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72/FAR/8

ARROW 2

AIRCRAFT 25211 - INSTRUMENTATION

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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

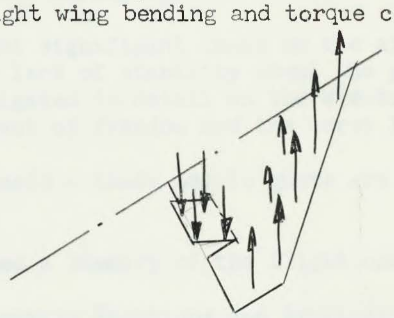
The symmetric flight manoeuvre envelope at this altitude is shown in figure 2. The fall off in design limit load factor with increasing temperature should be noted.

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.

Centre of Gravity Positions and Arbitrary Increments:

For all cases but Yaw Damper Failure, loads have been found for three c.g. positions

$.28$, $.30$, and $.32\ \bar{c}$

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 $\Delta C_{m_o} = .0075$ $\quad \Delta a.c. = -.025\bar{c}$

Max. nose down combination $\Delta C_{m_o} = -.0075$ $\quad \Delta a.c. = .025\bar{c}$

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 ~~possible~~ 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 planform 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

$$L_1 = a_{11}\alpha_1 + a_{12}\alpha_2 + a_{13}\alpha_3 \text{ ----}$$
$$L_2 = a_{21}\alpha_1 + a_{22}\alpha_2 + a_{23}\alpha_3 \text{ -----}$$

or in matrix notation

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

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

$$|L| = [A]|\alpha_0|$$

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

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

Elastic loads are

$$|L| = [A]\Big|\,|\alpha_0| + |\Delta\alpha|\,\Big| = [A]|\alpha_0| + [A][S]|L|$$
$$|L| - [A][S]|L| = [A]|\alpha_0|$$
$$|L| = \big[1 - [A][S]\big]^{-1}[A]|\alpha_0|$$

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 cause 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.

