72/LOADS/19

DOCUMENTS TABLED AT THE
MEETING ON 8TH MAY 1958
ON THE SUBJECT OF STRUCTURAL INTEGRITY
OF THE ARROW

May 1958

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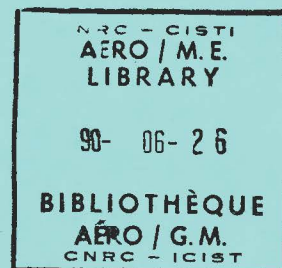
ON THE SUBJECT OF STRUCTURAL INTEGRITY OF THE ARROW

S E C R E T

72/LOADS/19

Avro Aircraft Limited

INTER-DEPARTMENTAL MEMORANDUM

Date   6th, May 1958   Reference Number: 8792/02E/J
  To   Mr. C. V. Lindow
From   S. Kwiatkowski
Subject MANOEUVRE LIMITING DEVICES IN THE ARROW DAMPER

The following devices limiting the pilot or automatic commands and/or protecting against damper failures resulting in manoeuvres causing large structural loads will be in operation with fully developed Arrow Damper:

1. Command limiter - pitch

2. "G" limiter (pitch)

3. Command limiter - roll

4. Roll rate limiter

5. Aileron limiter (transverse acceleration)

6. Transverse acceleration monitor

7. Rudder hinge moment limiter

8. Mode transfer switch (sideslip)

In this note operation and the basic principle of each of these devices is briefly discussed and types of failure against which protection is provided are indicated. Some of the above devices are not yet fully developed since additional information from flight test is required and therefore a large portion of numerical data necessary for full description of their operation is not available. Particularly full information is not yet available on amount of protection offered in areas of the flight enve-lope where combined effects of aerodynamic non-linearities, high effective airspeeds, aeroelasticity, aerodynamic cross-coupling, reduction in control effectiveness, control hinge moment limitations etc. may tend to produce critical loads in certain manoeuvres. However all these effects were con-sidered in the design of the limiters and satisfactory protection will be achieved when all the necessary information is available. All the limiters are designed to operate in the normal damper mode only with exception of the rudder hinge moment limiter which is available in both normal and emergency control modes.

1. Command Limiter - pitch

   Limits pilot's or any of the automatic commands to a fixed value
   of normal acceleration at approx. 4 to 5 positive "g's" (value
   not yet selected) and to negative one "g". Both will be constants
   throughout the flight envelope substantially below the structural
   limit and held to a close tolerance. Possible overshoots will be
   small because the damper will control them to approx. 10% in the
   worst case. The command limit is achieved by limiting the voltage
   output of the stick force transducer.

   Provides protection against any failure forward of the stick re-
   sulting in hard-over command e.g. in the autopilot, fire control
   etc.

   The command limiter can be overpowered intentionally by the pilot
   by applying stick force of approx. 75 - 90 lb which is substantially
   higher than stick force for maximum command (approx. 25 - 30 lb).

2. "G" Limiter

   The "g" limiter protects the aircraft against malfunctions of the
   damper such as:
                        runaway parallel servo
                        runaway differential servo
                        loss of pitch rate

   Partial protection is also offered for a simultaneous ramp inputs
   of both parallel and differential servos.

   The limits of protection are discussed in M-H Document R-ED-9240 MH
   64 G Limiter status dated January 17th, 1958 -
   The limit function is mechanised as follows:

$$\left(n_{z_{CG}} + \frac{162}{g} \dot{q} \right) \cdot \frac{1 + 02S}{1 + .1S} + 15.75 \ \delta_p \ \frac{.5s}{1 + .5s}$$

$$+ \ 12.5 \ \ \delta_D \ \frac{2S}{1 + 2S}$$

where $n_{z_{CG}}$ - normal acceleration at c.g.

              $\dot{q}$ - pitching acceleration

              δ_P - parallel servo deflection

              δ_D - differential servo deflection

The first two items are obtained by combining outputs of two normal
accelerometers suitably spaced along the longitudinal axis. The
remaining two inputs are virtually differentiated by electrical

2.   "G" Limiter (Continued)

networks such that the limit function contains in addition to
accelerations the directions and rates of servo motions.
Should the total exceed a predetermined value the pitch
axis will be disengaged by effectively disconnecting the
hydraulics in the servos. It can be seen from the limit
function that the actual value of "g" at which the disengage-
ment occurs will vary with the flight conditions, rates of
application of command and type of failure. However, the
resulting overshoot will be kept inside the structural limits
of the aircraft, since this was the basic aim in the design
of the limiter. On the other hand a so called "nuisance
disengagement" may occur in a manoeuvre inside the flight
envelope at a few specific flight conditions. Obviously the
number of "nuisance disengagement" is kept to the minimum but
it is not feasible to eliminate them completely in order not
to jeopardize the protection at other conditions.

If the "g" limiter action was caused by intentional pull on
the stick, at the disengagement of the pitch axis of the
damper there will practically be no change in the stick force,
therefore, higher load factors will not be pulled inadvertently.
If the limiter disengages with no force on the stick (e.g. in
the automatic mode) the aircraft will immediately be returned
to within $\pm \frac{1}{2}$ g of level flight by the centering action of the
servos.

3.   Command Limiter - roll

Limits pilot's or any of the automatic commands to 120°/sec.
of roll rate or the maximum roll rate available aerodynamically
if the latter is less than 120°/sec.

The command limit is achieved by limiting the voltage output
of the stick force transducer similar to pitch command
limiter.

Can be intentionally over-powered by applying approx. 40-50 lb.
at the stick. Protects against any malfunction forward of
the stick resulting in hardover signal.

