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LABORATORY TECHNICAL REPORT

LTR - ST - 1263

TRACKER FATIGUE TEST: LOAD ANALYSIS AND DISCUSSION

G.F.W. McCAFFREY

RAPPORT TECHNIQUE DE LABORATOIRE

ÉTABLISSEMENT AÉRONAUTIQUE NATIONAL

21 MAY 1981

OTTAWA, CANADA

TRACKER FATIGUE TEST - LOAD ANALYSIS AND DISCUSSION

1.0 Introduction:

In general, the planning and execution of this test are considered to have been conducted in a very satisfactory manner. The test concept can be considered an evolutionary development of those used in similar tests of the CF100 and CL41 airframes. The main purpose of the present report is to highlight some problems, particularly those stemming from the high-wing configuration of the aircraft and the method of programming the random loading. This should assist in interpreting the results of this test and planning future tests of a similar nature.

This will be accomplished by presenting a description of the nature of the investigation and an analysis of the measured loads, for comparison with the design loads, followed by a discussion of certain aspects of this test and similar tests of other high-wing aircraft.

2.0 Investigation and Analysis of Loads:

2.1 The general approach adopted has been to monitor the day-to-day operation of the test by observing a large number of outputs which are presented as pen recorder traces, typed print-outs or digital voltages. From time to time a "data set" which includes all loads, strains and deflection measurements is recorded. These are taken on an "as required" basis, such as before a scheduled wing removal or because of any unusual noise from, or cracking of, the structure or any unusual output from the monitoring instrumentation.

2.2 Normally a data set is taken by commanding a number of halts at "end-of-cycle" so as to obtain a reasonable dispersion of the flight load magnitudes over the available range (from -0.246g to 3.844g). Strain gauge circuits are set to zero at the lg (ground) condition. The load is held at each selected load for the approximately 2 minutes required to record data. Many load cycles (or even flights) may occur before another load is selected.

In the first few weeks of testing a number of problems were identified and eliminated. Among these were:

(a) An apparent asymmetry of over-wing load, of approximately 2 1/2%, which was the result of the use of an incorrect "shunt" in establishing the load-cell bridge sensitivity.

(b) Incorrect magnitude of the under-fuselage load which required re-calibration of the load-cell and provision of an independent shunting resistor for sensitivity calibrations, as well as a minor correction to the load equation for the actuator. (This load system is the only one which has the load cell not directly in series with its actuator).

(c) Loss of amplitude at the smaller flight loads of the under-nacelle loads. This was cured by the use of the re-set integrators which were provided on the control console but had not been used. Their purpose is to provide variable gain (with frequency) which is particularly necessary in "bucking" loads such as this which move through very small distances when the main loading actuators are cycling at high frequency (small flight loads).

2.3 As these, and some lesser problems, were eliminated and sets of "clean" data became available it was apparent that the over-wing load cells were consistently measuring loads which were some 5 percent lower than the design value, although they appeared to be linear. Initially this tended to be treated as inconsequential since the accuracy of the applied loads was confirmed to be excellent.

However, since the magnitude of the over-wing reaction (and to a lesser extent the under-wing load) determine the bending-moment at the (assumed) critical region of the wing i.e., wing stations 145 to 173 ins., further investigation seemed desirable.

In attempting to analyse the over-wing load it became necessary to also focus on the nose pivot load (the only other reaction) which was being measured but whose magnitude was known to be almost inconsequential. Closer examination of the collected data revealed that the recorded load was inclined to be rather erratic for any particular set of data and showed a steady drift to lower values (of compressive load) over consecutive sets of data.

After elimination of the latter problem by replacement of the amplifier used in the weighing circuit and re-establishment of the true zero load (by removal of the attachment pin) an investigation was undertaken to attempt to clarify the true value of the nose reaction over the range of applied loads. Since it was apparent that the dynamometer was being subjected to considerable bending moment as well as direct load (as a result of its installation geometry), the investigation initially was concentrated on the effects of this.

The original dynamometer was replaced by another which was considered less sensitive to the effects of bending and a series of ad hoc tests begun in which the location of the strain gauges on the dynamometer was systematically oriented to minimize and maximize the bending effect. No significant bending sensitivity was identified.