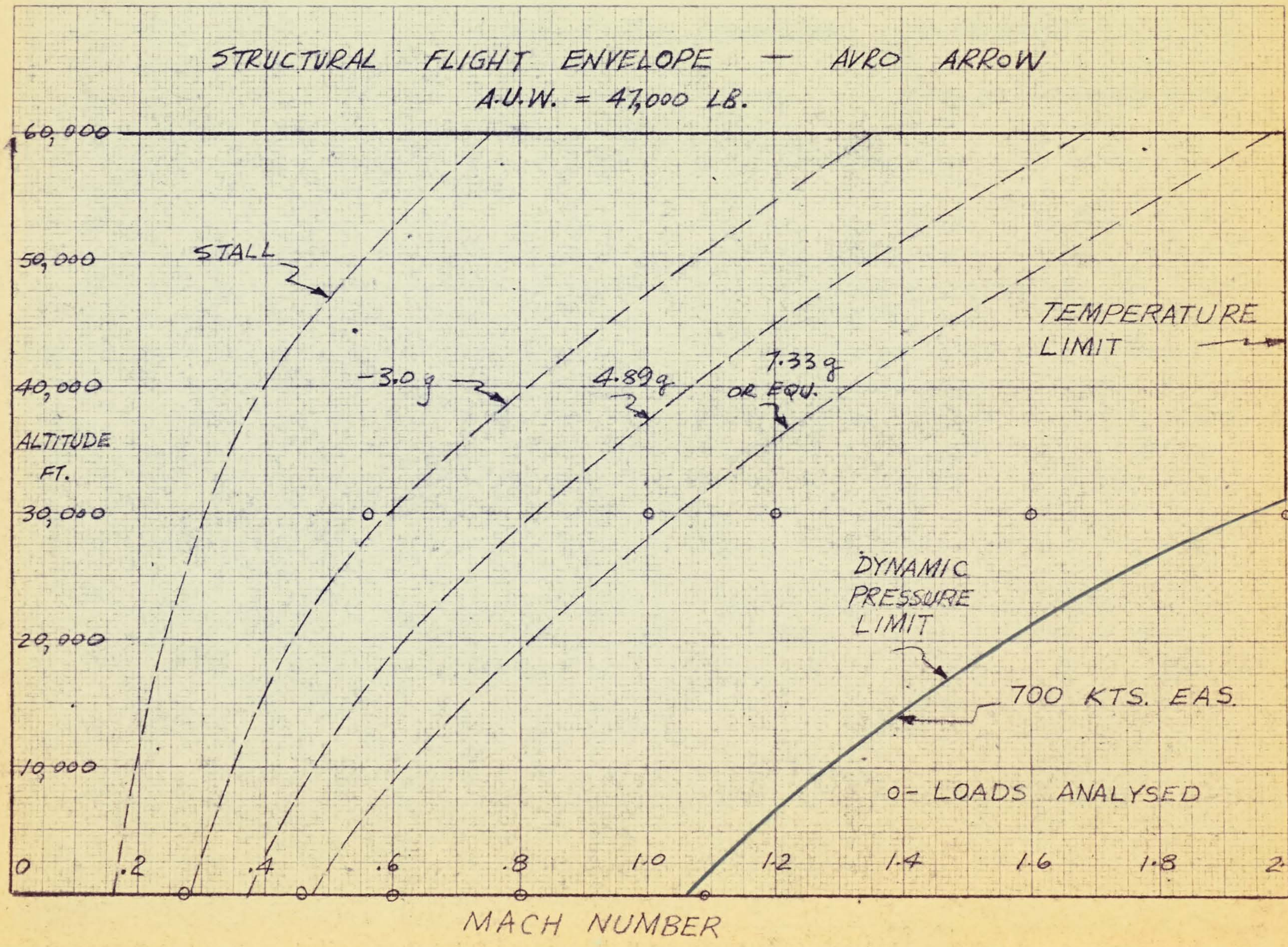
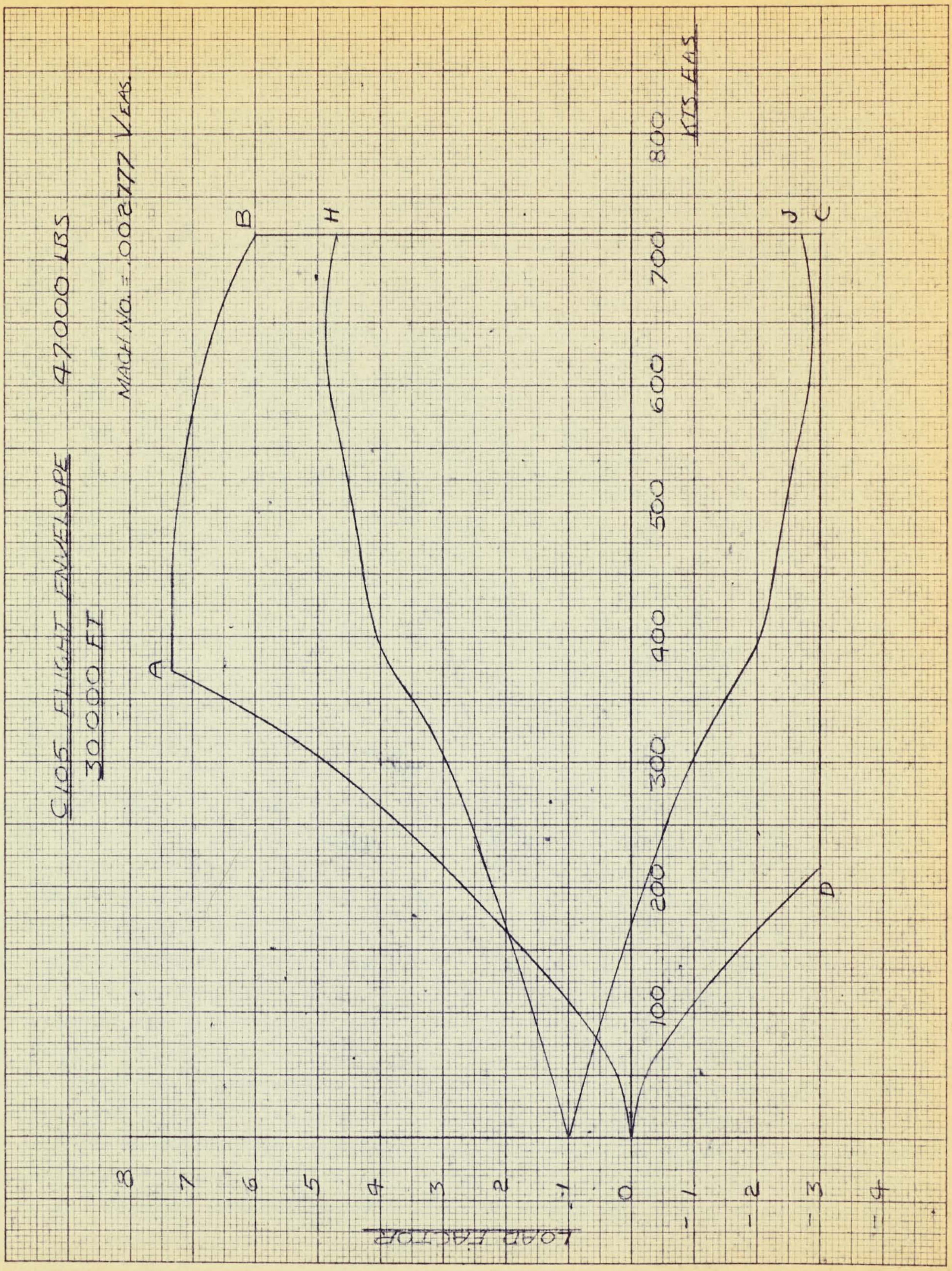

