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**U. S. A R M Y**  
**TRANSPORTATION RESEARCH COMMAND**  
**FORT EUSTIS, VIRGINIA**

477910

⑥ PRELIMINARY FLIGHT TEST DATA.  
XH-51A RIGID ROTOR HIGH SPEED FLIGHT PROGRAM.

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⑨ INTERIM REPORT, NO. 7

⑪ NOV ~~1964~~ 1964,

⑫ 36p.

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## SUMMARY

This report summarizes the flight test results of the Phase III four-bladed rotor testing on the XH-51A "rigid rotor" helicopter. The object of Phase III was to evaluate the four-blade rotor in comparison with the three-blade rotor. Ground tests commenced on 13 July 1964 with the first flight on 21 July 1964. The phase was terminated on 3 October 1964 after 49 flights and 14.9 flight hours.

## Results and Discussion

### General

Two problems were apparent as a result of early flights, blade stresses and vibration level - both were high. At this stage, the transmission was in soft suspension in pitch only with a spring rate of 14,400 pounds/inch; vertically and laterally the mount was solid. The fuselage vibration problem was dominantly 4P; it was considered that this resulted from the second and third harmonics of blade natural flapping frequency which were close to 3P and 5P. The 3P and 5P rotating inputs resulted in a 4P shaft oscillation which was transmitted to the fuselage. The fuselage responds rather readily to frequencies in this range. The transmission suspension was varied incrementally to tune it to a point of minimum transmissibility; this approach had not been completed at the close of Phase III. Additionally, experimental moves were made towards cutting down the input by reducing the blade natural flapping frequencies. The configuration at the termination of the phase, resulting from these two approaches, incorporated a 13-pound weight at station 6.0 on each of the four blades and the transmission soft mounted in pitch, roll, and vertically. Blade stresses were reduced to within the infinite life level and vibration, although unacceptable from a production viewpoint, was also considerably reduced.

Apart from the vibration level, no flying qualities problems were encountered. A marked increase in the Phase III flight envelope was achieved, 2.5 g being demonstrated from 0 to 120 knots CAS and a speed of 180 knots IAS flown in a shallow dive.

Phase III employed four blades of the same design used on the three-blade rotor; the hub blade attachment geometry was modified compared with that employed in the Phase II flying. The only significant problem apparent at the close of Phase III was fuselage vibration, none of the stability problems encountered with the three-blade rotor existed and stress levels, subject to fatigue test confirmation, indicate infinite life for the hub. With the reduction of the vibration level to within normal limits, a substantial speed envelope expansion should be possible. The phase is considered a considerable success in that a marked increase in

flight envelope was demonstrated, no stability problems existed, and no problems considered insurmountable were encountered. The stability characteristics confirmed the advisability of the choice of the four-blade rotor for the compound configuration.

### Configuration

Four-blade rotor - The blade design was that employed during Phase II, the three-blade rotor; however, the blade hub attachment was modified. The cone angle built into the hub was increased from 2.8 degrees to 3.2 degrees.

Main rotor gearbox suspension varied as described.

Tailplane angle of incidence  $-5\frac{1}{2}^{\circ}$  through test 271 and  $-3^{\circ}$  thereafter.

Blade weights 13 pounds each, fitted to each main rotor blade at station 6.0 for test 352 and remained on thereafter.

Four-arm gyro 7.3 slug ft<sup>2</sup>.

Arm incidence 30 degrees.

### Structures

Structural loads recorded during the program included main rotor hub and blades, gyro control arms, main rotor pitch link, tail rotor and tailplane loads. An incremental approach was employed during the tests, flight records being examined prior to further envelope expansion.

Figures 1 and 2 compare Phase II and III flight envelopes. At a C.G. of 1.5" forward, the envelope is substantially 2.5 to 0.2 g up to 120 knots CAS with reducing normal acceleration cut to 165 knots CAS. The aft envelope was opened up in only three or four flights and no specific attempt was made to exceed the Phase II envelope. While it is felt that there would be little point in extending the hover beyond the 0.15 to 2.7 g demonstrated, no structural, performance, or stability limits were encountered in forward flight.

### Rotor Stresses

A review of all structural data indicates that hub station 7.0 is the critical fatigue section of the rotor. It consists of three steel laminations bonded and bolted together. Assuming a stress concentration of 3, the endurance limit stress is 26,000 psi. The strain calibrations were affected in terms of bending moment rather than stress because the bending moment curve along the span of hub and blade is predictable. The conversion of bending moment to stress at station 7.0 is as follows:

Flapwise bending      Station 6.0 moment x 1.42 =  
station 7.0 stress

Chordwise bending      Station 6.0 moment x 1.152 =  
station 7.0 stress

Figure 3 shows that during the initial flights, the chordwise bending stress was 40 percent lower than that in the three-blade, but the flapping stress was up by 80 to 90 percent. At this stage, the vibration was very high and a series of changes in transmission suspension springs was initiated to reduce both vibration and stress. The pitch spring rate was varied first; the range covered was 6,400 pounds/inch to solid while both vertical and lateral remained solid. From the structural loads and vibration results 11,000 pounds/inch was selected as the pitch spring rate to be held constant during vertical spring variations. The vertical range covered was from solid down to 4,000 pounds/inch and 6,000 pounds/inch was selected from the results.

An analysis of the flapwise bending at station 6.0 during forward flight and flare showed considerable 3P and 5P content superimposed on the 1P. Note, 1P is main rotor rotational frequency, normally 5.9P cps (100T), 3P is 3 times rotor frequency, etc. The second and third harmonic blade natural flap bending frequencies were slightly below the 3P and 5P forcing frequencies. Tests conducted at 95 percent, 100 percent, and 105 percent illustrated the diminishing response of the blade as the forcing frequency was increased and separated from the natural. The 3P and 5P bending moments were reduced by approximately 45 and 55 percent respectively. The reduction of the blade natural frequencies had a similar effect; at 100 knots, the 3P bending moment was reduced from 4,090 inch pounds to 2,400 inch pounds and the 5P from 1,960 inch pounds to 740 inch pounds. In reducing these moments, the weights reduced the 3P and 5P driving forces which produced the 4P pitching and rolling moment into the fuselage. The vibration benefited to a minimum degree.

To reduce the 4P moment further, a transmission lateral spring was introduced with a rate of 19,000 pounds/inch; it did reduce the cabin vibration to a small degree, but did not affect the structural loads.

Records showed a 1P flapping component increasing with speed at high speed and producing a nose down moment and indicating that a reduction of tailplane negative incidence would be of value in reducing the total bending moment. The tailplane incidence was changed from  $-5\frac{1}{2}^{\circ}$  to  $-3^{\circ}$  for test 228. The change in slope of the flapwise bending moment curve between figures 4 and 5 at high speed is attributable to this change.

Figures 6 and 7 illustrate the affect of normal acceleration on the flapwise bending moment at station 6.0 at two centers of gravity. The average moment increased towards up flapping with increasing load factor due to lift on the rotor blade. Down flapping was recorded at 1.0 g due to the fact that the blade line was below the cone angle built into the hub. Zero moment was recorded when the blade at this station lined up with the hub cone angle. The smaller built-in cone angle on the three-blade rotor and the 30 percent greater lift per blade resulted in the different intercept shown in the graphs. The Phase III flapping cyclic stresses, figure 7, at the aft C.G. were of the order of 10 percent lower than those at a mid-C.G. on Phase II. At the forward C.G., figure 6, they were about 30 percent lower than the Phase II mid-C.G. Chordwise average and cyclic moments in maneuver were 50 percent lower than with the three-blade rotor except at high load factors where the reduction in average moment was about 10 percent. The cyclic flapwise and chordwise stresses at station 6.0 are the maximum that occurred during the maneuvers and do not necessarily coincide with the maximum load factor.

The spanwise bending moment distribution in high speed level flight, descent, and in maneuver are shown in figures 10 through 15.

#### Vibration

Vibration level in the cabin was measured for speeds up to 132 knots CAS in level flight and to 165 knots CAS in a descent. Tri-axis (vertical, fore and aft, lateral) measurements were made on the cabin floor at the pilot's seat. Vibration data in the 3 axes is plotted versus airspeed (CAS), figure 16.

The analysis of vibration for the various configuration was carried out in conjunction with the structural loads, because they were in most cases a function of each other. The comparison with the three-blade data is obvious, but it must be pointed out that the three-blade data represents only the soft cabin configuration. The high vibration level remaining at the termination of Phase III was the only problem of any significance and was the factor which limited the speeds attainable under this contract.

#### Stability

No adverse stability characteristics were recorded at any time throughout the Phase III flying. There was a tendency for the static

longitudinal stability to become neutral at high speed and some cyclic cross-coupling existed. The correction of the cross-coupling would have improved the stability which could have been made further positive by aerodynamic shaping of the gyro arms. For the purposes of the program, neither the stability nor the coupling warranted corrective action.

The cyclic control to trim is shown in figures 17 and 19 in terms of control position and maximum blade incidence. Figure 18 shows the results of constant collective static stick fixed and free longitudinal stability measurements.

Figures 20 and 21 illustrate the longitudinal and lateral control power and 22 compares Phase II and Phase III results. The lateral control power was increased by 14 percent to 12.0 degrees/second-inch and in both phases and was independent of speed. Longitudinally, the 20 to 25 percent increase apparent at the lower airspeeds decreased with speed to become zero at about 110 knots IAS. The data was obtained from longitudinal step inputs in the hover and at 70 and 110 knots IAS. The time histories showed compliance with MIL-H-8501A in that the angular velocity was in the proper direction within 0.2 seconds of the control displacement and the point of inflection of normal acceleration occurred within 1.0 second of the control displacement.

The stick force per g plots in figures 23, 24, and 25 illustrate the greatest single improvement of the four-blade rotor relative to the three-blade rotor. No pitch up, nor any tendency for the stick force/g to be other than positive was experienced during Phase III. The stick force/g has improved 150 to 200 percent, providing a high degree of confidence with regard to the anticipated outcome of maneuver tests at high airspeeds. An envelope of the bank angles flown during these maneuvers is presented in figure 26.

A number of entries and autorotations were affected in the range 80 to 120 knots CAS; the characteristics were normal and perfectly acceptable. The load factor to hold rotor speed presented in figure 27 stems from tests conducted to assist in the definition of the technique to be used following failure of the rotor engine on the compound helicopter.

#### Performance

Performance data presented in figure 29 is not representative of the four-bladed clean helicopter configuration in that at the time the tests were conducted, the blade weights were installed, and they represent a significant and unknown drag increment. The hover data presented in figure 28 is clean four-blade data.

DATA APPENDIX  
INDEX

<u>Figure</u>	<u>Title</u>
1	Flight Envelope - C.G. 1.46" Forward . . . . .
2	Flight Envelope - C.G. 2.15" Aft . . . . .
3	Moment at Hub Station 6.0 - Initial Phase III Data . . . . .
4	Moment at Hub Station 6.0 - Transmission Sus- pension Modified Blade Weights Incorporated . .
5	Moment at Hub Station 6.0 - Final Phase III Configuration - Tailplane -3° . . . . .
6	Flap Bending Moment at Station 6.0 vs. Load Fac- tor C.G. 1.46" Forward - Final Phase III Con- figuration . . . . .
7	Flap Bending Moment at Station 6.0 vs. Load Factor - C.G. 2.15" Aft - Final Phase III Con- figuration . . . . .
8	Chord Bending Moment at Hub Station 6.0 vs. Load Factor - C.G. 1.46" Forward - Final Phase III Configuration . . . . .
9	Chord Bending Moment at Hub Station 6.0 vs. Load Factor - C.G. 2.15" Aft - Final Phase III Configuration . . . . .
10	Flap Bending Moment at hub Station 6.0 vs. Span Level Flight - Final Phase III Configuration .

<u>Figure</u>	<u>Title</u>
11	Chord Bending Moment at Station 6.0 vs. Span Level Flight - Final Phase III Configuration
12	Flap Bending Moment at Hub Station 6.0 vs. Span in Dive - Final Phase III Configuration
13	Chord Bending Moment at station 6.0 vs. Span in Dive - Final Phase III Configuration .
14	Flap Bending Moment at Hub Station 6.0 vs. Span in Maneuver - Final Phase III Configuration . . . . .
15	Chord Bending Moment at Station 6.0 vs. Span in Maneuver - Final Phase III Configuration.
16	Cabin Vibration Level - Final Phase III Configuration . . . . .
17	Control to Trim in Level Flight . . . . .
18	Constant Collective Static Longitudinal Stability . . . . .
19	Blade Incidence to Trim in Level Flight . .
20	Longitudinal Control Power . . . . .
21	Lateral Control Power . . . . .
22	Control Power Comparison - 3-blade and 4-blade Rotors . . . . .
23	Maneuvering Stability - 4-Blade System . . .
24	Maneuvering Stability - 3-Blade Rotor . . .
25	Maneuvering Stability - 3-Blade Rotor . . .
26	Angle of Bank Velocity Envelope . . . . .
27	Load Factor - Rotor Speed in Autorotation .

Figure

Title

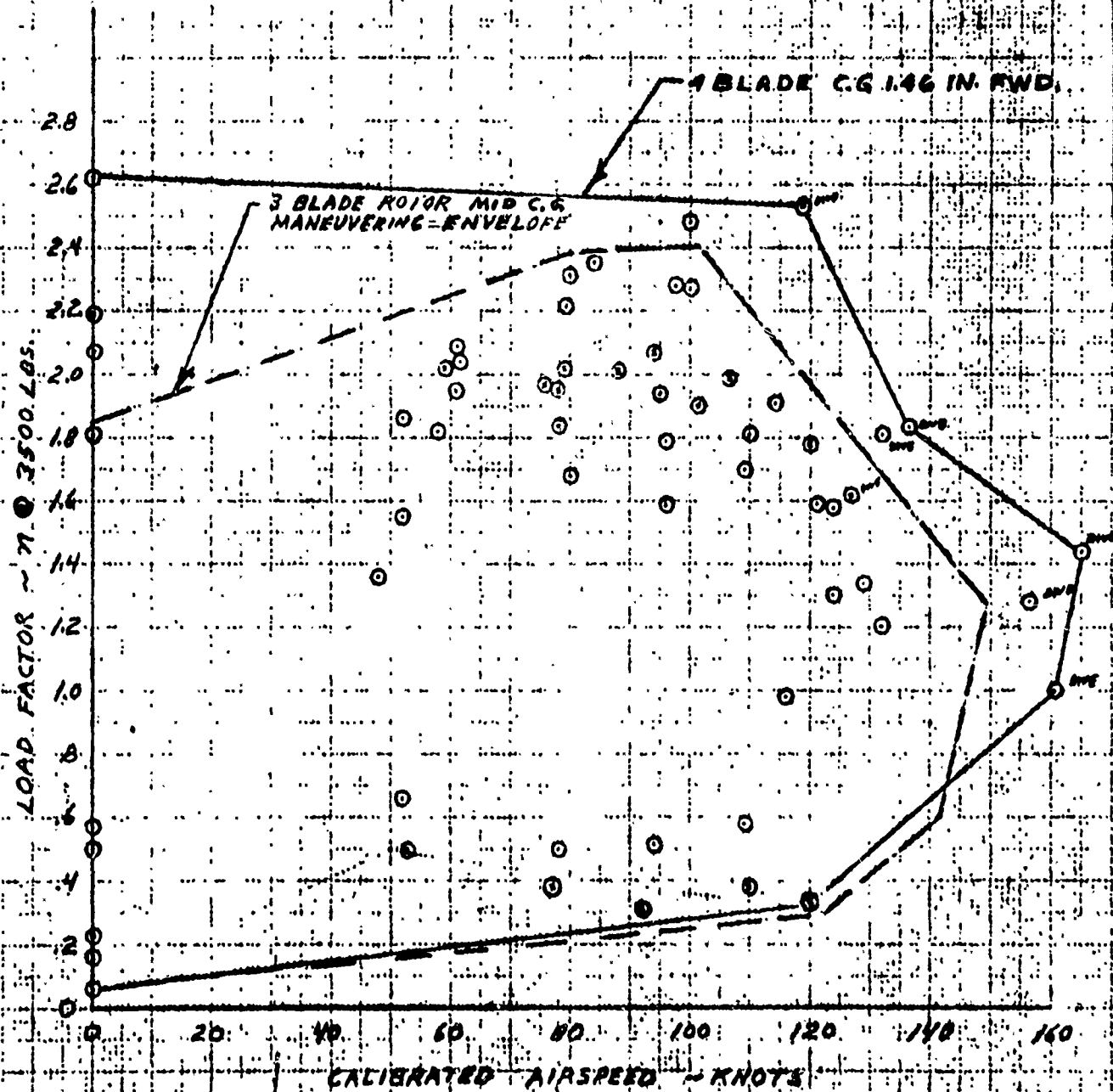
28	Hover Performance . . . . .
29	Forward Flight Performance . . . . .