From these tests a pattern of load versus flight g level emerged which showed two separate curved branches - one above 1g flight and the other below, with a very distinct discontinuity at 1g which was determined to depend on whether the 1g load was selected during "landing" or "taking-off." This finding, in conjunction with consideration of the nature of the loading programme confirmed that the two branches of the nose reaction curve were the result of only being able to select loads above 1g flight from an increasing load and those below 1g from a decreasing load i.e., from the upper and lower limbs respectively of a complete hysteresis loop.

2.4 An important subsidiary discovery made during the ad hoc tests described in section 2.3 was the existence of a programming error in the treatment of the "zero" load for the over-wing load cells. This became apparent when comparing manually-reduced loads to those produced by computer processing of the raw data.

It had been appreciated for some time, by visual observation, that the line of action of the over-wing reaction moved from a forward-of-vertical position at low flight loads to an aft-of-vertical position at the larger loads but attempts to achieve a balance of loads and moments had only achieved a limited degree of success due to the various discrepancies which have been discussed above. It now seemed likely that such a balance would be possible and with this in mind a larger than normal amount of load data was recorded in the next data set before the removal of the wings for the 50,000 hr. inspection. This set has been identified as Flight 13863→.

The analysis which is now described applies specifically to this set of data but, with minor alterations, is believed to be representative of all sets. The principal object of the analysis is to determine the variation with flight g level of the moment arm about the nose pivot of the over-wing reactions and from this and the known magnitude and moment arms of applied loads to deduce the over-wing reactions for comparison with the measured values. In addition by summing algebraically all vertical loads the theoretical nose reaction is determined. The effects of lack-of-verticality of applied loads has been ignored since the geometry changes are small and the magnitude of the load is maintained by closed-loop (active) systems.

Figure 1 depicts the geometry of the fuselage datum as it rotates about the nose pivot and the resulting horizontal movement of Point A (the theoretical resultant of the over-wing loads). Deflections of the under-fuselage actuator have been obtained from the linear variable differential transformer (LVDT) which is incorporated in the actuator and from dial gauge measurements of the deflection of the under-fuselage beam system. Figure 2 shows how the equation for the over-wing moment arm versus flight g level is derived from this.

The outer-wing load at zero g is derived from the measured outer wing loads at -0.246 and 0.352g (corrected for the zero error previously referred to) as follows:

$$\text{Load (per side)} = \frac{1}{2}\left[12850+\frac{.246}{.598}(17213-12850)\right] = 7322 \text{ lb.}$$

This is multiplied by 2 and by the moment arm at zero g to obtain the moment about the pivot at zero g:

i.e. $2\times7322\times158.01 = 2,313,898$ lb-ins.

The applied moment per g is derived by multiplying the appropriate moment arms by the measured load increment per g for each of the loading systems. That for the under-fuselage actuator (since the measuring load cell is at one end of the main loading beam) has been factored by dividing by 0.4605, the beam ratio.

Thus, the applied moment per g =

$$6541\times192.7+2\times1298\times97.09-2\times1293\times135.08=1,163,179 \text{ lb-in.}$$

The over-fuselage load is constant for all flight loads.

Figure 3 shows the resulting equation for the calculated load and the plotted curve which results. Superimposed on this are the discrete points measured in data set 13863 →, and the straight line 7300+3613n which is the predicted over-wing load from LTR-ST-1071. It can be seen that the discrepancy from the design load is significant particularly at the higher loads (3.69 percent at 3.84g) and when combined with the "zero" error, previously referred to, would amount to 5.43% in this data set (and up to 6.5% in other sets). Fortunately, the "zero error" portion is a computational error and does <u>not</u> represent a discrepancy of the test load from the design (predicted) load.

The wing bending moment at the critical station (assumed to be w.sta. 145 to 173) is produced by the over-wing load plus the under-wing load (which is considerably smaller) so that the bending moment discrepancy will be less than 3.69 percent, since the under-wing load has no appreciable error.

2.5 Once the over-wing load curve has been established the nose reaction can be calculated by deducting the calculated over-wing load from the sum of the applied load per g plus a constant which is the applied load at zero g plus the tare loads. The resulting equation is:

Nose Reaction = 24844+6551n - 7438-2 $^{O}/$w
= 17406+6551n - 2 $^{O}/$w

derived as follows:

<u>Load At Og</u>

$^{U}/$Fus. = 2572 lb.
$^{U}/$Nac (2) = 1342 lb.
$^{U}/$W (2) = 10360 lb.
Tare (Inc $^{O}/$Fus Beam) = <u>10570 lb.</u>
Total = 24844 lb.