STRUCTURAL FLIGHT ENVELOPE — AVRO ARROW
A.U.W. = 47,000 LB.
ALTITUDE FT.
60,000
50,000
40,000
30,000
20,000
10,000
0
.2
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER
STALL
-3.0 g
4.89 g
7.33 g OR EQU.
TEMPERATURE LIMIT
DYNAMIC PRESSURE LIMIT
700 KTS. EAS.
o- LOADS ANALYSED

FIG 1

# FIG 2

SHEET 4 P/LOADS/120 APPENDIX 1
JAN/57 R. LOKEN



March 18th, 1957.

# TABLE 1

| Flight Envelope Case | Mach No. | Normal Acceleration | Height | Surface Temp. ($^oR$) |
|---|---|---|---|---|
| 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. | $\Delta C_{M_0}$ | $\Delta$ 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. $T = 710^oR$

C.G. = 31% . $\Delta C_{M_0} = +.0075$ $\Delta$ a.c. = -2.5%

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

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 envelope 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 considered 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 resulting 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

$\delta_p$ - parallel servo deflection

$\delta_D$ - 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 disengagement 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 $120^{\circ}$/sec. of roll rate or the maximum roll rate available aerodynamically if the latter is less than $120^{\circ}$/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. $160^\circ$/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 $120^\circ$/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 $180^\circ$ 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. Rudder 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 $10^{\circ}$ is reached. This is applicable only to low speed range e.g. where the $10^{\circ}$ 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 $120^{\circ}$/sec. or approx. 35% of limit load.

4. Rolling pull-out to roll rates not exceeding $120^{\circ}$/sec. or approx. 50% of fin limit load, whichever occurs first.

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

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 air-frame. 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 air loads 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 aircraft. 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 equivalent 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 instrumentation is to monitor typical flight cases in order to assess 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 & the full quantity will only be available when expected 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 maneeuvres (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 devicesand 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.

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

ORIGINAL SIGNED BY
J. D. HODGE

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

JDH/mvs

8th, May 1958 Reference Number: 8858/38/J

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 reasonably 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 than 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).