# V-n DIAGRAM

GROSS WEIGHT = 3500 LBS

C.G. 1.46 IN. FWD

4 BLADE ROTOR



FORM 8879A

FLIGHT ENVELOPE C.G. 1.46" FWD.

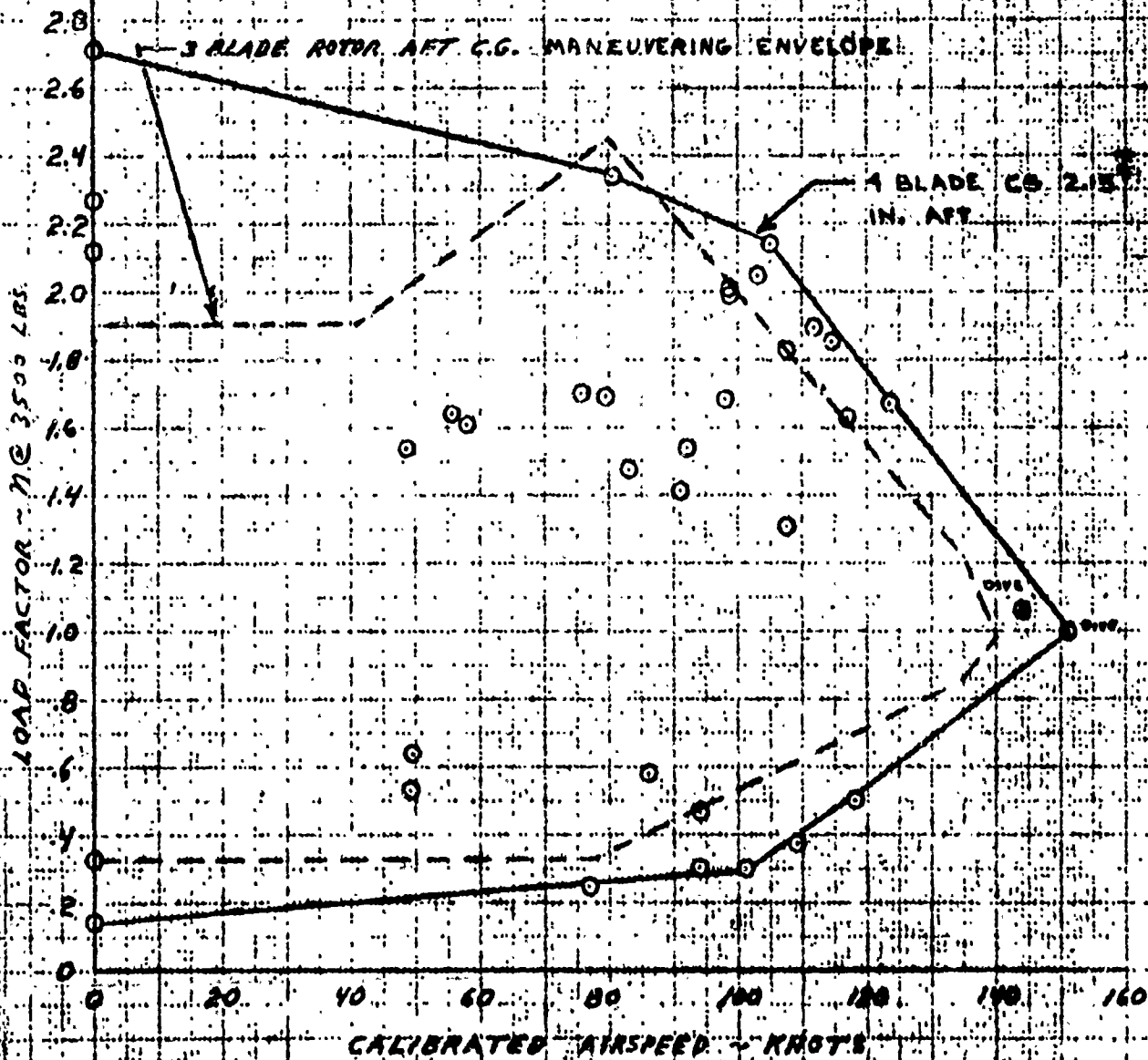
S.F.61.  
FIG. 1

### V-71 DIAGRAM

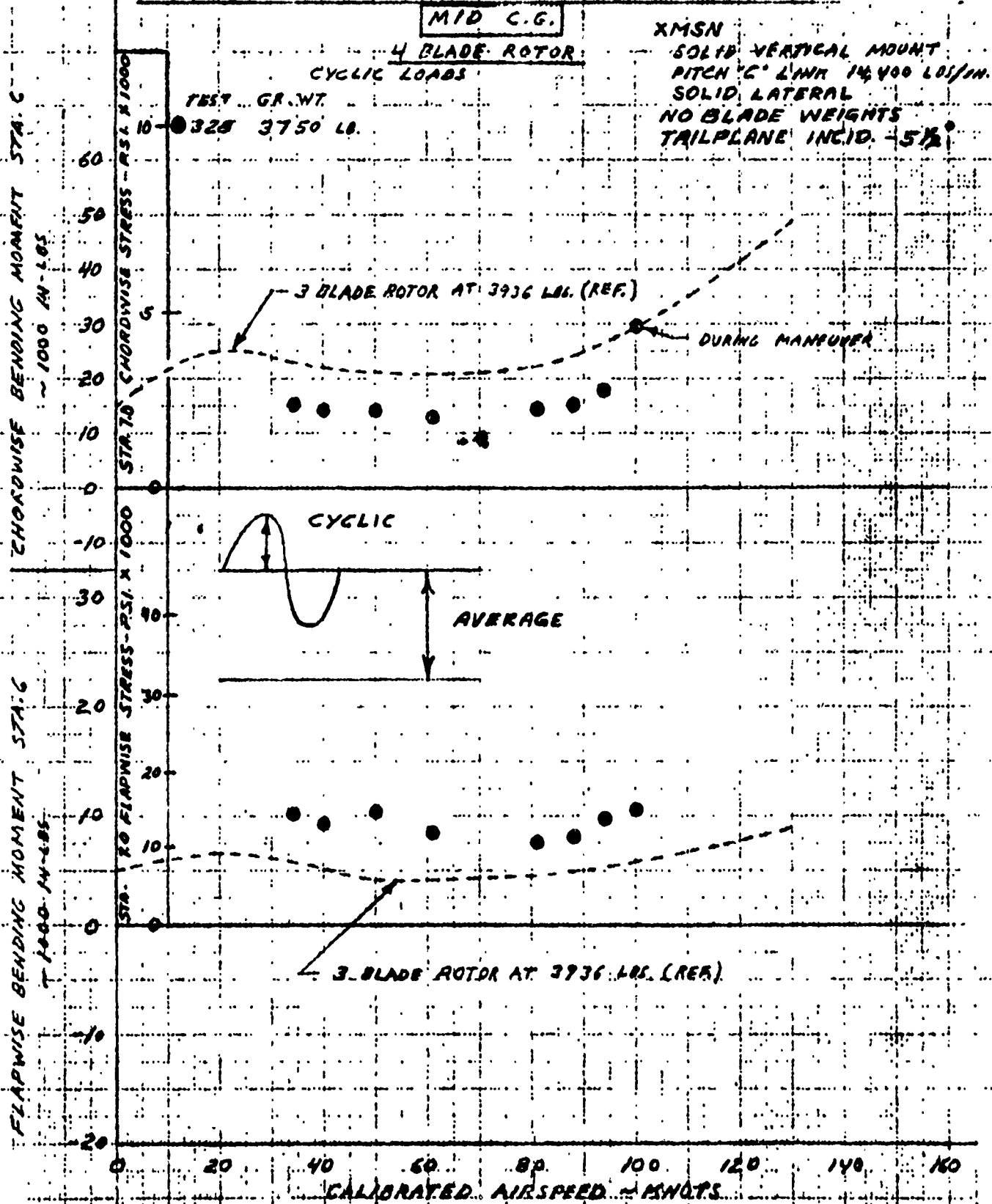
GROSS WEIGHT - 3500 LBS

C.G. 2.15" IN. AFT

4 BLADE ROTOR



MAIN ROTOR BLADE LOADS V. CALIBRATED AIRSPEED



MOMENT AT HUB STA. 6.0

INITIAL PHASE II DATA

EX 261  
FIG 3

MAIN ROTOR BLADE LOADS vs. CALIBRATED AIRSPEED

4 BLADE ROTOR

HORIZONTAL STABILIZER - 5 1/2 DEG

TEST	7.0 WEIGHT
359	4020
370	4010

XMSN

6000 VERTICAL

11000 PITCH

SOLID LATERAL TEST 359

19000 LATERAL TEST 370

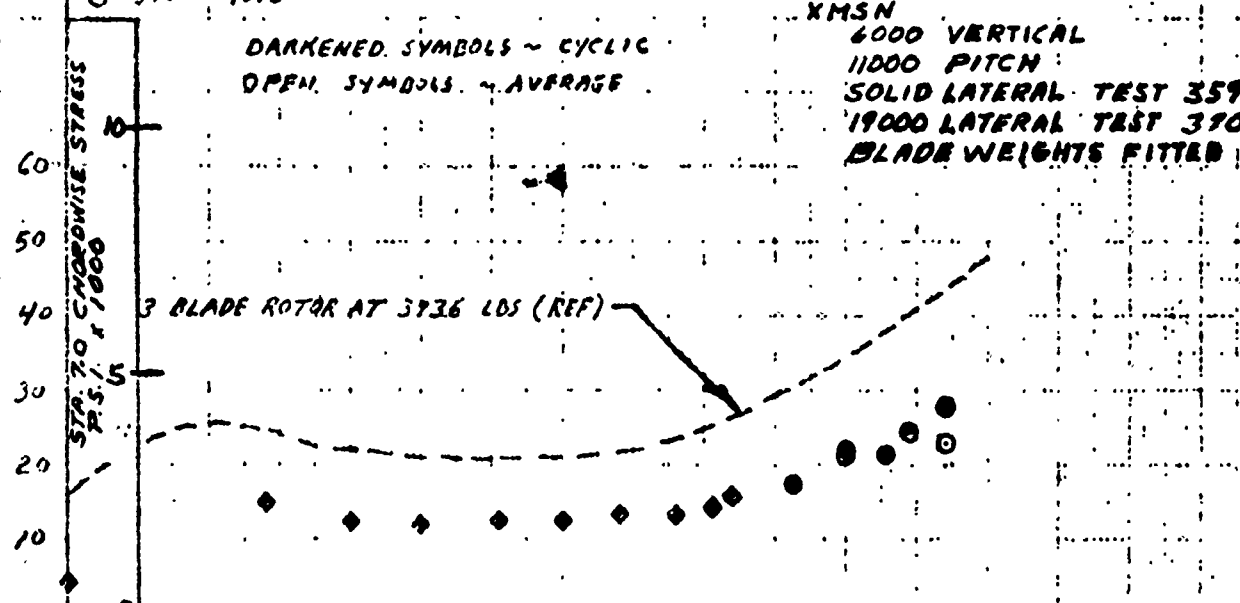
BLADE WEIGHTS FITTED

DARKENED SYMBOLS ~ CYCLIC

OPEN SYMBOLS ~ AVERAGE

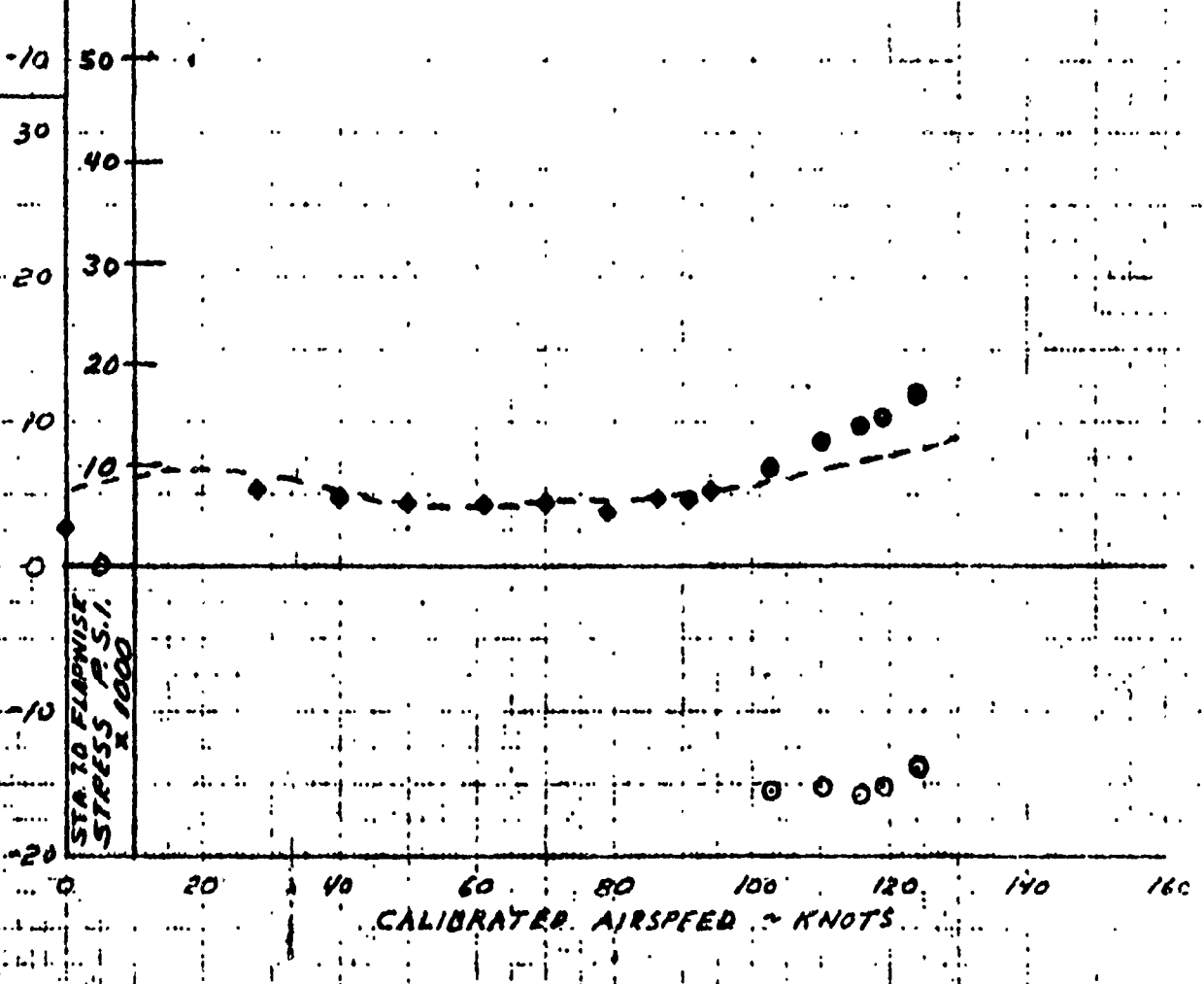
CHORDWISE BENDING MOMENT STA. 6

~ IN. LBS. x 1000



FLAPWISE BENDING MOMENT STA. 6

~ IN. LBS. x 1000

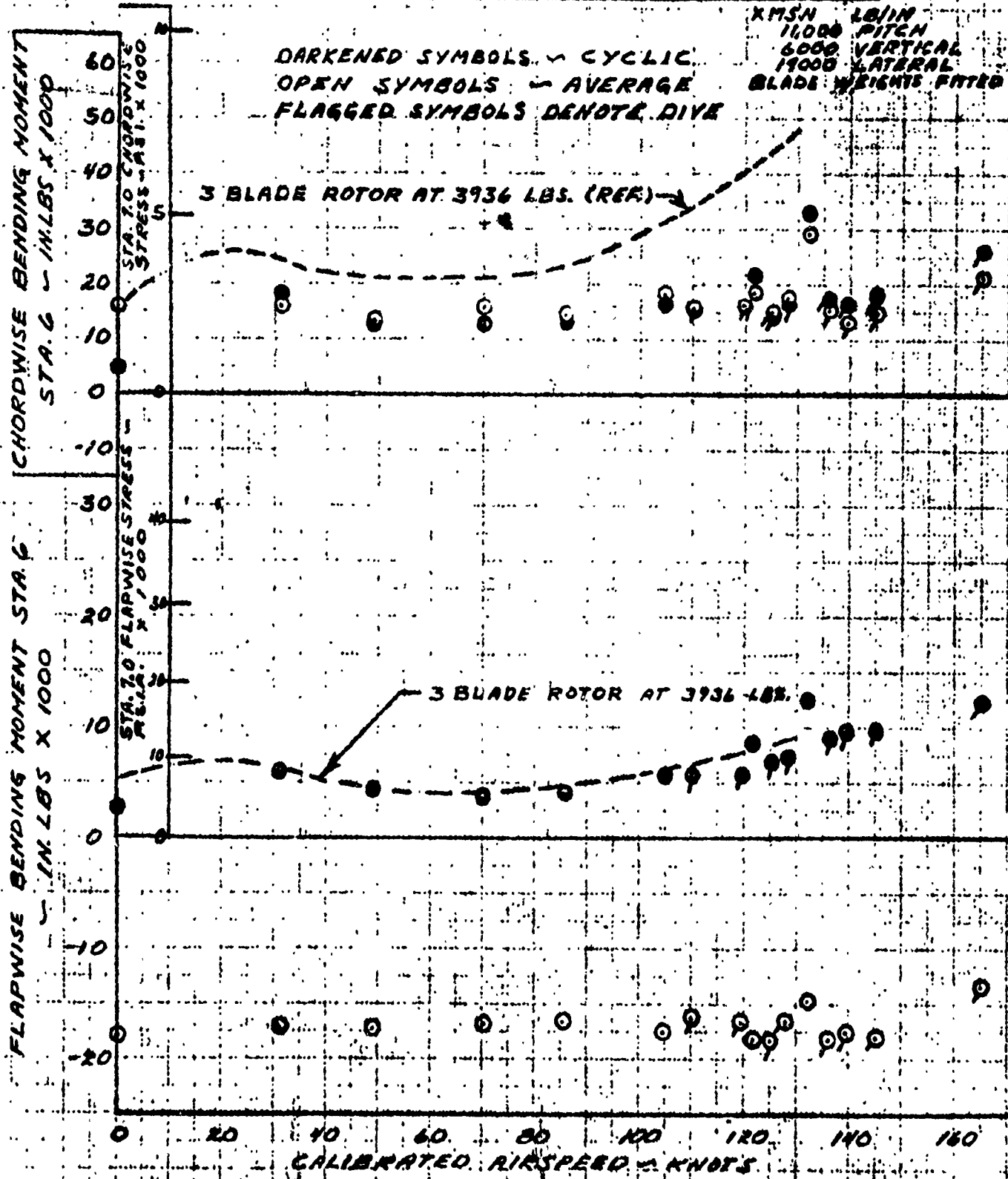


MOMENT AT HUB STA. 6.0 TRANSMISSION SUSPENSION MODIFIED BLADE WEIGHTS INCORPORATED

SF. 51.1  
FIG 4

**MAIN ROTOR BLADE LOADS VS. CALIBRATED AIRSPEED**

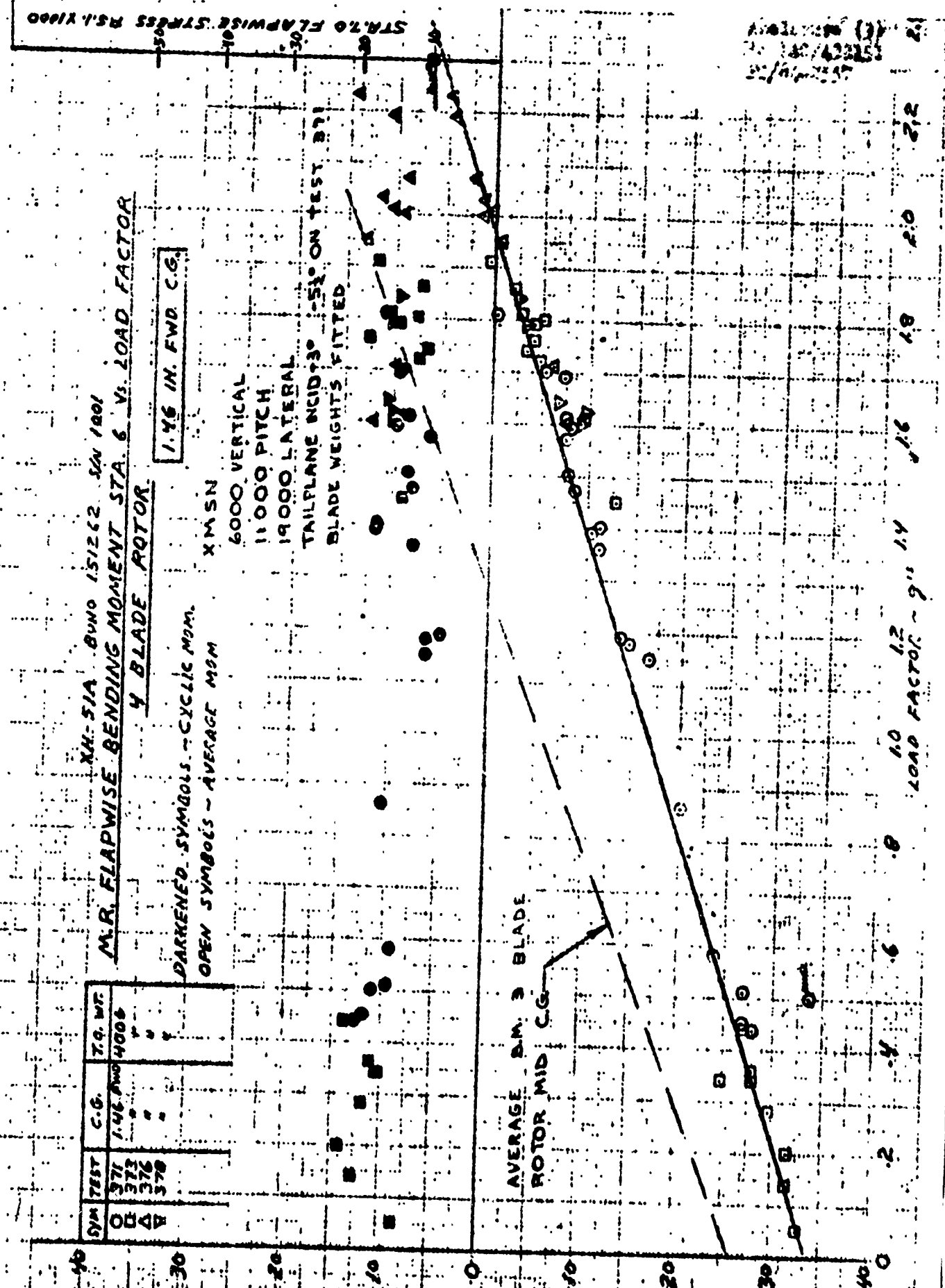
TEST 373 F 385 - 4 BLADE ROTOR  
 T.O. WT. - 4006 HORIZONTAL STABILIZER - 3 DEG.



FORM 8878A

MOMENT AT HUB STA. 6.0 FINAL PHASE II CONFIG. TAILPLANE -3°

S.F. 62.1  
