2172-2181	90	40	-4	—	OFF	Pitch Sweep
153	2182-2192	"	60	"	—	"	"
154	2193-2201	"	80	"	—	"	"
155	2207-2217	"	40	0	—	"	Pitch Sweep
156	2219-2225	"	"	"	—	"	Yaw Sweep
157	2226-2236	"	60	"	—	"	Pitch Sweep
158	2237-2244	"	"	"	—	"	Yaw Sweep
159	2245-2255	"	80	"	—	"	Pitch Sweep
160	2256-2264	"	"	"	—	"	Yaw Sweep
	Run 137-160 tail-on		Run 167-176 tail-off				
161	—	90	40	—	—	OFF	Pitch Sweep
162	—	"	"	—	—	"	Yaw Sweep
163	—	"	60	—	—	"	Pitch Sweep
164	—	"	"	—	—	"	Yaw Sweep
165	—	"	80	—	—	"	Pitch Sweep
166	—	"	"	—	—	"	Yaw Sweep
167	—	75	40	—	—	"	Pitch Sweep
168	—	"	"	—	—	"	Yaw Sweep
169	—	"	60	—	—	"	Pitch Sweep
170	—	"	"	—	—	"	Yaw Sweep
171	—	"	80	—	—	"	Pitch Sweep
172	—	"	"	—	—	"	Yaw Sweep
173	—	60	80	—	—	"	Pitch Sweep
174	—	"	"	—	—	"	Yaw Sweep
175	—	30	80	—	—	"	Pitch Sweep
176	—	"	"	—	—	"	Yaw Sweep
	End of Sting Test						

APPENDIX B

Sting Test Balance Data

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45/25°
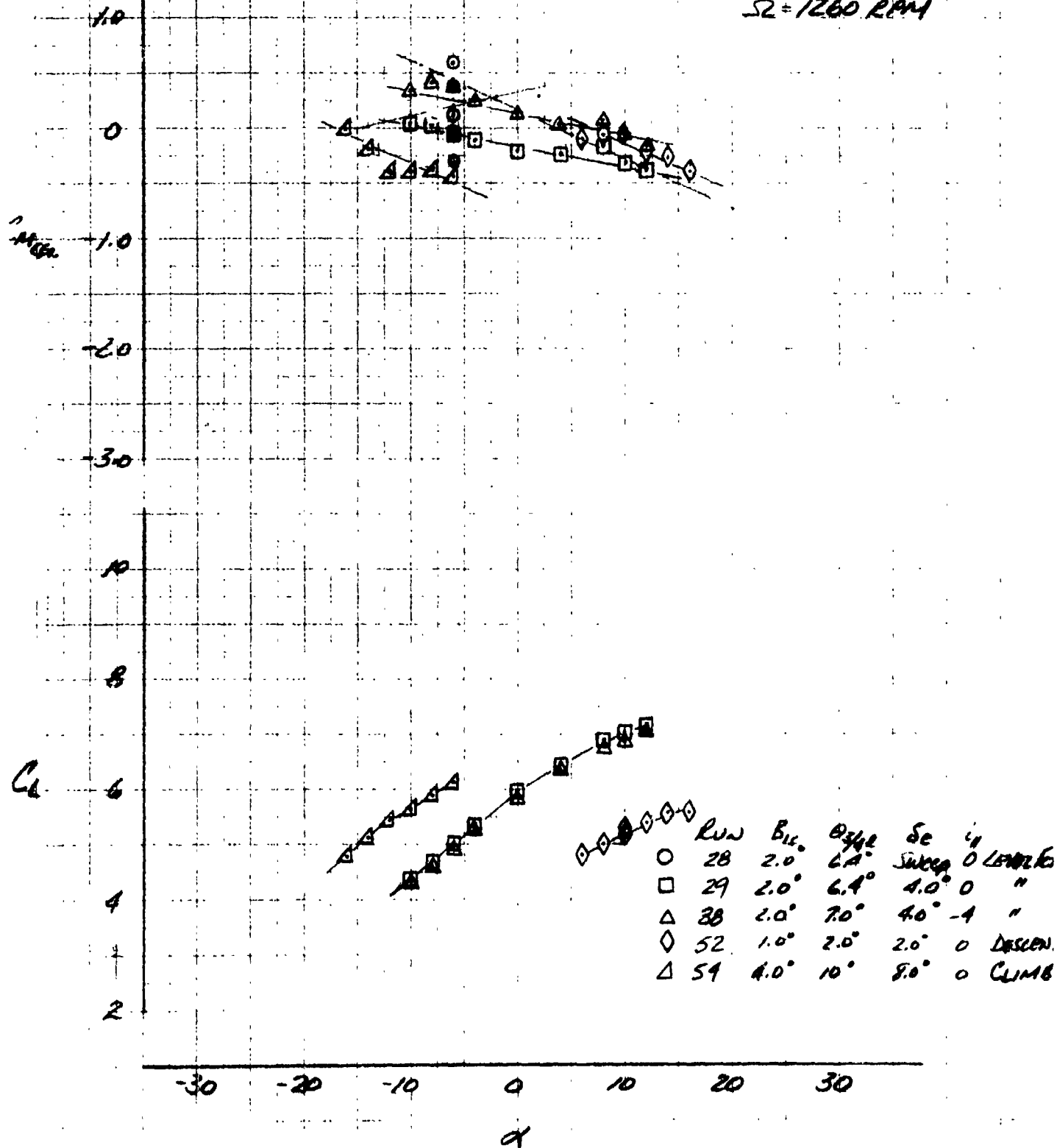
60 KTS
Ω = 1260 RPM

$C_{M_{GEO}}$

C_L

Run	B_{1c}	$\Theta_{3/4}$	δ_e	i_t	
○ 28	2.0°	6.4°	Sweep	0	LEVEL FLT
□ 29	2.0°	6.4°	4.0°	0	"
△ 38	2.0°	7.0°	4.0°	-1	"
◇ 52	1.0°	2.0°	2.0°	0	DESCEN.
△ 54	4.0°	10°	8.0°	0	CLIMB

α

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

60 KTS
Ω = 1260 RPM

$C_{m_{EG}}$

2.0

1.0

0

1.0

C_L

.8

.6

.4

.2

0

Run	b_{1s}	θ_{MR}	δ_e	
○ 107	2.0	6.5	—	LEVEL, TAIL OFF
□ 113	1.0	2.8	—	DESCENT, "
△ 121	4.0	10.0	—	CLIMB, "

-30   -20   -10   0   10   20   30

α

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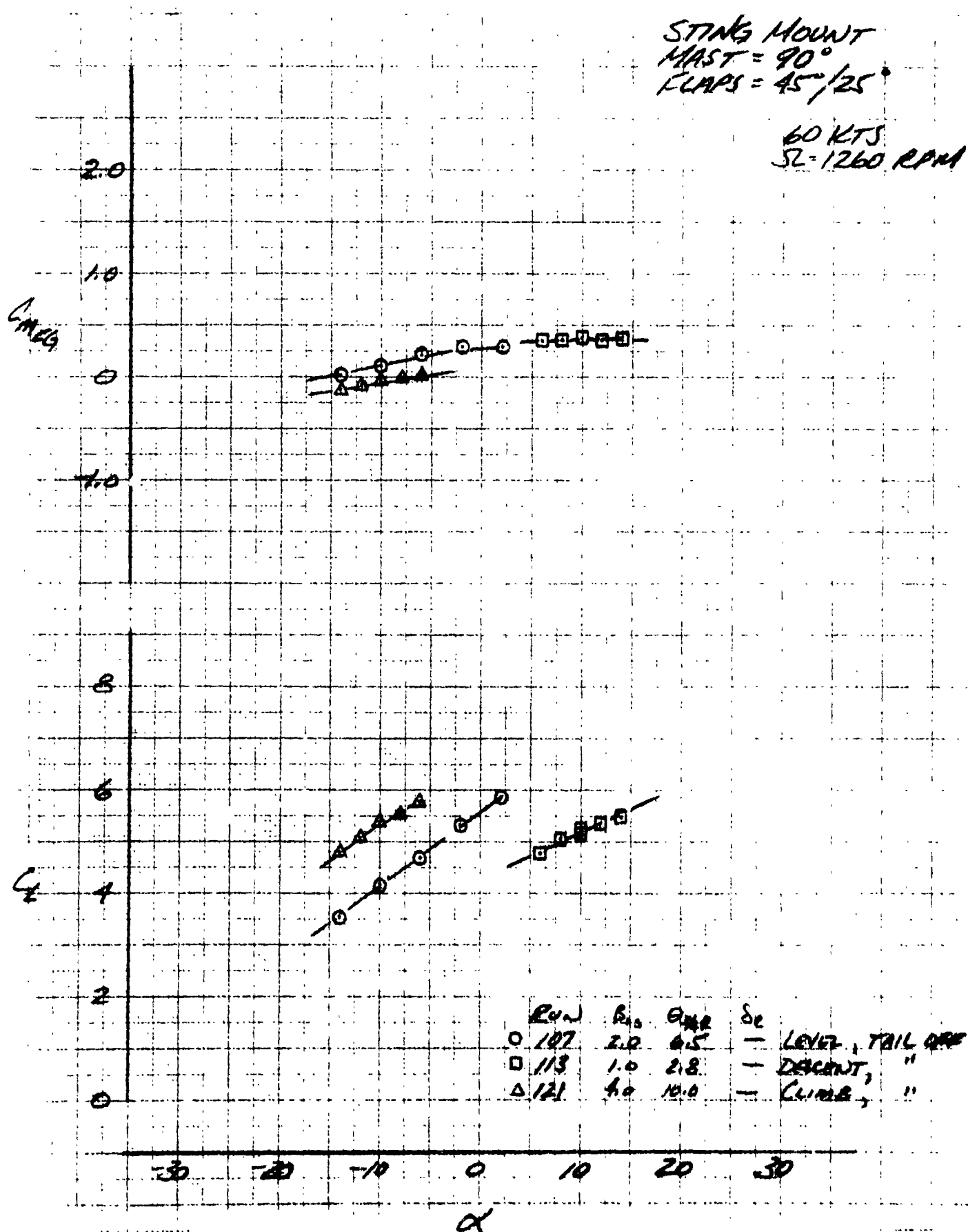
LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45/25°

40 KTS
$\Omega = 1260$ RPM

$B_1 = 16°$
$\Theta_{3/4r} = 7.0$
$\delta e = 3.0°$

RUN	α	
□ 7	-2°	Level
◇ 8	+0°	
D 9	-4°	
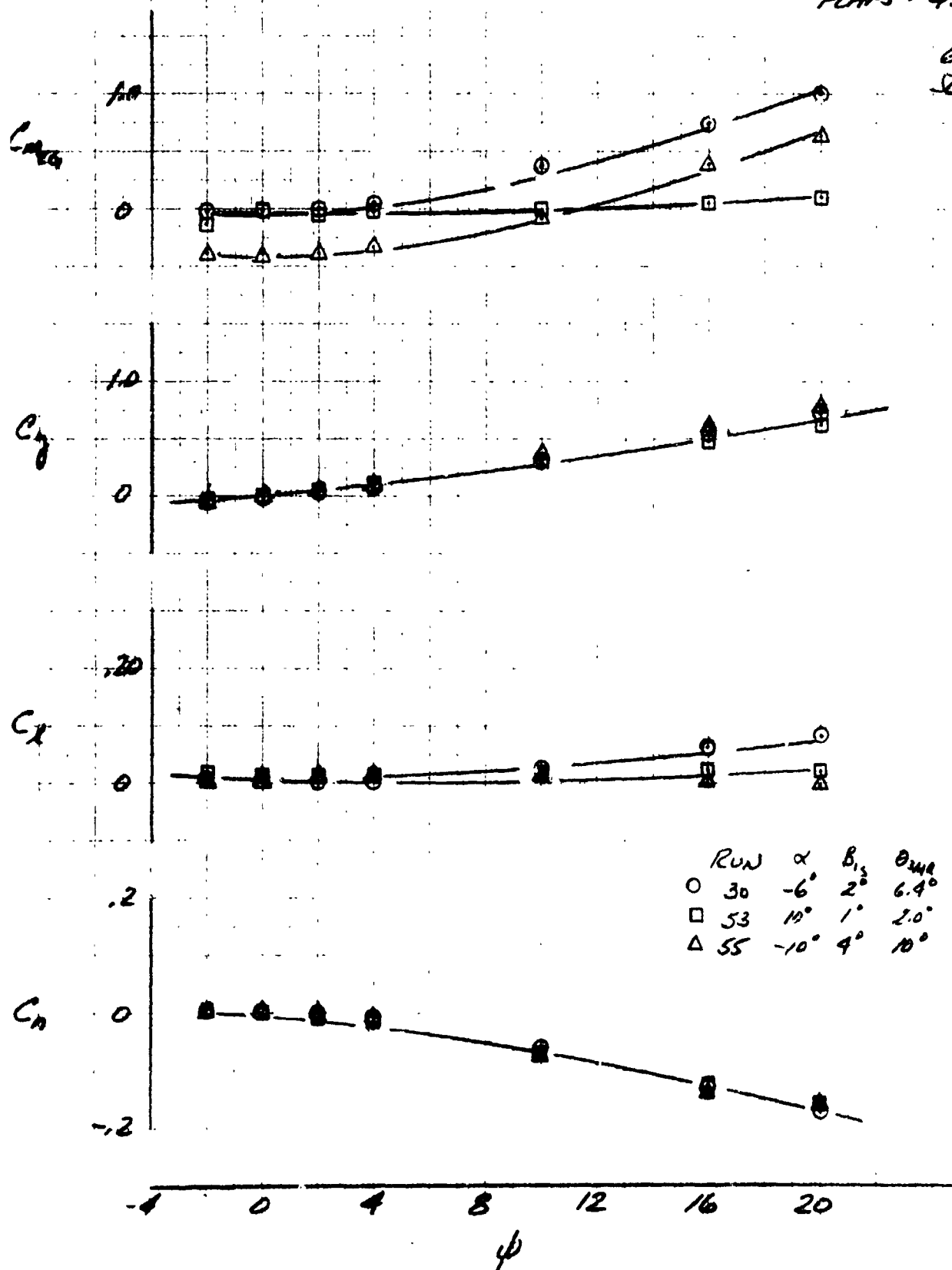
$C_{m_{c.g.}}$

C_y

C_ℓ

C_h

ψ

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A-16

BELL
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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

60 KTS
Ω = 1260

RUN	α	B_{1s}	$θ_{3/4R}$	$δ_e$	
○ 30	-6°	2°	6.4°	3.0°	Level
□ 53	10°	1°	2.0°	2.0°	Descent
△ 55	-10°	4°	10°	8.0°	Climb

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A-17

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

65 KTS
Ω = 1260 RPM

$C_{m_{cg}}$

.40

.20

0

C_y

.10

0

C_ℓ

.04

.02

0

C_n

.02

0

-.02

RUN	α	B₁	θ₇₅	δe	
☐ 108	-6°	2.0	6.5	—	LEVEL, TAIL OFF

-4   0   4   8   12   16   20

ψ

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

80 KTS
Ω = 1260 RPM

Run	B_{1s}	$\theta_{3/4R}$	δe	i_w	
○ 31	3.5	7.5	Sweep	0	Level
□ 32	"	"	7.0	0	
△ 34	2.5	4.0	5.0	0	Descn
◇ 36	5.5	10.0	11.0	0	Climb
▽ 39	3.5	7.6	7.0	-4	Level

LSWT 421

STING MOUNT
MAST = 90°

BO KTS
$\Omega = 1260$ RPM

	Run	B_{1_3}	$\theta_{3/4R}$	δ_e	FLAPS	
△	34	2.5°	4°	5°	45/5°	Down
○	58	2.5°	4°	5°	0/0	"
□	59	0	0	0	0/0	"

C_{mcg}

C_L

α
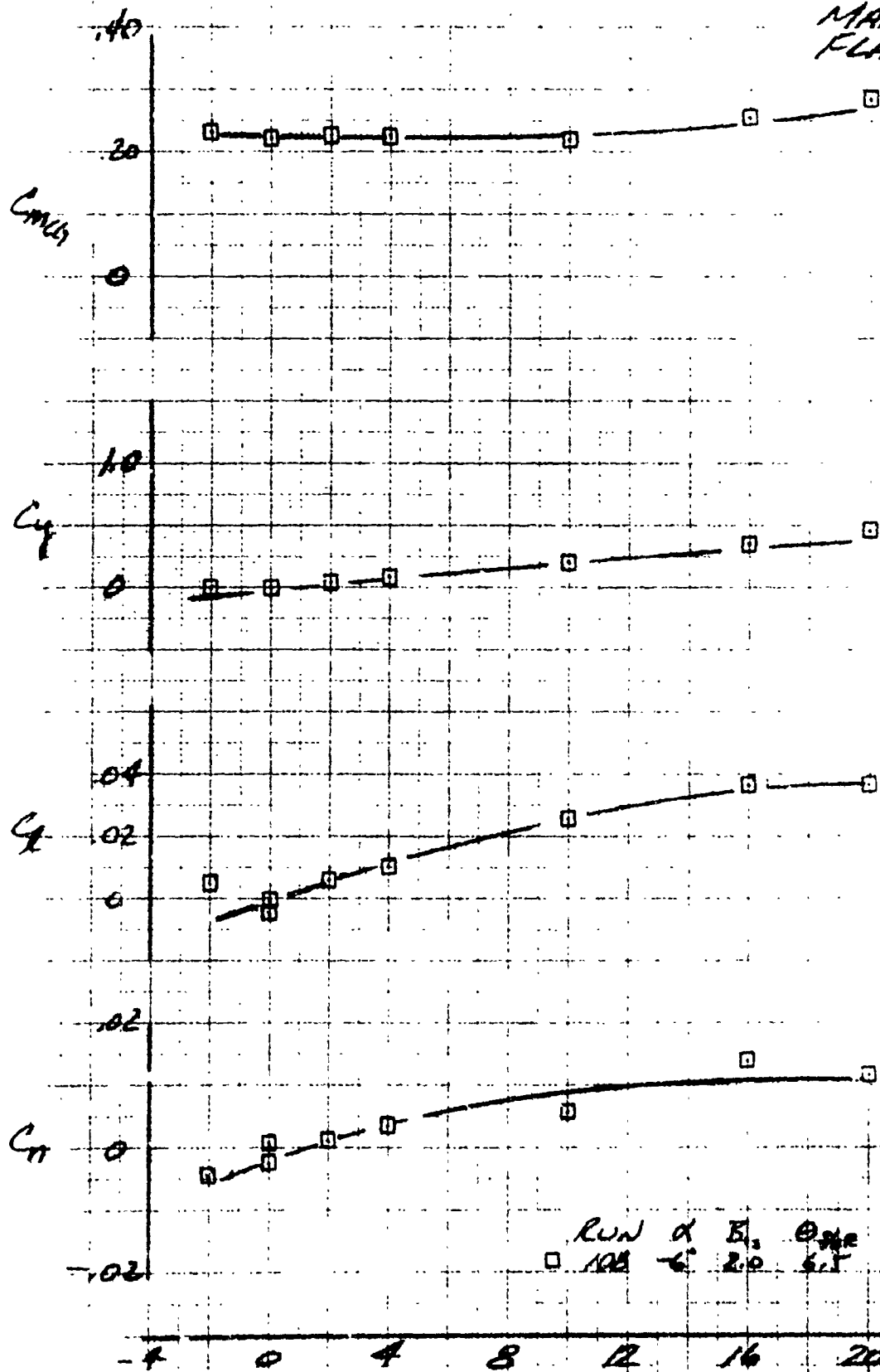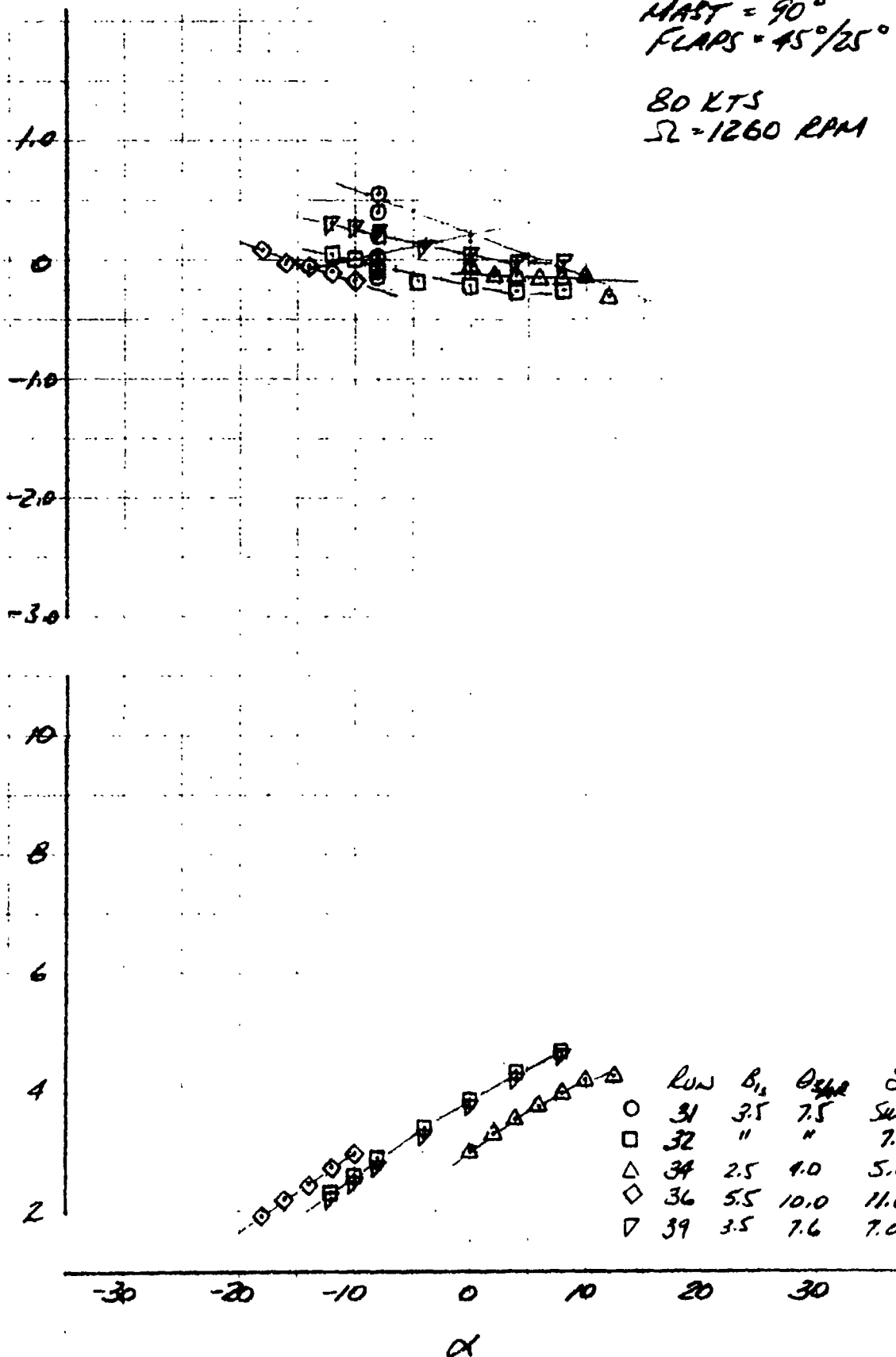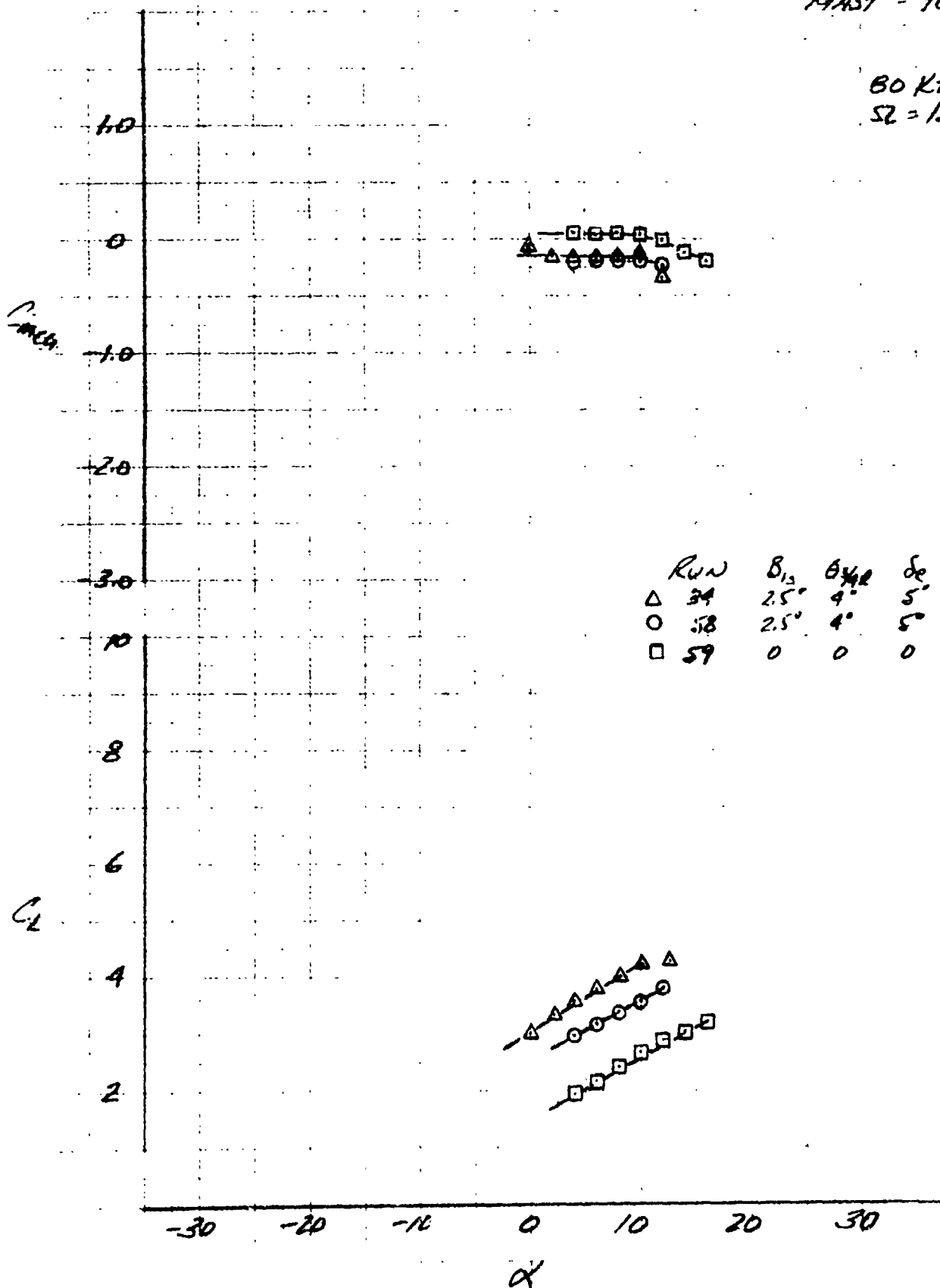
LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45/25°