(h) Flaps down pull-out.
Not applicable to Arrow.

(i) Abrupt uncoordinated rolling pull-out.
See item (d).
Any increase of the vertical tail loads over 50% of limit load due to abruptness of the manoeuvre will automatically be restricted by aileron limiter. Only mild manoeuvres of this type are possible with emergency mode within a small part of the flight envelope.

(j) Rudder manoeuvre, high speed, steady sideslip, rudder reversed.

Critical amount of sideslip cannot be developed in this manner in the normal mode.
Manoeuvre restricted in emergency mode.

(k) Rudder manoeuvre, high speed, abrupt kick with abrupt return.

Critical amount of tail load cannot be produced in this manoeuvre in normal mode.
In emergency mode manoeuvre will be restricted although critical load cannot be produced.

(l) Rudder manoeuvre - landing approach.
For controllability reasons sideslip is limited to $10^{\circ}$.
Tail loads within this limit could be tested with proper precautions.

S. Kwiatkowski,
Chief of Stability and Control.

SK/g

c.c. -Messrs. C. Lindow
G. Watts
J. Hodge

(24)



ENGINEERING DIVISION

January 31st 1958
Ref: 5997/01/J

W/C G.B. Waterman
Officer Commanding
Technical Services Detachment
Royal Canadian Air Force
Avro Aircraft Limited
MALTON, Ontario

SUBJECT: STRUCTURAL INTEGRITY AIRCRAFT - ARROW

Dear Sir:

The enclosed note and report are our recommendations regarding the instrumentation for the structural integrity aircraft. We are of the opinion that the method proposed will give adequate coverage of the critical structural areas during flight testing. We have allocated aircraft #25211 for use on this program and on receipt of a decision from the R.C.A.F., will proceed with the installation design.

Prior to a decision being made however, we would like to discuss some aspects of the installation, in particular the problems of aircraft safety when using strain gauges in the wing fuel tanks.

Please advise when you are ready to discuss the matter.

Yours very truly

AVRO AIRCRAFT LIMITED

A R Buley

A.R. Buley
Project Designer - ARROW 2

C Lindow

C.V. Lindow
Engineering Project Manager - ARROW

Encls:
(72/FAR/8 & Drg. 7-4400-15007)

/ch

GENERAL:

The applicable specification for the Avro Arrow relative to the requirements for the flight determination of air loads and structural demonstration of piloted aircraft is MIL-S-5711.

It is to be noted here, that the concept of MIL-S-5711 cannot be directly applied to a low aspect ratio wing such as the Arrow. Paragraph 3.6.1.1. of the specification states:-

> "Unless otherwise stated, the airload instrumentation shall consist of systems capable of measuring the spanwise distribution of bending moment, shear and torque on the principal lifting surfaces".

Further to this quote, paragraph 3.6.1.1.1. of the specification states:-

> "Unless otherwise agreed, wing load distribution measurements by strain gauges method, shall include at least four stations spanwise, for shear and bending and two for torque measurements, on small to medium size airplanes".

In addition, the following is a statement quoted from Avro Report GEN/1090/336 relative to the stressing of the Arrow wing:-

> "The stress analysis of high aspect ratio subsonic wings was a development of the beam theory, and ——— the whole analysis was basically one dimensional, with the span coordinate as an independent variable. With the advent of the low aspect ratio supersonic wing, the analysis has become a two dimensional problem. In these wings, stress distribution is defined in terms of two local bending moments acting at different angles and of a local torque, all these depending on spanwise and chordwise coordinates."

It follows therefore, that in low aspect ratio delta wings, the notion of spanwise bending moments and shear acting at certain span stations has little meaning. Moreover, in tail-less aircraft the lift produced by the fuselage and its distribution are as important as wing or elevator loads. Having regard to the foregoing, there is little value in establishing a procedure of flight load determination as no use can be made of terms such as, wing bending moment, or wing torque, at certain span stations.