 FIG 5



SYM TEST	C.G.	T.G. WT
□	1.46 FWD	4006
△	"	"
○	"	"

M.R. FLAPWISE BENDING MOMENT STA. 6.0 VS. LOAD FACTOR  
 4 BLADE ROTOR  
 1.46 IN. FWD C.G.

DARKENED SYMBOLS - CYCLIC MOM.  
 OPEN SYMBOLS - AVERAGE MOM

6000 VERTICAL  
 11000 PITCH  
 19000 LATERAL  
 TAILPLANE INCID<sup>3</sup> - 51° ON TEST B71  
 BLADE WEIGHTS FITTED

AVERAGE B.M. 3 BLADE  
 ROTOR MID C.G.

STA. 7.0 FLAPWISE STRESS PSI, 1/1000

FLAP B.M. AT STA. 6.0 VS. LOAD FACTOR C.G. 1.46" FWD. FINAL PHASE CONFIG. FIG 6

XH-51A. BUONO 151262. 3/4 1031  
 M. R. FLAPWISE BENDING MOMENT STA. 6.6 VS. LOAD FACTOR  
 4 BLADE ROTOR

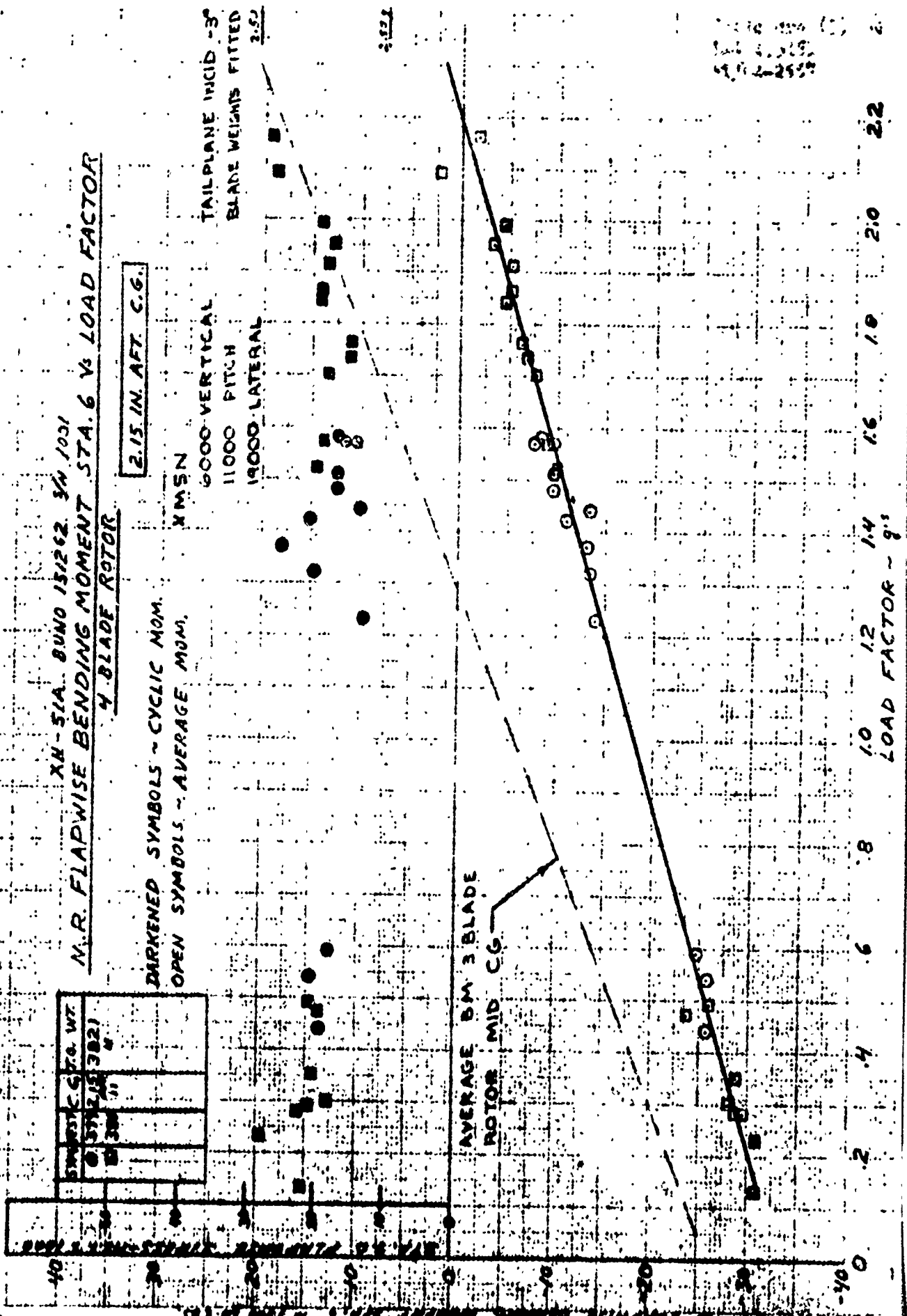
2.15 IN. AFT. C.G.

DARKENED SYMBOLS ~ CYCLIC MOM.  
 OPEN SYMBOLS ~ AVERAGE MOM.

XMSN  
 6000 VERTICAL  
 11000 PITCH  
 19000 LATERAL

TAILPLANE INCID -3°  
 BLADE WEIGHTS FITTED  
 2.50

SYMBOL	C.G. WT.
○	3821
●	3821
□	3821
■	3821



FLAP B.M. AT STA. 6.6 VS. LOAD FACTOR C.G. 2.15' AFT FINAL FRAME FIG 7

XN-57A BU No 151262 SN 1001

M.R. CHORDWISE BENDING STA. 6 VS. LOAD FACTOR

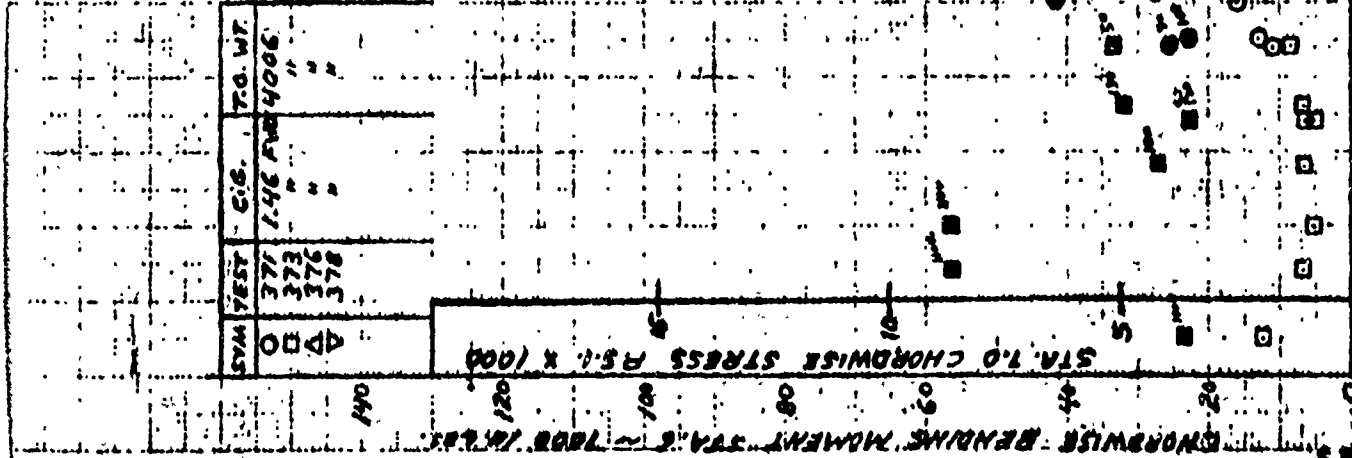
4 BLADE ROTOR

1.46 W. FWD. C.G.

SYM TEST	C.G.	TO WT
△	1.46 FWD	HOGG
□	"	"
○	"	"
◇	"	"

DARKENED SYMBOLS - CYCLIC MOM.  
 OPEN SYMBOLS - AVERAGE MOM.  
 SMALL NUMBERS AT RIGHT OF  
 DARKENED SYMBOLS ARE  
 INDICATED AIRSPEED.