$^{O}/$Fus. = 7438 lb. (Inc. $^{O}/$Fus. Beam)---Constant for all Flight conditions. Net applied load per g = 6541+2(1298)-2(1293) = 6551 lb./g

This curve is plotted in Figure 4 as well as a dashed curve (below 1g flight) which is drawn 250 lb. below the calculated curve to reflect the fact that the measured over-wing hysteresis is 250 lb. at 1g flight which is the only condition at which it can be measured directly, as explained in Sec. 2.3.

The plotted points from data set 13863→ as well as another set measured with the dynamometer strain gauges rotated 90 degrees (gauges on the neutral axis) are shown superimposed on the plotted curves. Considering the scale of this plot and the magnitude of the applied loads the agreement is considered reasonable. The straight line identified as 3062-662n is the nose reaction predicted in LTR-ST-1071, which assumed a linear over-wing reaction versus g level relationship.

3.0 Discussion:

A number of important findings have flowed from this investigation and analysis which are recorded here for the benefit of anyone wishing to examine the matter further and to assist in planning similar tests and in interpreting the outcome of this test upon its completion.

3.1 This section deals with general points that are not specific to the Tracker (or other high-wing aircraft) test.

(a) Desirably, provision should be made for achieving a "free-in-air zero" condition for all dynamometers in addition to a sensitivity calibration, by shunting the bridge. As an alternative, some well-defined loaded condition, which can be accurately repeated, may be used for routine checks provided a true zero load condition is available when required.

(b) Dynamometers should not be used in compression except as a last resort; but, if they are the design of the installation should ensure that the measuring element is not subjected to bending. This, of course, also applies to tension dynamometers but, in general, is more easily achieved.

(c) In planning programmed loading, provision should always be made for reversion to unrestricted, incremental loading, as and when required. This should include the ability to set any load (except the highest and lowest) from any higher or lower load.

3.2 The following comments are specific to the Tracker fatigue test (or tests of similar high-wing configurations).

(a) As a result of the high-wing configuration of the aircraft and the use of the nose landing gear pivot axis as a fixed reaction point, the line of action of the over-wing force undergoes a substantial angular change with change in applied load, as the fuselage rotates about the pivot. In the case of this aircraft, this results in a change in the moment arm of this force about the pivot of some 8.7 inches over the flight spectrum, with the direct result that the over-wing reaction is reduced by some 5.4% as the moment arm increases even though the applied loads (and hence moments) are as specified.

A number of alternative arrangements could be adopted to obviate this problem, such as:

(i) Provision of an active (actuator and load cell) nose loading point

(ii) Provision of a passive nose loading point with fore and aft freedom (e.g. hanging by cable anchored overhead)

(iii) Provision of (b) and in addition a "soft" spring against vertical load to maintain an essentially level fuselage attitude. For example, this could also be accomplished by leaving the nose landing gear and wheels in place, with the tire and oleo pressures suitably adjusted and by provision of a "weighing" platform allowing the wheels to move fore and aft to compensate for any inadvertent horizontal loading while measuring the vertical reaction.

N.B.

For any of (a), (b) and (c) a method of resisting loads in the opposite direction should be provided to facilitate erection and maintenance. This could be simply a cable tie or a slotted bar or rod.

(iv) Provision of the same set-up as in the Tracker test with careful design of the nose attachment so that fore and aft loads are resisted on the centre line of the pivot axis and therefore do not result in bending of the dynamometer. This approach also requires that the deflection of the wing relative to the fuselage be known with reasonable accuracy so that the applied loads can be adjusted to compensate for non-verticality of the over-wing reaction force during loading.

(v) Adoption of the traditional approach (which is not without penalties) of anchoring the fuselage in a fixed attitude and applying all other loads through active systems.

3.3 The second source of difficulty which was revealed in the analysis might be described as more apparent than real although its existence could only be identified by the very careful examination of the data sets required to balance the measured loads and moments.