It is therefore proposed that a procedure based on strain gauge readings in response to pre-established combination of loads be considered. Of necessity, these will include both air and inertia loads. The inertia loads will then have to be eliminated by means of acceleration measurements.

continued/2

~~RESTRICTED~~

METHOD A

The determination of actual airloads can be accomplished by using the results of the static test of the complete aircraft. The static test aircraft is extensively strain gauged and will be acted upon by inertia loads and calculated airloads. If the structural integrity aircraft is, to a degree, identically strain gauged and flown under the same flight conditions as those calculated for the static tests, the two results can be compared. If the flight test results are compatible with those for the static test, it can be concluded that the actual airloads agree with the calculated airloads and provide a measure of assurance for the structural integrity of the aircraft.

If the test results however are not compatible, a very approximate check can be made on the airload distribution by examination of the strain gauge results. This check would only be of value if there existed a recognizable pattern in the strain gauge results and must take into account the marginal accuracy of the strain gauges or possible inaccuracies produced in the pre-calculated load distribution.

An additional 200 strain gauges will be employed serving as a back-up system. These additional 200 gauges would be utilized, on a selective basis, to provide a more detailed sampling of measurements in the event that the results of the initial examination proved inconclusive. The instrumentation requirements for this type of program have been listed in Arrow 2 Report 72/FAR.8.

METHOD B

A more exact method of determining the distribution of the airloads could be carried out by calibrating the aircraft, strain gauged on the ground, with unit loads applied using jacks. The proposed breakdown of airloads would consist of 13 loads distributed over 13 areas. Two of these loads can be determined by conditions of equilibrium, subsequently, the loads are combined in 11 self-equilibrating groups. The number of strain gauges employed must exceed the number of groups to be determined for two reasons:-

(a) It may be difficult to obtain selectiveness of readings with respect to load locations.

(b) Strain gauges readings are seldom accurate to more than two significant figures and the averaging of many readings necessary.

continued/3

METHOD B (Continued)

It must be emphasized that this is a very extensive program. This method would require a re-design of the static test rig and the specimen calibrated as a large number of test cases must be examined.

RECOMMENDATIONS:

It is recommended that METHOD A be considered for this program. A total of 400 strain gauges would be necessary to measure stress levels at critical areas in the structure. In addition, instrumentation to record measurements of the following would be required.

1) Airspeed
2) Pressure Altitude
3) Control surface forces - five positions
4) Rates of roll, pitch and yaw
5) Accelerations in roll, pitch and yaw
6) Angle of attack
7) Angle of sideslip

~~RESTRICTED~~

INTER-DEPARTMENTAL MEMORANDUM

Ref 5516/02A/J
Date Jan. 15, 1958
To S. E. Harper
From J. D. Hodge
Subject ARROW 2 - AIRCRAFT 25211 - INSTRUMENTATION

Herewith 72/FAR/8 "Instrumentation - Arrow 2 - Aircraft 25211" which lists the instrumentation requirements for Aircraft 25211.

J D Hodge

J. D. Hodge
Technical Flight
Test Co-ordinator

AA/bb

c.c.

Messrs J. Chamberlin
F. Brame
C. Lindow
D. Scard (5)
A. Buley
J. Ames
J. Booth
A. Stenning (4)
S. Kwiatkowski
C. Marshall
J. Lucas
H. Malinowski
R. Young
J. Scott
S. Whiteley (2)
W. Alford
R. Wade
J. McKillop
J. Gale
Central Files

~~RESTRICTED~~

Aircraft: Arrow 2
A/C 25211

Report No 72/FAR/8

No of Sheets:

INSTRUMENTATION

ARROW 2 - AIRCRAFT 25211

Prepared By A.T. Anderson Date 15th JAN 1958

Checked By Wm C. Ettwright Date 17 Jan. 1958

Supervised By J Hodge. Date

Approved by Date

~~RESTRICTED~~

INSTRUMENTATION - ARROW 2 - AIRCRAFT 25211

This report is issued to cover instrumentation requirements for Aircraft 25211. This aircraft will be primarily used for Structural Integrity Testing.