XMSN  
 6000 VERTICAL  
 11000 PITCH  
 19000 LATERAL  
 BLADE WEIGHTS FITTED  
 TAILPLANE INCID - 3°  
 - 5 1/2° ON TEST 371



FORM 8270A  
 CHORD B.M. AT HUB STA. 6.0 VS. LOAD FACTOR C.G. 1.46" FWD  
 FINAL PHASE III CONFIG.

FIG 8

CHORD B. M. AT HUB STA. 6.0 VS LOAD FACTOR C. G. 2.15" AFT FINAL PHASE II CONFIG

SYM TEST	C. G.	T. W.
○ 379	2.15 AFT	3821
□ 380	"	"

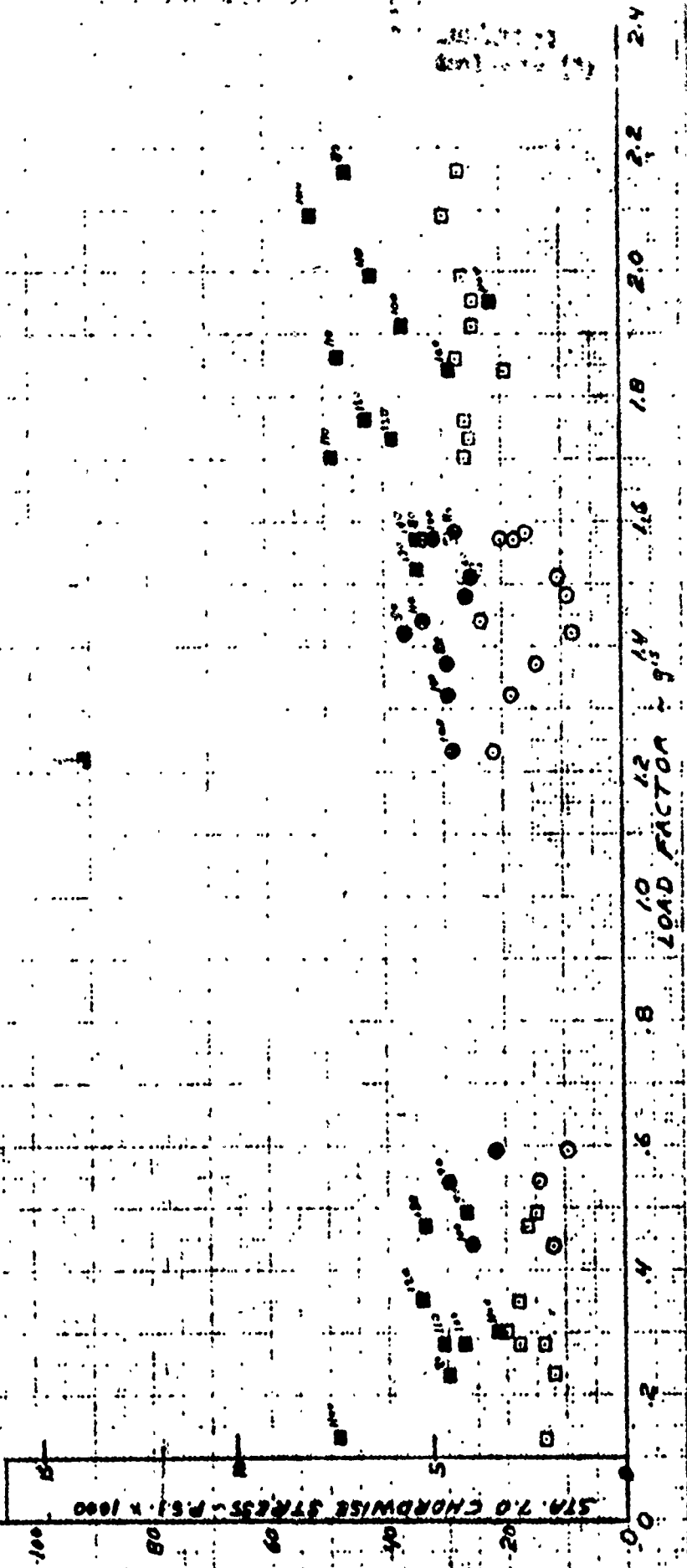
XN-51A BUHO 151262. 1/12/01.  
 M.R. CHORDWISE BENDING STA. 6.0 VS. LOAD FACTOR  
 4 BLADE ROTOR  
 2.15 IN. AFT C.G.

DARKENED SYMBOLS - CYCLIC MOM.  
 OPEN SYMBOLS - AVERAGE MOM.  
 SMALL NUMBERS AT RIGHT OF  
 DARKENED SYMBOLS ARE  
 INDICATED AIR SPEED

XMSN 45/IN  
 6000 VERTICAL  
 11000 PITCH  
 14000 LATERAL  
 BLADE WEIGHTS FITTED  
 TAILPLANE INCID - 3°

STA 7.0 CHORDWISE STRESS - PSI x 1000

CHORDWISE BENDING MOMENT STA. 6.0 - 1000 IN. LB.



**M.R. FLAPWISE BENDING MOMENT VS. ROTOR SPAN**

AN-51A BUND 151262 3/4/1961

4-BLADE ROTOR - HIGH SPEED LEVEL

TEST 385 10-2-64 OSC. CTR. 253

CG. 1.45 IN. FND. 70. GR. WT. 4006 LBS.

LEVEL FLIGHT AT 132.0 KNOTS C.A.S.

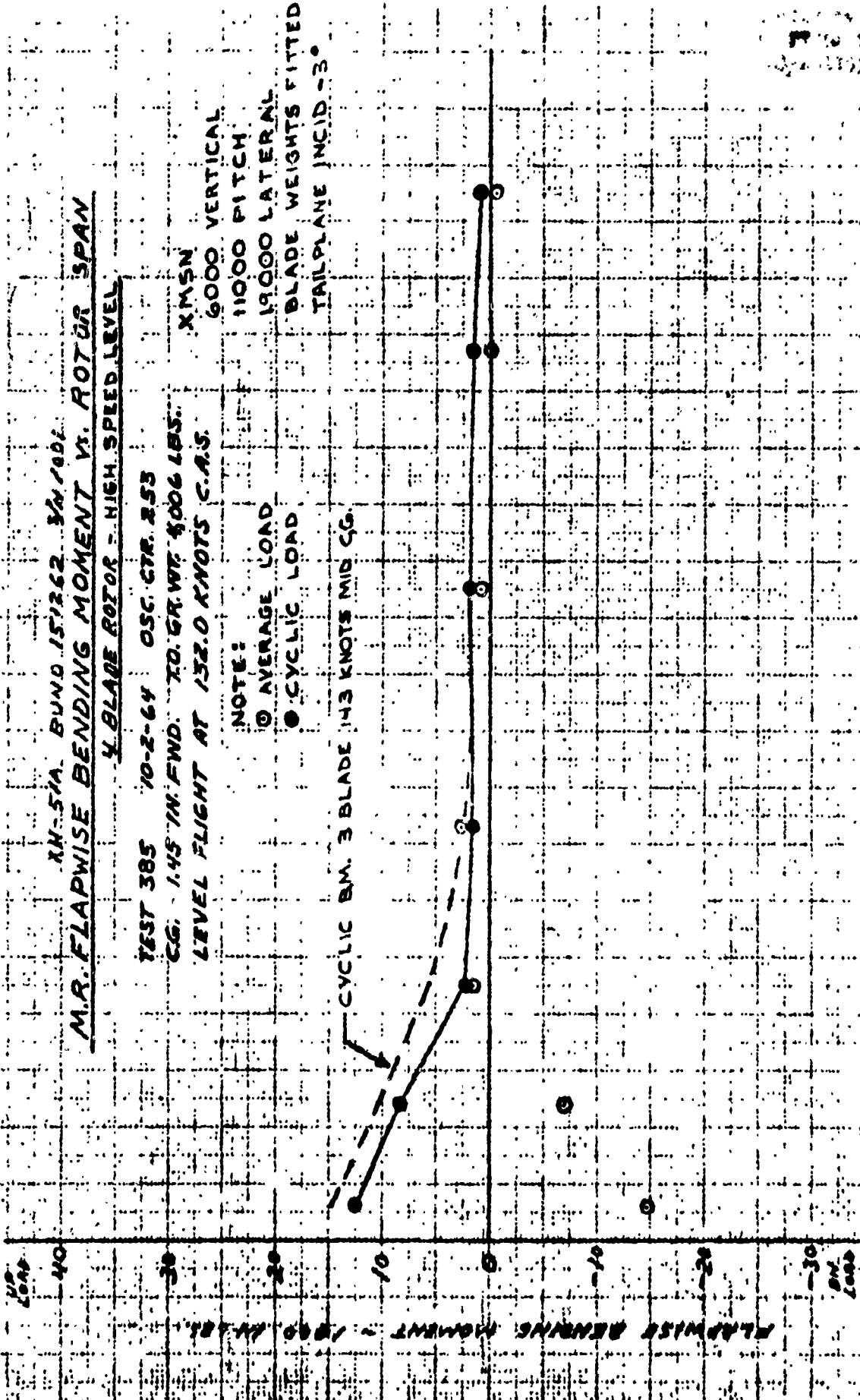
XMSN  
6000 VERTICAL  
11000 PITCH  
19000 LATERAL  
BLADE WEIGHTS FITTED  
TAILPLANE INCID. -3°

NOTE:

○ AVERAGE LOAD

● CYCLIC LOAD

CYCLIC BM. 3 BLADE 143 KNOTS MID CG.



FLAP BENDING MOMENT VS. SPAN-LEVEL FLT. FINAL PHASE III CONFIG.

S.F.62.1  
FIG 10

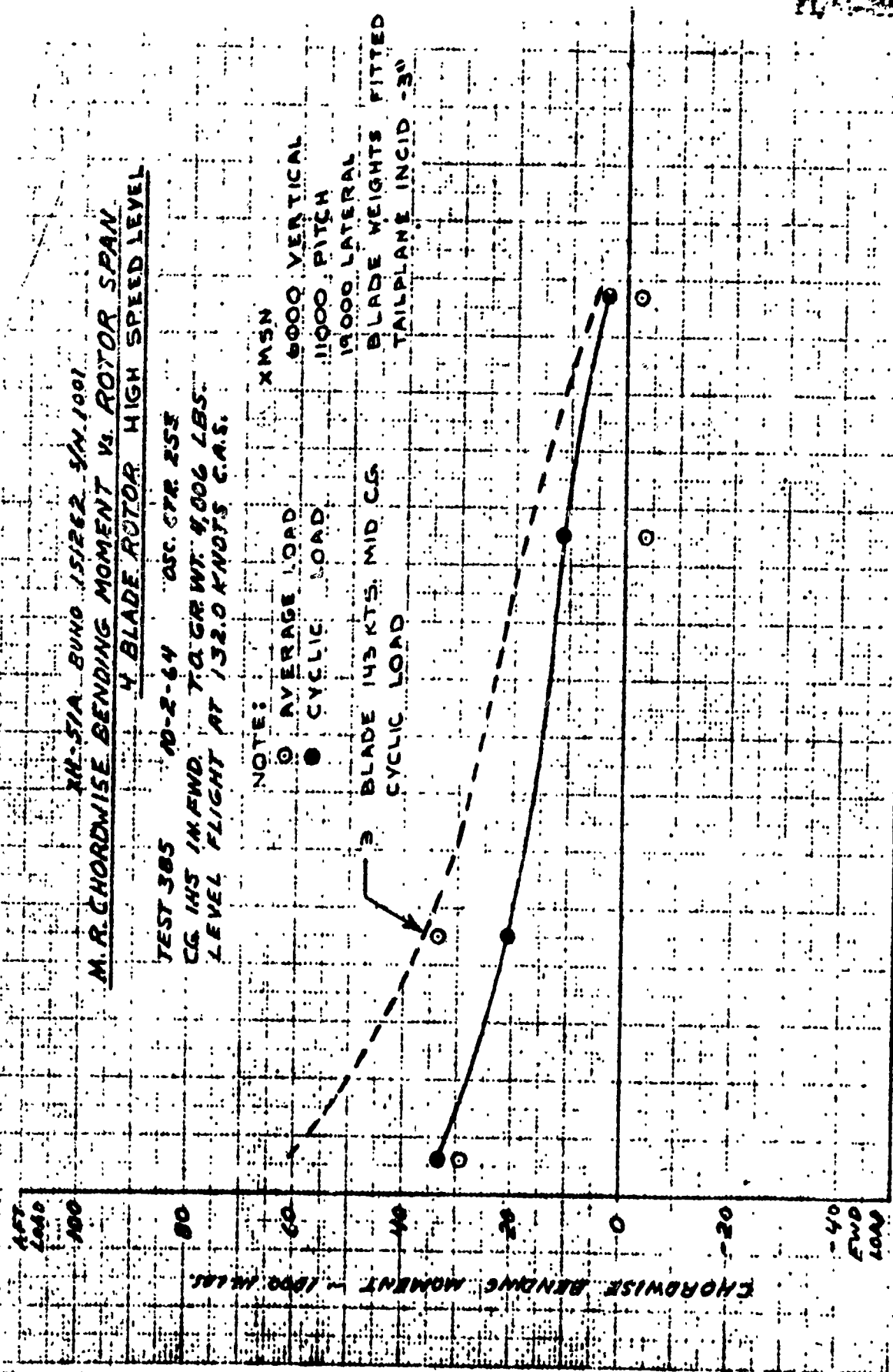
See also  
 62-141/13/11  
 PL/1-109

**M.R. CHORDWISE BENDING MOMENT VS. ROTOR SPAN  
 4 BLADE ROTOR HIGH SPEED LEVEL**

TEST 385 10-2-64 OSC. CTR. 253  
 CG 145 IN FWD. TO GRWT. 4,006 LBS.  
 LEVEL FLIGHT AT 132.0 KNOTS C.A.S.

NOTE: XMSN

- AVERAGE LOAD 6000 VERTICAL
- CYCLIC LOAD 11000 PITCH
- BLADE 143 KTS. MID CG 19000 LATERAL
- CYCLIC LOAD BLADE HEIGHTS FITTED
- TAILPLANE INCID -30



CHORD BEND. MOM. VS. SPAN LEVEL FLT. FINAL PHASE 22 CONFIG.