4. **Roll Rate Limiter**

The action of the roll rate limiter is based on a wing tip accelerometer set to disengage the roll axis at approx. 160°/sec. of roll rate. Therefore, it protects against any malfunction of the damper resulting in a hardover parallel or differential servo signal.

At the instant of disengagement there generally will be a change in stick force, but in the majority of cases the emergency mode stick force will be higher than in normal mode.

5. **Aileron limiter (transverse acceleration)**

This limiter protects the fin in rolling manoeuvres resulting in high fin loads. The action of this limiter is based on a transverse accelerometer located 30 ft. forward from aircraft c.g.

In the normal mode the ratio of aileron stick force to roll rate is a constant throughout the flight envelope approx. 20 lb. of stick force per 120°/sec. of roll rate. This ratio is monitored by the aileron limiter in such a manner that whenever a transverse acceleration of .6 "g" is reached the stick force transducer output becomes zero e.g. no roll rate can be commanded. The rate of increase of aileron stick force per unit of roll rate is linear with transverse acceleration, with a shallow slope applying up to .2g and steeper slope up to .6g, the latter corresponding approx. to 50% of fin limit load. This limiter is independent of other damper actions and will limit the aileron output no matter what has caused the increase in transverse acceleration.

The action of this limiter is particularly important in prolonged rolling manoeuvres e.g. in excess of 180° of bank angle or at high roll rates in rolling pull-outs where cross -coupling effects are particularly significant and generally in any rolling manoeuvre causing saturation of the rudder servo to deflection or hinge moment limit.

6. **Rudder Monitor**

Rudder monitor provides protection against any malfunction of yaw axis, resulting in hardover rudder signal automatically in level flight. The normal damper is disengaged and emergency damper engaged when transverse acceleration reaches .4g as measured 40 feet ahead of c.g.

6. **Rudder Monitor (Continued)**

The limit function of the accelerometer is as follows

$$\ddot{y}_{40} = \ddot{y}_{C.G.} - 40\,(pq + \dot{r})$$

It can be seen that in manoeuvering some anticipation is obtained due to yaw acceleration and roll rate and pitch rate product. The actual fin load at disengagement will depend on flight condition and type of manoeuvre. The limiter was designed to disengage at approx. 50% of fin load but in a few conditions at high E.A.S. (above 650 knots) this is exceeded and switching does not occur until 80% of fin load is reached. The limiter is designed to cater for max. rudder rate of 50°/sec. This device is not fully developed yet and some uncertainty exists about the amount of protection provided when failures occur during extreme manoeuvres, e.g. a rolling pull-out.

It was not feasible to design a relatively simple emergency damper which would provide coverage of all possible manoeuvres involving sideslip. Therefore if due to malfunction of the normal damper, the rudder monitor engages the emergency damper during extreme manoeuvre fin limit loads may be exceeded unless a prompt pilot's action will minimise occuring sideslip. These conditions occur mostly at high E.A.S. and in extreme manoeuvres. At the present time not enough information is available to define in detail extent of the protection provided for manoeuvres involving high roll rates and normal accelerations. These new problems involving cross-coupling effects are common to most aeroplanes operating in the flight envelopes similar to that of the Arrow and further development time will be needed to obtain sufficient information permitting redesign or modifications of the rudder monitor to obtain a maximum possible protection.

7. **Hinge moment limiter**

A hinge moment limiter combined with pilot's trim and feel unit in the rudder performs the following functions:

1. Provides variable feel with speed and altitude.

2. Limits the pilot's input into the rudder to values such that 150 lb of pedal force will not exceed the fin limit load, with dampers off.

3. Ascertains sufficient servo authority independent of the pilot particularly in areas where total rudder deflection is heavily restricted by the available hinge moment.

The hinge moment limiter consists of variable linkage, and springs and trim motors, common to any artificial feel and trim unit. The variable linkage is driven by an electric actuator receiving signals from the dynamic pressure sensor.

In normal damper gear-up mode the use of rudder bar is necessary only to trim out an engine out condition or any other aerodynamic assymetry, otherwise the aeroplane is basically a two-control aircraft. In gear-down mode rudder is used in a conventional manner for landing and take-off. It may be used as an alternative way of correcting minor damper malfunctions. In the emergency mode of control rudder may be used to help co-ordinate manoeuvres which are not adequately co-ordinated by the damper.

The failure of hinge moment limiter may cause inadvertent too large pilot's corrections resulting in high fin loads at high speeds.

8.   Mode Transfer Switch (sideslip)

This switch operated by the relative wind sensor switches over normal damper into emergency mode whenever a sideslip angle of 10° is reached. This is applicable only to low speed range e.g. where the 10° of sideslip produce less than 50% of the fin limit load.

Protects against hardover type of malfunctions in the low speed region.

CONCLUSION

It can be seen from the above description that with normal damper in flightworthy condition the aircraft manoeuvres are positively limited to:

1.   Straight pull-ups to 4 - 5 positive "g's" or approx. 80% of limit load depending on aircraft weight and c.g. position. In the operational range e.g. above 40000 ft. at any speed the maximum positive "g's" are limited by elevator deflection or hinge moment.

2.   Straight push-downs to one absolute "g" negative which is well inside the structural envelope.

3.   Maximum roll rate to 120°/sec. or approx. 35% of limit load.

4.   Rolling pull-out to roll rates not exceeding 120°/sec. or approx. 50% of fin limit load, whichever occurs first.

5.   Fin load to approx. 50% of limit load in any manoeuvre.
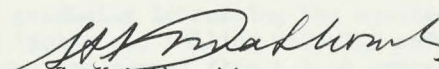
UNCLASSIFIED

CONCLUSION (Continued)

6. "Nuisance" disengagements of any axis will produce loads generally smaller than quoted above.