# ARROW 2 - AIRCRAFT 25211

## STRUCTURAL INTEGRITY

### Structural Strain Gauges

Strain gauges are required at 400 points on the aircraft structure, plus such spares and duplicates as the Flight Test Department considers necessary. The basic distribution of these gauges is shown below and their detailed positioning has been agreed to by the Flight Test Department and the Stress Department.

These gauges will give strain monitoring of major structural members and an approximate distribution of wing loads.

### DISTRIBUTION

| Component | Gauges |
|---|---|
| Forward Fuselage | 40 |
| Aft. Fuselage | 60 |
| Inner Wing L.H. | 160 |
| Inner Wing R.H. | 40 |
| Outer Wing L.H. | 45 |
| Outer Wing R.H. | 15 |
| Fin | 40 |
| Total | 400 |

ARROW 2 - AIRCRAFT 25211

STRUCTURAL INTEGRITY

Vibration Pick up Accelerometers

As in aircraft 25201 and 25202, figure 1 shows the approximate location of 57 vibration pick-up accelerometers. The precise location of these may be obtained from Flight Test Drawings Nos.

7-0782-1
7-0782-2
7-0774-1
7-0774-2
7-0764-1
7-0764-2
7-0762-1
7-0762-2
7-0784-1
7-0783-1
7-0759-1
7-0758-1
7-0756-1
7-0754-301
7-0752-1
7-0751-51

The required range is -10g to +10g with an accuracy of ± 0.25g and the instruments should be capable of recording frequencies up to 60 cycle/sec. Under normal flight test conditions it will not be necessary to record any information from the accelerometers, but should any flutter problem arise it will be necessary to provide telemetering or continuous trace recording.

ARROW 2 - AIRCRAFT 25211

STABILITY AND CONTROL INSTRUMENTATION

1. Ambient Conditions

| ITEM | RANGE | ACCURACY | ACCURACY % OF FULL RANGE | SAMPLING FREQUENCY |
|---|---|---|---|---|
| Aircraft Static Pressure | 0 - 2160 $lb/ft^2$ | ± 15 $lb/ft^2$ | ± .75% | 2/sec |
| Differential Pressure | 0 - 2880 $lb/ft^2$ | ± 20 $lb/ft^2$ | ± .75% | 2/sec |
| Free Air Total Temp. | -65 to +350°F | ± 2°F | ± 0.5% | 2/sec |

2. Motion of Aircraft

| ITEM | RANGE | ACCURACY | ACCURACY % OF FULL RANGE | SAMPLING FREQUENCY |
|---|---|---|---|---|
| Angle of Attack $\alpha$ | -6 to +30° | ±0.1° | ± 0.3% | Cont. |
| Angle of Sideslip $\beta$ | -15 to +15° | ±0.1° | ± 0.5% | Cont. |
| Rate of Roll $\dot{\phi}$ | -300 to +300° | ±2.0° | ± 0.5% | Cont. |
| Normal Acceleration (near C.G.) | -3 to +8g | ±.06g | ± 0.5% | Cont. |
| Normal Acceleration (fwd) | -3 to +8g | ±.06g | ± 0.5% | Cont. |
| Lateral Acceleration (near C.G.) | -1 to +1g | ±.01g | ± 0.5% | Cont. |
| Lateral Acceleration (fwd) | -1 to +1g | ±.01g | ± 0.5% | Cont. |