S.F. 13.1  
 FIG 11

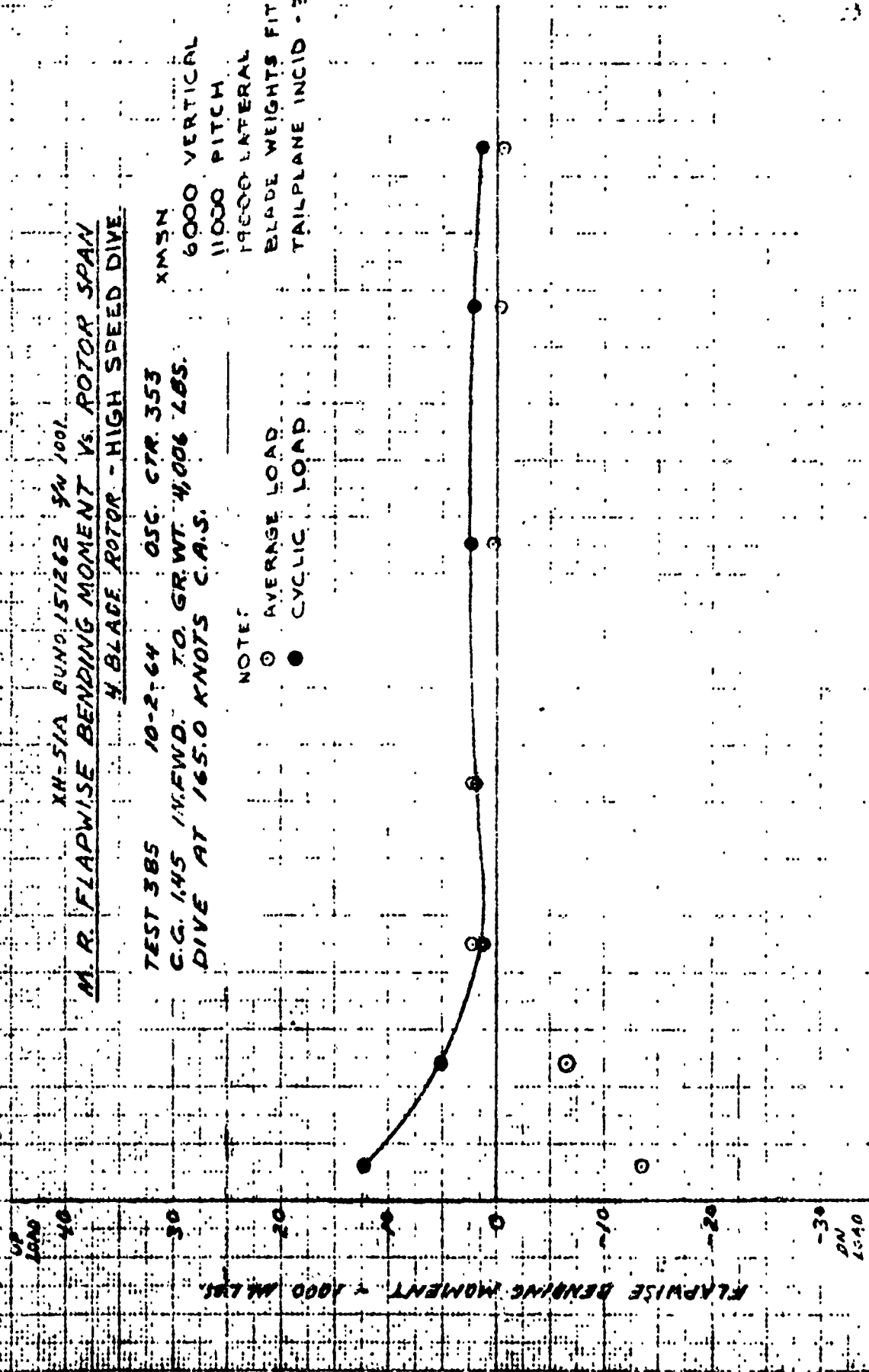
**M. R. FLAPWISE BENDING MOMENT VS. ROTOR SPAN**  
**H. BLADE ROTOR - HIGH SPEED DIVE**

TEST 385 10-2-64 OSC. CTR. 353  
 C.G. 145 IN. FWD. TO. GR. WT. 4,006 LBS.  
 DIVE AT 165.0 KNOTS C.A.S.

XMSN  
 6000 VERTICAL  
 11000 PITCH  
 19000 LATERAL  
 BLADE WEIGHTS FITTED  
 TAILPLANE INCID - 3°

NOTE:

- AVERAGE LOAD
- CYCLIC LOAD

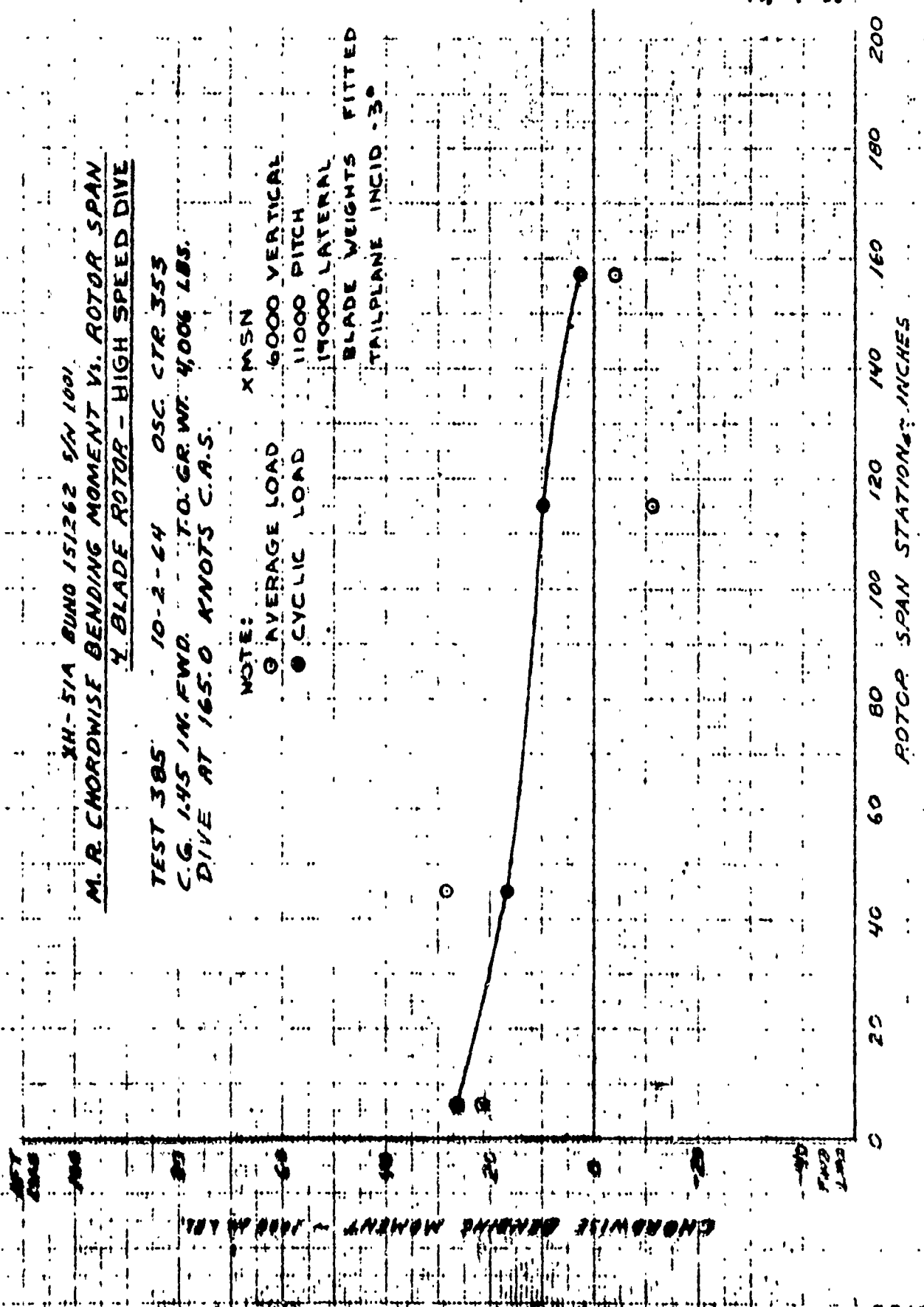


FLAP BENDING MOMENT VS. SPAN IN DIVE FINAL PHASE III CONFIG.

XH-51A BUONO 151262 SN 1001  
 M. R. CHORDWISE BENDING MOMENT VS. ROTOR SPAN  
 Y. BLADE ROTOR - HIGH SPEED DIVE

TEST 385 10-2-64 OSC. CTR 355  
 C.G. 1.45 IN. FWD. T.O. GR. WT. 4,006 LBS.  
 DIVE AT 165.0 KNOTS C.A.S.

NOTE: XMSN  
 ○ AVERAGE LOAD 6000 VERTICAL  
 ● CYCLIC LOAD 11000 PITCH  
 17000 LATERAL  
 BLADE WEIGHTS FITTED  
 TAILPLANE INCID. 3°



CHORD BENDING MOMENT VS. SPAN IN DIVE FINAL PHASE II CONFIG. SP. 15.1  
 FIG 13

XH-51A BUNO 151262 SIN 1001  
**MR. FLAPWISE BENDING MOMENT VS. ROTOR SPAN**

**4 BLADE ROTOR - HIGH 'G' TURN IN DESCENT**

TEST 376 9-23-64 OSC. CTR. 564  
 CG. 146 IN. FWD. GR. WT. 3,965 LBS.  
 DESCENT AT 118.5 KNOTS C.A.S.  
 2.23 G'S AT 3,965 LBS.  
 2.53 G'S CORRECTED TO 3,500 LBS.

XMSN

6000 VERTICAL  
 11000 PITCH  
 19000 LATERAL  
 BLADE WEIGHTS FITTED  
 TRAILPLANE INCID -3°

NOTE: ○ AVERAGE LOAD  
 ● CYCLIC LOAD

UP  
 LOAD

FLAPWISE BENDING MOMENT - 1000 IN. LB.

-30  
 DN  
 LOAD

0 20 40 60 80 100 120 140 160 180 200  
 ROTOR SPAN STATION - INCHES

FLAP BENDING MOMENT VS. SPAN IN MANEUVER FINAL PHASE II CONFIG. S.F.57. FIG 1

XH-51A BUNO 151262 SIN 1001

M/R CHORDWISE BENDING MOMENT VS. ROTOR SPAN

4-BLADE ROTOR HIGH G TURN IN DESCENT

TEST 376 9-23-64 OSC. CTR. 564  
C.G. 1.16 IN. FWD. GR. WT. 3,965 LBS.  
DESCENT AT 118.5 KNOTS C.A.S.  
2.23 G'S AT 3,965 LBS. XM SN  
2.53 G'S CORRECTED TO 3500 LBS.

6000 VERTICAL  
11000 PITCH  
19000 LATERAL  
BLADE WEIGHTS  
FITTED  
TAIL PLANE INCID -3'

NOTE:

- AVERAGE LOAD
- CYCLIC LOAD

CHORDWISE BENDING MOMENT - 1000 IN. LBS.  
AFT LOAD  
100  
80  
60  
40  
20  
0  
-20  
-40  
DN. LOAD

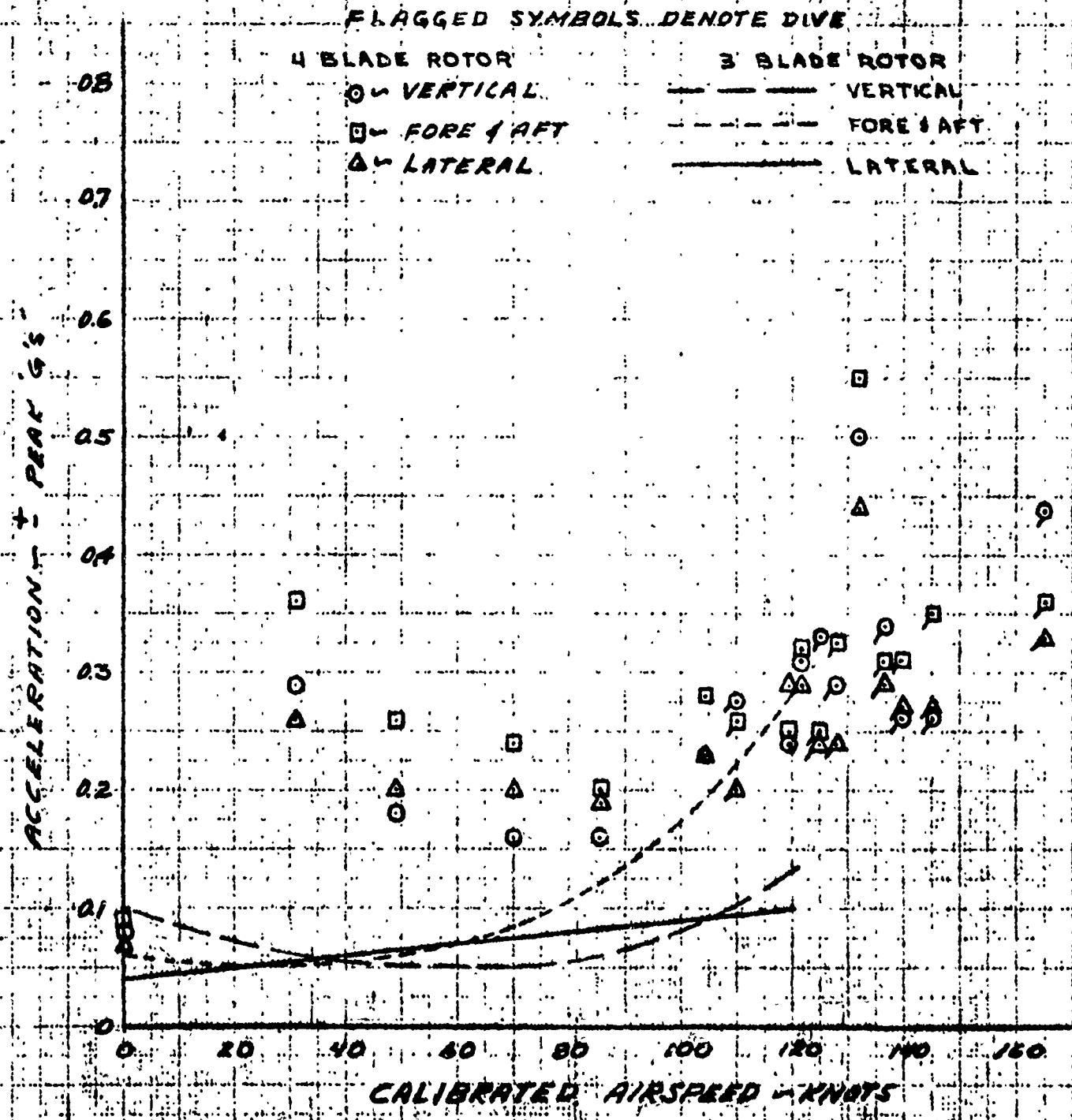
0 20 40 60 80 100 120 140 160 180 200  
ROTOR SPAN STATION - INCHES

CHORDWISE BENDING MOM. VS. SPAN IN MANEUVER FINAL PHASE II CONFIG.

S.F. 58.1  
FIG 15

# CABIN VIBRATIONS VS. CALIBRATED AIRSPEED

TEST 373 1385  
4 PER REV.



FORM 5476a

CABIN VIBRATION LEVEL

FINAL PHASE & CONFIG.

5.59  
FIG 16

# CYCLIC CONTROL POSITIONS IN LEVEL FLIGHT

4-BLADE ROTOR SYSTEM

FWD C.G. LOCATION

SHIP: BUNO 151262

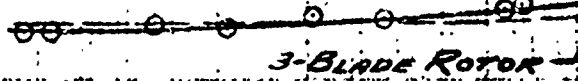
CYCLIC STICK ROLL POSITION - IN

RIGHT  
2  
1  
0  
1  
2  
LEFT

AVERAGE TEST CONDITIONS

SYM	TEST	WAVE LEN	DEN REFST	RPM	LONG. MOM.	LAT. MOM.
0	384	3910	1250	22.5	-61.0 IN-LB FUL	+755 IN-LB FT.

TRAVEL LIMIT: 2.81 IN. RIGHT - 2.81 IN. LEFT.



3-BLADE ROTOR

CONFIGURATION NOTES:

1. CYCLIC STICK PITCH SENSITIVITY = 100%.
2. LANDING GEAR UP.
3. SPEED SENSOR OFF.

CYCLIC STICK PITCH POSITION - IN

AFT  
1  
0  
1  
2  
3  
4  
5  
6  
FWD

3-BLADE ROTOR 100% SENSITIVITY

AFT TRAVEL LIMIT = 4.75 IN.

FWD TRAVEL LIMIT = 6.125 IN.