It is intended (LTR-ST-1071) that the over-wing load cells indicate zero load when carrying the weight of the over-wing beam system. Since there is no method provided to check this precisely or to check the free-in-air zero of the load cell, the reduction of data (by computer) had been programmed to treat the "1g" ground condition as zero and to deduct this reading from the data before conversion of the bridge voltage to pounds (force). Fortunately, the zero of the system appears to have remained relatively stable with the indicated "1g" ground load ranging from 200 to 600 lb. (16 to 48 millivolts) per side over the recorded sets of data. The "correct" 1g ground load should be (LTR-ST-1071 - Table X) 1864-996-499 = 369 lb. which is equivalent to 29.5 millivolts. Thus, in general, a better approximation to the "true" load would be to either ignore the 1g ground reading completely (since it is often close to 29.5 millivolts) or alternatively to revise the data-reduction programme, to correct only for departures from 29.5 millivolts at the 1g ground condition. The second approach has more technical merit and has been adopted. For practical reasons, it has been accomplished by subtracting 60 from the combined 1g (ground) bridge voltages, dividing by 2 and treating the resultant as a zero correction for each of the over-wing loads.

3.4 The third source of confusion in analysis of the tracker data has to do with our inability to command load levels to permit incremental loading for data recording purposes i.e., applied loads, reactions and strain gauge and deflection measurements. In practice, these are read by halting the test at the end of a cycle. This means that flight loads less than 1g are always approached from a higher load and those above 1g are always approached from a lower load and that 1g is the only flight condition which can be set from a higher or a lower load. In the case of the former it is usually only a very slightly higher load simply because of the very large numbers of these. A slightly more subtle source of confusion is that due to the pairing of loads in the programme, some loads are always or almost always preceded by the same load which may be close to it.

For example, 0.97g always follows 1.303g and because of the high relative frequency of this load pair, 1.303g almost always follows 0.979.

In effect, this means that loads above 1g flight are read from the upper limb of the hysteresis loop and loads below 1g from the lower limb, with 1g flight the only one capable of directly showing the magnitude of the hysteresis. In the data recorded as FL 13863, for example, both 1g "taking-off" and 1g "landing" were recorded and while the recorded applied loads are identical except for a difference of 11 lb., in the under-west-wing load, there are substantial differences in the measured over-wing and nose dynamometer loads and in the over-fuselage and under-fuselage deflections and, as would be expected, in the strain gauge readings. These can be used to provide some "scale" to the width of the hysteresis loop which appears to be approximately equivalent to 0.1g.

3.5 Early in the investigation, plotting of the recorded nose dynamometer load had revealed a steady decline from a mid-range (1.5g Flight) value of approximately 2400 lb., to 600 lb., with roughly the same shape prevailing. The exact cause of this is not known but is presumed to have resulted from drift of the amplifier over many months (from FL 2731 to FL 13075). The zero of the system was not checked during this period. The dynamometer and amplifier were changed during the ad hoc investigation carried out in February/March 1981 and a technique developed for checking the true zero of the load.

4.0 Conclusions:

4.1 The Tracker Fatigue Test loading system has performed in a very satisfactory manner over 50,000 simulated flight hours of testing.

4.2 The investigation and analysis reported herein has examined a number of anomalous findings which were identified during the course of the test and the results are included.

4.3 The report also presents some comments, for future reference, regarding the Tracker test and full scale fatigue tests, in general, with particular reference to tests of shoulder-wing (or high-wing) aircraft.

REFERENCE: National Research Council of Canada (N.A.E.)
LTR-ST-1071 - Tracker Fatigue Test Analysis of Loading Systems - J. F. deWaal, March, 1979.