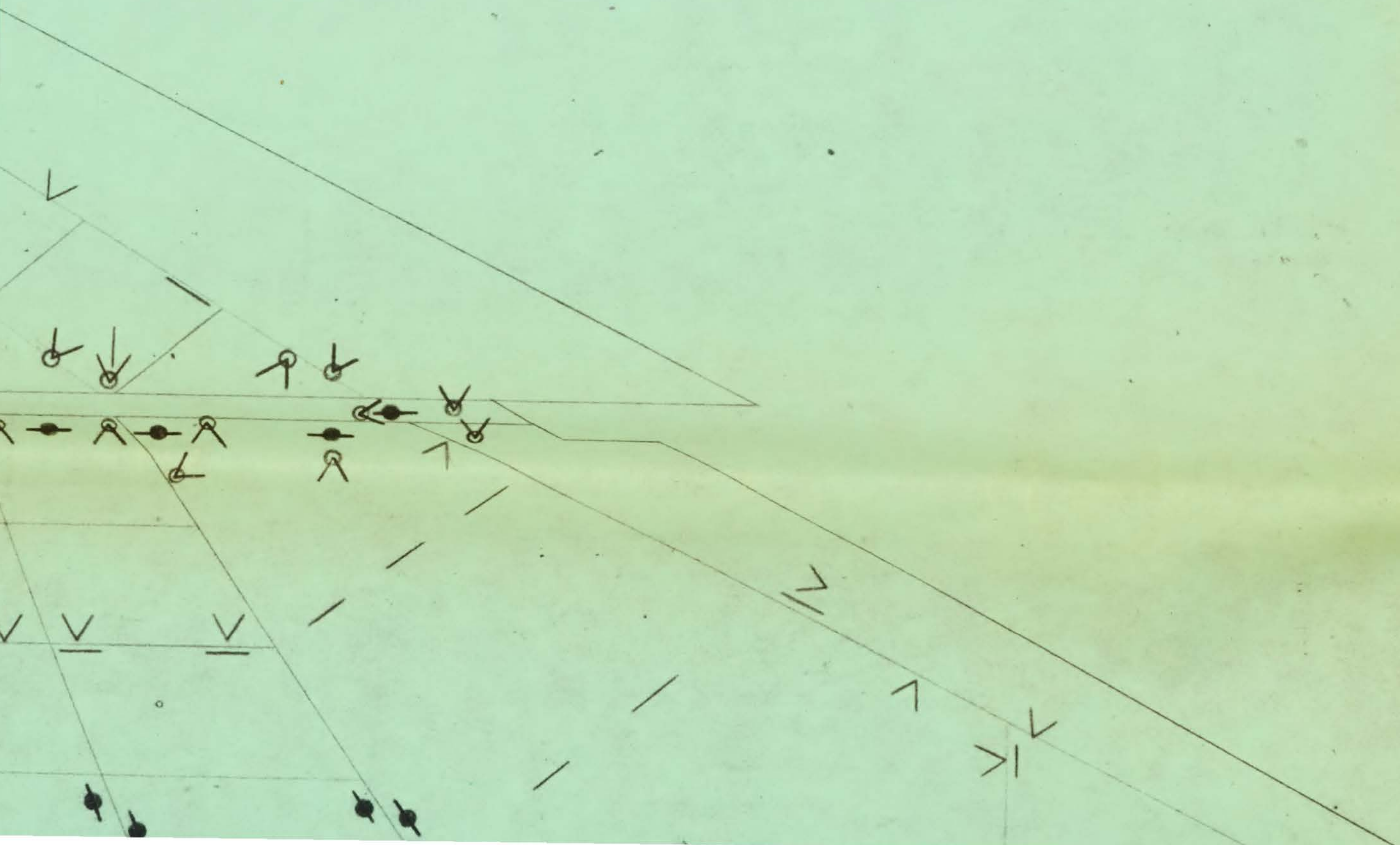
### 3. Control Surface Motion

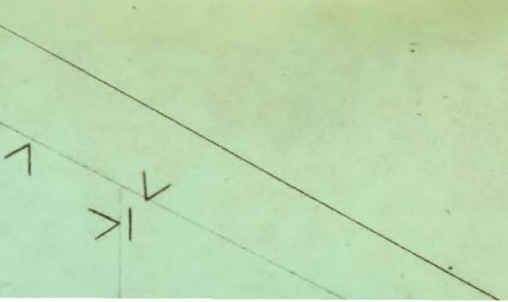
| ITEM | RANGE | ACCURACY | ACCURACY % OF FULL RANGE | SAMPLING FREQUENCY |
|---|---|---|---|---|
| Left Elevator Angle $\delta e$ | -30 to +20° | ± 0.3° | ± 0.5% | Cont. |