7. With the hinge moment limiter operating application of 150 lb force to the rudder bar will produce only small fin loads.

8. Overshoots in pitch and roll axes for abrupt applications of controls will be much smaller than in unaugmented airframe. Representative magnitude of overshoots in both axes is 10%.

The above loads can be exceeded only by intentional overpowering of the parallel servos requiring very large control forces or due to malfunctions of the damper system. The emergency mode of control will be limited (by pilot's instructions) to manoeuvres and speeds fully covered by emergency damper.

Failures of the normal damper with limiting devices operating will result in loads inside the structural flight envelope with exception of failures occuring during extreme manoeuvres.

The policy of structural integrity testing should be reviewed bearing in mind the presence of limiters described above. Furthermore it has not been established yet that loads in excess of normal command limits can be tested in a manner offering a reasonable amount of safety from controllability point of view.

SK/g

S. Kwiatkowski
Chief of Stability and Control

cc Messrs R.N. Lindley
          J.A. Chamberlin
             F. Brame
             G. Watts
             R. Carley

AVRO AIRCRAFT LIMITED

STRUCTURAL INTEGRITY OF THE ARROW AIRCRAFT

A NOTE ON THE PROPOSED FLIGHT PROGRAM

1. Introduction

This note discusses the purpose and intent of the flight program for demonstration of the structural integrity of the Arrow Aircraft.

It is proposed that the program be carried out in two distinct parts, the first using an Arrow I, either 25201, 2 or 3 and the second using an Arrow II which is at present No. 25215.

2. Purpose

The purpose of this program is to confirm, by strain gauge measurements, the accuracy of airloads calculations. Flight strain measurements will be compared with calculations for similar manoeuvres. The validity of these calculations will be demonstrated by use of a static ground test for particular design cases.

Associated with this program will be a demonstration of the airworthiness of the aircraft to the satisfaction of the company and the R.C.A.F.

The proposed program is planned with a view to the requirements of MIL-S-5711 being a considerable increase on the simple demonstration but somewhat less than the full flight loads survey.

3. Program

3.1 Part I

This part may be carried out on any of the first three Arrow I air-craft. The object will be to cover that part of the design flight envelope required for the Phase I program. It is expected that this will approach 80% of the design flight envelope.

The program will be carried out by gradually increasing the equiva-lent airspeed and allowable normal load factor and rolling velocity and monitoring the results continuously during the normal flight test program. This will be implemented with the use of about 57 strain gauged positions located at critical points in the structure together with other instrument-ation to indicate local accelerations, pressures and control surface hinge moments. The purpose of this instrumentation is to monitor typical flight cases in order to assess stress levels at these singular position. No loads analysis is possible.

Before an increase in the envelope is attempted typical flight cases will be analysed on the simulator and these will be processed for flight loads and stresses using the available aerodynamic and stressing matrixes.

3.1 Part I (Cont'd)

Towards the end of this part of the program, the limiting devices on the damping system will be investigated. However, these will be set at falsely low values in order that the factors may be kept below those already tested in this part of the program. Examination of the resulting manoeuvres will give checks on the accuracy of simulation and allow for the more stringent tests of the Part II program.

3.2 Part II

Aircraft 25215 has been allocated solely for this program. It is intended that 400 strain gauged positions will be available together with other instrumentation as in Part I.

The extent of this instrumentation is such that comparison with design cases is more complete than in Part I testing. However, it is proposed that only 200 of these quantities will normally be used for conformation of airloads and the full quantity will only be available when unexpected results appear.

The object of this part of the program is to increase the flight envelope to the design limits and to demonstrate the flight worthiness of the aircraft during typical manoeuvres which may be expected to achieve these design limits. The basic program will consist of carrying out normal manoeuvres (pull ups, turns, rolling pull outs etc.) to the design limits, or to such limited manoeuvres as are possible for safe operation of the aircraft. In order to complete these tests some parts of the damping system will be inoperative, e.g. limiting devices and pitch and roll damping. It must be mentioned that owing to the low design weight certain parts of the aircraft which depend on n rather than nW cannot be demonstrated to the full factors.

Following this, the damper system limiting devices will be demonstrated at their design settings and damper system failure cases will be examined. Typical of this latter test are hard over control signals during various manoeuvres. The latter part of the program will be preceded by simulator and loads analysis in order to approach the limiting cases in a safe manner. The aircraft will be demonstrated using the normal damping system. Of course, the emergency damping system will automatically come into use during the tests on limiting devices and failure cases.

4. Conclusion

It must be emphasised that this program constitutes a considerable task. In order to conserve flying time and to achieve success in the limited time available very close liaison between Flight Test Engineering, Technical Design Department, and the Stress Office is necessary.

4. Conclusion   (Cont'd)

       It is suggested that the detailed program method of data handling and associated manpower requirements be examined and prepared as soon as possible.

ORIGINAL SIGNED BY
J. D. HODGE

JDH/mvs

J.D. Hodge
Technical Co‑ordinator
Arrow I and II

AIRLOADS - AVRO ARROW

The structural flight envelope is shown in figure 1. As can be seen, the lowest altitude at which the full range at supersonic Mach is met is 30,000 ft. At this altitude airload calculations were carried out for the full speed range ie.

M=2.0 , 1.6 , 1.2, 1.0 and .56

The symmetric flight manoeuvre envelope at this altitude is shown in figure 2. The full off in design limit load factor with increasing temperature should be noted.

The subsonic and low supersonic loads were investigated under high air density conditions at sea level. The following speeds were considered:

M=1.09 , .80 , .60 , .453 and .272

Figure 3 shows the sea level manoeuvre envelope. It should be noted that all load factors quoted are based on an A.U.W. of 47,000 lb. and for higher weights " n W " is kept constant. That is the design load factor is lower.