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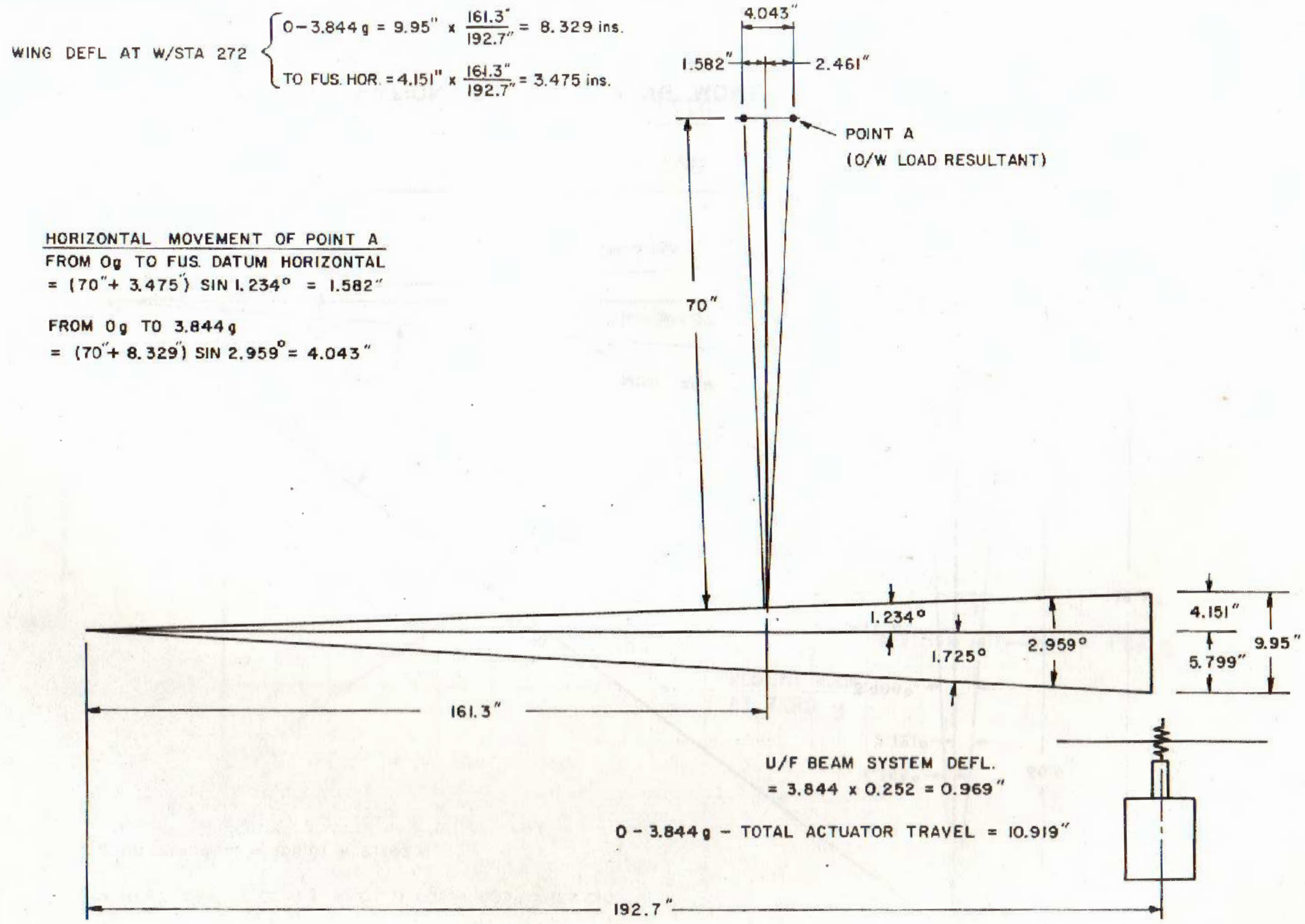


WING DEFL AT W/STA 272

$0-3.844\,g = 9.95'' \times \frac{161.3''}{192.7''} = 8.329$ ins.

TO FUS. HOR. $= 4.151'' \times \frac{161.3''}{192.7''} = 3.475$ ins.