FWD 20

40

60

80

100

120

140

TRUE AIRSPEED - KNOTS

CONTROL TO TRIM IN LEVEL FLIGHT

FIGURE 17

# STATIC LONGITUDINAL STABILITY

FWD C.G. LOCATION

FOUR BLADE ROTOR SYSTEM

SNIP: BUWD 151262

PULL  
10  
5  
0  
5  
10  
PUSH

TEST	WAVE NO.	DN ASMT	LONG. MON.	LAT. MON.
386	3588	4200	-330 44-10 PWD	737 44-10 R

○△ - TRIM  
○△ - RETURN

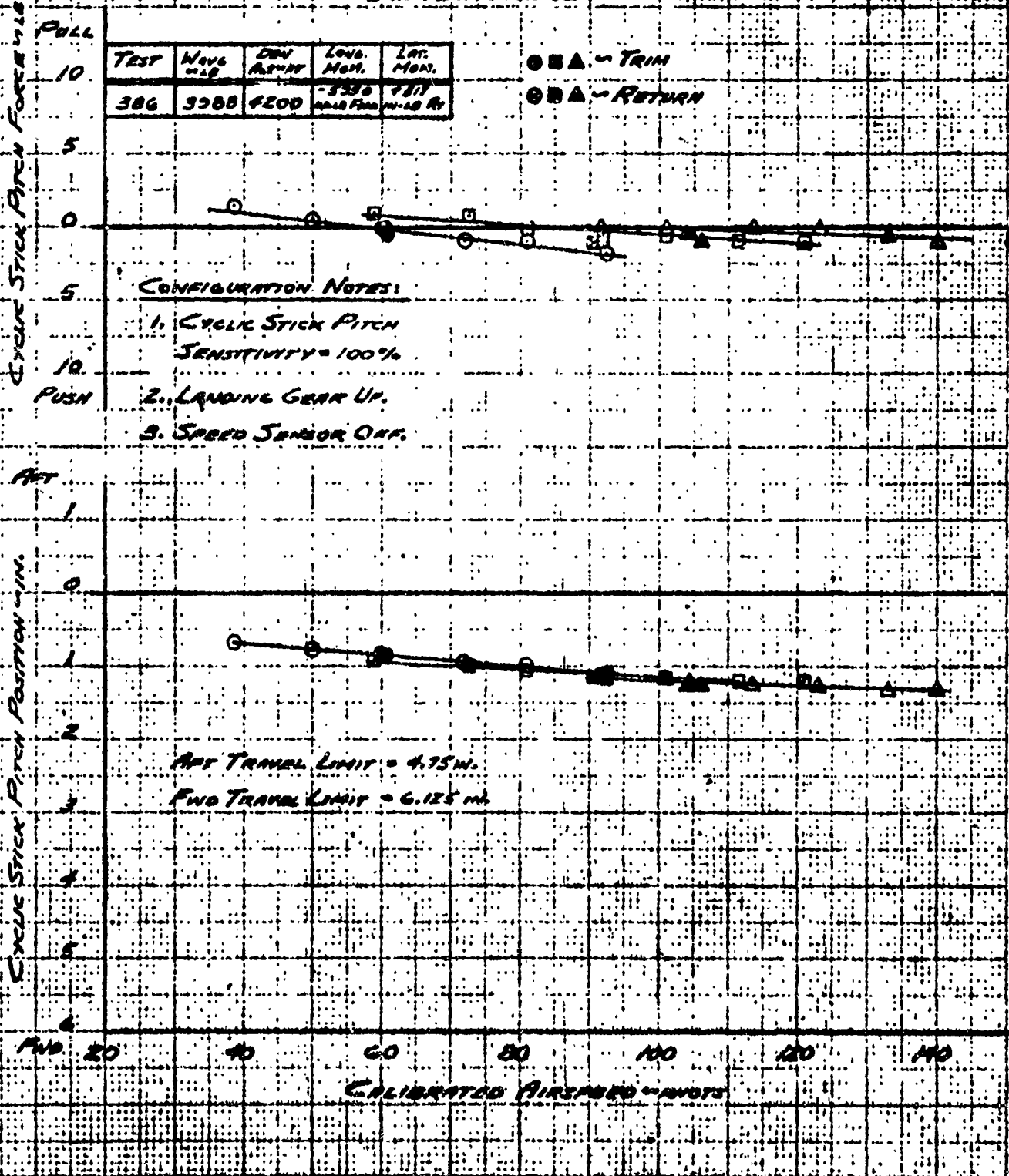
**CONFIGURATION NOTES:**

1. CYCLIC STICK PITCH SENSITIVITY = 100%
2. LANDING GEAR UP.
3. SPEED SENSOR OFF.

AFT  
1  
0  
1  
2  
3  
4  
5  
6

AFT TRAVEL LIMIT = 4.75 IN.  
FWD TRAVEL LIMIT = 6.125 IN.

20 40 60 80 100 120 140  
CALIBRATED AIRSPEED - KNOTS



FORM 837C

CONSTANT COLLECTIVE STATIC LONGITUDINAL STABILITY

(1000)

993A

FIGURE 18

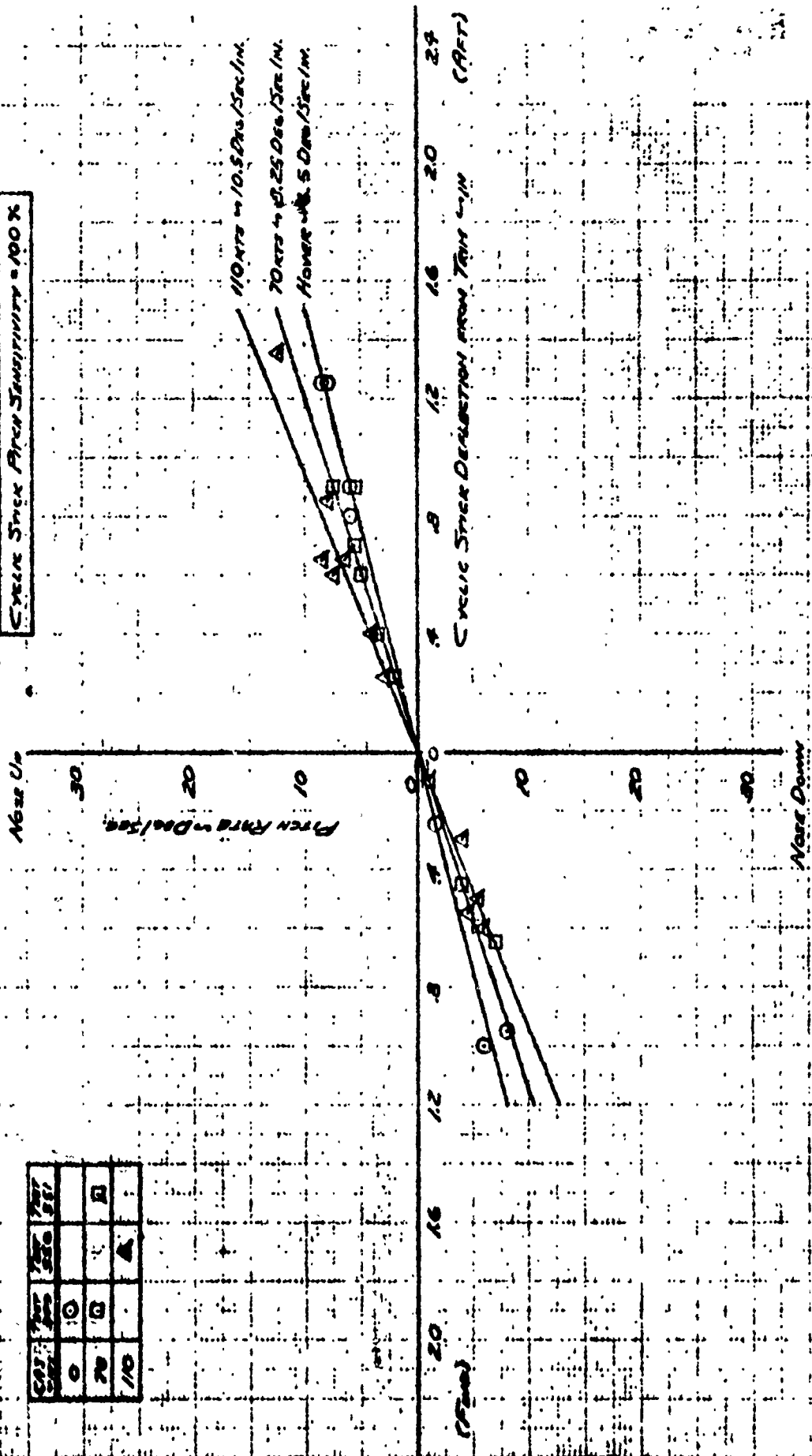
LOCKHEED HELICOPTER  
Model XH-51A

LONGITUDINAL CONTROL POWER  
Four Blade Main Rotor

SNIP: 6810 15126Z

Cyclic Stick Pitch Sensitivity = 100%

COL. 1	COL. 2	COL. 3	COL. 4	COL. 5	COL. 6
0	10	20	30	40	50
0	10	20	30	40	50
0	10	20	30	40	50



LONGITUDINAL CONTROL POWER

FIGURE 20

(100)

979A

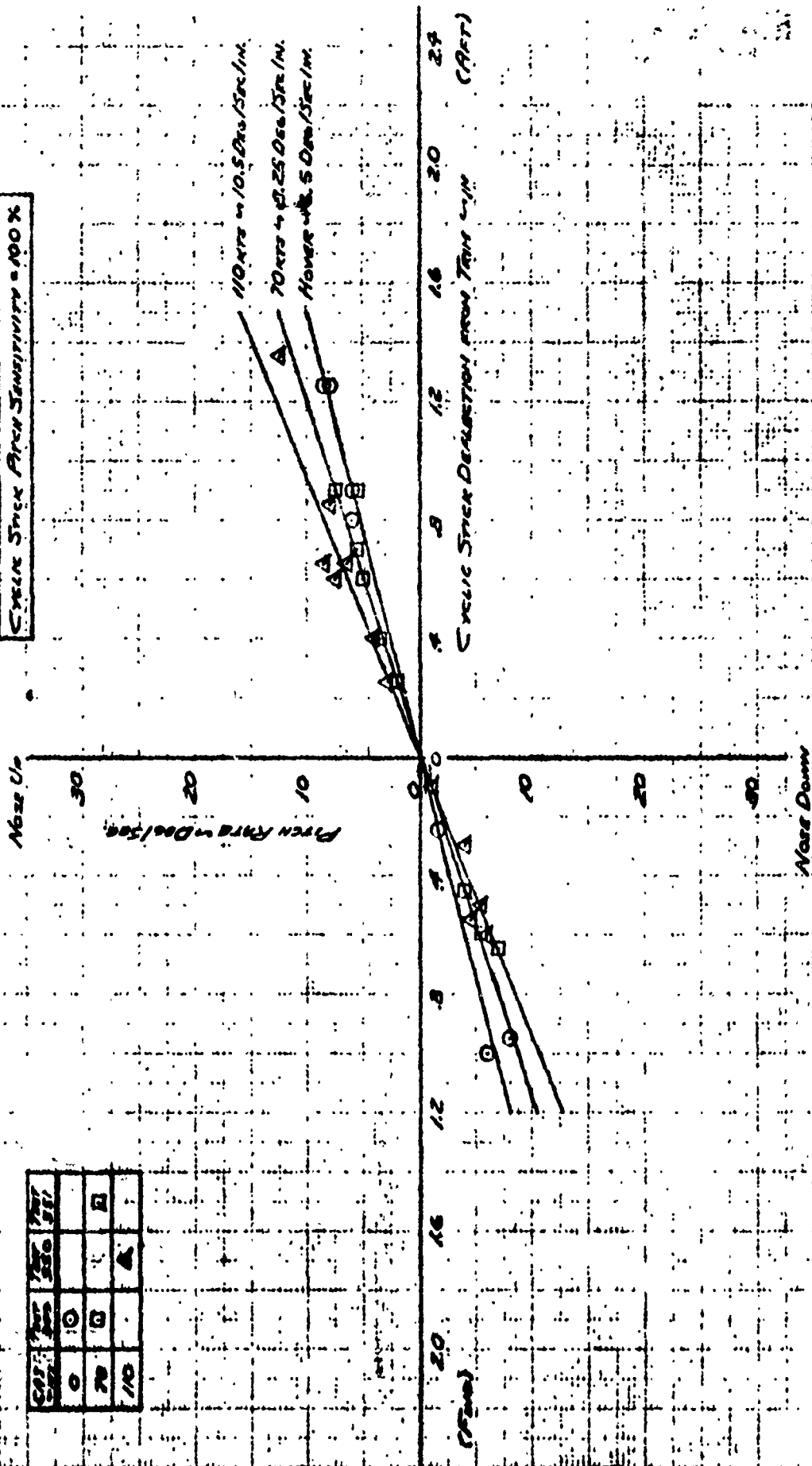
LOCKHEED HELICOPTER  
Model XH-51A

LONGITUDINAL CONTROL POWER  
Four Blade Main Rotor

SNIP: BUANO 151262

Cyclic Stick Pitch Sensitivity = 100%

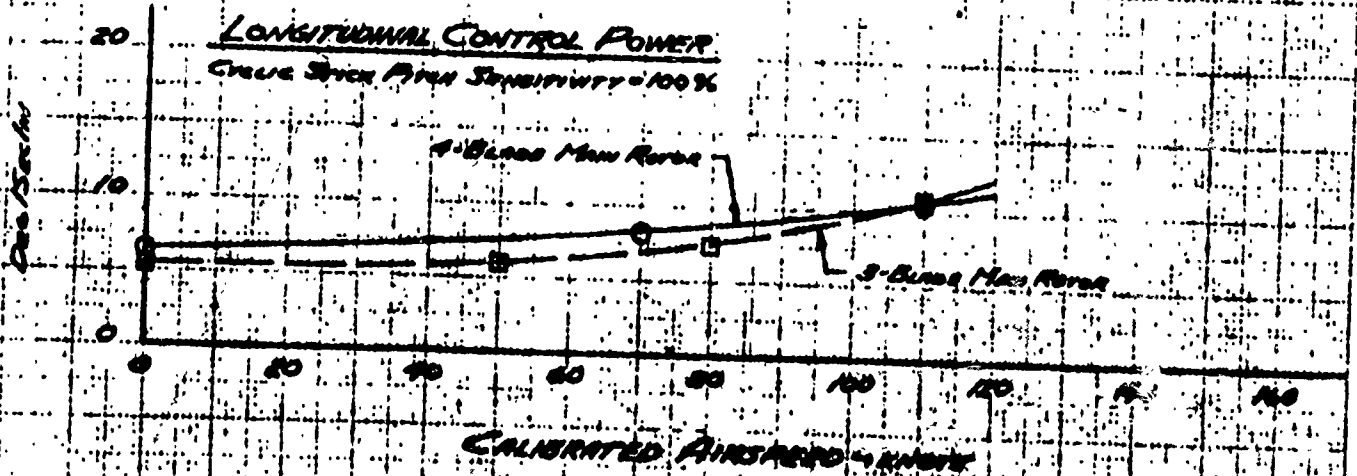
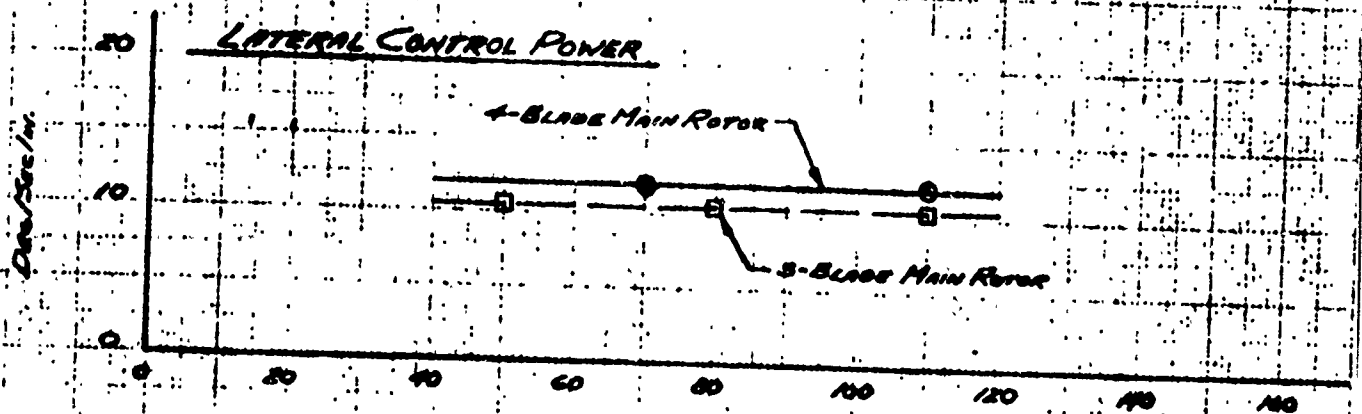
CTP	Max Stick Defl	Max Stick Defl	Max Stick Defl
0	0	0	0
75	0	0	0
100	0	0	0