At each altitude and Mach Number mentioned above airloads were calculated for some of the following manoeuvres:

(a) Steady pull-up to limit load factor -
Because of the special aircraft configuration this produced the largest in flight wing bending and torque conditions.
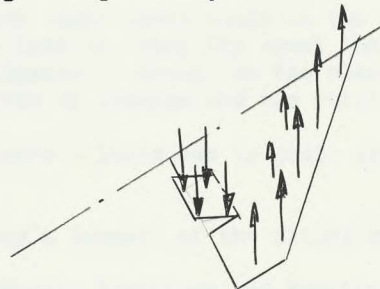
In achieving limit load factor the wing had to produce additional lift to overcome the large elevator balancing down load. In this manoeuvre pitching acceleration is zero.

(b) Steady push-down to negative limit load factor-
Not critical for design because at low load level.

(c) Checked pull-up to limit load factor-
Similar to the balanced pull-up except that the elevator load is dropped to the neutral level and the unbalanced aircraft allowed to pitch nose-down. Without the down elevator load the wing lift is somewhat smaller to reach the same normal load factor. This case produces large up bending in the rear of the wing structure. It also designs the fuselage rearend because the inertia effects due to pitching acceleration and normal acceleration add.

(d) Checked push-down - Not critical for design.

(e) Rolling pull-out - Consists of rolling the aircraft at the maximum allowable rate while pulling 2/3 of the maximum load factor. On old fashioned airplanes this manoeuvre was used to design the vertical tail. However, on aircraft with low aspect ratio wings and long high density fuselages the fin side load so induced is prohibitively large.

The problem is met by use of automatic stability augmentation which precludes the possibility of attaining large fin loads.

The rolling pull-out is still of importance due to wing loads. The combination of loads due to 2/3 the normal load factor with those due to aileron and damping in roll produce design torque shear and B.M. for the outer wing.

(f) Rudder damper failure - The use of automatic artificial stability solves one problem but brings another in its wake. Depending on electronic and other devices rather than on unfailing air flow about air foils raises the possibility of a failure in the system. The worst possible thing which could occur would be a runaway of the control surface to either hinge moment max. or to the control stop. To guard against this possibility, safety devices are built into the system which switch the runaway system off and a duplicated emergency system on, which then brings the aircraft back under control. However, signifigant loads may occur during the manoeuvre.

The most signifigant loads on the aircraft occur during a rudder runaway due to lack of stability about the yaw axis. These cases have been investigated in detail on the electronic analogue (flight simulator) in 5 degrees of freedom and the worst loads used for fin strength checking.

(g) Gust loads - loads due to gusts are much smaller than manoeuvre loads.

Table 1 gives a summary of the flight conditions for which loads were issued.

<u>Centre of Gravity Positions and Arbitrary Increments:</u>

For all cases but Yaw Damper Failure, loads have been found for three c.g. positions
.28 , .30 , and .32 $\overline{c}$

Following the practice of AP 970 , arbitrary increments of pitching moment and aerodynamic centre shift have been included in each case.

Max. nose up combination ΔC_{m_o} = .0075 Δa.c. = -.025$\overline{c}$

Max. nose down combination ΔC_{m_o} = -.0075 Δa.c. = .025$\overline{c}$

A consequence of taking the two arbitrary increments acting together has been to make elevator loads to balance the aircraft larger than those available from the jacks. However, to retain the conservatism (and prepare for unforeseen problems) elevator loads were shifted forward arbitrarily to reduce the hinge moments to available limits while still balancing the aircraft.

Methods of Airloads Analysis:

From the outset an attempt has been made to keep the determination of overall aircraft airloads on an analytic basis. The powerful linearized supersonic flow techniques of J.C. Evvard have been extended by F. Woodward and Prof. B. Etkin to the solution of loads over wings of arbitrary platform and camber. These methods have been used to compute the rigid wing and control surface loads for the three supersonic Mach Numbers.

An interesting extension to account for the effects of elasticity has been made by F. Woodward. In this technique the wing is assumed to be composed of a number at control surface shaped panels free to take up any incidence to the wind. The deflection of any one panel causes a load on itself and loads on other panels in its Mach cone. These loads form a column of an aerodynamic matrix. Written in algebraic form the loads on the wing for a certain incidence distribution would be

$$L_1 = q_{11}\alpha_1 + q_{12}\alpha_2 + q_{13}\alpha_3 \; - - - -$$
$$L_2 = q_{21}\alpha_1 + q_{22}\alpha_2 + q_{23}q_3 \; - - - -$$

or in matrix notation

$$|L| = [A]|\alpha|$$

where $|\alpha_0|$ is the rigid camber and incidence distribution of the wing, panel rigid loads are

$$|L| = [A]|\alpha_1|$$

to find elastic loads use is made of the structural slope matrix

$$|\Delta\alpha| = [s]|L|$$

Elastic loads are

$$|L| = [A]||\alpha_0| + |\Delta\alpha|| = [A]|\alpha_0| + [A][s]|L|$$
$$|L| - [A][s]|L| = [A]|\alpha_0|$$
$$|L| = [I - [A][s]]^{-1}[A]|\alpha_0|$$

The implementation of this simple method of course requires the use of computing facilities capable of rapidly inverting matrices of the order of 50 x 50.

From unit elastic loads found for angle of attack, control surface deflection, roll rate etc. complete elastic cases have been synthesized.