80 KTS.
SL = 1260 RPM

	RUN	B_{1s}	θ_{75}	S_e	
○	109	3.5	8.0	—	LEVEL, TAIL OFF
□	111	3.5	9.0	—	DESCENT, "
△	112	3.5	10.0	—	CLIMB, "

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A-21

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

80 KTS
ℓ = 1260

B_{15} = 3.5°
$\theta_{3/4R}$ = 7.5°
δ_e = 7.0°
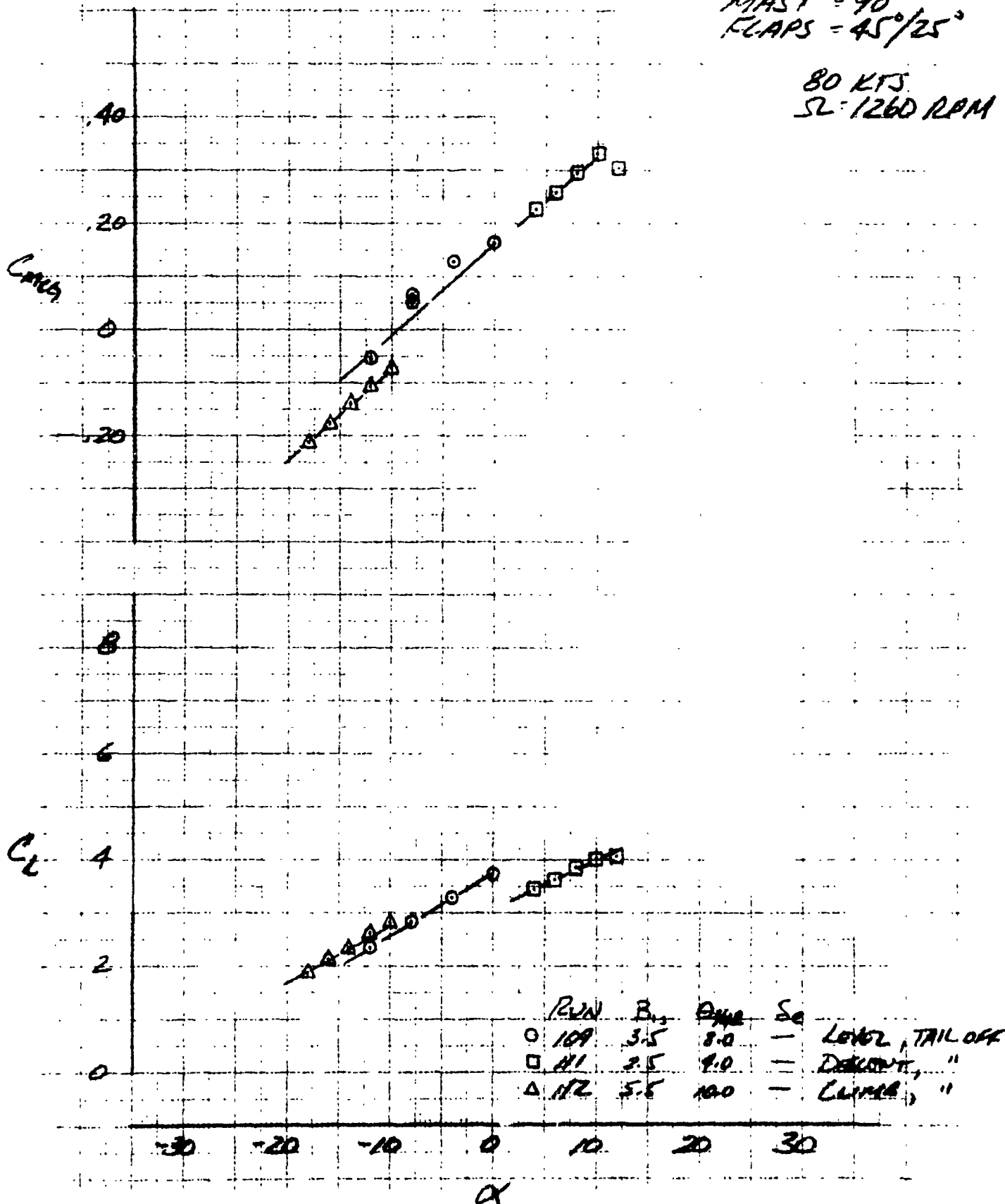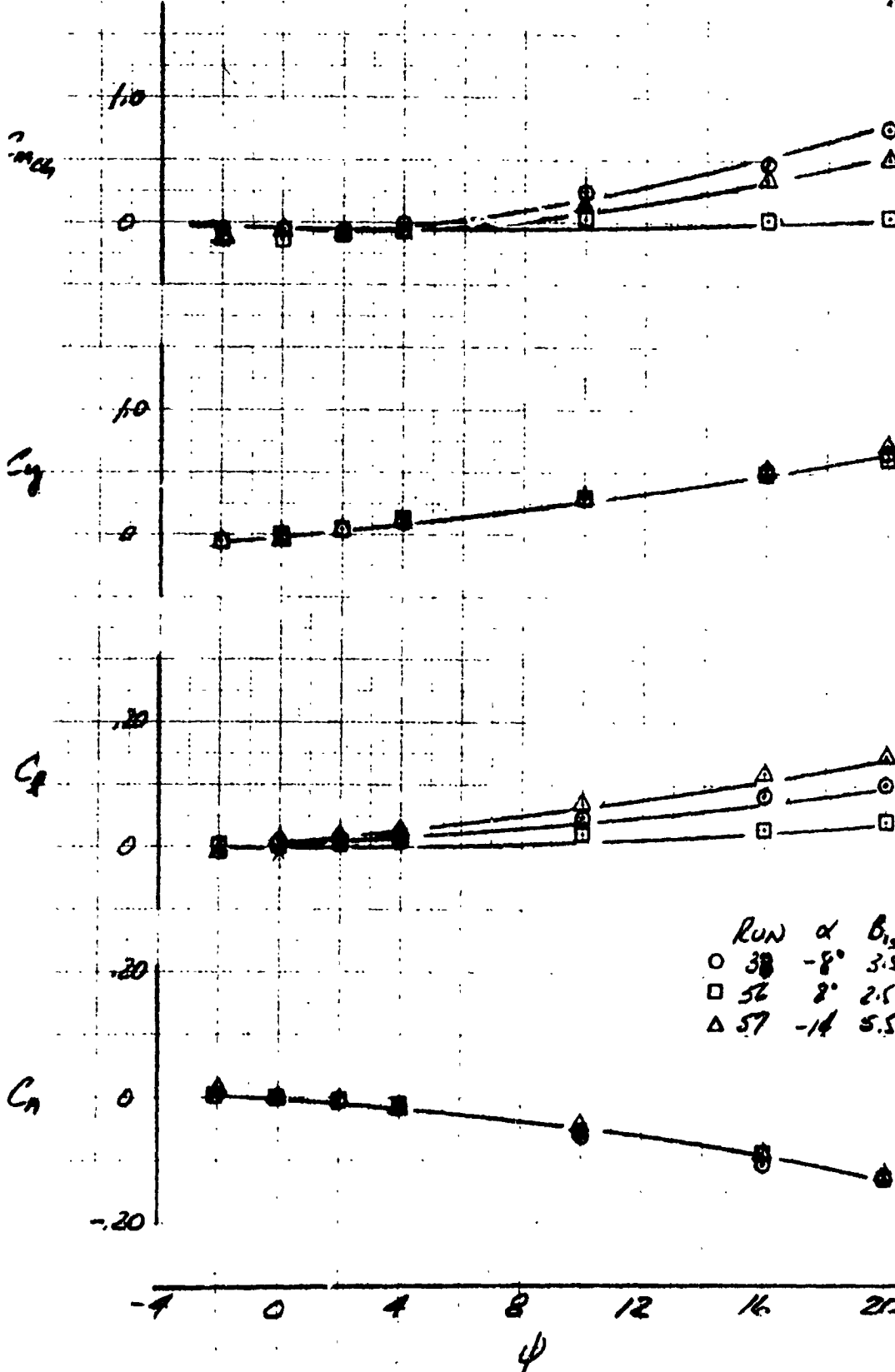
Run	α	B_{15}	$\theta_{3/4R}$	δ_e	
○ 38	-8°	3.5°	7.5°	7.0°	LEVEL
□ 56	8°	2.5°	4.0°	5.0°	DESCENT
△ 57	-14	5.5°	10°	11°	CLIMB

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A-22

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

80 KTS
Ω = 1260 RPM

Run	α	B_{1s}	θ_{75}	δ_e	
110	-8°	3.5°	8.0°	—	LEVEL, TAIL OFF

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

40 KTS
ROTORS OFF

Run	δ_e	i_H	
○ 155	0	0	TAIL ON
□ 152	0	-4	" "
△ 161	—	—	TAIL OFF

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

40 KTS
ROTORS OFF

RUN	δ_e	i_N	
○ 156	0	0	TAIL ON
□ 162	—	—	TAIL OFF

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A-25

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45/25°

60 KTS
ROTORS OFF

Run	δ_e	i_H	
○ 157	0	0	TAIL ON
□ 153	0	-4	" "
△ 163	—	—	TAIL OFF

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

60 KTS
ROTORS OFF

RUN	δ_e	i_H	
O 158	O	O	TAIL ON
□ 164	—	—	TAIL OFF

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

80 KTS
ROTORS OFF

Run	δ_e	i_H		
○ 159	0	0	TAIL ON	
□ 154	0	-4	"	"
△ 165	—	—	TAIL OFF	

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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

80 KTS
ROTORS OFF

$C_{m_{cg}}$

C_y

C_ℓ

Run	δ_e	i_H	
⊙ 160	0	0	TAIL ON
☐ 166	—	—	TAIL OFF

C_n

ψ

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A-29

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

16 KTS
Ω = 1260 RPM

RUN	B_{1S}	$θ_{75}$	$δ_c$	
☐ 24	2.0°	10°	4°	LEVEL FLT

LSWT 421

STING MOUNT
MAST = 9°
FLAPS = 45°/25°

16 KTS
Ω = 1260

$B_{1s} = 2.0$
$θ_{75R} = 10°$
$δe = 4.0°$

RUN   α
○   25   0°

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A-31

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45/25

20 KTS
S_c = 1260 RPM
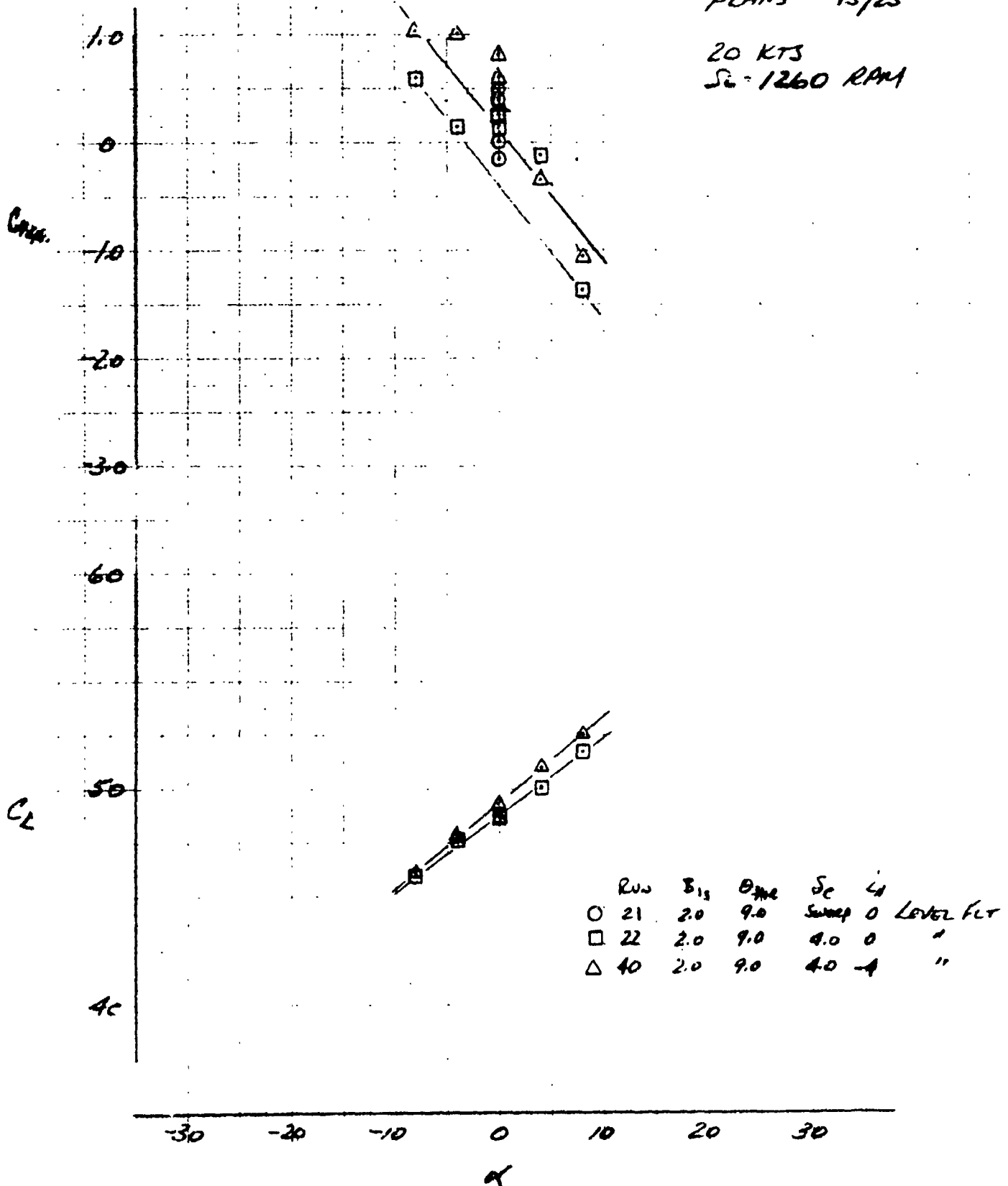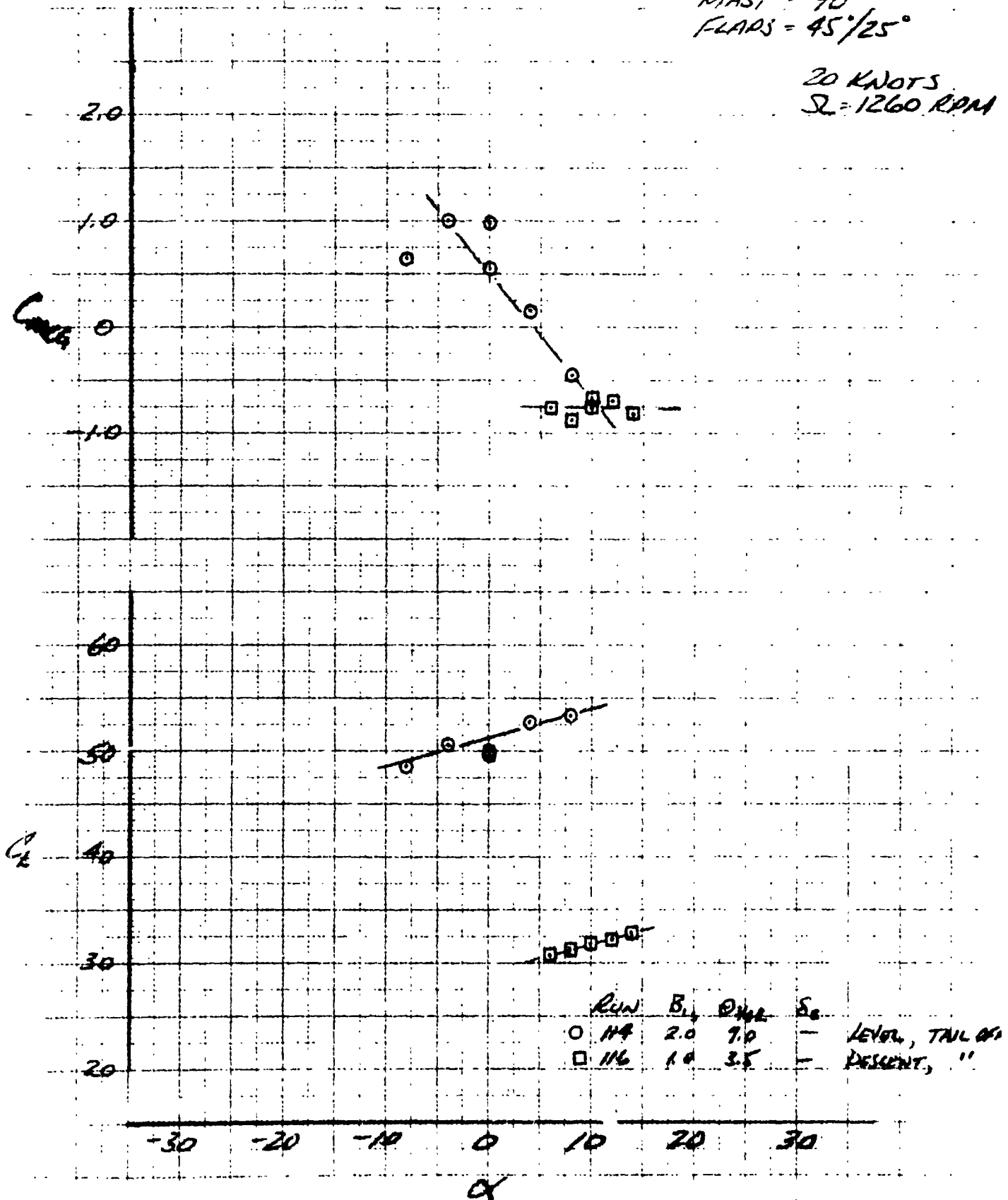
$C_{M\!A\!C}$

C_L

Run	B_{1s}	$\theta_{\!\!\!TR}$	S_c	i_c	
O 21	2.0	9.0	Sweep	0	LEVEL FLT
□ 22	2.0	9.0	4.0	0	"
△ 40	2.0	9.0	4.0	-4	"

α

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A-32

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

20 KNOTS
Ω = 1260 RPM

Run	B_1	Θ_{75R}	δ_e	
○ 114	2.0	7.0	—	LEVEL, TAIL ON
□ 116	1.0	3.5	—	DESCENT, ''