LONGITUDINAL CONTROL POWER

# CONTROL POWER COMPARISON BETWEEN THE 3 AND 4 BLADE ROTOR SYSTEMS

SIN: BUND 15828



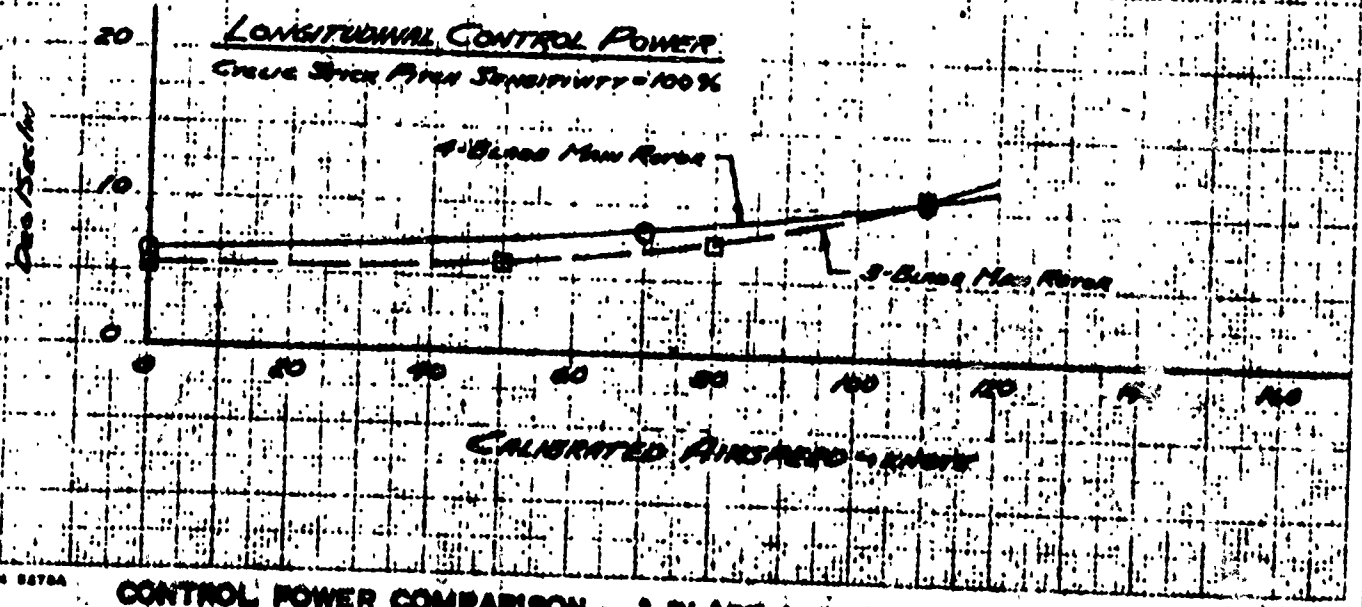
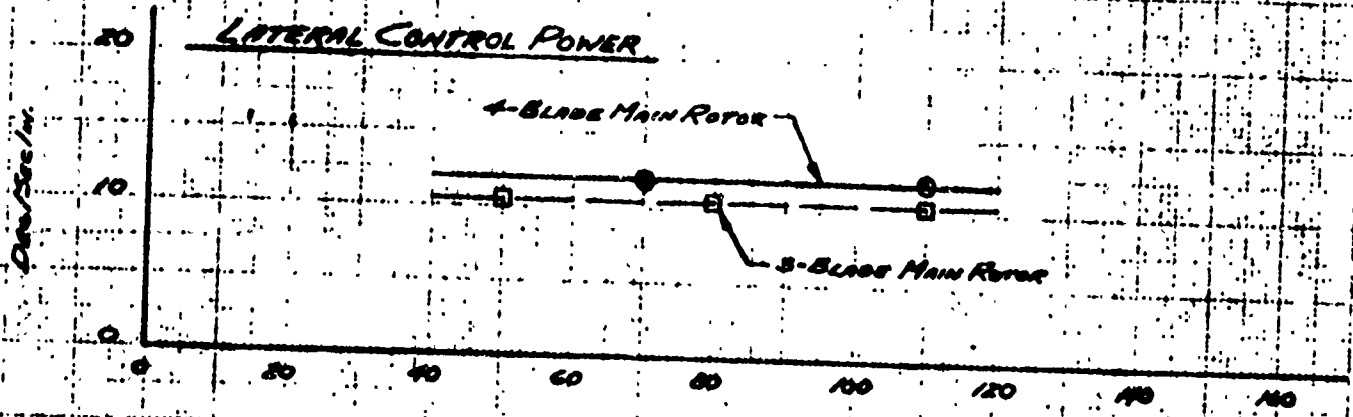
CONTROL POWER COMPARISON - 3 BLADE & 4 BLADE ROTORS

FIGURE 22

2809

# CONTROL POWER COMPARISON BETWEEN THE 3 AND 4 BLADE ROTOR SYSTEMS

SIN: BWD 1582



CONTROL POWER COMPARISON - 3 BLADE & 4 BLADE ROTORS

FIGURE 22

280A

# MANEUVERING STABILITY

FWD C.G. LOCATION

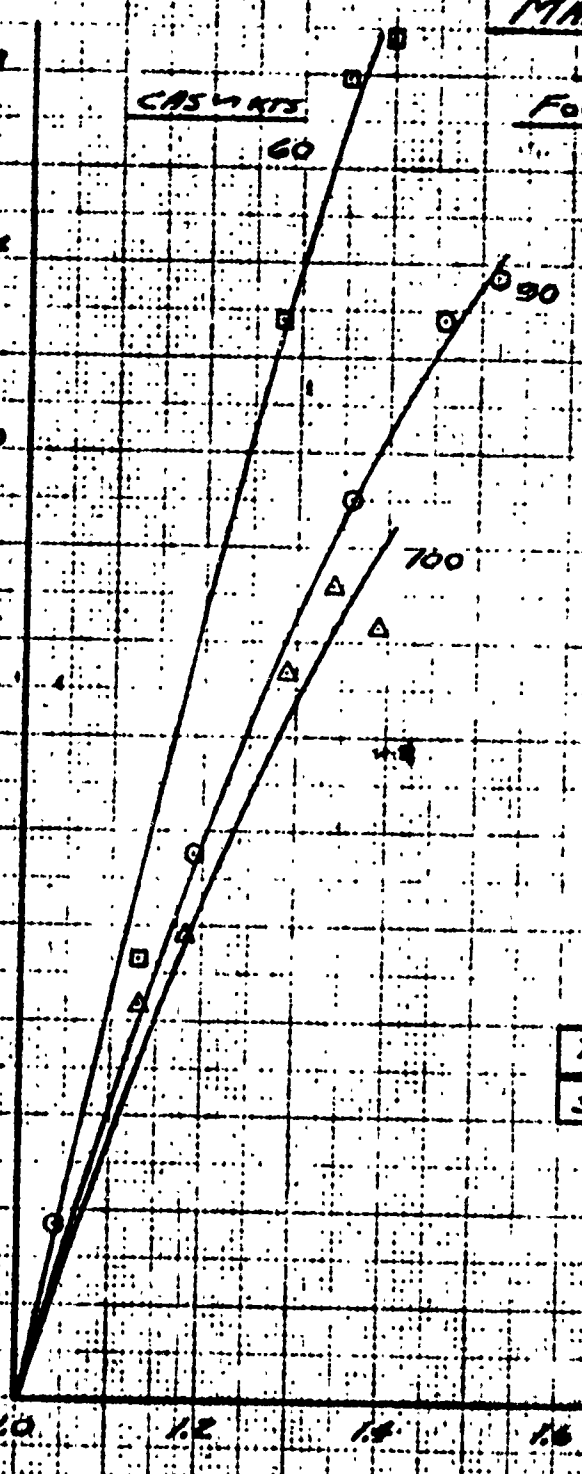
FOUR BLADE ROTOR SYSTEM

SHAW-BUNO SIZE 2

Cyclic Stick PITCH FEEDBACK

## CONFIGURATION NOTES:

1. CYCLIC STICK PITCH SENSITIVITY = 100%
2. EXTERNAL TUNING WEIGHTS ON MAIN ROTOR BLADES.
3. LANDING GEAR UP.
4. SPEED SENSOR OFF.
5. 31.5 LB BOE-WEIGHT INSTALLED.



TEST	FLY	WIND	DEN	LONG.	LAT.
		W/LE	ALTIM	MM.	MM.
386	244	9388	1300	-8330	1517
				W/LE	W/LE

MANEUVERING STABILITY 4 BLADE SYSTEM

FIGURE 23 994A

81800

FORM 8278

# MANEUVERING STABILITY

MID C.G. LOCATION

SHIP: BUNO 151262

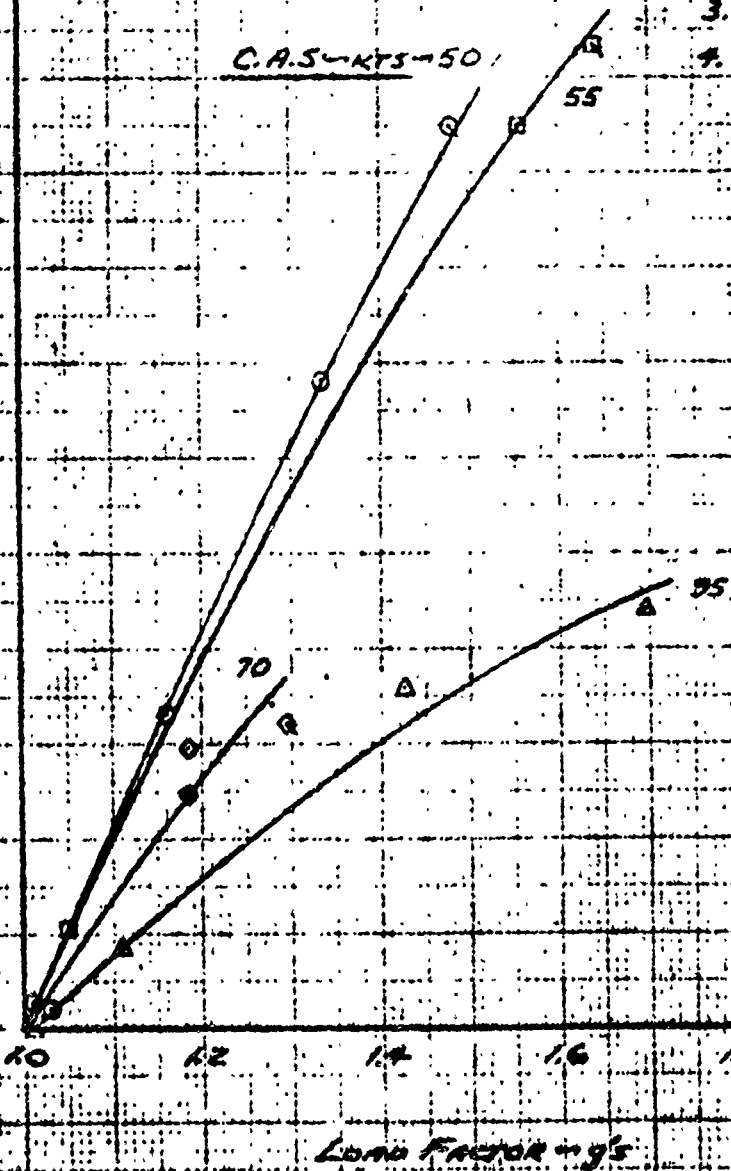
THREE BLADE ROTOR SYSTEM

SYM	TEST	WAVE WLG	GEN ALT-F'	LONG MSA.	LAT. MM.
○ △ □	279	3710	5600	0	2-87 IN-18-0
○ △ □	303	3430	2350	588 W-LB-AT	2109 -RT.

**CONFIGURATION NOTES:**

1. CYCLIC STICK PITCH SENSITIVITY = 100%
2. LANDING GEAR RETRACTED
3. SPEED SENSOR OFF.
4. 31.5 LB BOB-WEIGHT INSTALLED.

CYCLIC STICK PITCH FORCE IN LB



FORM 487B

MANEUVERING STABILITY 3 BLADE ROTOR  
FIGURE 24

879A  
(1001)

# MANEUVERING STABILITY

FWD. C.G. LOCATION

SHIP: BUONO 151262

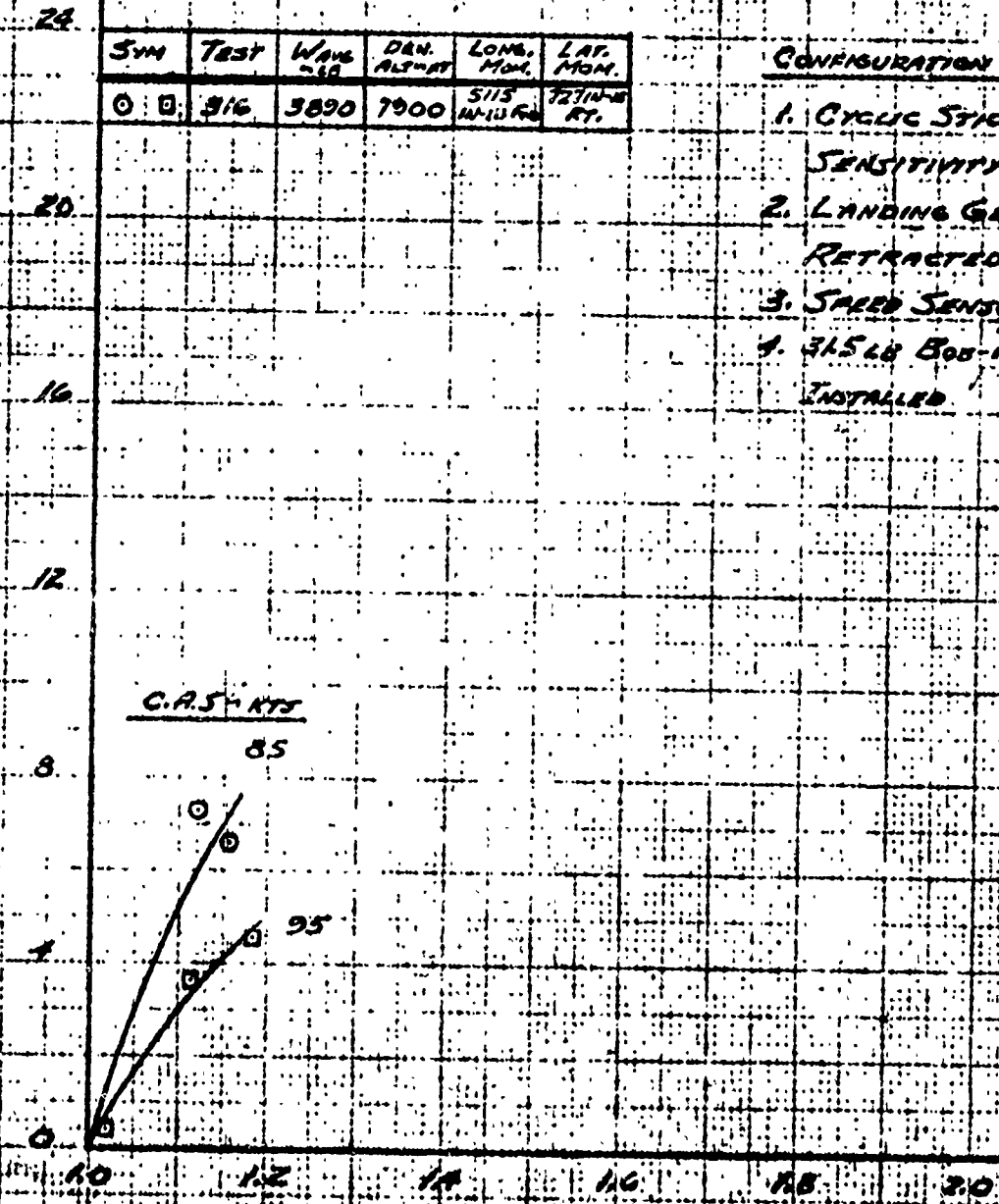
THREE BLADE ROTOR SYSTEM

CYCLIC STICK PITCH FORCE - LB

SYM	TEST	WAVE NO.	DRN. ALTITUDE	LONG. MOG.	LAT. MOG.
○ □	316	3890	7900	5115 W-13 K8	7271N-15 RT.