The elastic loads study showed the effects of elasticity to be insignifigant up to all but the highest speed. At M=2.0 at 30,000 ft. the symmetric pull-up loads were of the order of 10% more severe than the rigid loads. Investigation of the case of this revealed that if it had not been for the arbitrarily high elevator hinge moment the elastic load would have been signifigantly below the rigid one. Since it was deemed not reasonable to double penalize the struture for the high elevator load, rigid loads were used throughout for symmetric manoeuvres.

For the rolling pull-out case at M=2.0 Alt. = 30,000 ft. full effects of elasticity have been considered in loads evaluation and used for wing stressing.

Subsonic airloads have been calculated using the methods of Lawrence, Multhopp and Faulkner. Comparison with wind tunnel tests on the Arrow for aerodynamic centre position and lift curve slope showed the method of Faulkner to be most representative and it was used for all design cases. Elevator load distributions were obtained from work done by N.A.C.A. on an almost identical wing. The distributions were altered slightly to agree with pitching moment, hinge moment and lift due to elevator deflection obtained from wind tunnel tests on the Arrow.

Elastic effects were neglegible for the subsonic cases.

STRUCTURAL FLIGHT ENVELOPE — AVRO ARROW
A.U.W. = 47,000 LB.

FIG

FIG 2

C105 FLIGHT ENVELOPE 47000 LBS
30,000 FT

MACH NO. = .002777 VEAS.

LOAD FACTOR
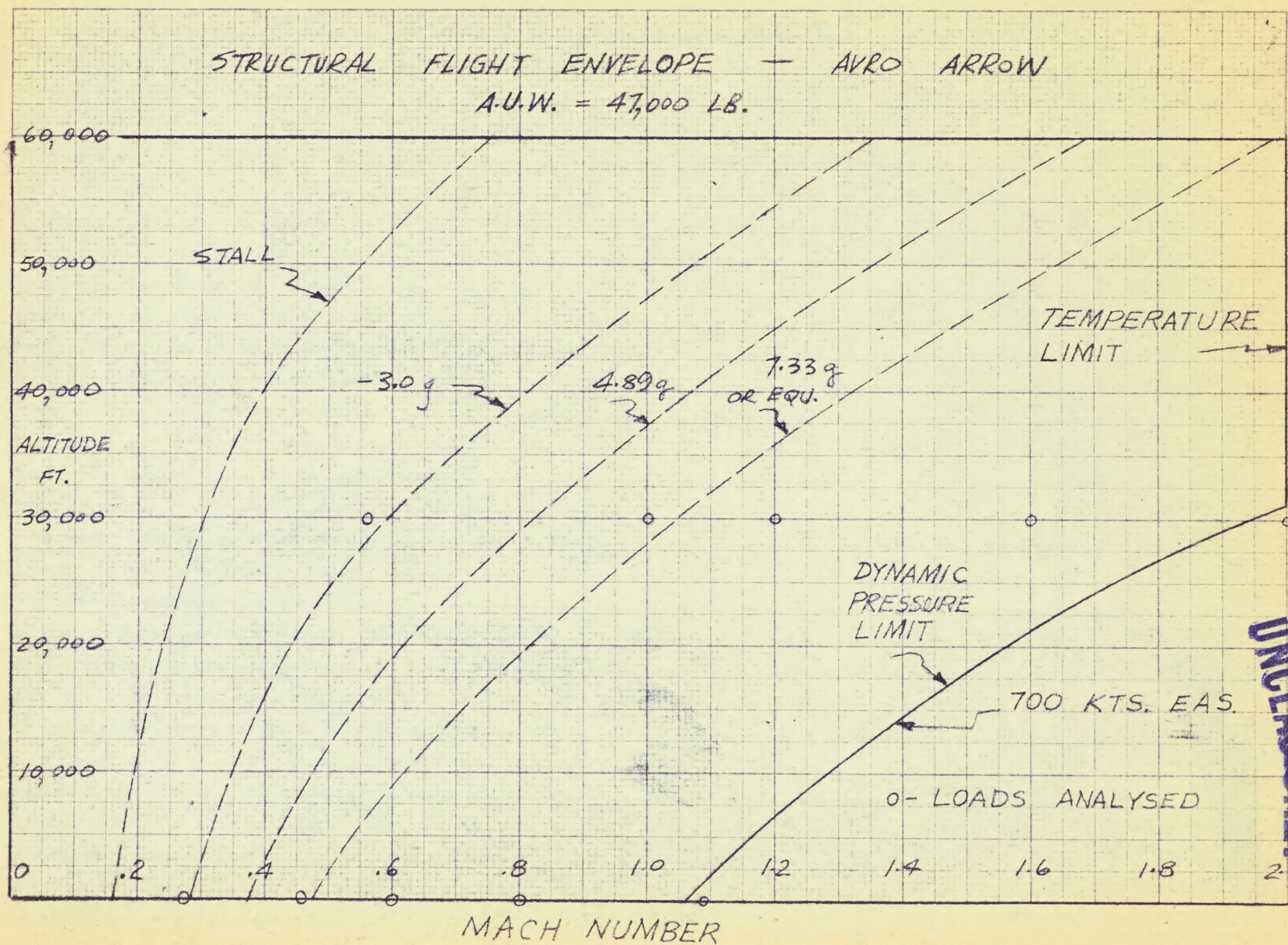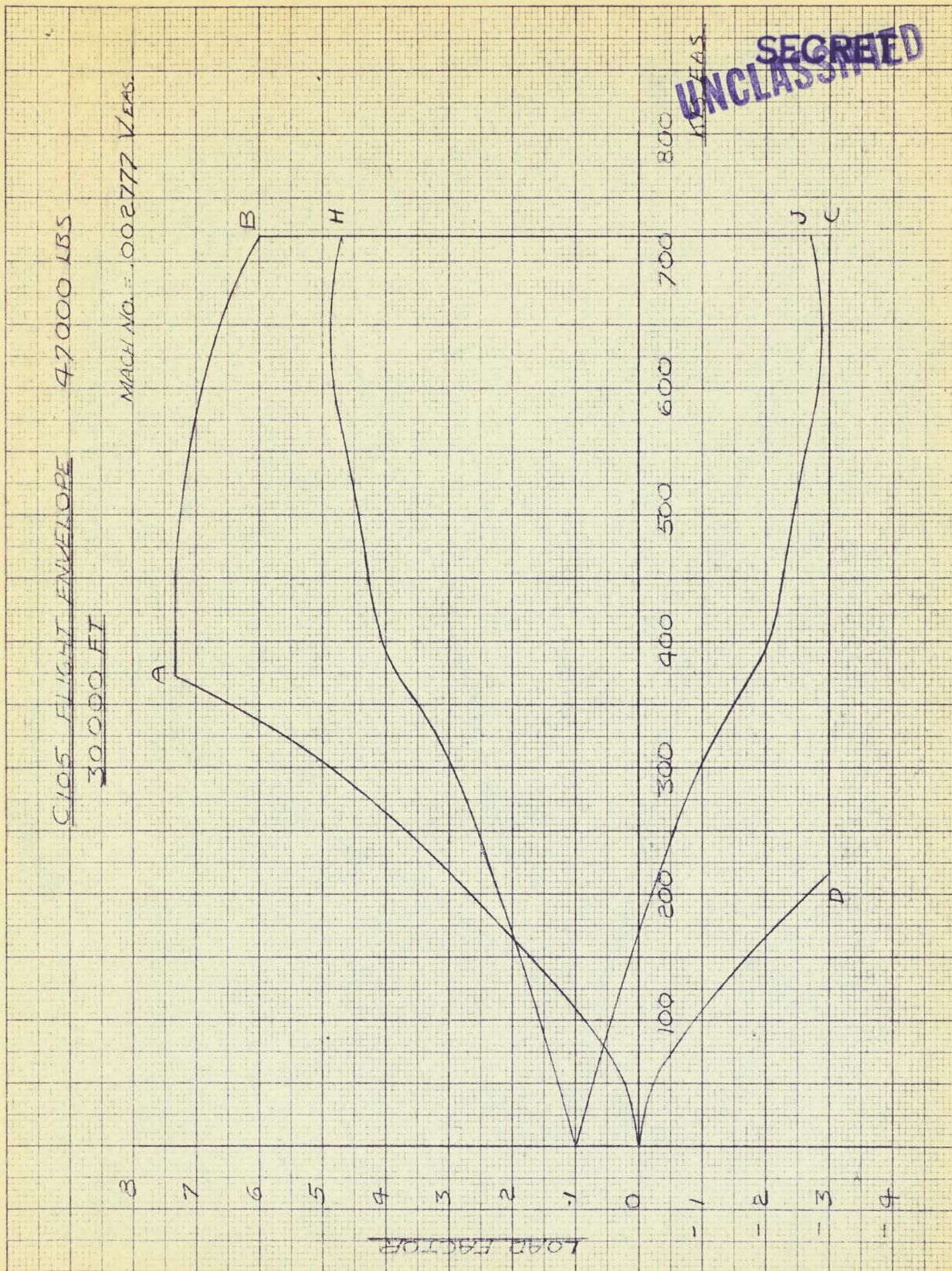
FIG 3