HORIZONTAL MOVEMENT OF POINT A

FROM 0g TO FUS. DATUM HORIZONTAL

$= (70'' + 3.475'')$ SIN $1.234° = 1.582''$

FROM 0g TO 3.844g

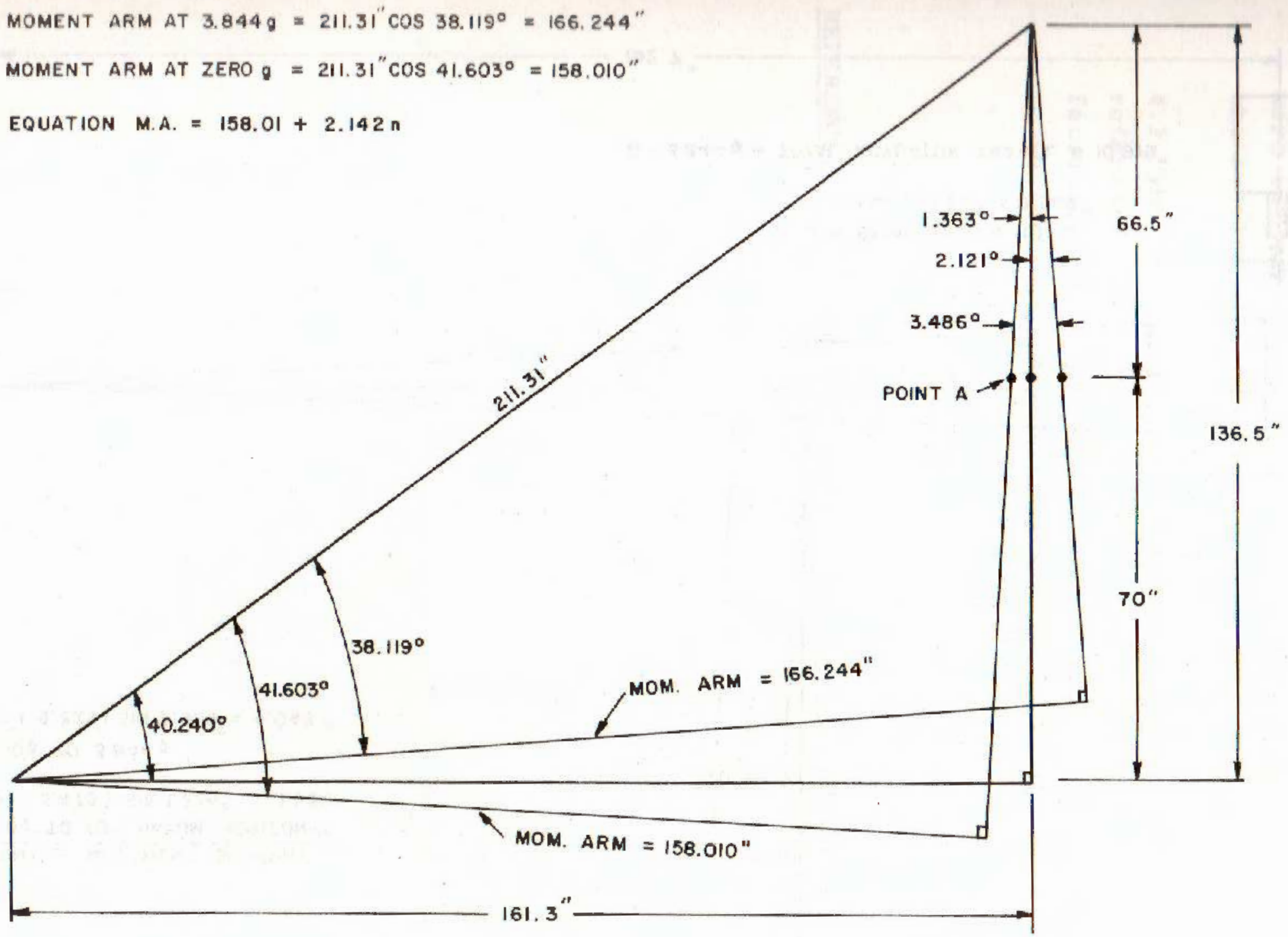
$= (70'' + 8.329'')$ SIN $2.959° = 4.043''$

U/F BEAM SYSTEM DEFL.

$= 3.844 \times 0.252 = 0.969''$

0 - 3.844g - TOTAL ACTUATOR TRAVEL = 10.919"

FIG. I GEOMETRY OF POINT "A"



MOMENT ARM AT 3.844 g = 211.31" COS 38.119° = 166.244"

MOMENT ARM AT ZERO g = 211.31" COS 41.603° = 158.010"