LSWT 421

STAXS MOUNT
MAST = 90°
FLAPS = 45°/25°

20 KTS
Ω = 1260

$B_{1S} = 2.0°$
$θ_{3/4R} = 9.0°$
$δ_C = 4.0°$

Run   α
○   23   0°

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A-34

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45/25°

20 KTS
Ω = 1260 RAM

$C_{m,cg}$

C_{x}

Run	α	B_{1s}	$\theta_{3/4R}$	δ_e	
□ 115	0	20	9,0	—	LEVEL, TAIL OFF

C_{ℓ}

C_{n}

ψ

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A-35

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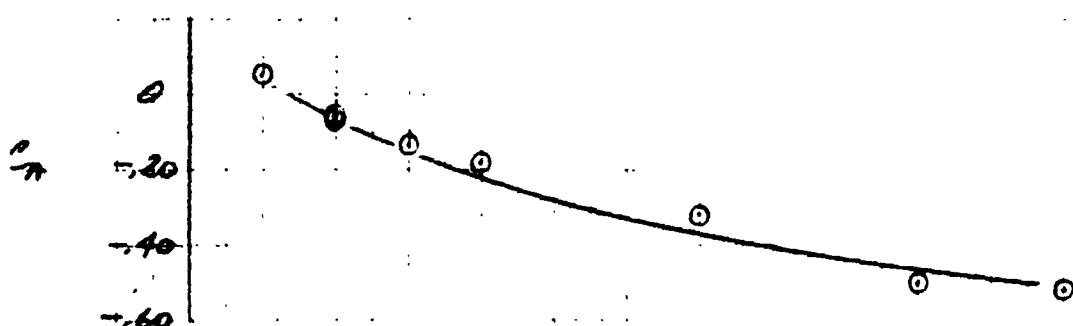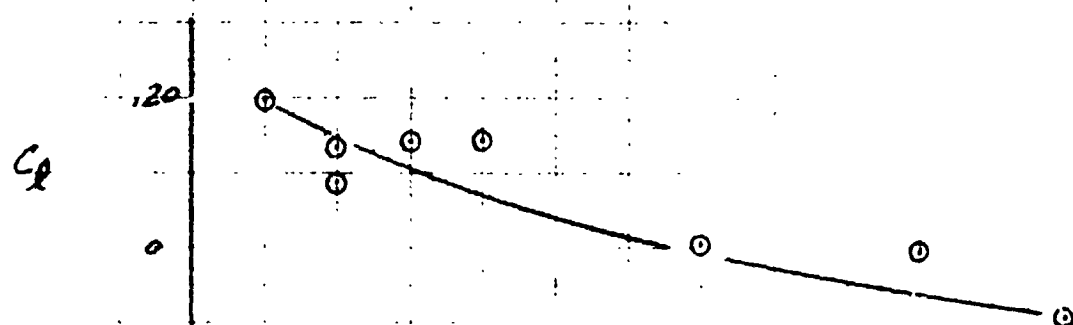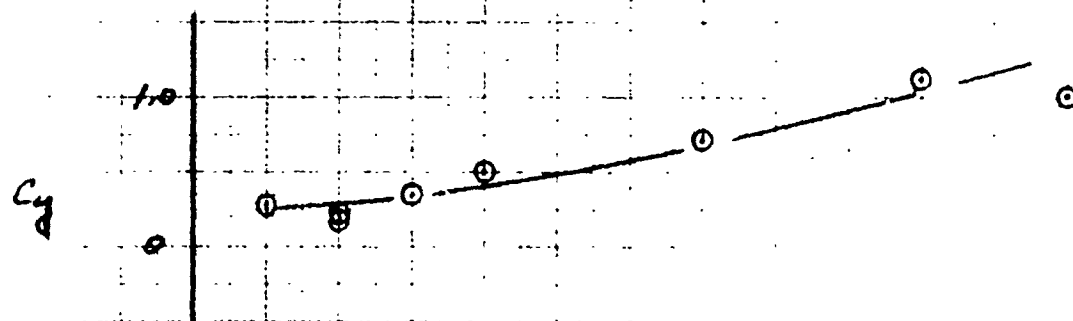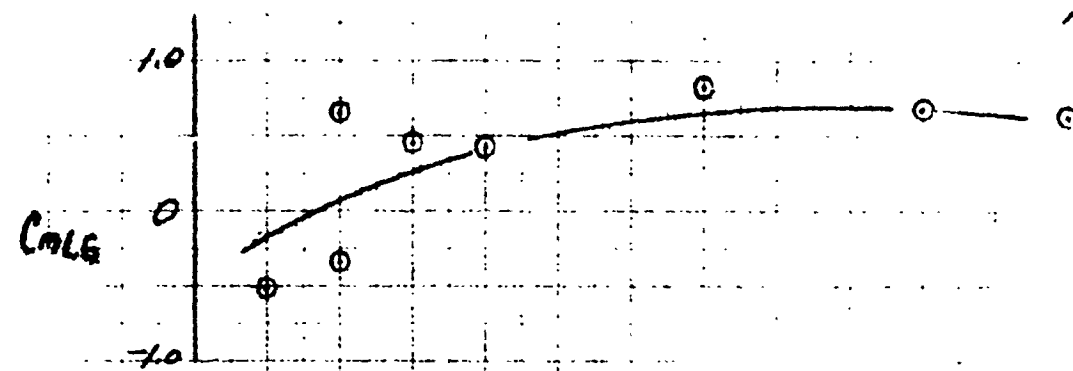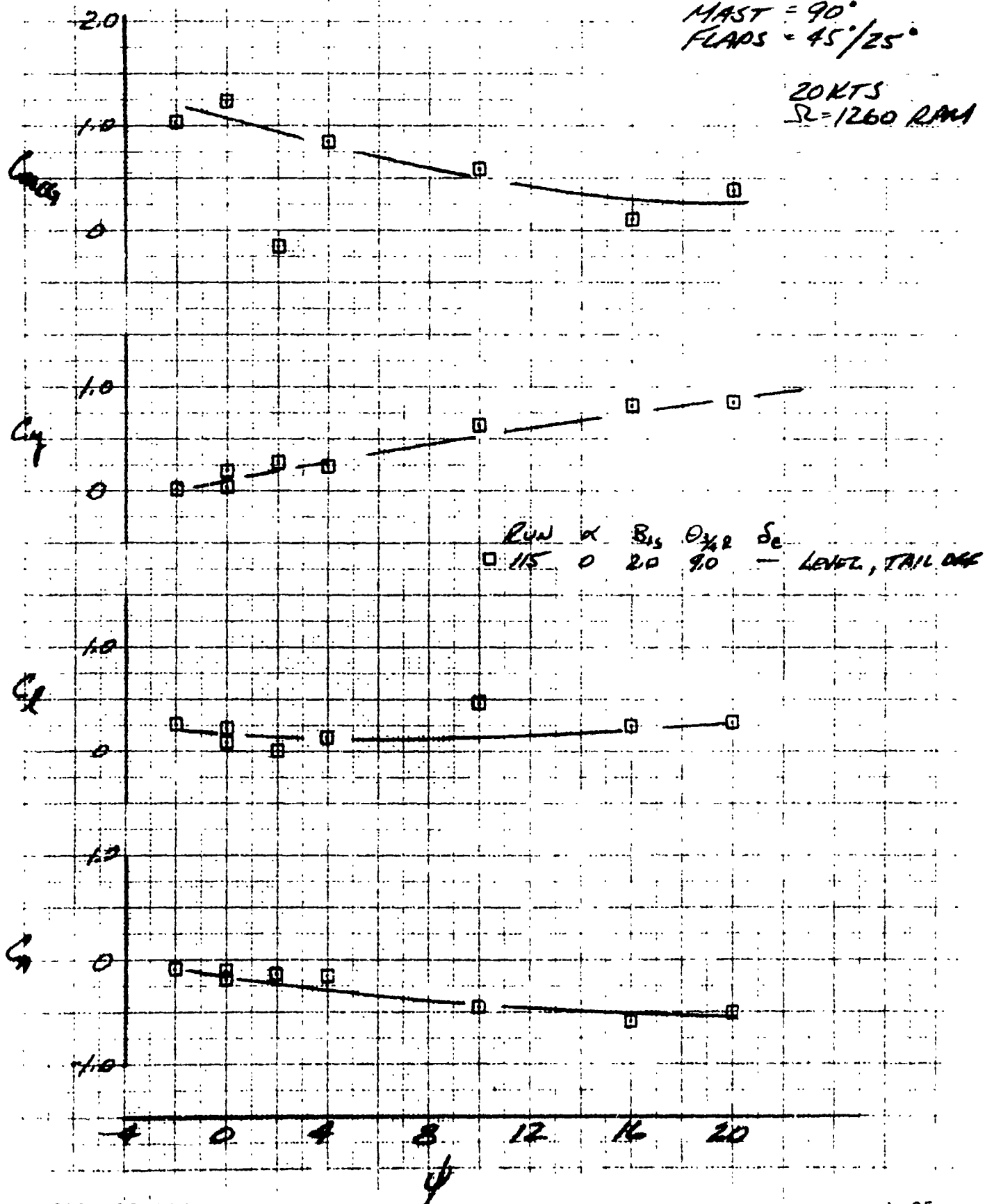
STING MOUNT
MAST = 90°
FLAPS = 45/25°

25 KTS
$\Omega = 1260$ RPM

C_{mcg}

C_L
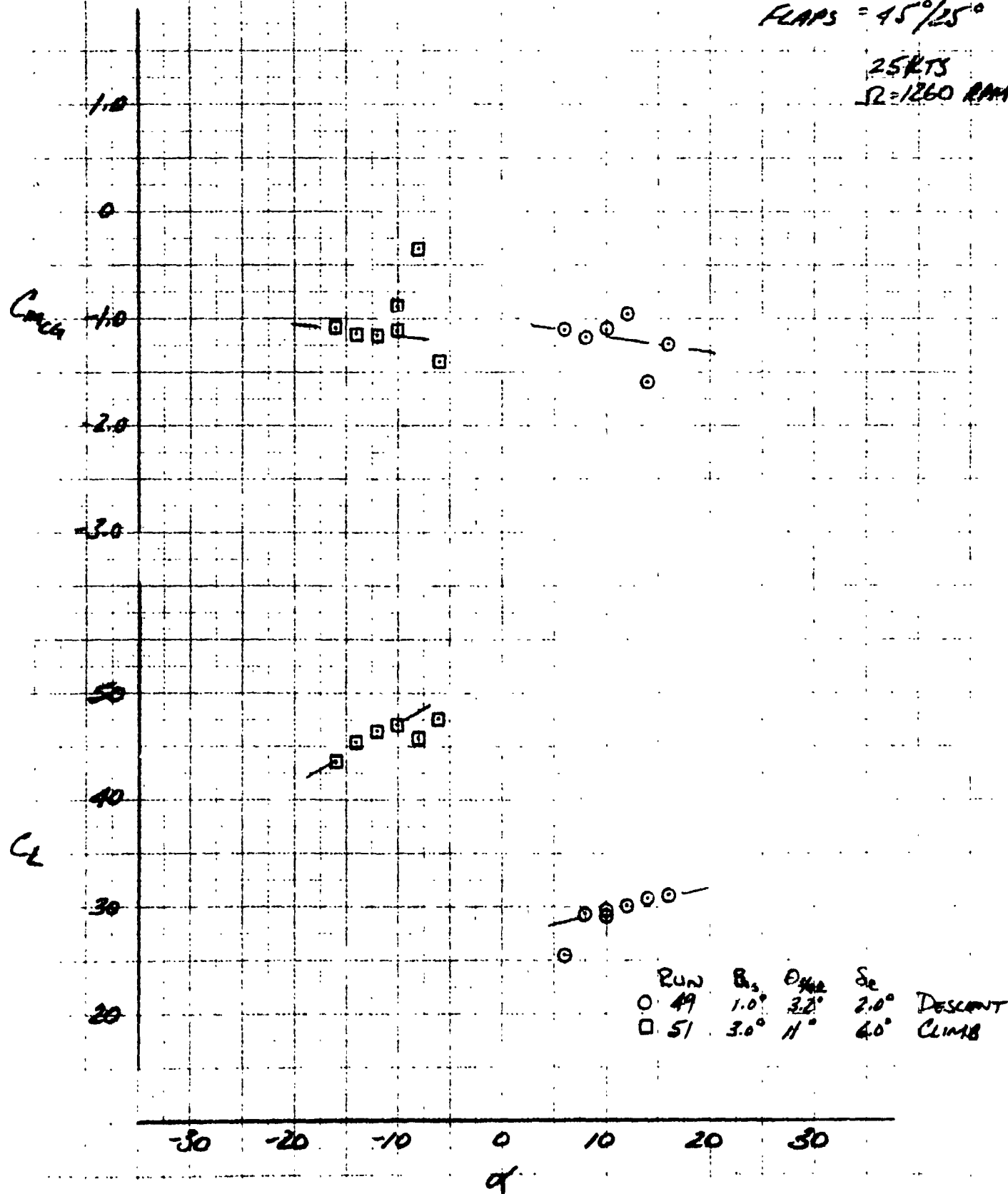
Run	B_s	θ_{75R}	δ_e	
O 49	1.0°	3.2°	2.0°	DESCENT
□ 51	3.0°	11°	4.0°	CLIMB
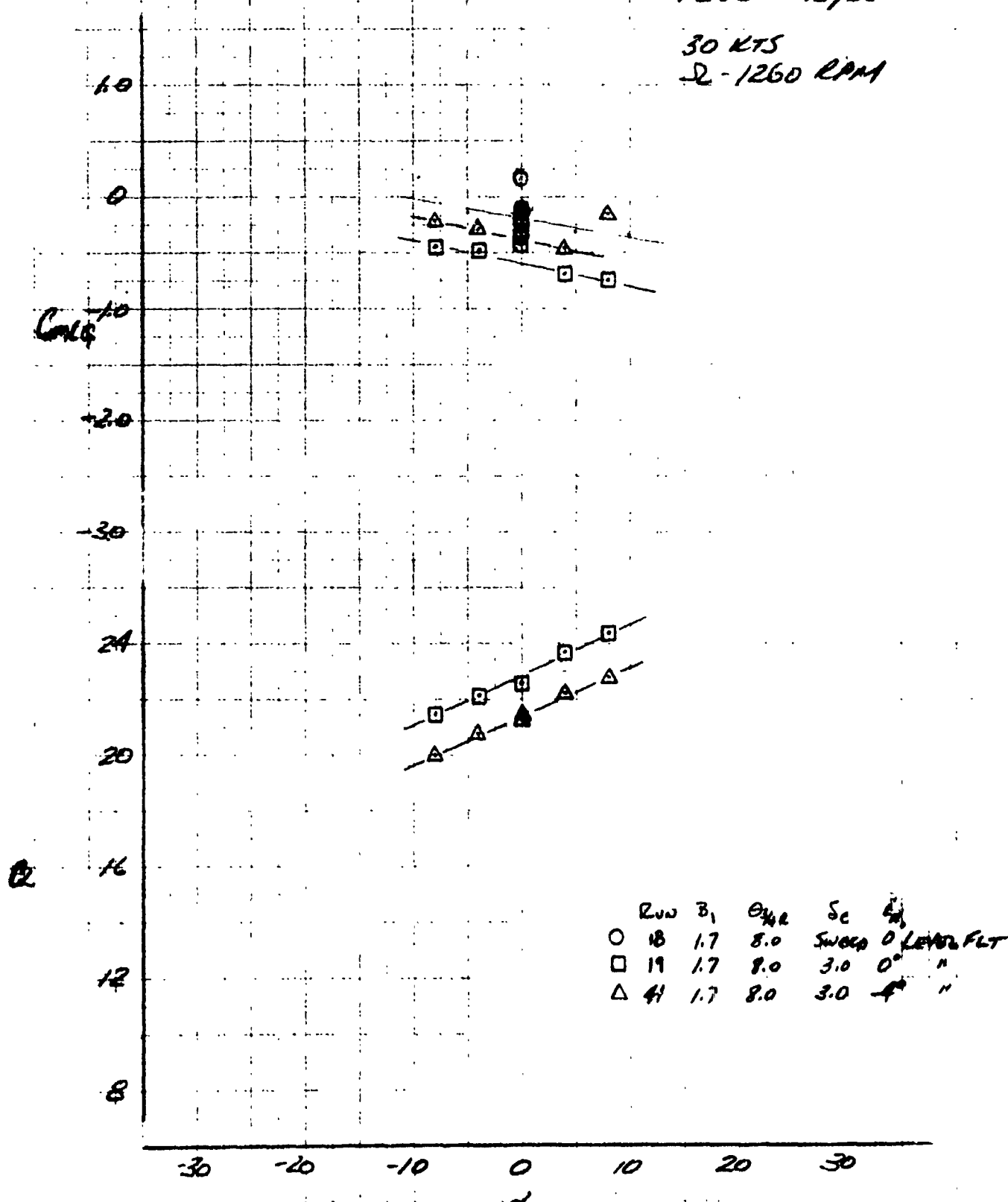
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LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

25 KTS
Ω = 1260

B_{ts} = 1.0
$\theta_{3/4R}$ = 3.2
δ_e = 2.0°

Run α
○ 50 10° CLIMB

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A-37

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 90°
FLAPS - 45/25°

30 KTS
Ω - 1260 RPM

C_{MCG}

α

	Run	B_1	$\theta_{3/4R}$	δ_c	δ_H	
O	18	1.7	8.0	SWCP	0	LEVEL FLT
□	19	1.7	8.0	3.0	0°	"
△	41	1.7	8.0	3.0	4°	"

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST 90°
FLAPS = 45°/25°

30 KTS
Ω = 1260 RPM

Run	B₁	θ₇₅	δₑ	
○ 103	1.7	8.5	— LEVEL, TAIL OFF
□ 117	.7	4.0	— DESCENT, "
△ 118	2.7	10.0	— CLIMB, "

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A-39

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

30 KTS
Ω = 1260 RPM

Run	B_{1s}	$\theta_{.75}$	δ_c			
△	19	1.7°	8.0°	3.0°	LEVEL FLT
O	45	.7°	3.0°	1.0°	DESCENT
□	47	2.7°	10°	5.0°	CLIMB

LSWT 421

BELL
HELICOPTER COMPANY

STING MOUNT
MAST = 90°
FLAPS 45/25

36 KTS
Ω = 1260

Run	α	B₁ₛ	θ₃/₄R	δe	
◯ 20	0°	1.7°	8°	4.0°	Level
☐ 46	10°	.7°	3°	1.0°	Descent
△ 48	-10°	2.7°	10°	5.0°	Climb

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A-41

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

30 KTS
Ω = 1260 RPM

RUN	α	B_{1s}	$Θ_{74R}$	$δ_e$	
104	0°	1.7°	0°	—	LEVER, TAIL OFF
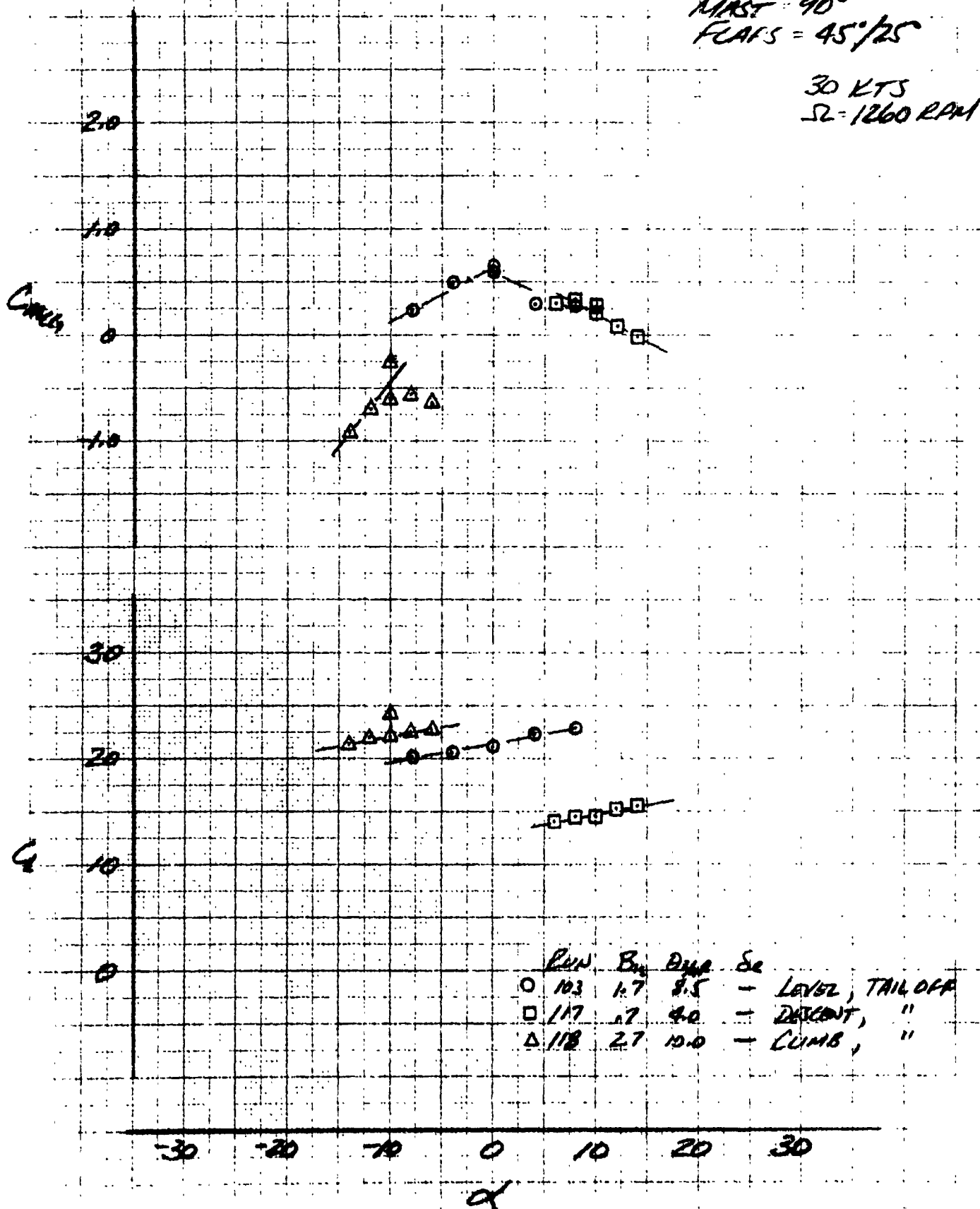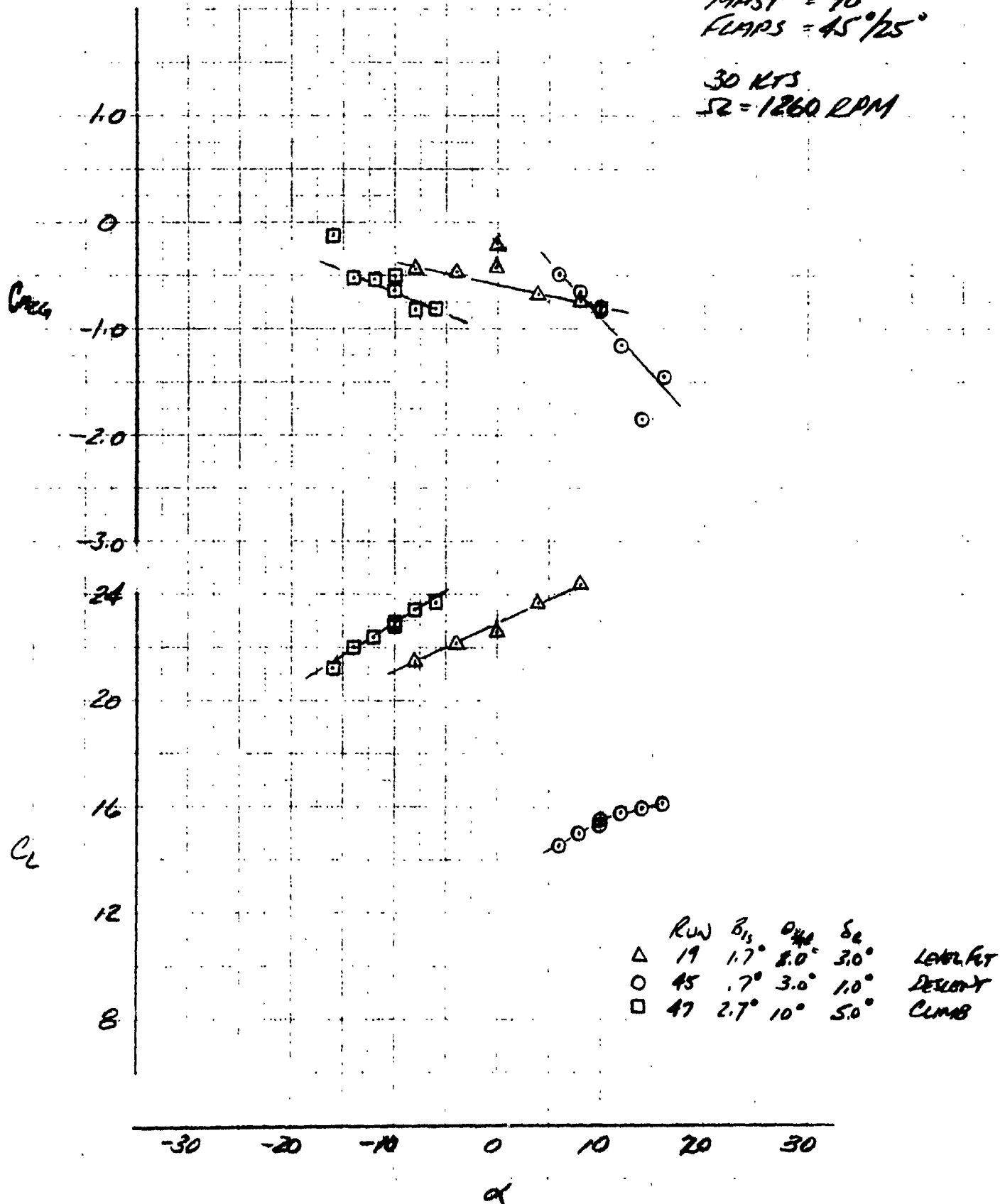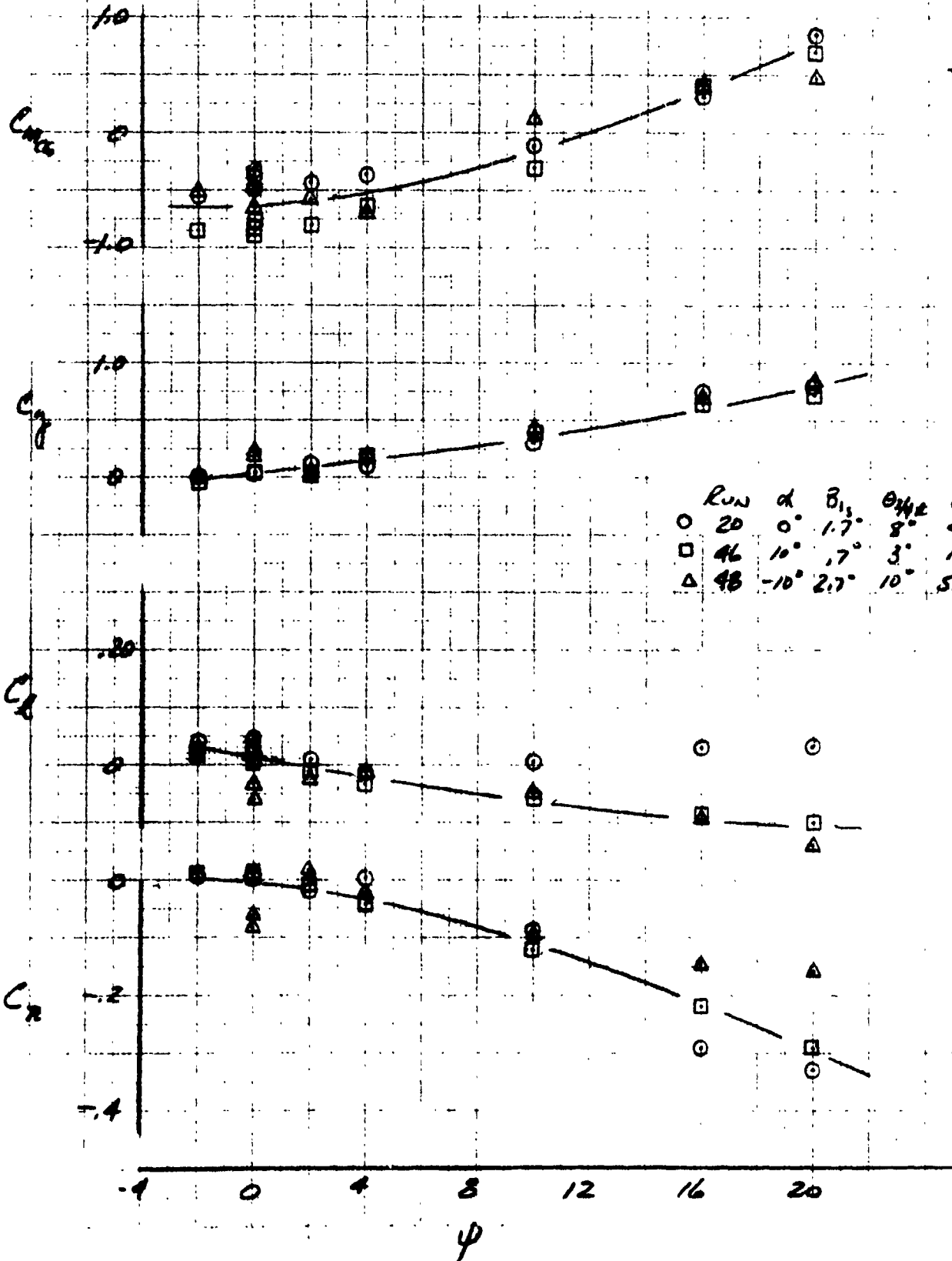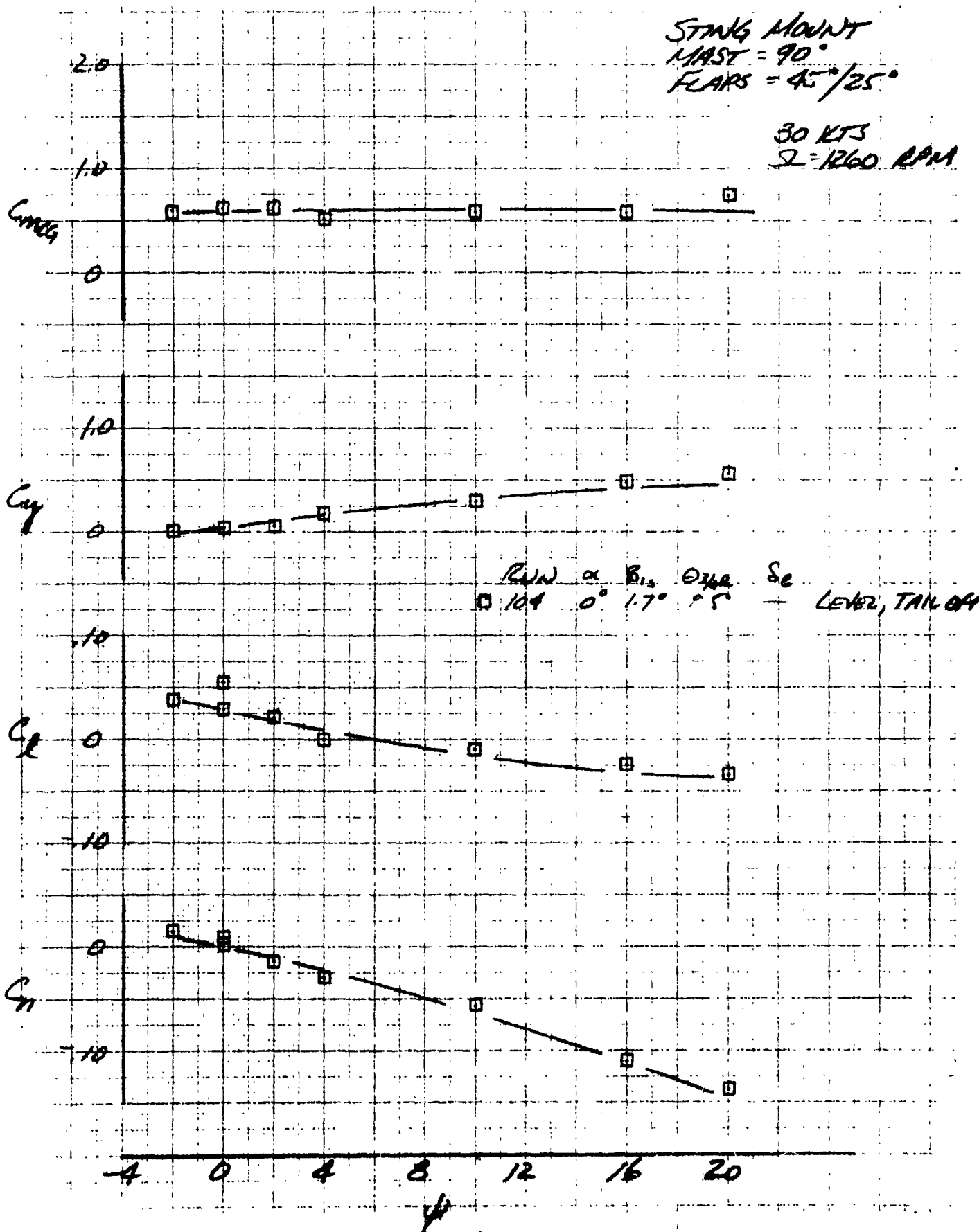
LSWT 421