SHEET I P/LOADS/120 APPENDIX I
JAN/57 R. LOKEN

C-105 FLIGHT ENVELOPE 47000 LBS

SEA LEVEL

MACH NO = .001512 VEAS

LOAD FACTOR
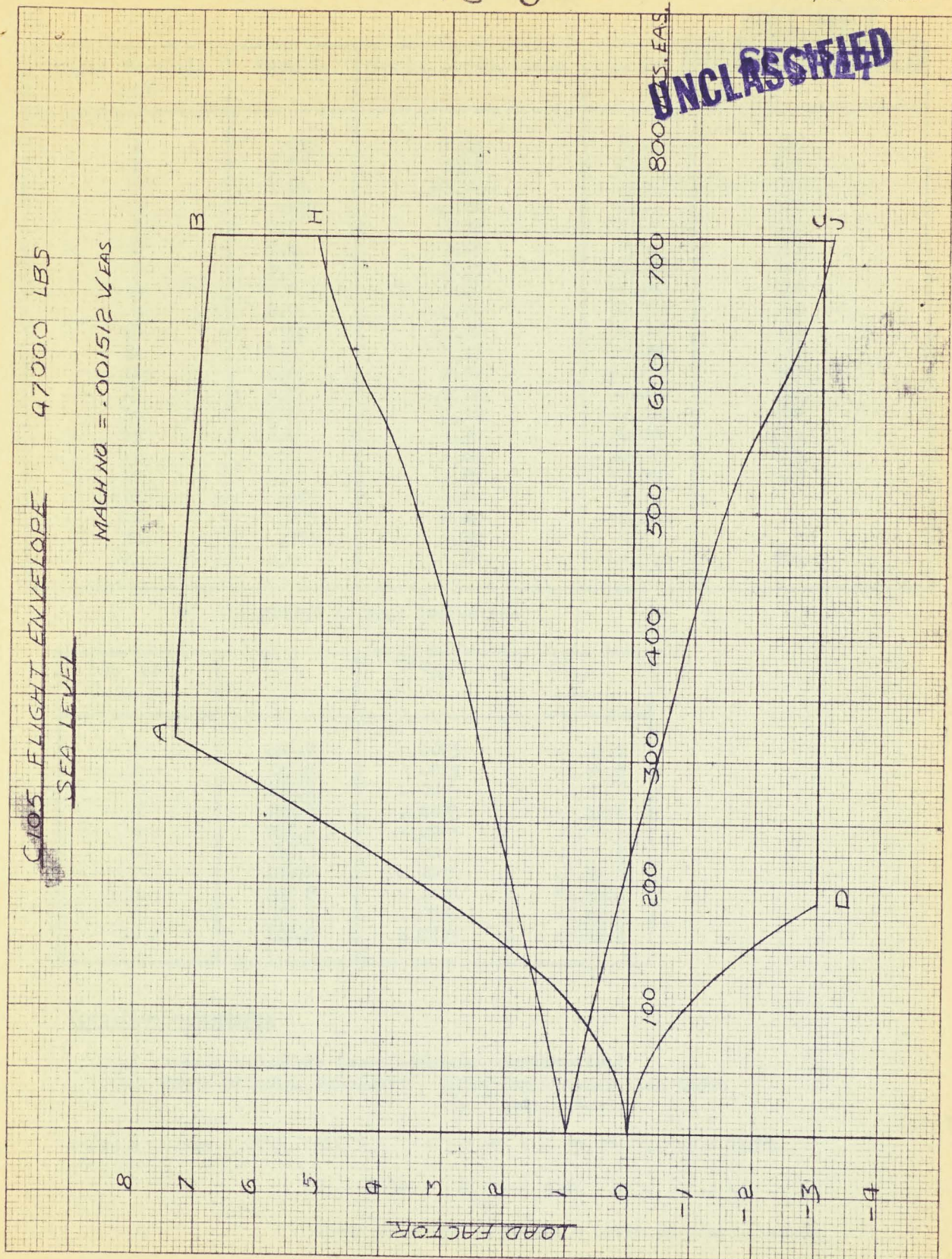
TABLE 1   March 18th, 1957.