**CONFIGURATION NOTES:**

1. CYCLIC STICK PITCH SENSITIVITY = 100%.
2. LANDING GEAR RETRACTED.
3. SPEED SENSOR OFF.
4. 315 LB BOB-WEIGHT INSTALLED.



FORM 5278

**MANEUVERING STABILITY 3 BLADE ROTOR  
FIGURE 25**

80/A  
(100)

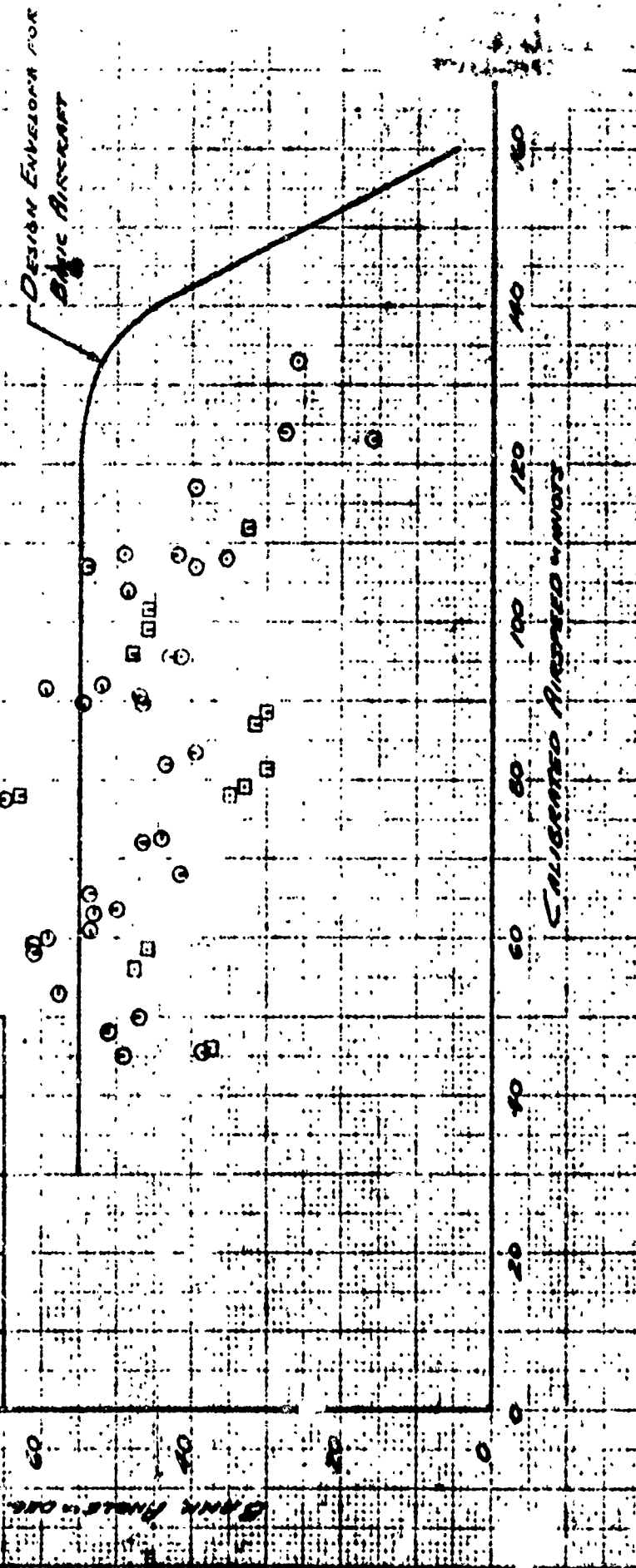
LOCKHEED HELICOPTER  
MODEL XH-51A

**BANK ANGLE - VELOCITY ENVELOPE**

SNIP: BUWG 151262  
FWD AND AFT C.G. LOCATION  
FOUR BLADE MAIN ROTOR

SIN	TEST	Max V <sub>100</sub>	Low V <sub>100</sub>	LAC Aft V <sub>100</sub>
①	371	3965	3680	+778
②	373	3875	3630	+778
③	376	3998	3720	+778
④	378	3975	3700	+778
⑤	386	3988	3850	+817

SIN	TEST	Max V <sub>100</sub>	Low V <sub>100</sub>	LAC Aft V <sub>100</sub>
⑥	379	3765	3750	+778
⑦	380	3785	3780	+778



ANGLE OF BANK VELOCITY ENVELOPE

FIGURE 28

(100)

28

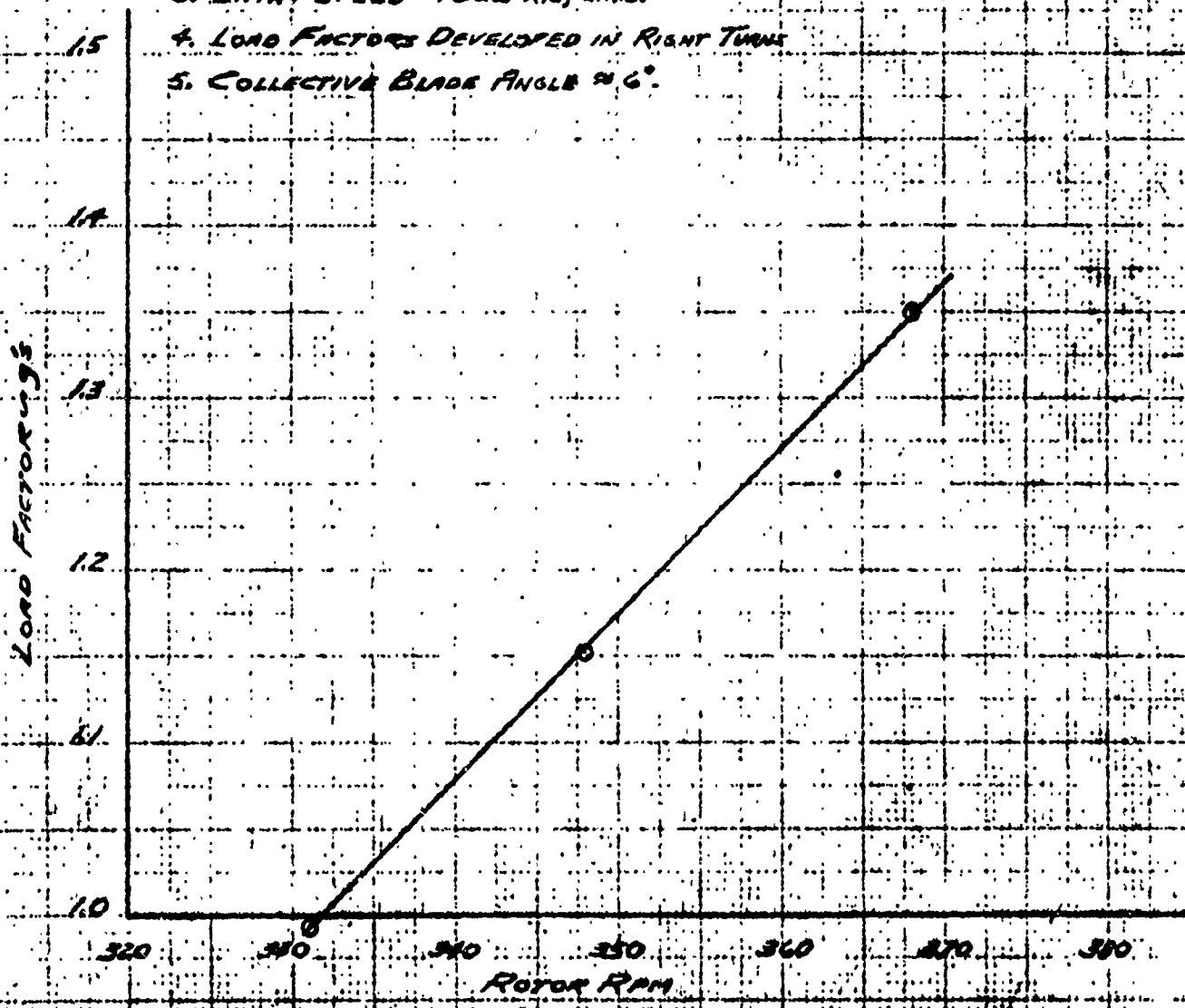
# LOAD FACTOR REQUIRED TO MAINTAIN A GIVEN ROTOR RPM IN AUTOROTATION

SHIP: BUONO 151262      TEST 383, FLIGHT 291

NOTES:

1. 4 BLADE MAIN ROTOR WITH EXTERNAL TUNING WEIGHTS AT THE 5-FT RADIUS.
2. LANDING GEAR RETRACTED.
3. ENTRY SPEED = 7622 KTS, C.A.S.
4. LOAD FACTORS DEVELOPED IN RIGHT TURN
5. COLLECTIVE BLADE ANGLE  $\approx 6^\circ$ .

W = 4000 LB  
 LONG. MOM = 4720 IN-LB FROM  
 LAT. MOM = 198 IN-LB TO R.



LOAD FACTOR - ROTOR SPEED IN AUTOROTATION  
 FIGURE 27

(100)  
 386

FIGURE 20

COMPARISON OF 3-BLADE AND 4-BLADE ROTOR  
HOVER PERFORMANCE

SEA LEVEL STD. DAY 100% RPM

SMP: BUNG 15126R

WGT: 4050 LB

TEST 329

28 July 67

NO EXTERNAL TUNING WEIGHTS  
ON MAIN ROTOR BLADES.

5000  
4000  
3000  
2000  
1000  
0  
500  
1000  
1500  
2000  
2500  
3000  
3500  
4000  
4500  
5000  
5500  
6000  
6500  
7000  
7500  
8000  
8500  
9000  
9500  
10000

3 BLADE MAIN ROTOR  
(PREVIOUS TEST RESULT)

4 BLADE MAIN ROTOR

240 280 320 360 400 440 480

HOVER PERFORMANCE

FIGURE 20

2547

FORM 8870

GRAPHIC CASE

LOCKHEED HELICOPTER  
MODEL XH-51A

LEVEL FLIGHT PERFORMANCE

COMPARISON OF 3-BLADE AND 4-BLADE ROTOR SYSTEMS

IMP: BUWD 151262  
SEA LEVEL STANDARD DAY -100% RAN  
LANDING GEAR RETRACTED

Average Test Conditions

Test	Wing Area	Sea Level Altitude	Sea Level Altitude %	Car Weight
①	3970	1250	92.5	6760 lbs. (incl. rotor)
②	4000	1400	93.0	6925 lbs. (incl. rotor)

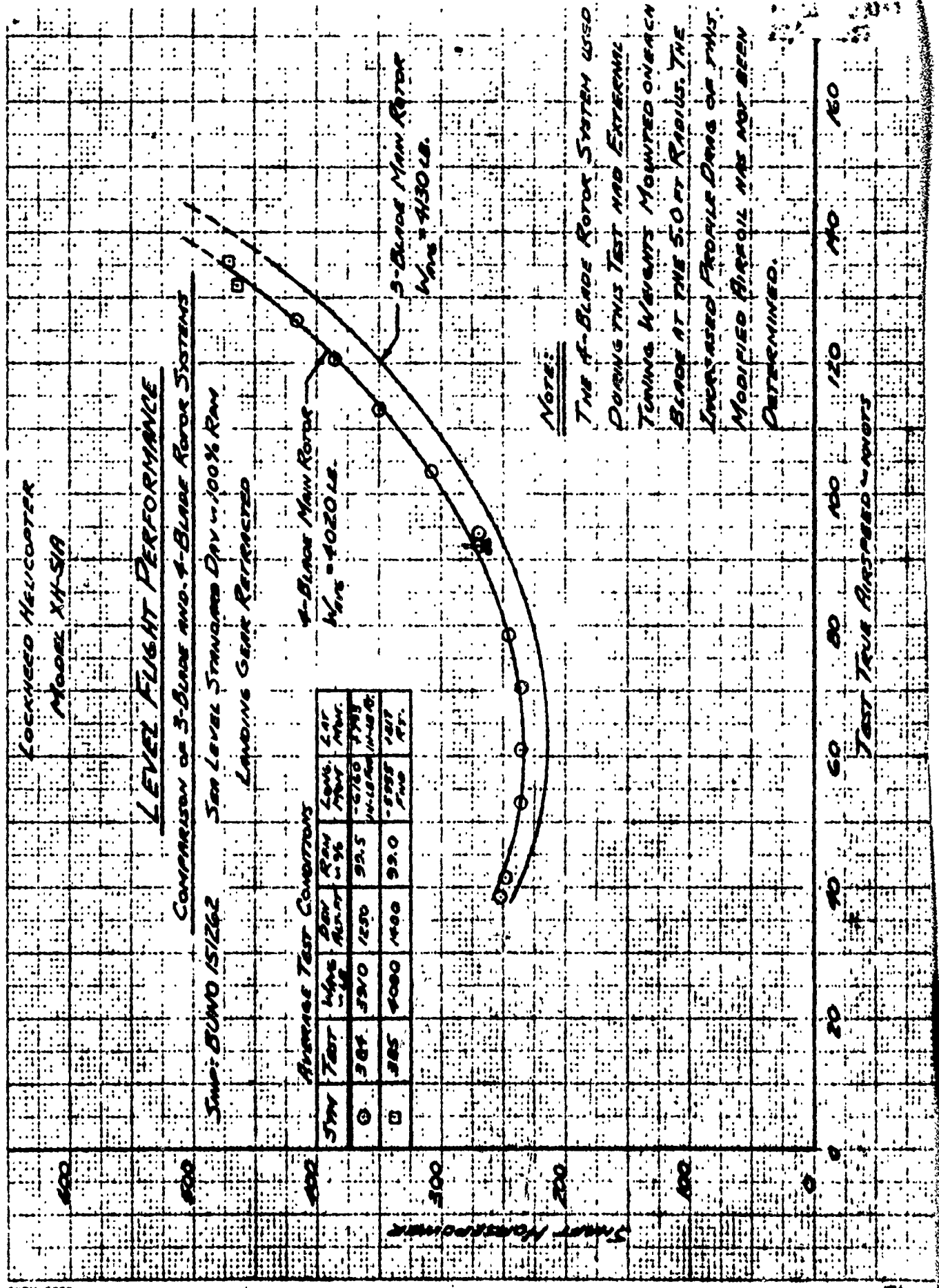
4-Blade Main Rotor  
Wing = 4020 LB.

3-Blade Main Rotor  
Wing = 4130 LB.

True Airspeed

Note:

THE 4-BLADE ROTOR SYSTEM USED DURING THIS TEST AND EXTERNAL TOWING WEIGHTS MOUNTED ON EACH BLADE AT THE 50 FT RADIUS. THE INCREASED PROFILE DRAG OF THIS MODIFIED AIRFOIL HAS NOT BEEN DETERMINED.



FORWARD FLIGHT PERFORMANCE

FIGURE 29