STING MOUNT
MAST = 90°
FLAPS - 45°/25°
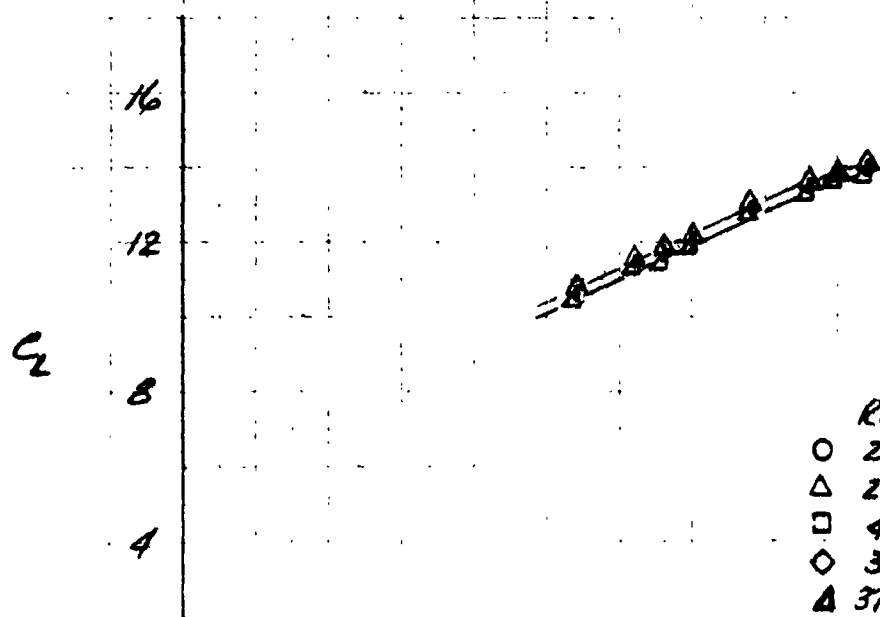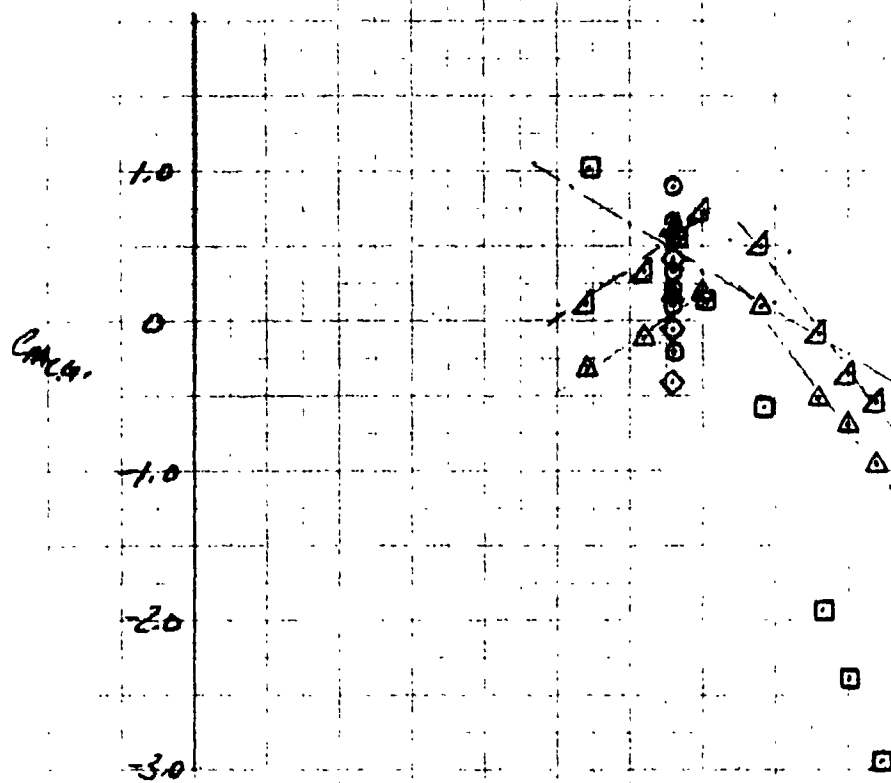
40 KTS
$\Omega = 1260.RPM$

Run	B_1	$\theta_{3/4R}$	δ_e	i_t
○ 26	1.6	7.2	Sweep	0
△ 27	1.6	7.2	3.0	0
□ 4	1.6	7.5	3.0	0
◇ 3	1.6	7.5	Sweep	0
▲ 37	1.7	7.2	3.0	-4°

BELL HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

40 KTS
Ω = 1260 RPM

Run	B_{1S}	θ_{75R}	$\dot{\omega}_c$	
○ 105	4.6	7.5	—	LEVEL, TAIL OFF
□ 119		3.5	—	DESCENT, "
△ 120	2.4	9.8	—	CLIMB, "

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 40/25°

40 KTS
Ω = 1260 RPM

C_{mCG}

C_L

	RUN	B_1	$\Theta_{3/4}$	δ_c	
□	4	1.6	7.5	3.0	LEVEL FLT
◇	3	"	"	"	"
◨	10	.6	7.5	1.0	DESCENT
◨	11	"	5.5	"	"
◨	12	"	3.5	"	"
◁	13	2.6	7.5	5.0	CLIMB
◀	14	"	8.0	"	"
◁	15	"	9.8	"	"

LSWT 421

BELL HELICOPTER company

STING MOUNT
MAST = 90°
FLAPS = 45°/25°

40 KTS
Ω = 1260

Run	α	β_{1s}	$\theta_{3/4R}$	δ_e	
O 43	14°	1°	3.5°	1.0	DESCENT
□ 44	-14°	2.6°	9.8°	5.0	CLIMB

LSWT 421

STING MOUNT
MAST = 90°
FLAPS = 45/25°

40 KTS
Ω = 1260 RPM

Run	α	β₇₅	Θ₃/₄ᵣ	δₑ	
□ 106	-2°	13	7.5	— LONG, TAIL OFF

LSWT 421

BELL HELICOPTER COMPANY

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

16 KTS
Ω = 1260 RPM

C_{mcg}

C_L

RUN	B_{1s}	$\theta_{3/4}$	S_e	
○ 60	-1.8°	9.5°	SWEPT LEVEL	
□ 61	-1.8°	9.5°	-3.6°	"

α

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 55°/25°

16 KTS
Ω = 1260 RPM

$\beta_{1S} = -1.8°$
$\theta_{75R} = 9.5°$
$\delta_\beta = -5.6°$

Run α
O   67.   12°

-4   0   4   8   12   16   20

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LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

20 KTS
Ω = 1260 RPM

C_{MCG}

3.0

2.0

1.0

0

C_L

55

50

45

40

Run	B_{1S}	$θ_{HP}$	$δ_e$	i_N	
○ 63	-1.6	9.2°	Sealed 0°	LEVEL
□ 64	-1.6	9.2°	-3.2	0°	"
△ 78	-1.6	10°	-3.2	-4°	"

-30   -20   -10   0   10   20   30

α

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

20 KTS
Ω = 1260 RPM

RUN α B₁ₛ θ₄₅ δe
○ 65 12° -1.6° 9.2° -3.2° LEVEL

C_{req}

C_q

C_l

C_n

ψ

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

30 KTS
Ω = 1260 RPM

C_{LG}

C_{D}

RUN	b_{15}	$θ_{HR}$	$δ_e$	i_0	
○ 66	-1.6°	7.6°	Surop	0	Levsr
□ 67	-1.6°	7.6°	-3.2°	0	"
△ 79	-1.6°	8.0°	-3.2°	-4	"

α

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

30 KTS
Ω = 1260 RPM

$C_{m_{cg}}$

3.0

2.0

1.0

0

C_L

26

24

22

20

18

Run	B_{1s}	$\theta_{3/4R}$	δ_e	
○ 67	-1.6	7.6	-3.2	LEVEL
□ 101	-1.6	8.2	—	LEVEL, TAIL OFF

-30 -20 -10 0 10 20 30

α

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A-53

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

30 KTS
$\Omega = 1260$ RPM
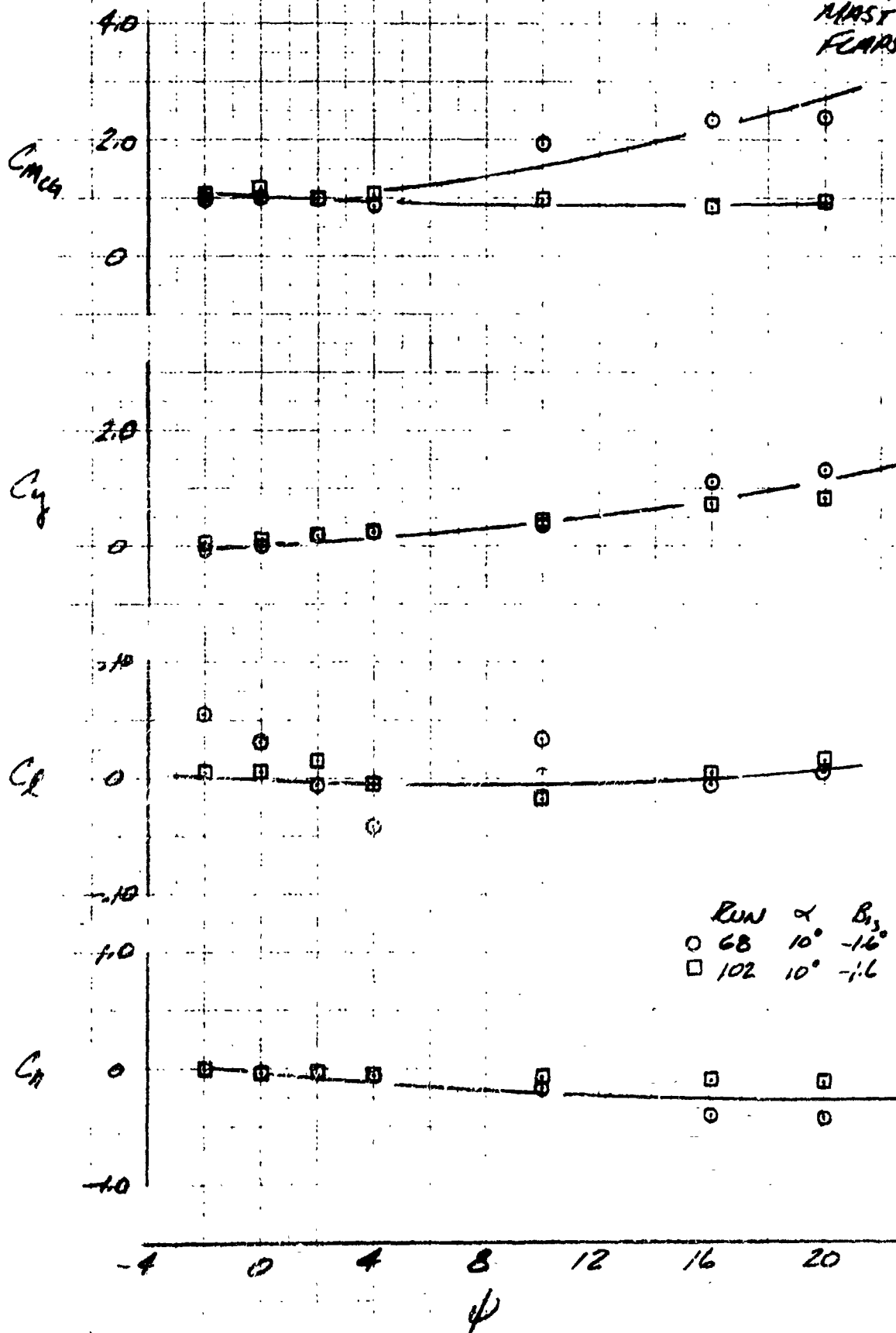
4.0

2.0 C_{MCG}

0

2.0 C_y

0

.10 C_ℓ

0

-.10

4.0

C_n 0

-4.0

Run	α	B_{1}	$\theta_{.75R}$	δ_e	
O 68	10°	-1.6°	7.6°	-3.6°	LEV'R
□ 102	10°	-1.6	8.2	—	", TAIL OFF

-4   0   4   8   12   16   20

ψ

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A-54

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

40 KTS
Ω = 1260 RPM

$C_{m_{CG}}$

3.0

2.0

1.0

0

Run	B_{1s}	$\Theta_{.75R}$	δ_e	i_H	
○ 69	-1.8°	5.2	Swept	0°	Level
□ 70	-1.8°	5.2	-3.6°	0°	"
△ 80	-1.8°	7.0	-3.6°	-4°	"
◇ 83	-2.8°	1.5°	-5.6°	0°	Descent
▽ 85	-.8°	9.5°	-1.6	0°	Climb

14

C_L

12

10

8

6

-30   -20   -10   0   10   20   30

α

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A-55
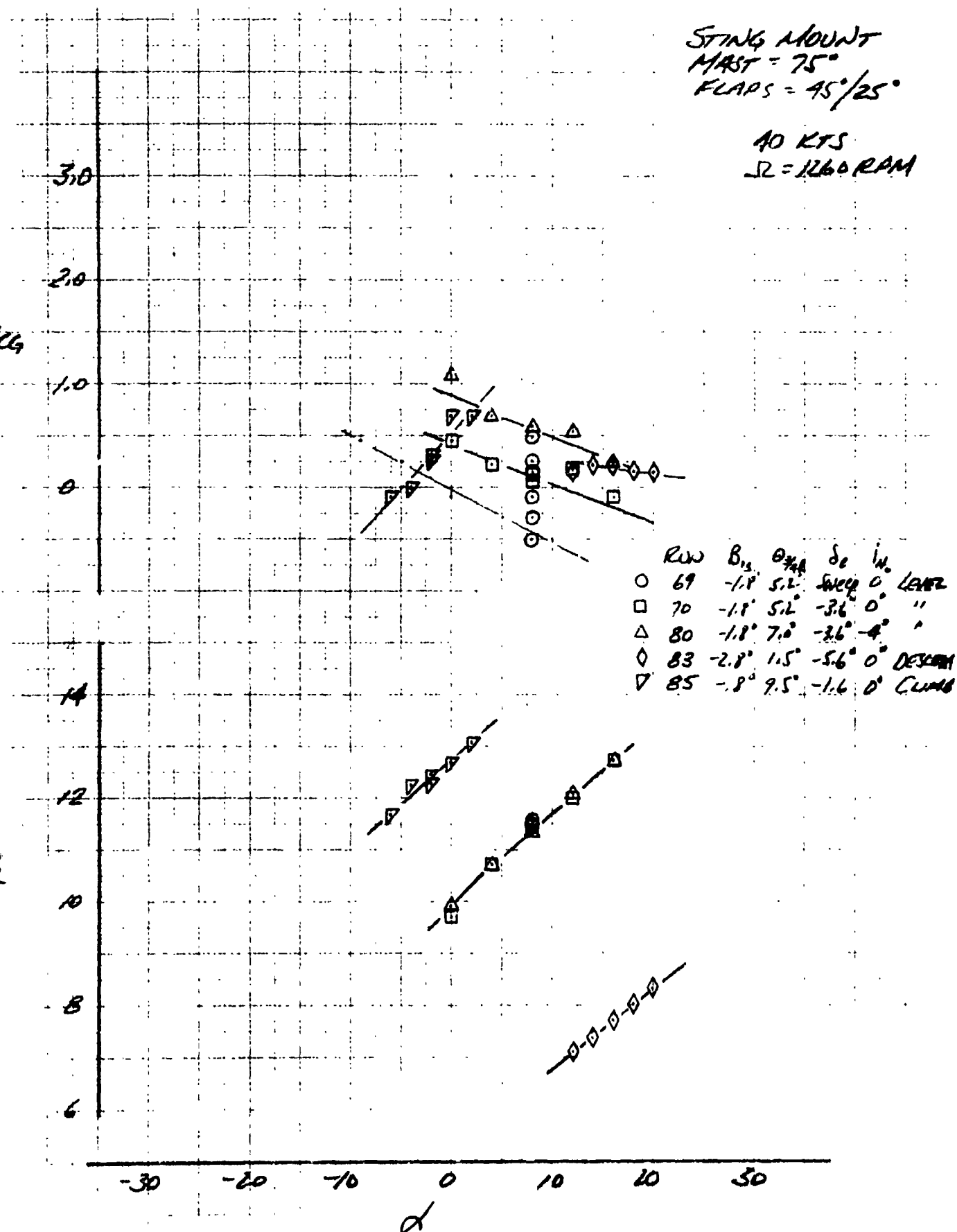
LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

40 KTS
52 = 1260 RPM

Run	B₁₅	Θ₇₄ᵣ	δₑ	
○ 70	-1.8	5.2	-3.6	LEVEL
□ 99	-1.8	6.6	—	LEVEL, TAIL OFF

C_{MEG}

C_L

α

LSWT 421

STATIC MOUNT
MAST = 75°
FLAPS = 45°/25°

40 KTS
Ω = 1260 RPM

Run	α	b_{1s}	$θ_{.75}$	$δ_e$	
○ 71	8	-1.8°	5.2	-3.6	LEVEL
□ 84	16	-2.8°	1.5	-5.6	DESCENT
△ 86	-2	-.9	9.5	-1.6	CLIMB

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 75°°
FLAPS = 45°/25°

40 KTS
$\Omega = 1260$ RPM

C_{m_g}

C_g

C_l

RUN	α	β_{1s}	$\theta_{.75}$	δ_e	
○ 71	8.0	+.8	5.2	-3.6	LEVEL
□ 100	8.0	-1.8	6.6	—	LEVEL, TAIL OFF

C_n

ψ

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

60 KTS
Ω = 126.0 RPM

C_{MCG}

C_L

RUN	B_{1S}	$θ_{3/4R}$	S_e	i_w	
○	72	-1.7°	4.0°	Sweep	0°	LEVEL
□	73	-1.7°	4.0°	-3.4°	0°	"
△	81	-3.4°	4.0°	-3.4°	-4°	"
◇	87	-2.7°	0°	-5.4°	0°	DESCENT
▽	89	.3°	8.5°	.6°	0°	CLIMB