Flight Envelope Case	Mach No.	Normal Acceleration	Height	Surface Temp. ($^\circ$R)
1	.453	7.33	S.L.	538
2	1.09	6.80	S.L.	631
3	1.09	-3.00	S.L.	631
4	.272	-3.00	S.L.	526
5	0.6	7.28	S.L.	553
6	0.8	7.12	S.L.	579
7	1.0	7.33	30,000	487
8	2.0	6.00	30,000	710
9	2.0	-3.00	30,000	710
10	.56	-3.00	30,000	435
11	1.2	7.33	30,000	520
12	1.6	7.02	30,000	603
13	1.2	-3.00	30,000	520
14	1.6	-3.00	30,000	603
15	2.0	4.89	30,000	710
16	1.6	4.89	30,000	603
17	1.2	4.89	30,000	520
18	.825	4.89	S.L.	
19	.825	4.89	30,000	
20	.375	4.89	S.L.	
21 *	2.0	6.40	45,000	710

*   Symetrical Aileron Deflection

Sub Case	C.G.	$\triangle C_{M_O}$	$\triangle$ a.c.
1	27%.$\bar{c}$	+.0075	-2.5%.$\bar{c}$
2	31%.$\bar{c}$	+.0075	-2.5%.$\bar{c}$
3	27%.$\bar{c}$	-.0075	+2.5%.$\bar{c}$
4	31%.$\bar{c}$	-.0075	+2.5%.$\bar{c}$
5	28%.$\bar{c}$	+.0075	-2.5%.$\bar{c}$
6	30%.$\bar{c}$	+.0075	-2.5%.$\bar{c}$
7	32%.$\bar{c}$	+.0075	-2.5%.$\bar{c}$
8	28%.$\bar{c}$	-.0075	+2.5%.$\bar{c}$
9	30%.$\bar{c}$	-.0075	+2.5%.$\bar{c}$
10	32%.$\bar{c}$	-.0075	+2.5%.$\bar{c}$

Specified Manoeuvres

(a)   Steady manoeuvre (zero pitching acceleration)
(b)   Check pitching manoeuvre - (pitching acceleration shown
                              on sheets issued).

Example - Case 9.2(b):   M = 2.0   n = -3.00   h = 30,000 ft.   T = 710°R

C.G. = 31% $\triangle C_{M_O}$ = +.0075 $\triangle$ a.c. = -2.5%

Check pitching manoeuvre $\ddot{\theta}$ shown on sheets issued.

SECRET

Reference Number: 8858/38/J

8th May, 1958
Mr. F. Brame
S. Kwiatkowski
STRUCTURAL INTEGRITY MANOEUVRES AS REQUIRED BY

SPECIFICATION MIL-S-5711 (U.S.A.F.)

The manoeuvres listed below are required to demonstrate structural integrity to specification requirements. Due to the presence of dampers in the Arrow some of these manoeuvres are not representative of design loads, others are limited to values below structural integrity limits by limiters and other protective devices as described in Memo 8792/02E/J. The following remarks apply to manoeuvres listed in section 4.2.2.1.2.2 of the specification.

(a) Normal symmetrical pull-out
Maximum value limited by command limiter and "g" limiter in normal mode. In emergency mode of control can be performed in areas where adequate controllability exists at high angles of attack up to reasonable high normal accelerations but not necessarily equal to the structural integrity limits.

(b) Normal symmetrical push-down
Push-down in normal mode to -3 g will cause nuisance disengagements in normal mode. Safe limit has not yet been established in the emergency mode (cross-coupling effects).

(c) Gust load factor simulation manoeuvre
Not required for the Arrow because manoeuvre load factors are always higher then gust load factors.

(d) Normal uncoordinated rolling pull-out.
In normal mode of control pilot's coordination is not required and rolling pull-outs will be coordinated automatically. These in some conditions are limited to values well below the structural integrity limits.