EQUATION M.A. = 158.01 + 2.142 n

FIG. 2 DETERMINATION OF OVER-WING MOMENT ARM

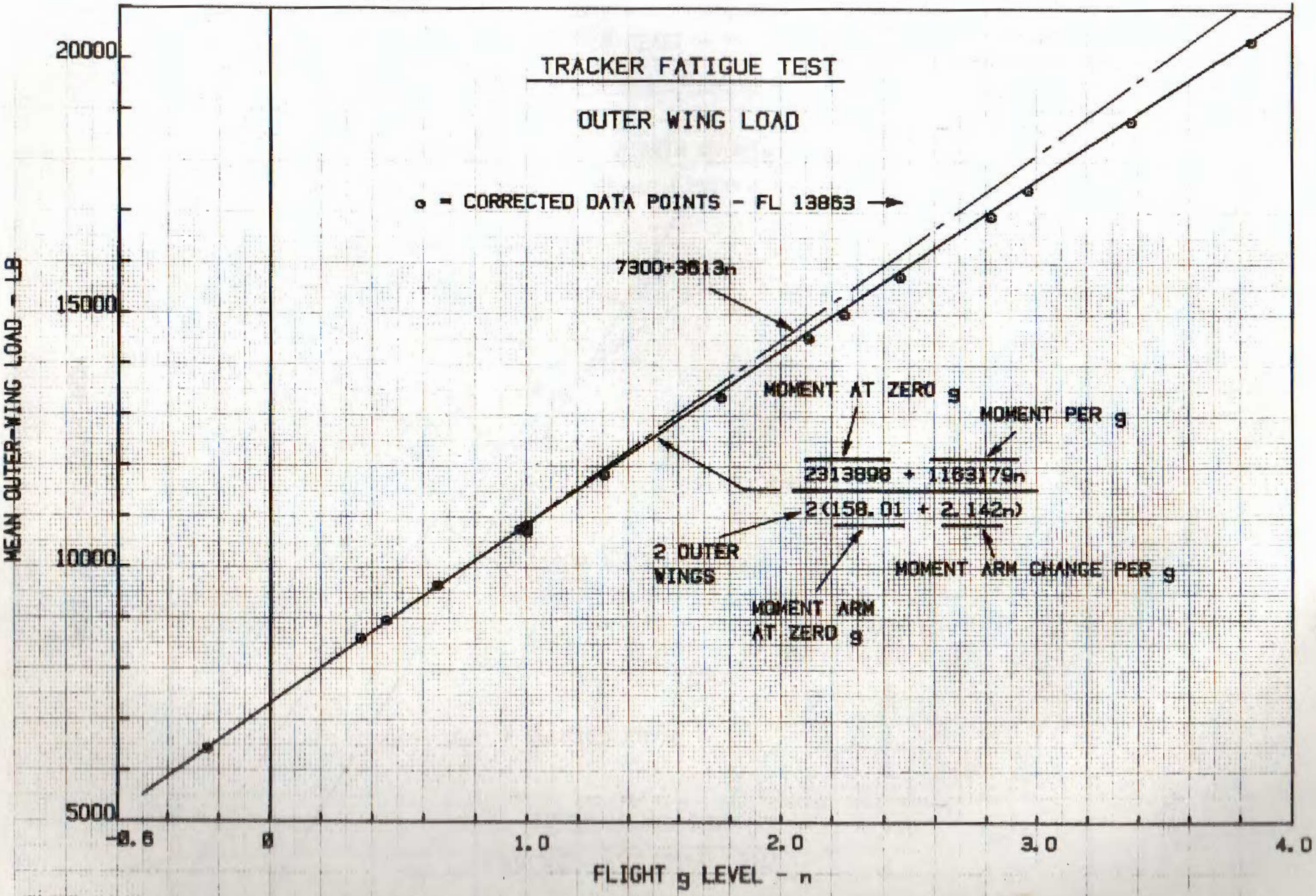

TRACKER FATIGUE TEST
OUTER WING LOAD
○ = CORRECTED DATA POINTS - FL 13863 →
7300+3613n
MOMENT AT ZERO g
MOMENT PER g
2313898 + 1163179n
2(158.01 + 2.142n)
2 OUTER WINGS
MOMENT ARM CHANGE PER g
MOMENT ARM AT ZERO g
MEAN OUTER-WING LOAD - LB
20000
15000
10000
5000
-0.6
0
1.0
2.0
3.0
4.0
FLIGHT g LEVEL - n