α

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

60 KTS
Ω = 1260 RPM

C_{mcg}

C_L

RUN	B_{1s}	θ_{x4z}	δ_e	
○ 73	-1.7	4.0	-3.4	LEVEL
□ 97	-1.7	4.0	—	LEVEL, TAIL OFF

α

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A-60

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

60 KTS
Ω = 1260 RPM

Cm_{CG}

C_y

Run	α	B_{1s}	θ_{tail}	δ_e	
○ 74	4	-1.7	4°	-3.4°	LEVEL
□ 88	14	-2.7	0	-5.4°	DESCENT
△ 90	-6	.3	8.5	-.6°	CLIMB

C_ℓ

C_n

ψ

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

60 KTS
Ω = 1260 RPM

FLAPS	α	B_{1_s}	$θ_{3/4}$	$δ_{1c}$	
74	4	-17	4.0	-34	LEVEL
90	4	-17	4.0	—	LEVEL, TAKEOFF

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

80 KTS
Ω = 1260 RPM

RUN	B_{1S}	θ_{75R}	δ_E	i_H	
○ 75	1.1°	5.5°	SWEEP	0°	LEVEL
□ 76	1.1°	5.5°	2.2°	0°	"
△ 82	1.1°	6.0°	2.2°	-4°	"
◇ 91	.2	1.0	4.0	0	DESCENT
▽ 93	3.2	10	6.4	0	CLIMB

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

80 KTS
$\Omega = 1260$ RPM

$C_{m_{cg}}$

C_ℓ
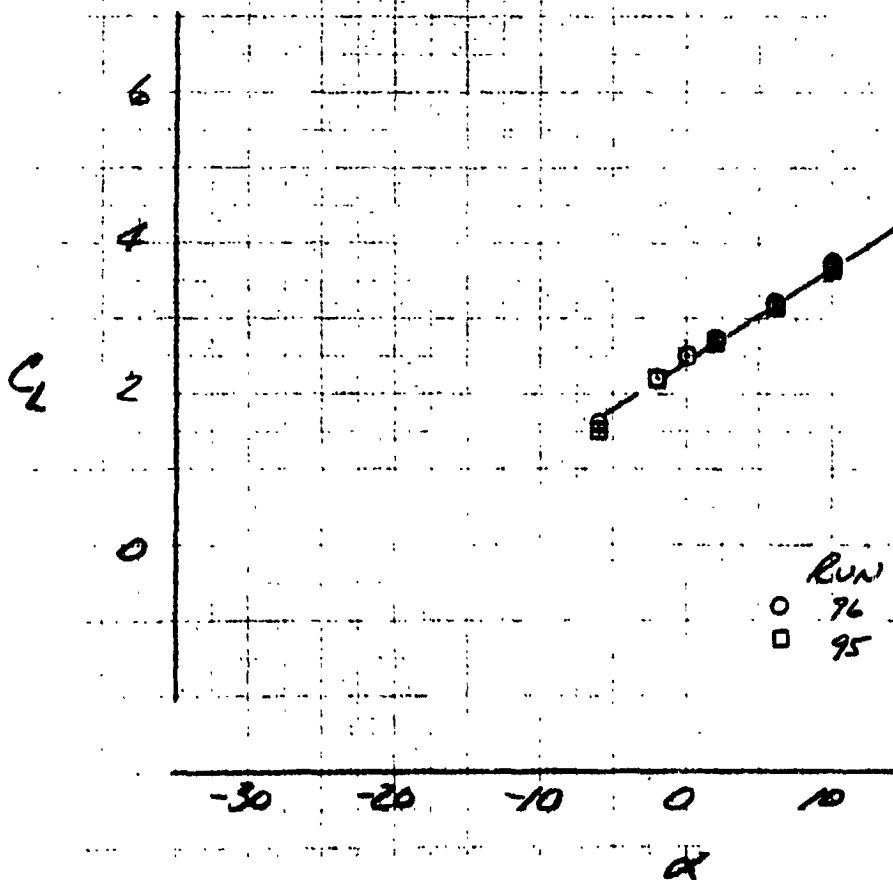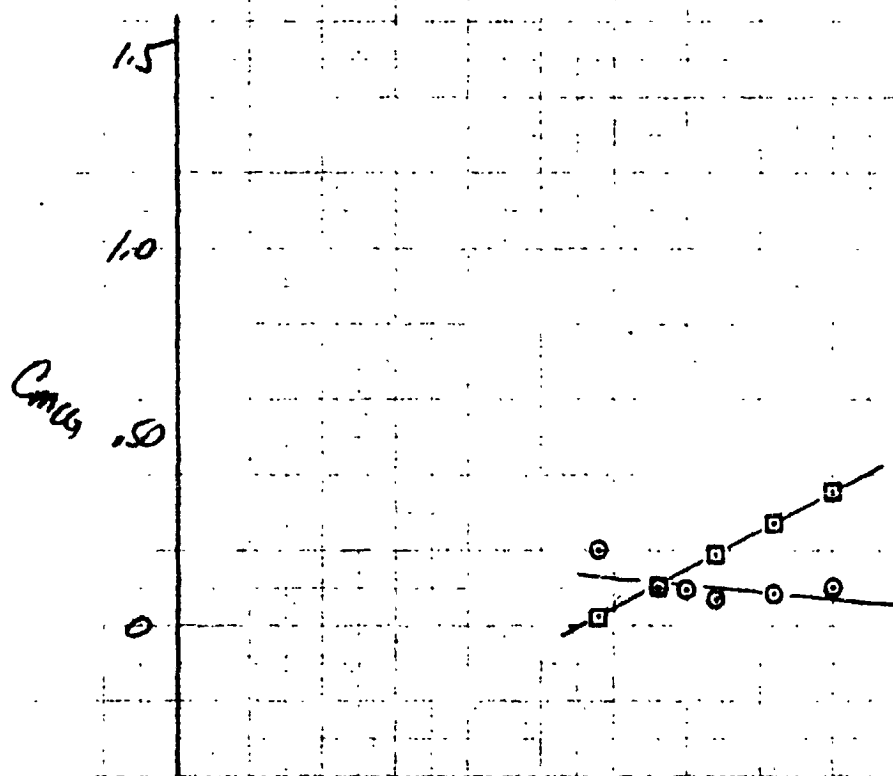
Run	B_{1s}	θ_{7R}	δ_e	
O 96	1.1	5.5	2.2	LEVEL
□ 95	1.1	5.5	—	LEVEL, TAIL OFF
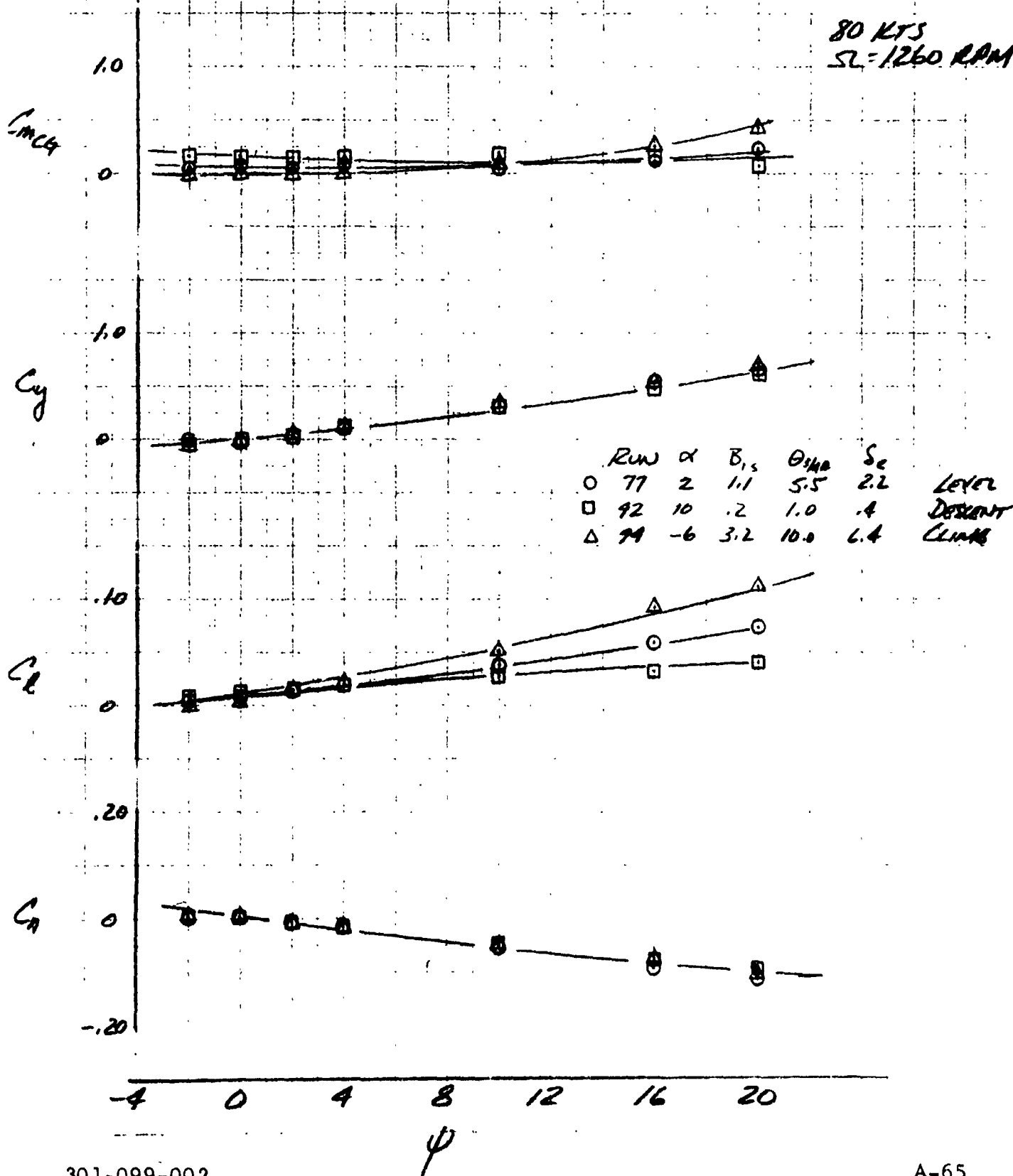
α

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A-64

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

80 KTS
Ω = 1260 RPM

C_{mc_4}

1.0

0

C_y

1.0

0

Run	α	B_{1s}	$\Theta_{3/4}$	S_e	
○ 77	2	1.1	5.5	2.2	LEVEL
□ 92	10	.2	1.0	.4	DESCENT
△ 74	-6	3.2	10.0	6.4	CLIMB

C_l

.10

0

C_A

.20

0

-.20

-4   0   4   8   12   16   20

ψ

BELL HELICOPTER COMPANY

LSWT 42'

STING MOUNT
MAST = 75°
FLAPS = 15°/25°

80 KTS
Ω = 1260 RPM

Run	α	B₁ₛ	θ₇₅	δₑ	
○ 77	2.0	1.1	5.5		LEVEL
□ 96	2.0	1.1	5.5	—	LEVEL, TAIL OFF

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°
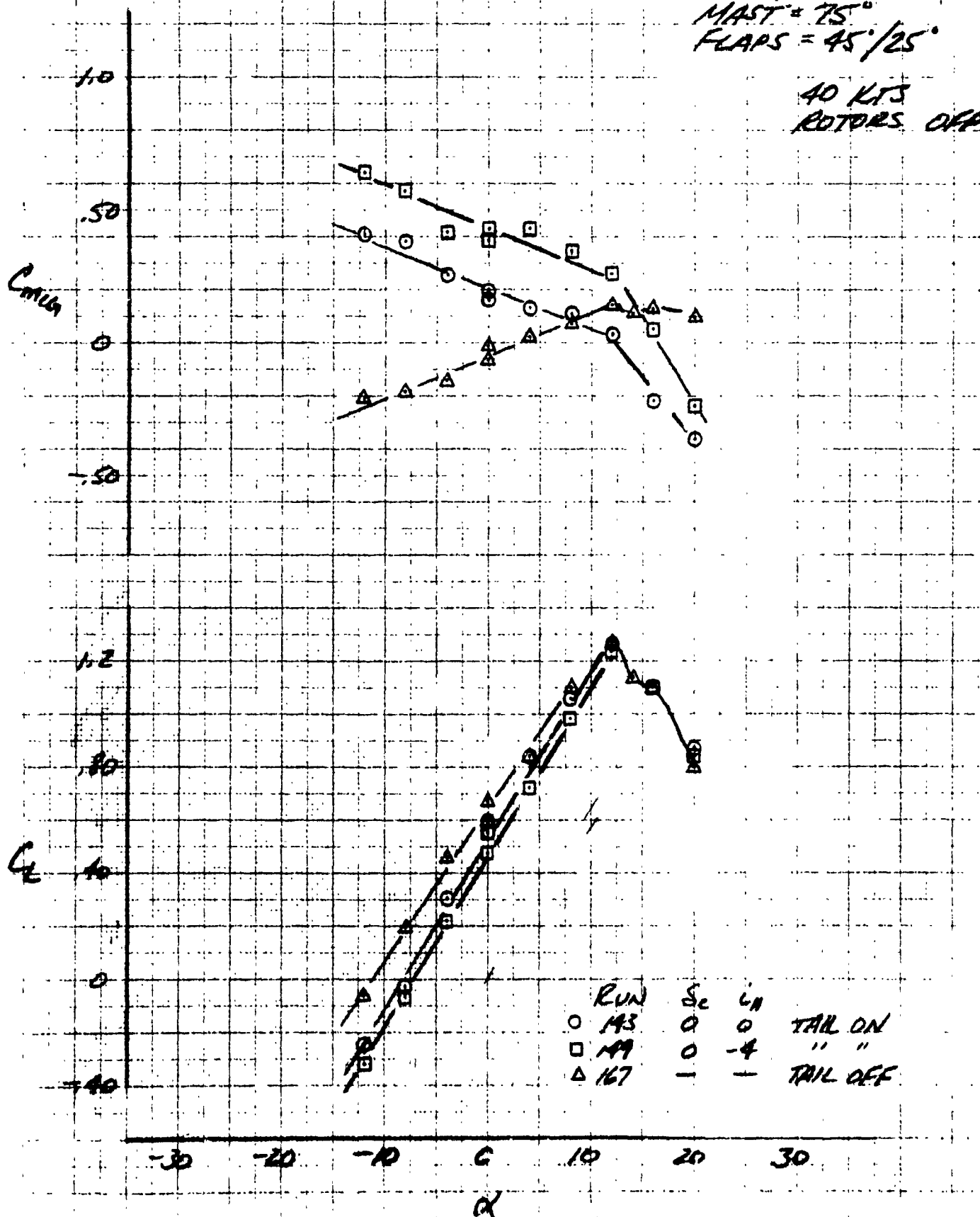
40 KTS
ROTORS OFF

Run	δ_e	i_H	
○ 143	0	0	TAIL ON
□ 149	0	-4	" "
△ 167	—	—	TAIL OFF

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LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

40 KTS
ROTORS OFF

RUN	δ_e	i_n	
○	144	0	0	TAIL ON
□	168	—	—	TAIL OFF

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LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45/25°

60 KTS
ROTORS OFF

Run	δ_e	i_H	
○ 145	0	0	TAIL ON
□ 150	0	-4	" "
△ 169	—	—	TAIL OFF

α
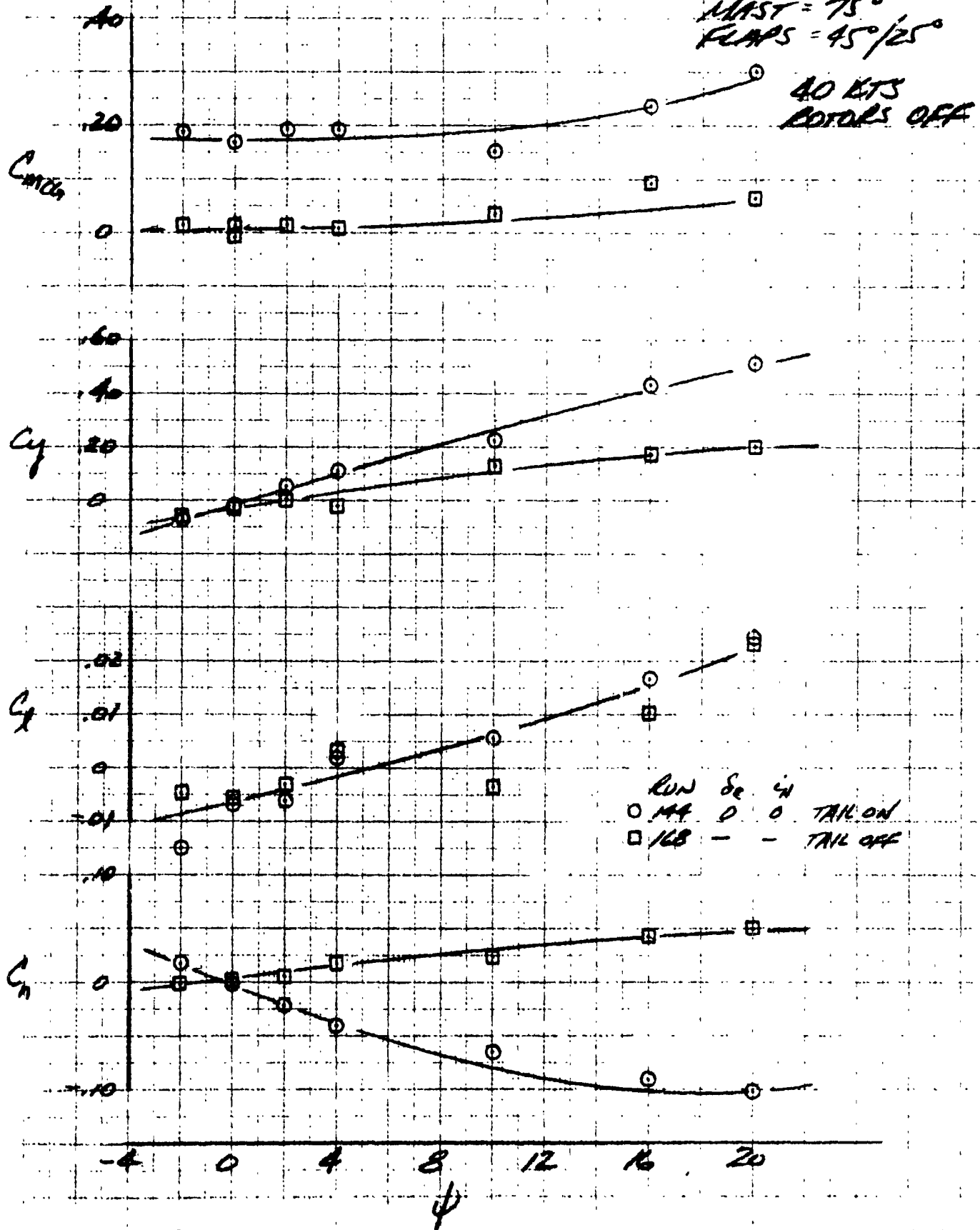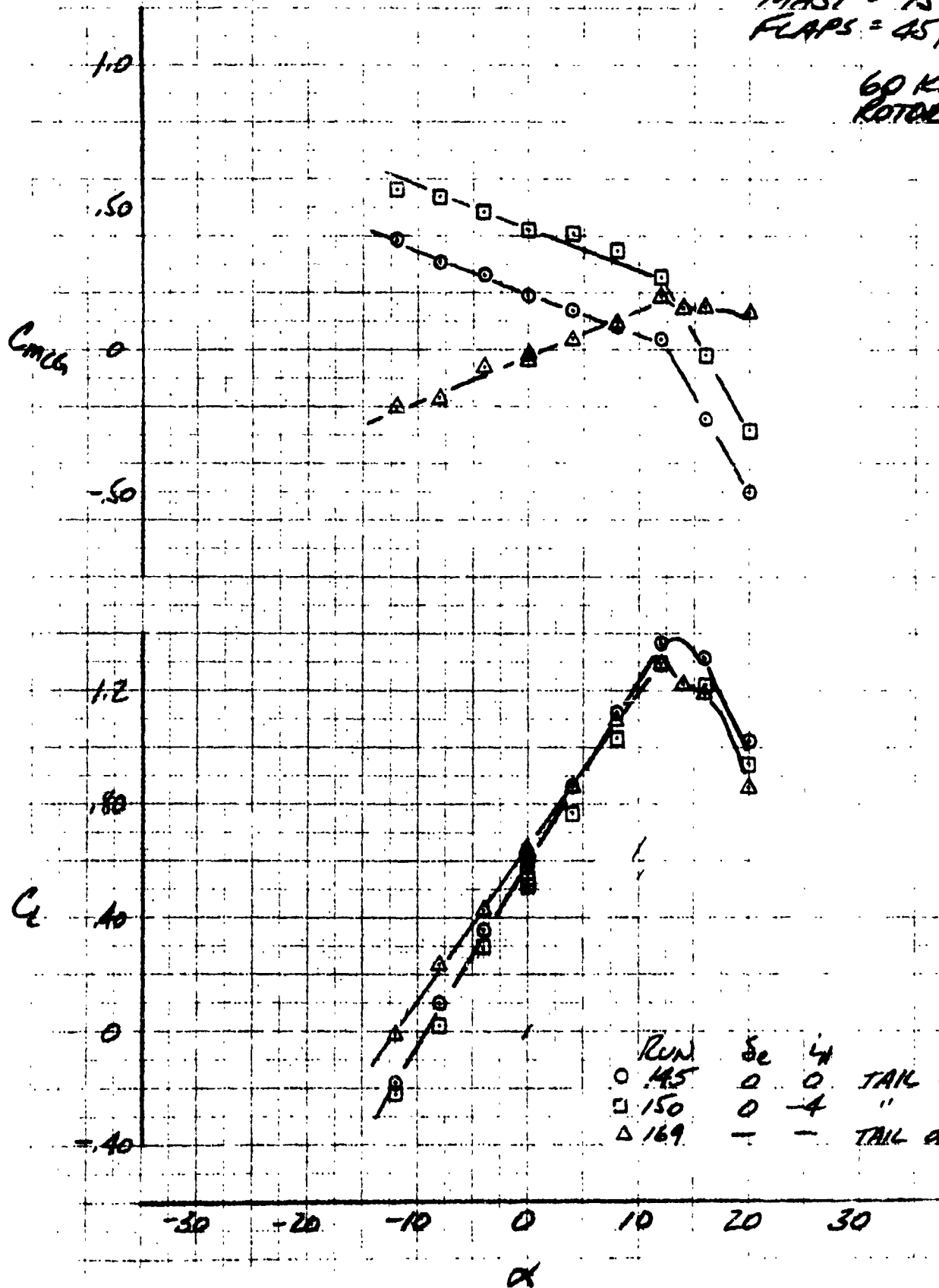
LSWT 421