(e) Abrupt symmetrical pull-out
Abruptness of pull-outs is smoothed out by damper action. Case nearly equivalent to (a).

(f) Abrupt symmetrical pull-out with checking.
It is not possible to check the design case because damper action will tend to oppose initially the abrupt checking.

(g) Abrupt symmetrical push-down with abrupt checking (see item f).

USE AND PROCESSING OF FLIGHT DATA

A. 1. 200 channels will be used to confirm the flight air loads.

   2. Distribution of gauges.

O/Wing	22 plus 10	)	65% Wing
I/Wing	78 plus 20	)	
Fin	20		10% Fin
Fwd. Fuse	20	)	25% Fuse
Aft. Fuse	30	)	

   3. Readings taken in flight 20 times/sec. and recorded on tape
       for strain gauges.

   4. Accelerometer readings continuous recording.

   5. Scan accelerometer readers and select point closest to static
       test condition for similar case.

   6. Extract from magnetic tape punch card strain gauge information
       at same time interval as accelerometer.

   7. Compare strain gauge results to static test strain gauge results –
       note static test gauges are zeroed to zero load; flight strain gauges
       are zeroed to 1g inertia load.

       (a) Direct comparison
       (b) Plot of results round a section
       (c) Plot of results along spar or longeron

   8. Declare whether in general results are out by more than 10%

   9. Discuss any variation with Technical Design.

B. 1. If variation extensive use 400 gauges and calibration of wing.

   2. Distribution of gauges

O/Wing	45	15	)	65% Wing
I/Wing	160	40	)	
Fin	40			10% Fin
Fwd. Fuse.	40		)	25% Fuse.
Aft. Fuse.	60		)	

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(a) Direct comparison
(b) Plot of results round a section
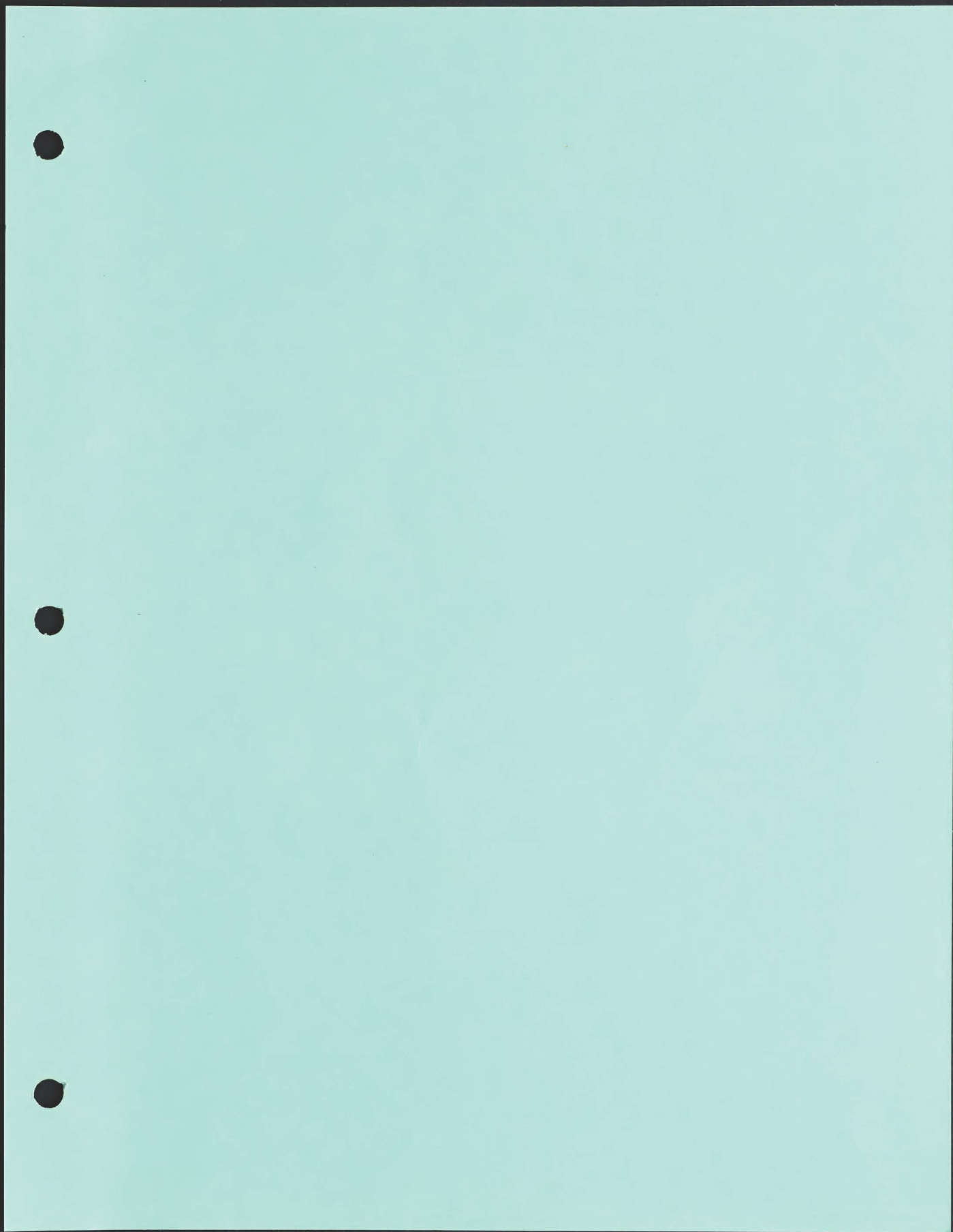(c) Plot of results along spar or longeron

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