FIG. 3

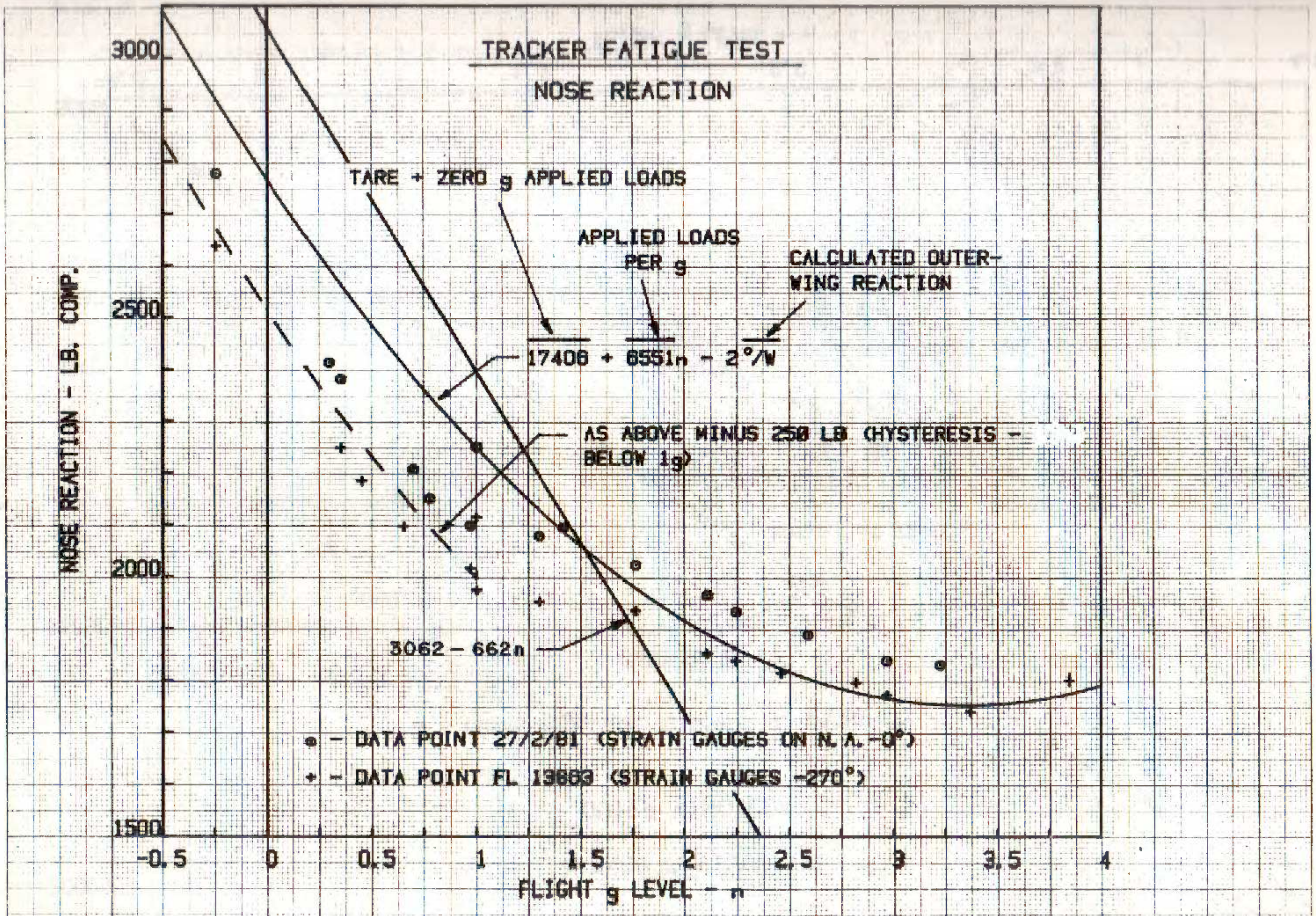

TRACKER FATIGUE TEST
NOSE REACTION
TARE + ZERO g APPLIED LOADS
APPLIED LOADS PER g
CALCULATED OUTER-WING REACTION
17406 + 6551n - 2°/W
AS ABOVE MINUS 250 LB (HYSTERESIS -
BELOW 1g)
3062 - 662n
● - DATA POINT 27/2/81 (STRAIN GAUGES ON N.A. -0°)
+ - DATA POINT FL 13663 (STRAIN GAUGES -270°)
NOSE REACTION - LB. COMP.
3000
2500
2000
1500
-0.5
0
0.5
1
1.5
2
2.5
3
3.5
4
FLIGHT g LEVEL - n

FIG. 4