STING MOUNT
MAST = 15°
FLAPS = 45/25°

60 KTS
ROTORS OFF

Run	δ_e	i_H	
○ 146	0	0	TAIL ON
□ 170	--	--	TAIL OFF
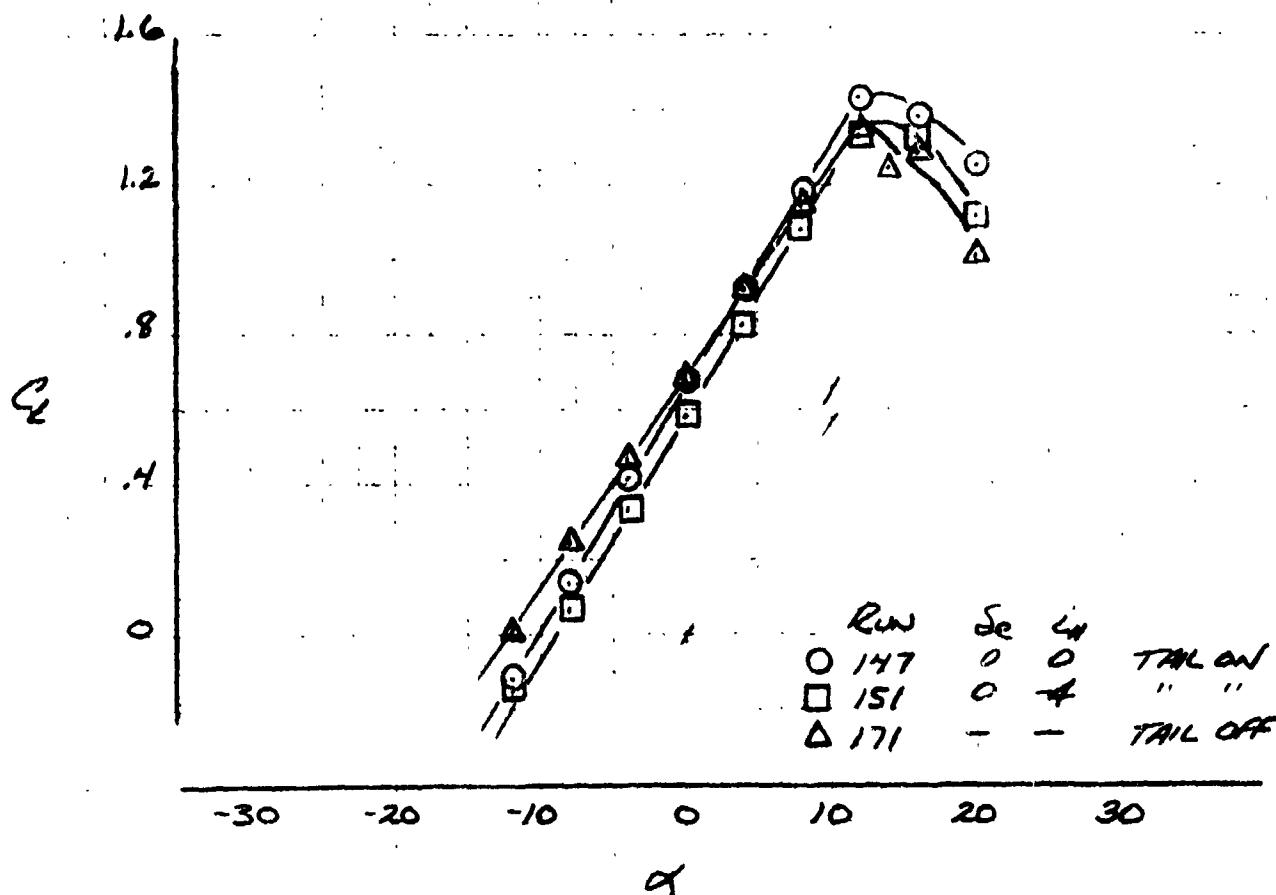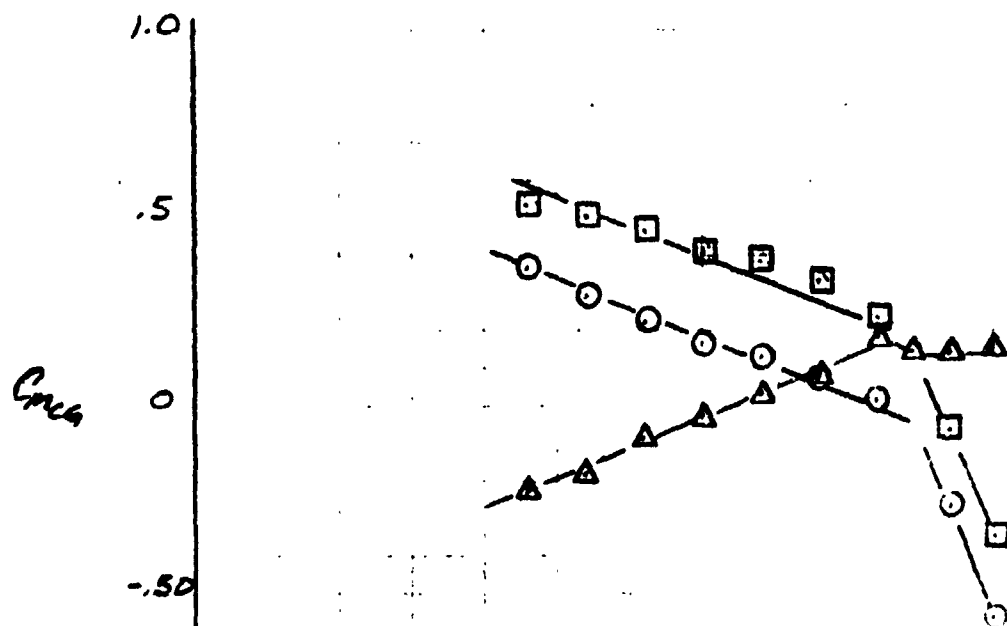
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A-70

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

80 KTS
ROTORS OFF

Run	δ_e	i_H	
◯ 147	0	0	TAIL ON
☐ 151	0	4	" "
△ 171	—	—	TAIL OFF

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A-71

LSWT 421

STING MOUNT
MAST = 75°
FLAPS = 45°/25°

80 KTS
ROTORS OFF

RUN	δ_e	i_h	
○ 198	0	0	TAIL ON
□ 172	—	—	TAIL OFF

ψ

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 60°
FLAPS - 45°/25°

80 KTS
SZ - 1260 RPM

RUN	B_{15}	Θ_{34R}	δ_c	
O 132	-1.7	7.0	-4°	LEVEL
□ 122	-1.7	7.0	—	LEVEL, TAIL OFF

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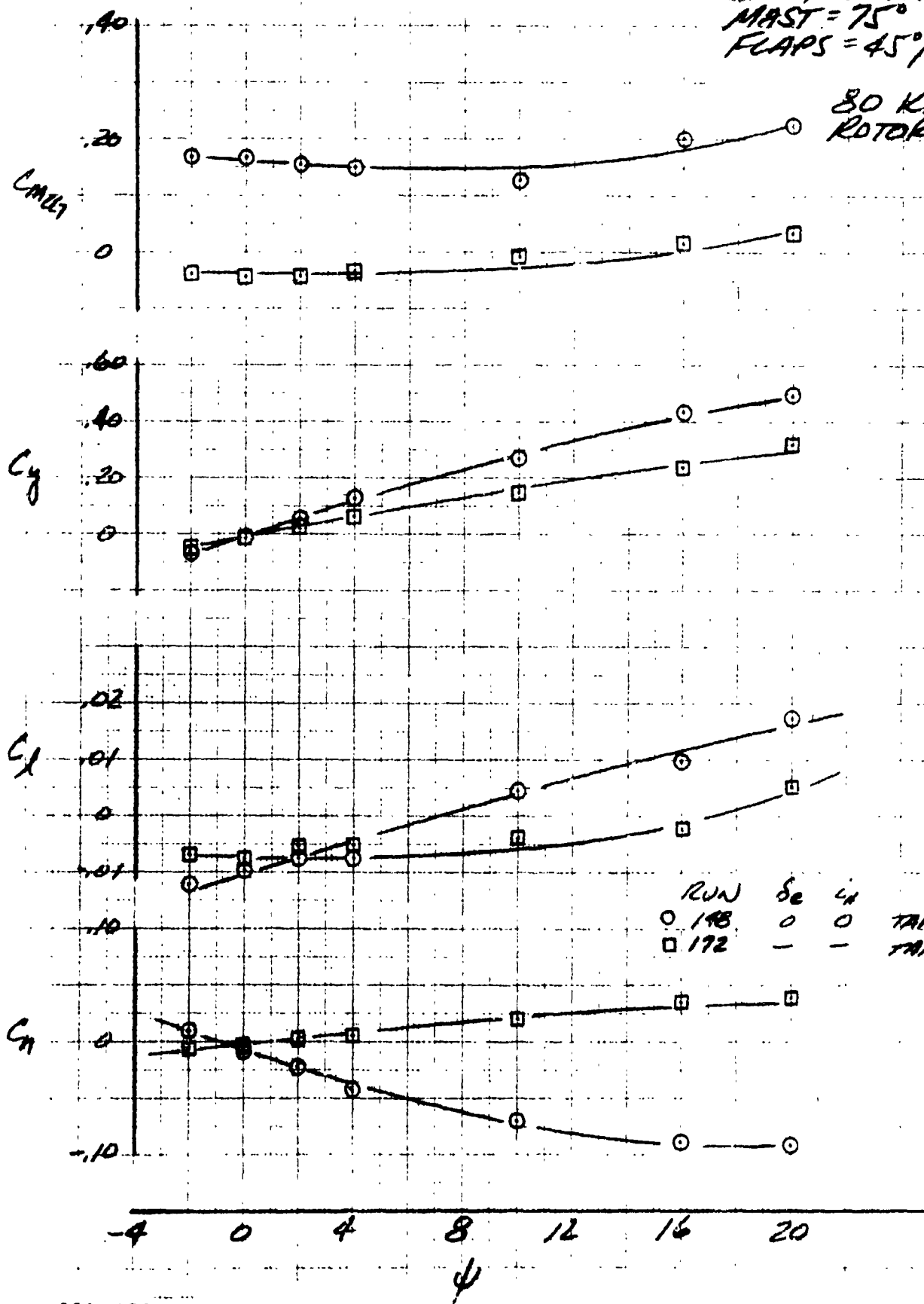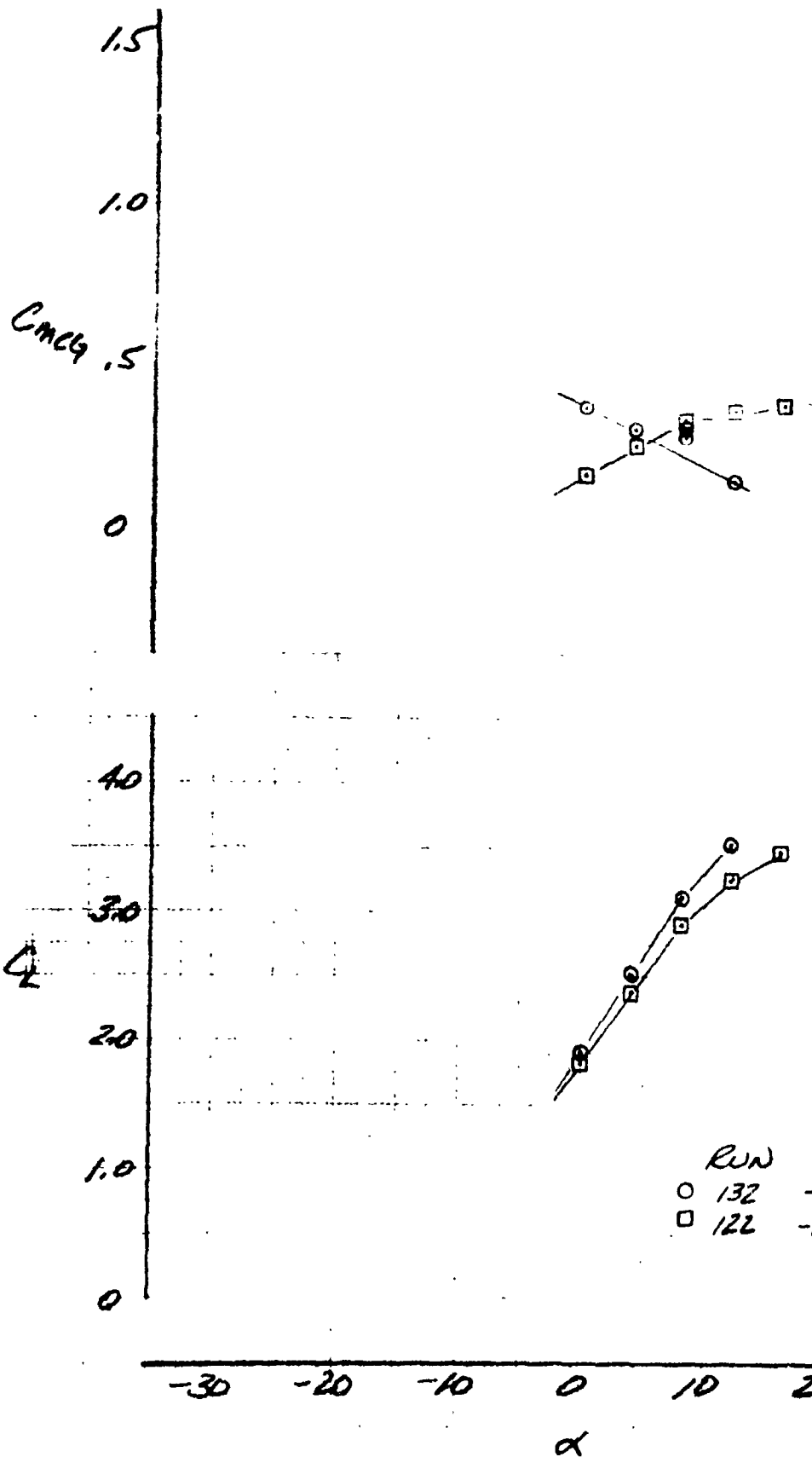
BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 60°
FLAPS = 45°/25°

80 KTS
Ω = 1260 RPM
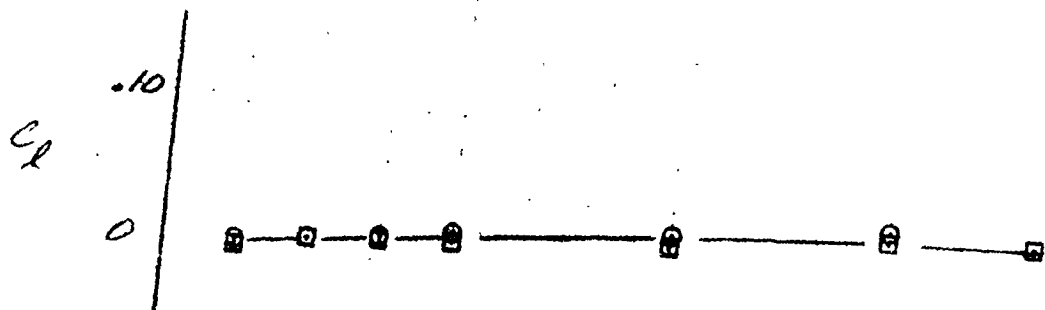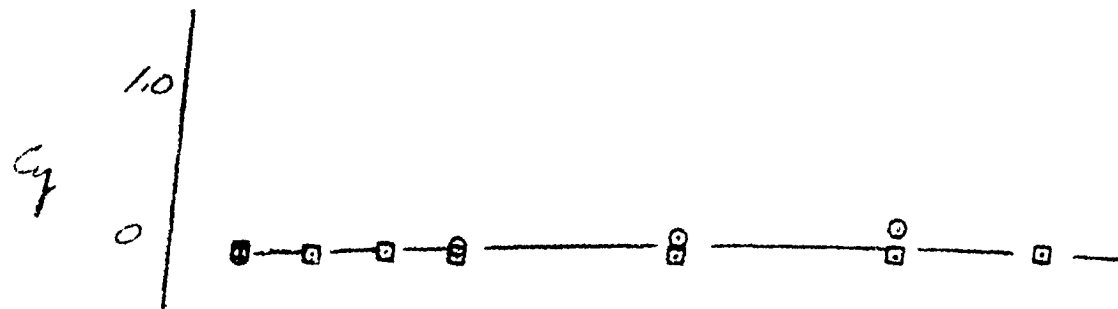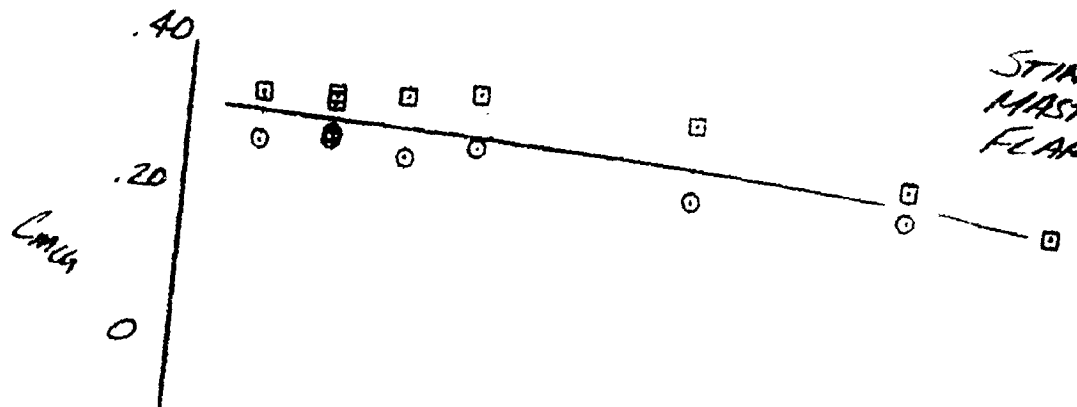
RUN	α	B_{1S}	θ_{34R}	δ_e	
○ 133	8	-1.7	7.0	-4	LEVEL
□ 123	8	-1.7	7.0	—	LEVEL, TAIL OFF
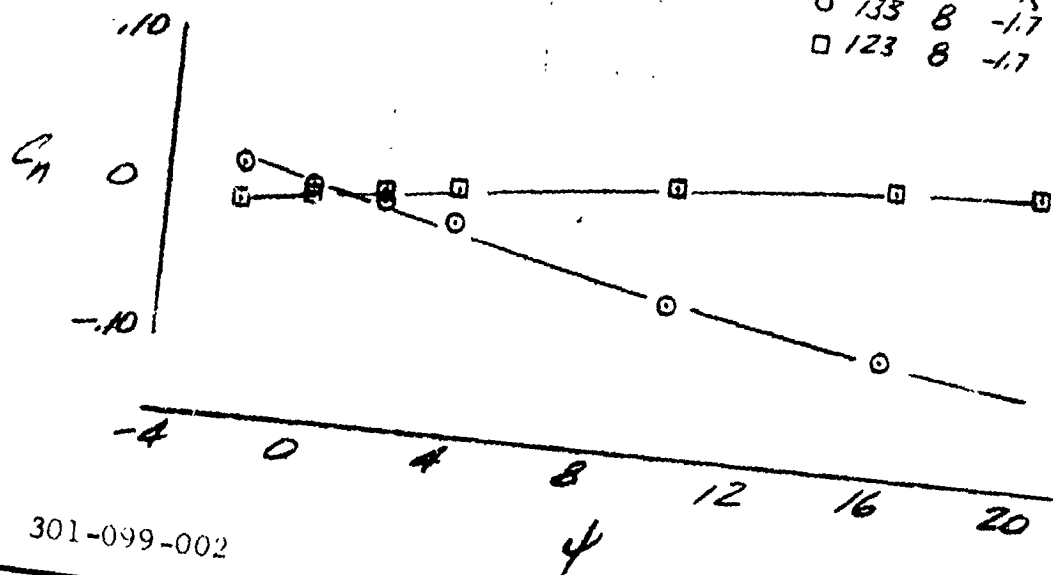
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LSWT 421

STING MOUNT
MAST = 60°
FLAPS = 45°/25°
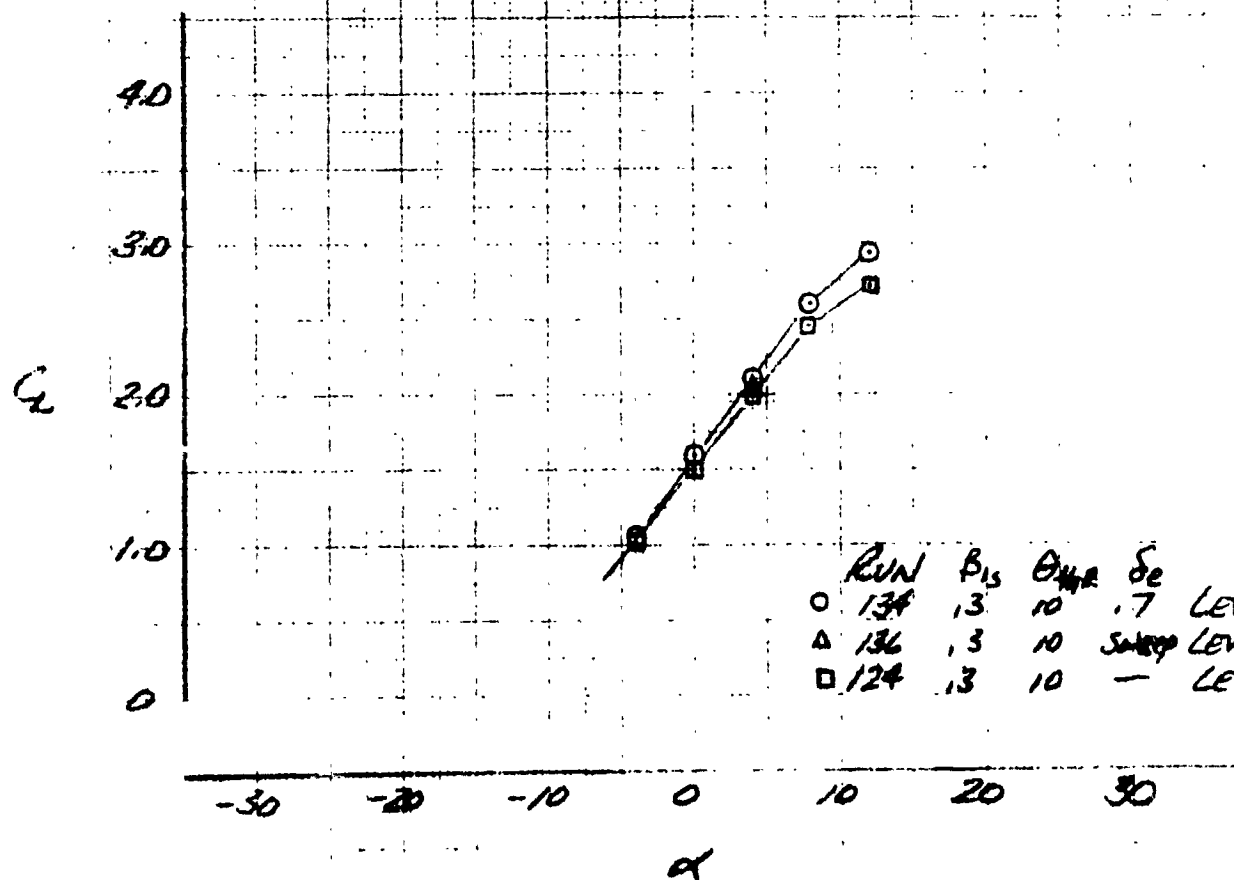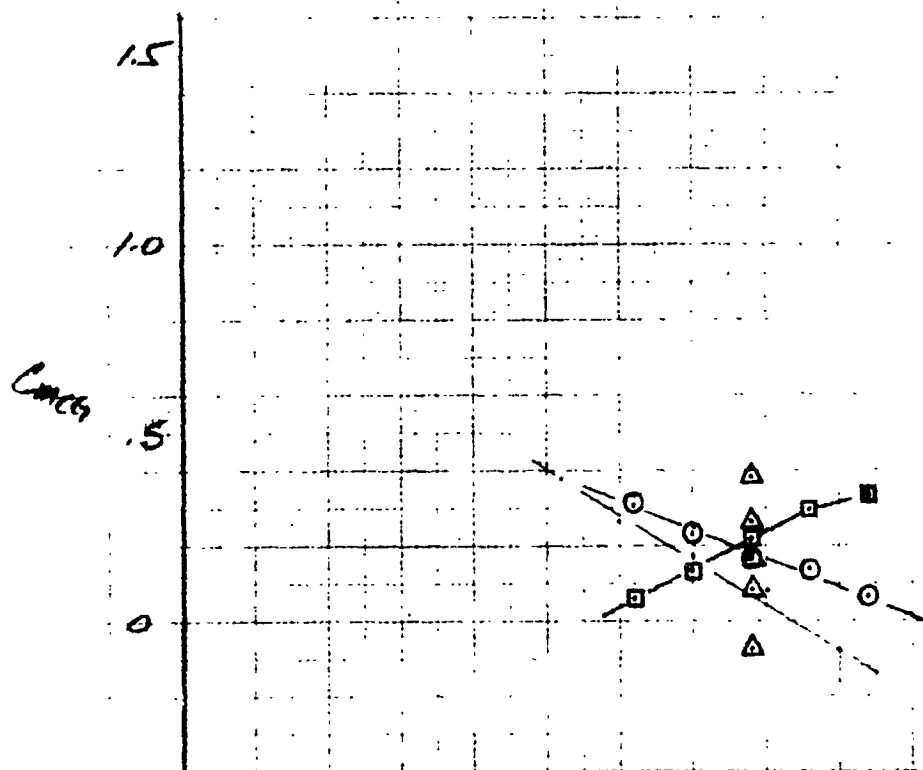
100 KTS
Ω = 1260 RPM

RUN	B₁ₛ	θ₇₅	δₑ	
○ 134	.3	10	.7	LEVEL
△ 136	.3	10	Sweep	LEVEL
□ 124	.3	10	—	LEVEL, TAIL OFF

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 60°
FLAPS = 45°/25°

100 KTS
$\Omega = 1260$ RPM
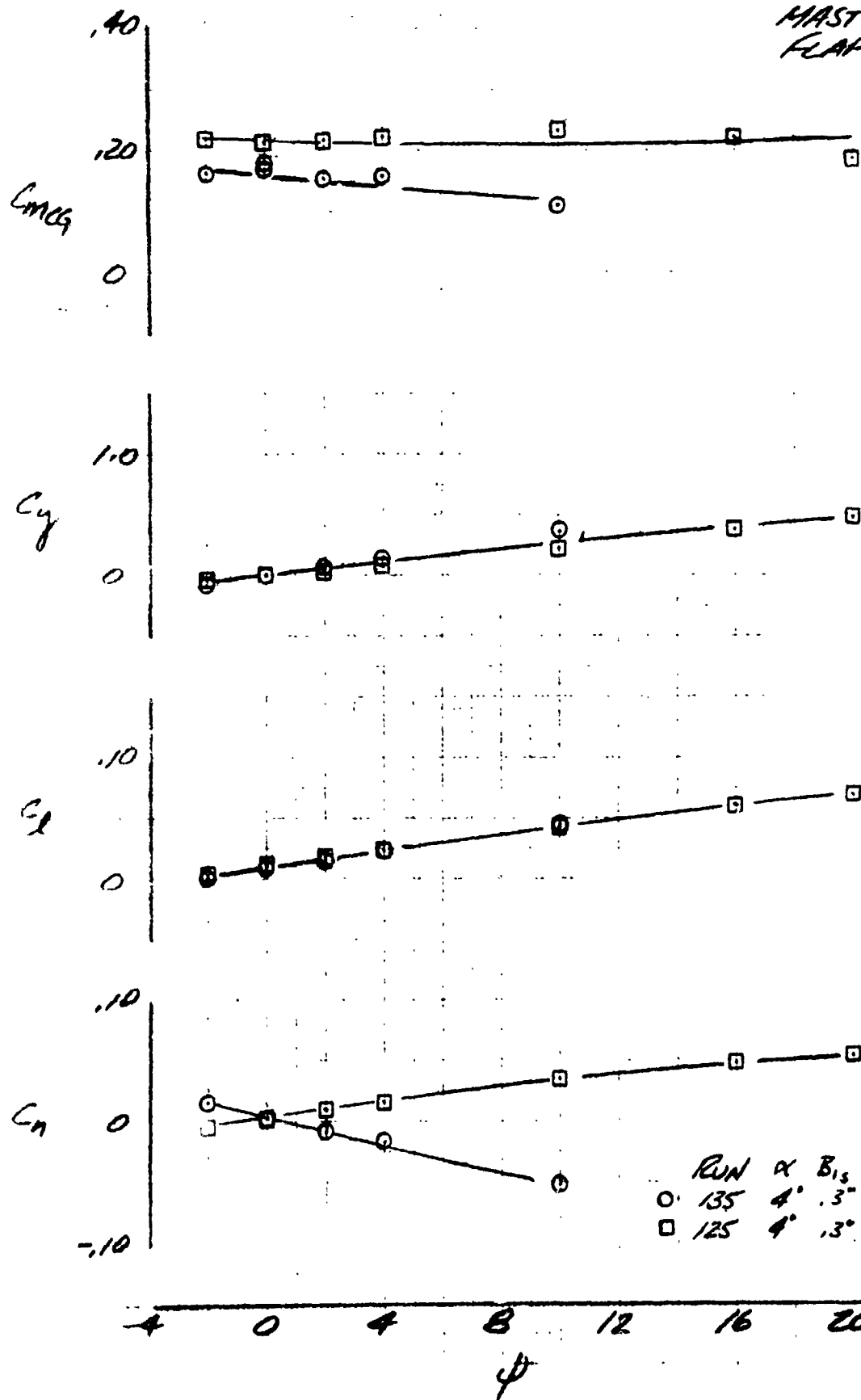
RUN	α	B_{1S}	$\theta_{3/4R}$	δ_e	
O 135	4°	.3°	10°	.7°	LEVEL
□ 125	4°	.3°	10°	—	", TAKOFF

**BELL
HELICOPTE** COMPANY

CSWT 421

STING MOUNT
MAST = 60°
FLAPS = 45°/25°
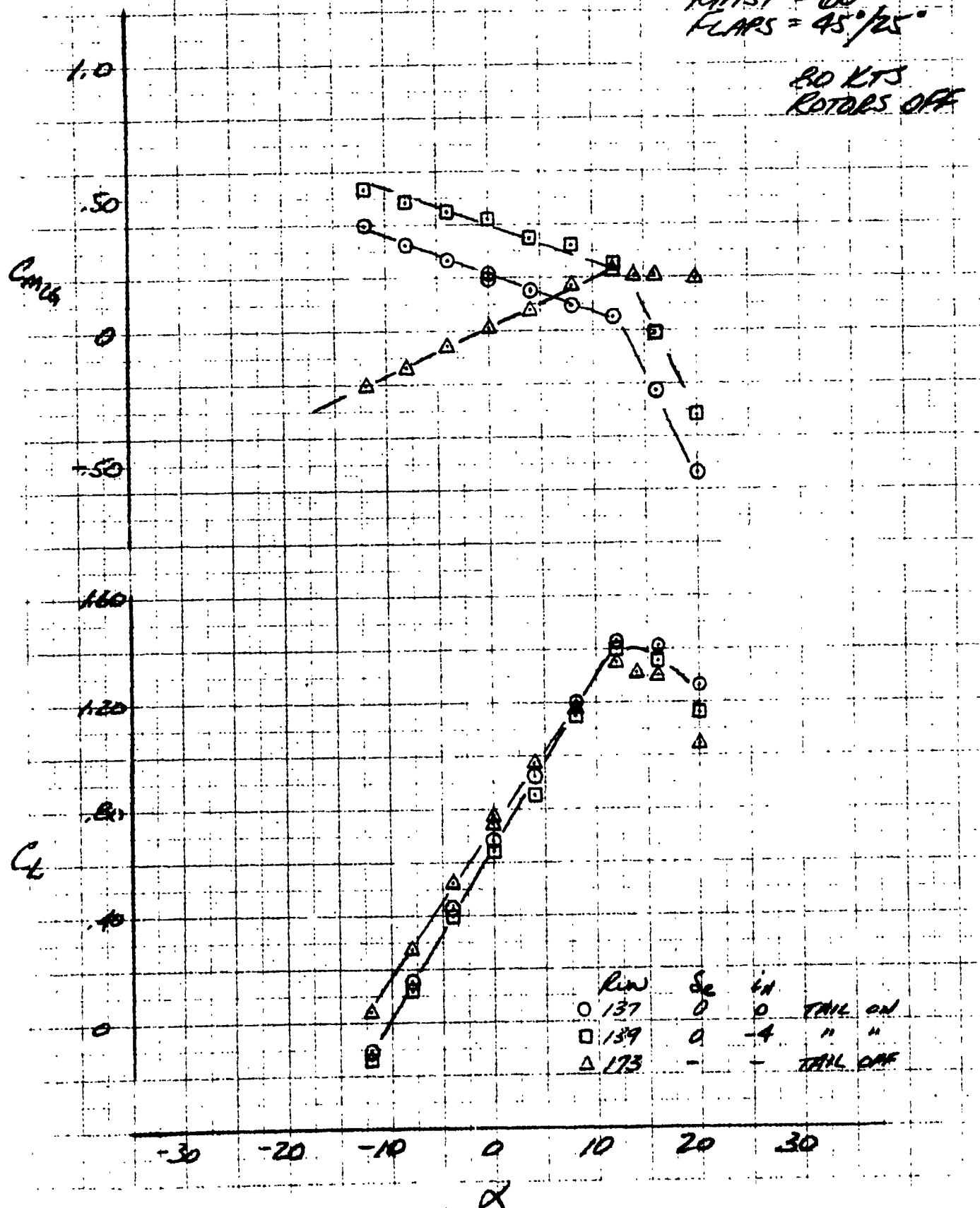
80 KTS
ROTORS OFF

Run	δe	in	
○ 137	0	0	TAIL ON
□ 139	0	-4	" "
△ 173	—	—	TAIL OFF
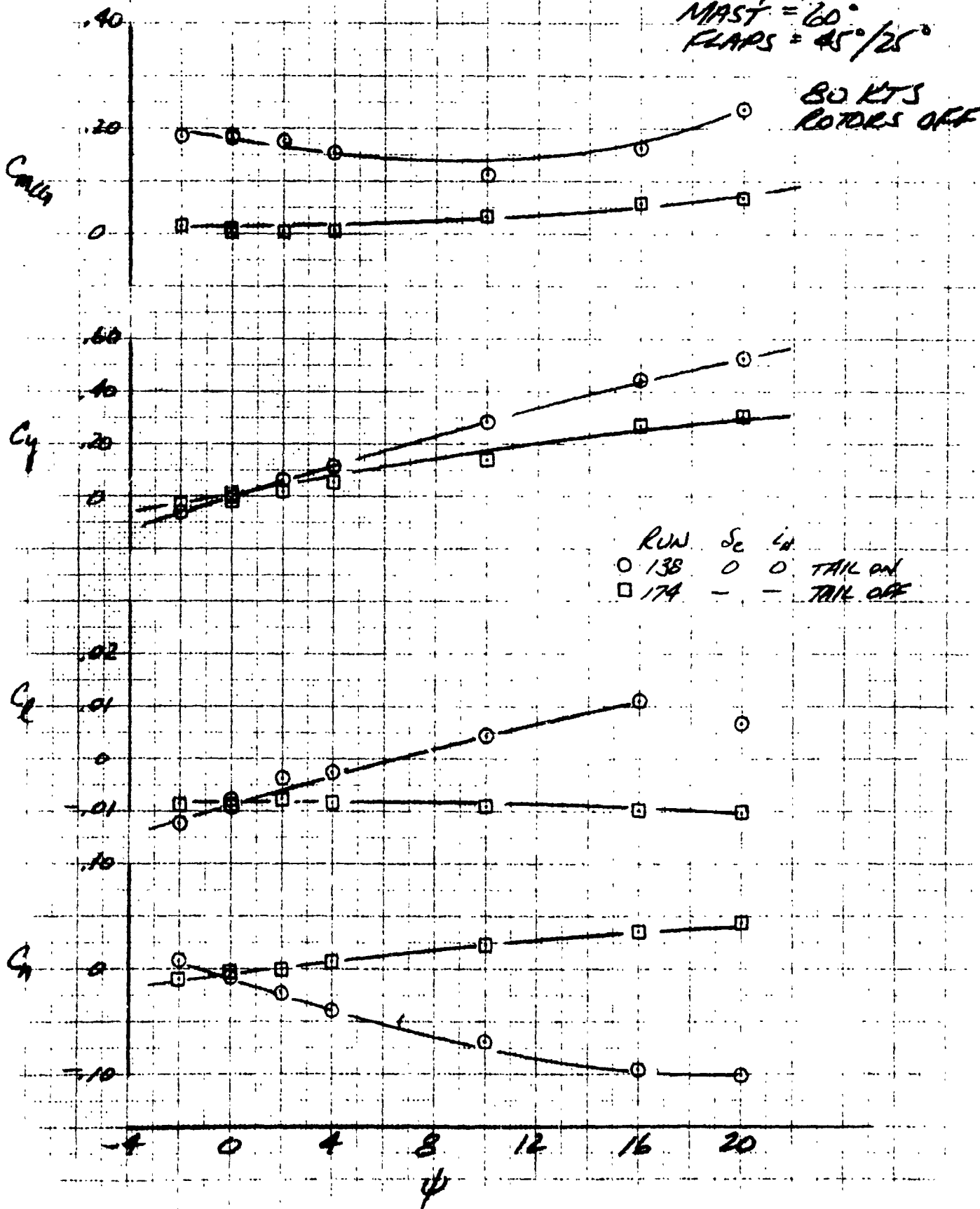
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BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 60°
FLAPS = 45°/25°

80 KTS
ROTORS OFF

RUN	δ_e	i_H	
○ 138	0	0	TAIL ON
□ 174	—	—	TAIL OFF

ψ

BELL HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 30°
FLAPS = 45°/25°

80 KTS
Ω = 1260 RPM

C_{MCG}
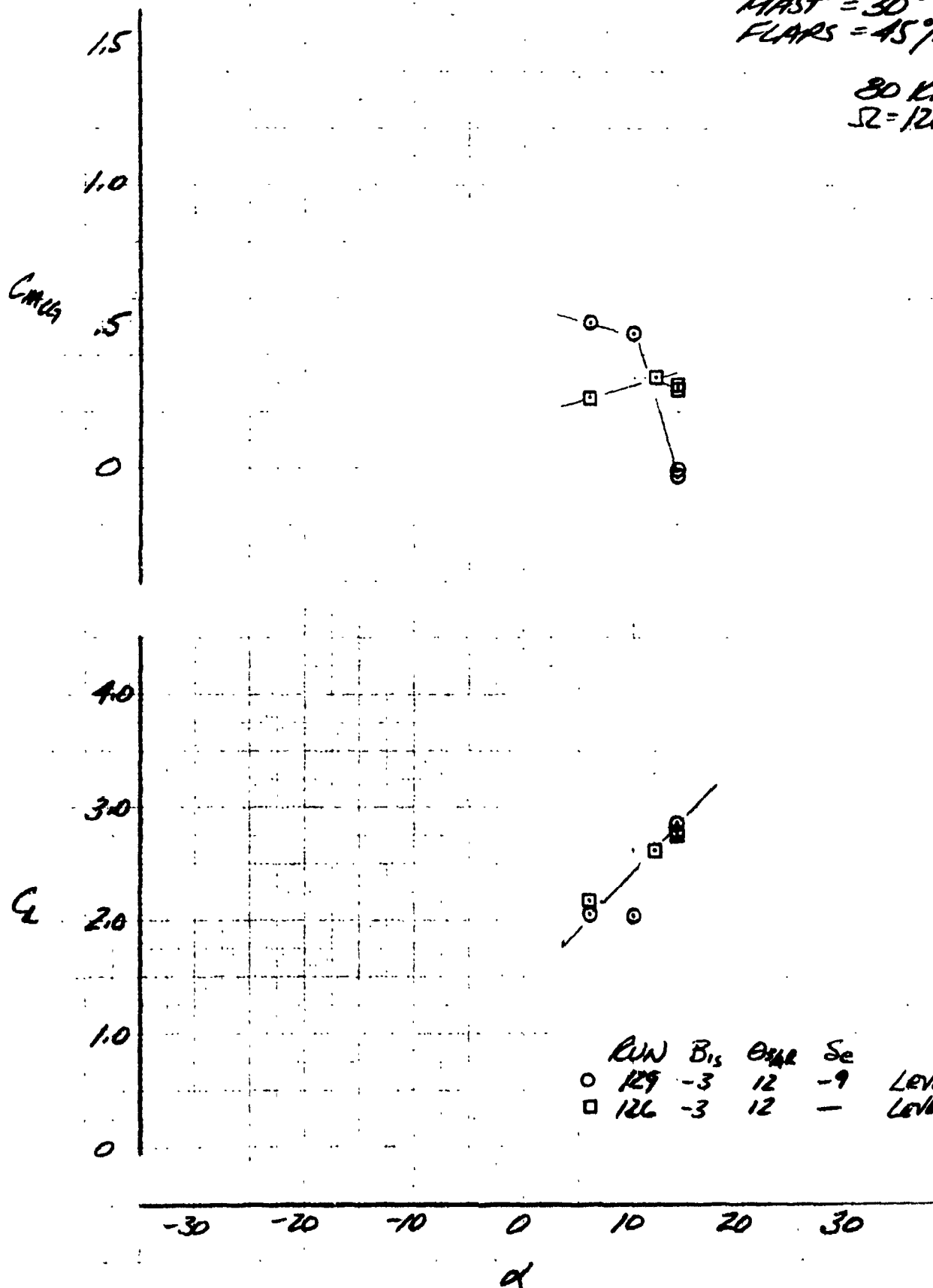
RUN	B_{1S}	θ_{TAIL}	S_e	
○ 129	-3	12	-9	LEVEL
□ 126	-3	12	—	LEVEL, TAIL OFF

C_L

α

LSWT 421

STING MOUNT
MAST = 30°
FLAPS = 45°/25°

100 KTS
Ω = 1260 RPM

$C_{m_{cg}}$

C_L

RUN	B_{1S}	$\theta_{3/4R}$	δ_e	
O 130	-1.0	12.5	-4	LEVEL
□ 127	-1.0	12.5	—	LEVEL, TAIL OFF

α

301-099-002

A-80

BELL
HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 30°
FLAPS = 45°/25°

100 KTS
Ω = 1260 RPM

C_{mcg}

C_y

C_l

C_n

RUN	α	B_{1s}	$\theta_{3/4R}$	δ_c	
151	10°	-1.0	12.5	-4	LEVEL
211	10°	-1.0	12.5	—	LEVEL, TAIL OFF

ψ

301-099-002

A-81

BELL HELICOPTER COMPANY

LSWT 421

STING MOUNT
MAST = 30°
FLAPS = 45°/25°

80 KTS
ROTORS OFF

RUN	δe	iH	
O M2	O	O	TAIL ON
□ 176	—	—	TAIL OFF

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APPENDIX C

Tuft Grid Photographs
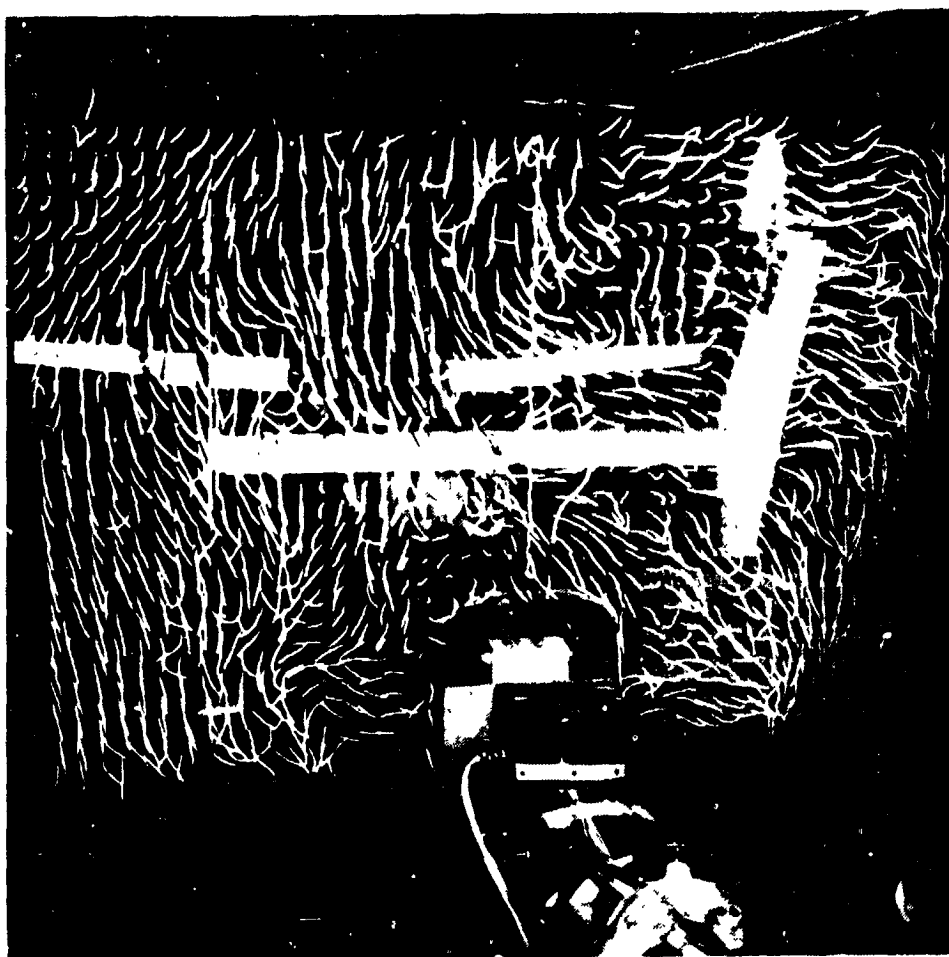
16 KTS, $\Omega = 1260$ RPM, $\alpha = 0$, $\psi = 0$, $\ell = 30$ in lb
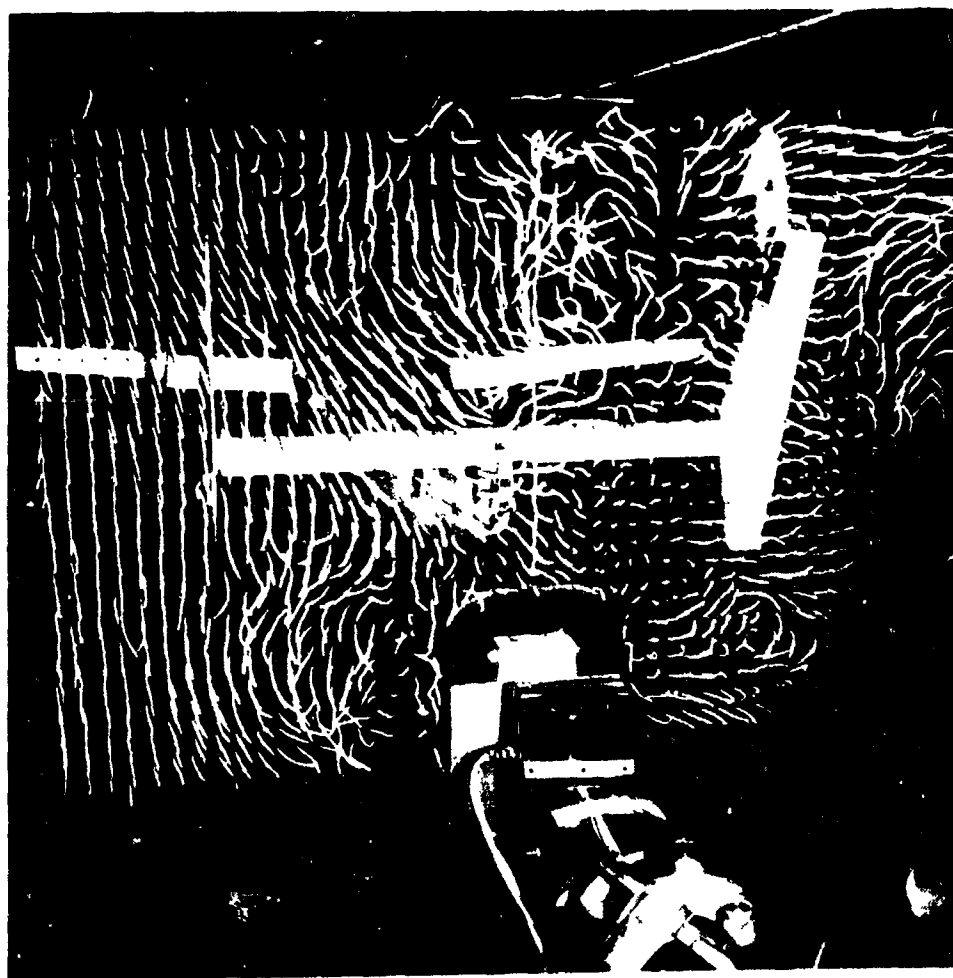
301-099-002

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HELICOPTER COMPANY

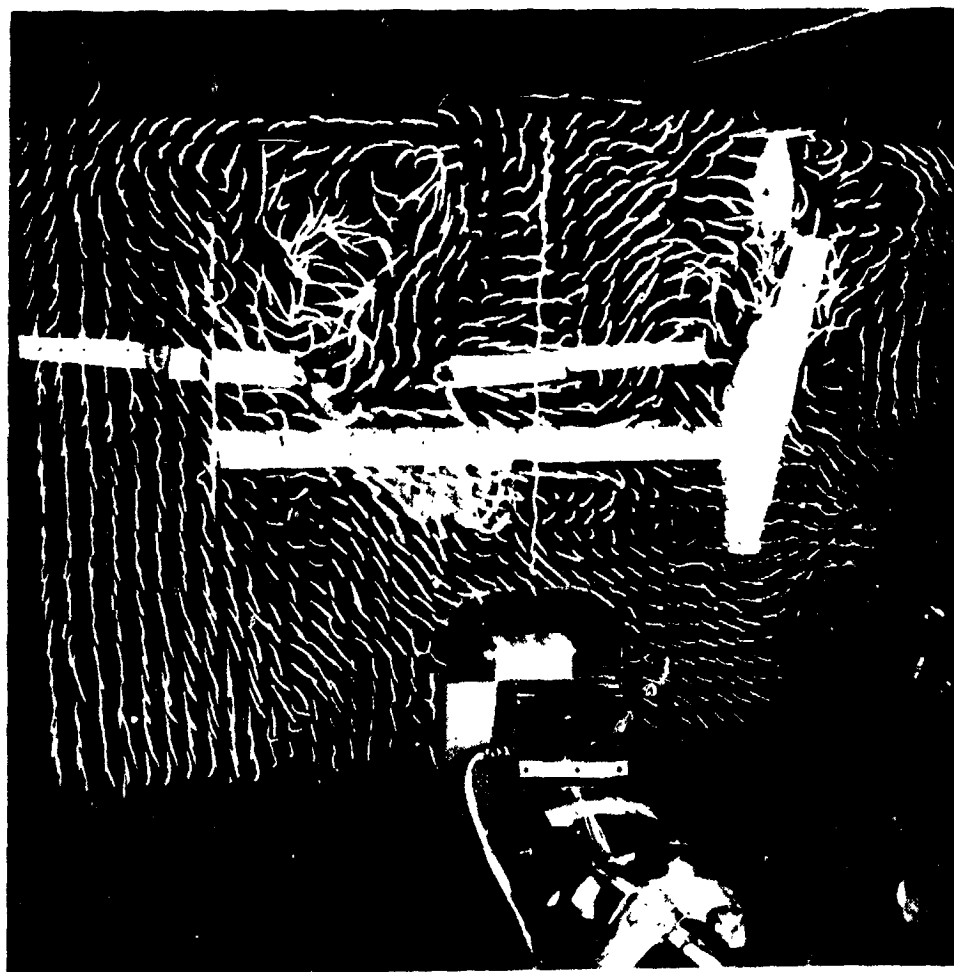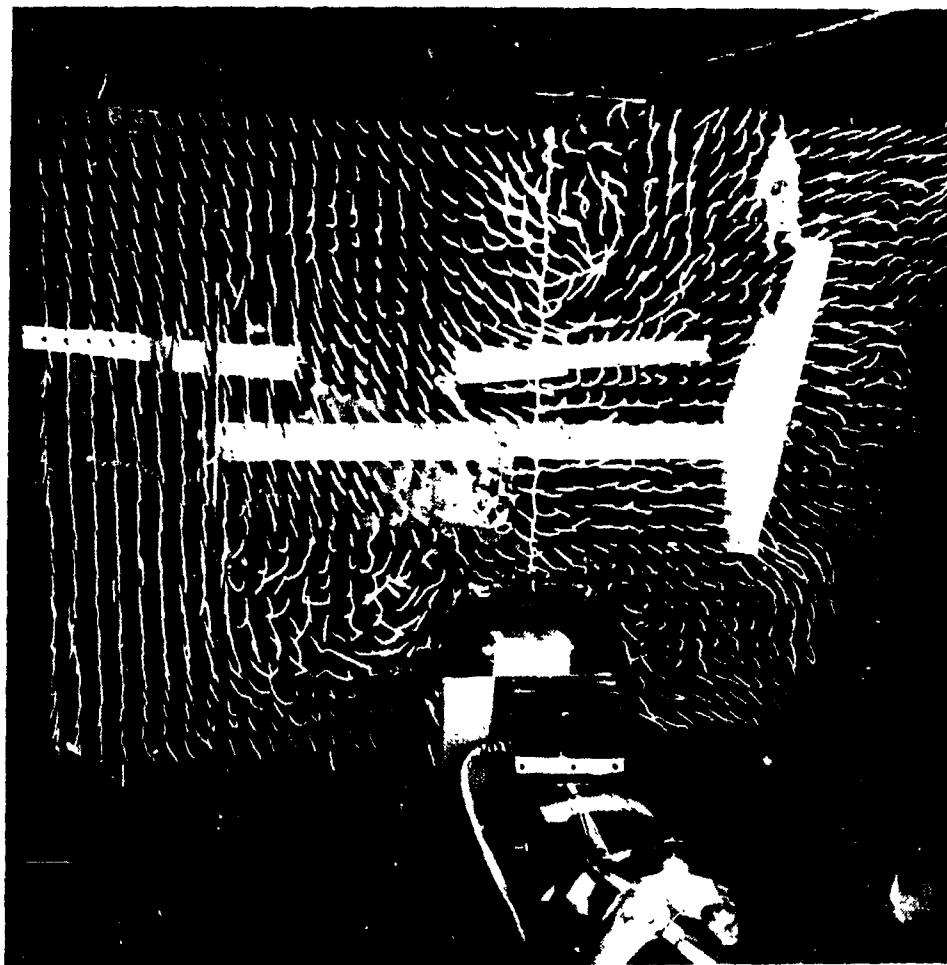16 KTS, $\Omega = 1260$ RPM, $\alpha = 0$, $\psi = 10$

301-099-002
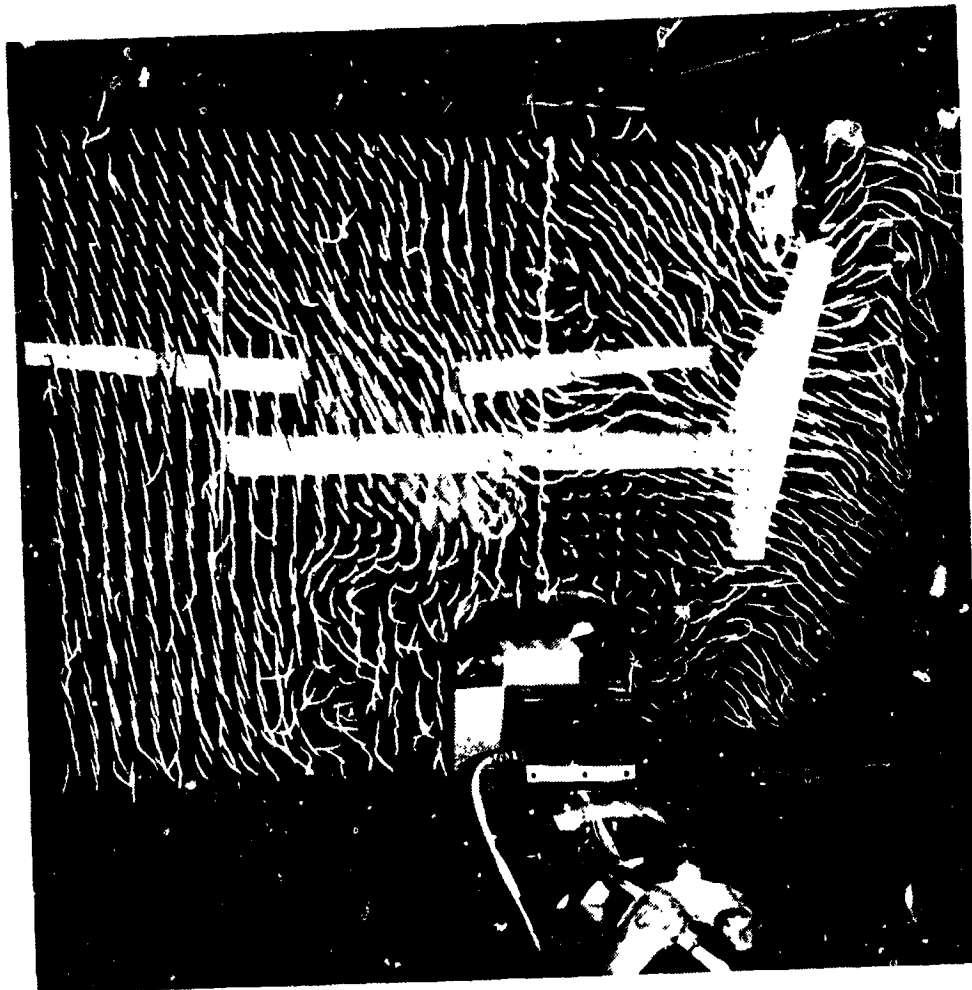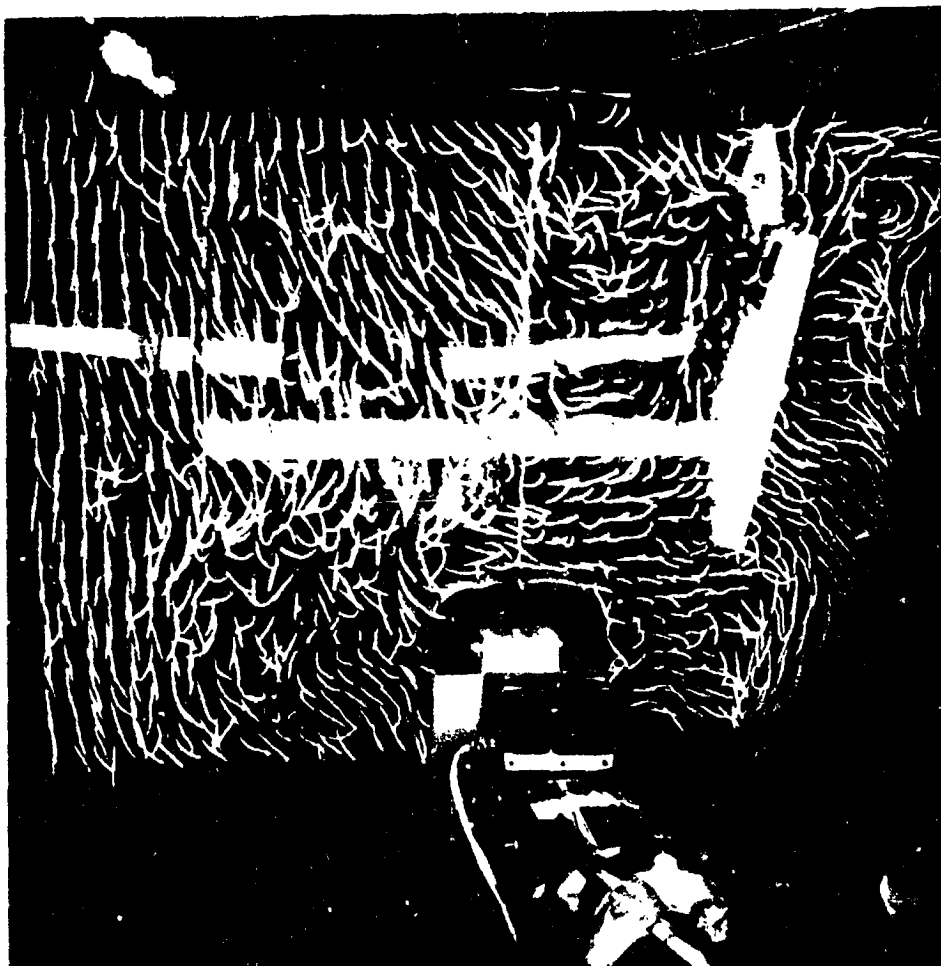
20 KTS, $\Omega = 1260$ RPM, $\alpha = 0$, $\psi = 0$, $Q = 120$ in lb

A-87
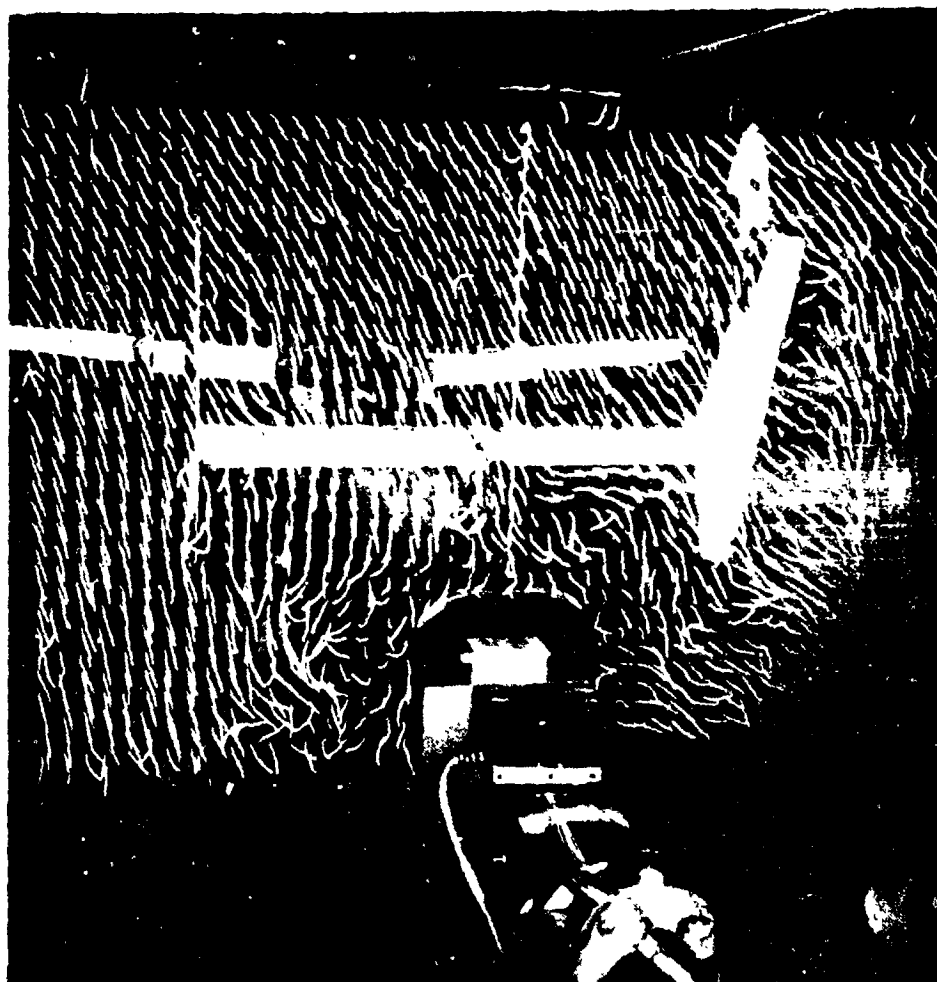
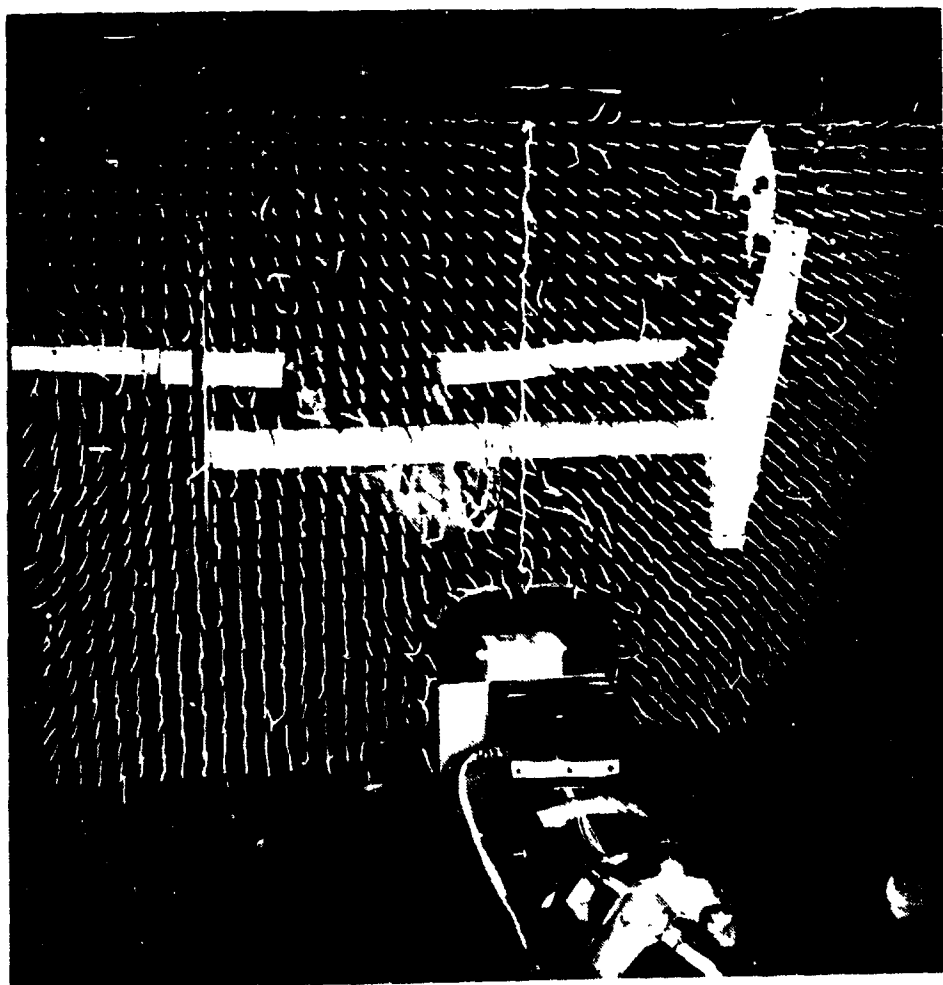20 KTS, $\Omega = 1260$ RPM, $\alpha = 0$, $\psi = 10$
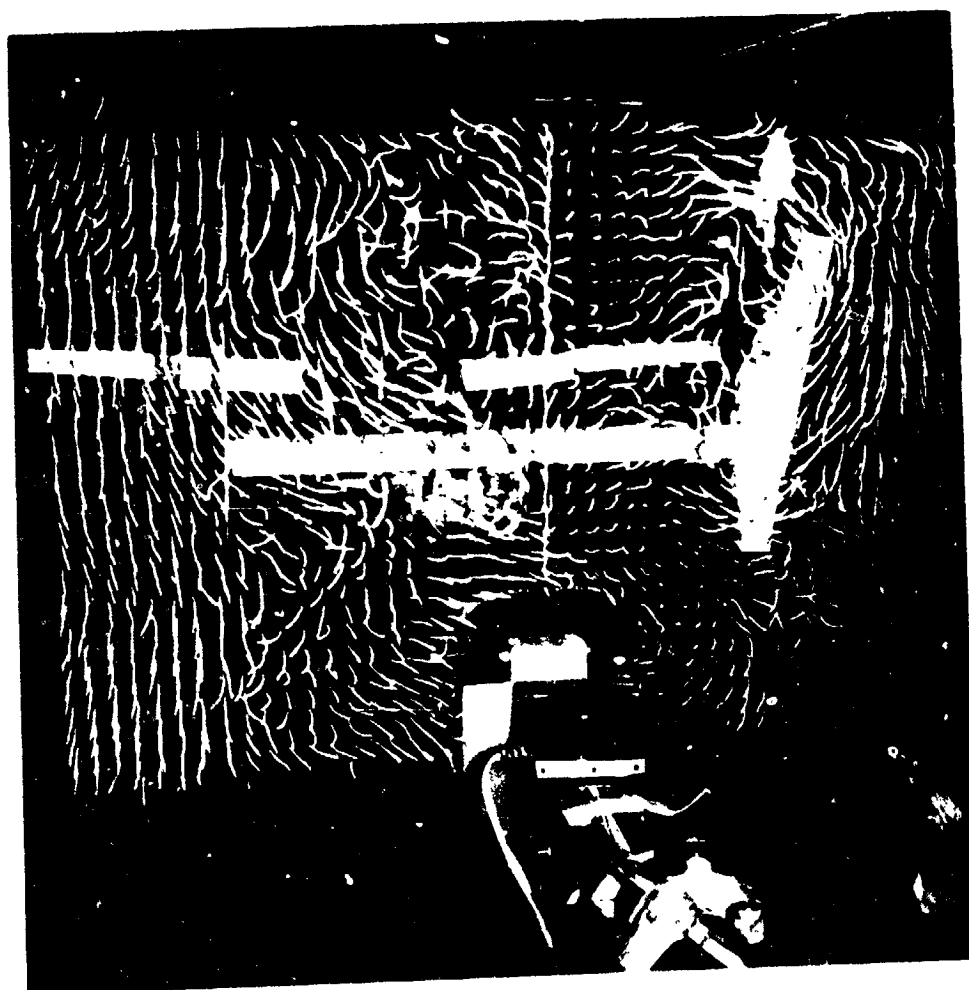
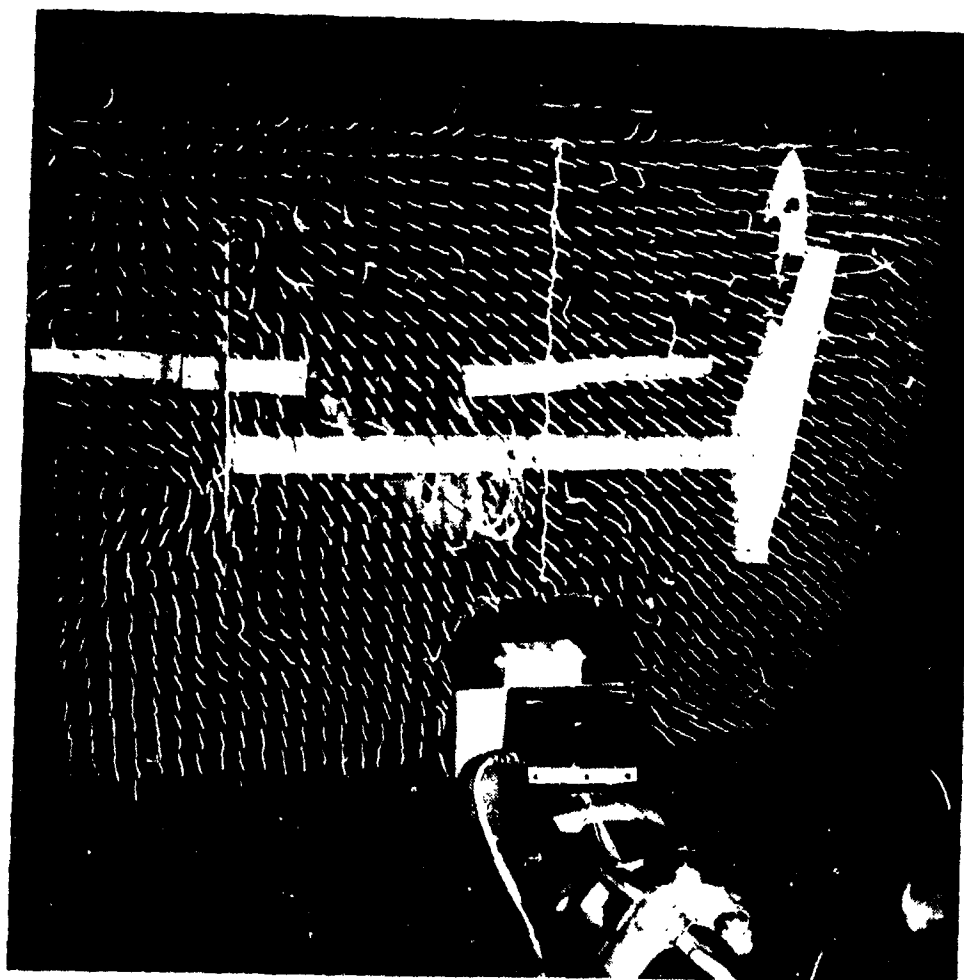

30 KTS, Ω = 1260 RPM, α = 0, ψ = 0, Q = 103 in lb

30 KTS, $\Omega = 1260$ RPM, $\alpha = 0$, $\psi = 10°$

40 KTS, Ω = 1260 RPM, α = -2, ψ = 0, Q = 90 in lb

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40 KTS, $\Omega = 1260$ RPM, $\alpha = -2$, $\psi = 10$

40 KTS, $\Omega = 0$ RPM, $\alpha = -2$, $\psi = 0$

40 KTS, $\Omega = 0$ RPM, $\alpha = -2$, $\psi = 10$

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