| Left Aileron Angle $\delta a$ | -19 to +19° | ± 0.2° | ± 0.5% | Cont. |
| Angle of Rudder $\delta k$ | -20 to +30° | ± 0.3° | ± 0.5% | Cont. |
| Left Aileron Angular Acceleration $\ddot{\delta} a$ | -200 to +200°/sec² | ±2°/sec² | ± 0.5% | Cont. |

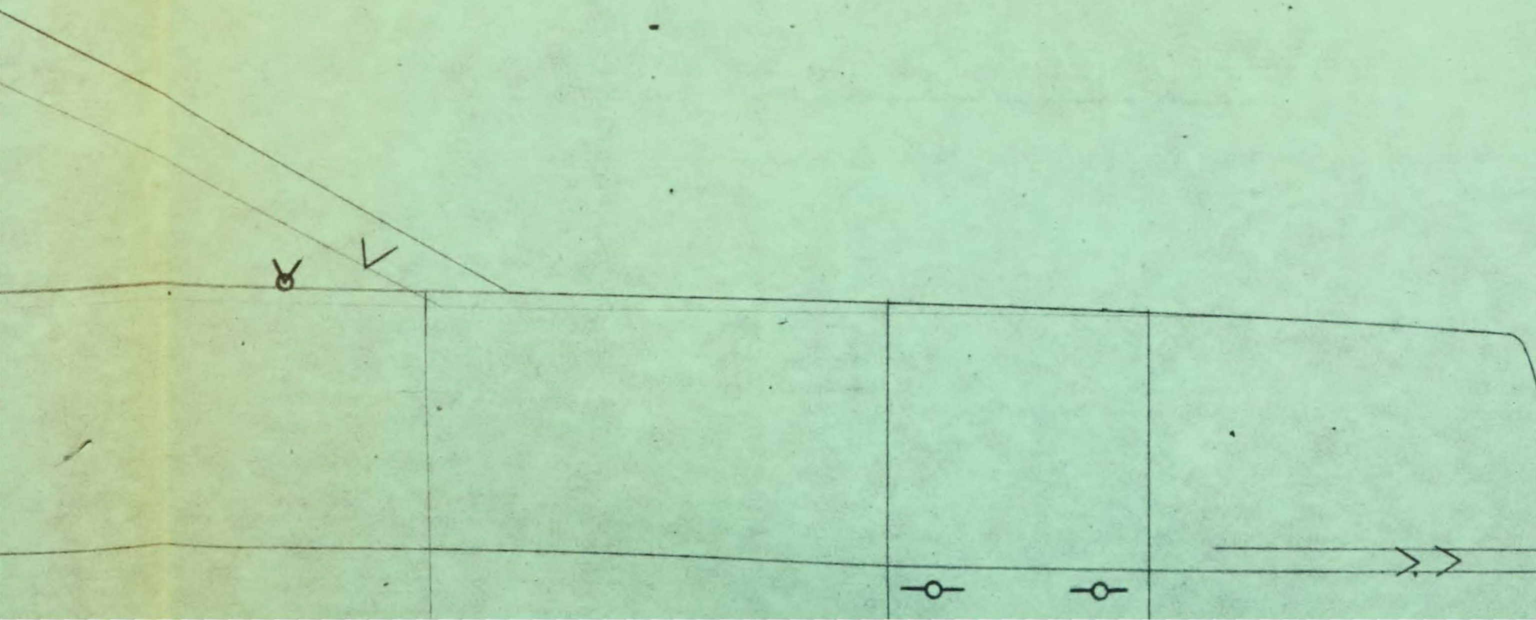
### 4. Control Mechanism

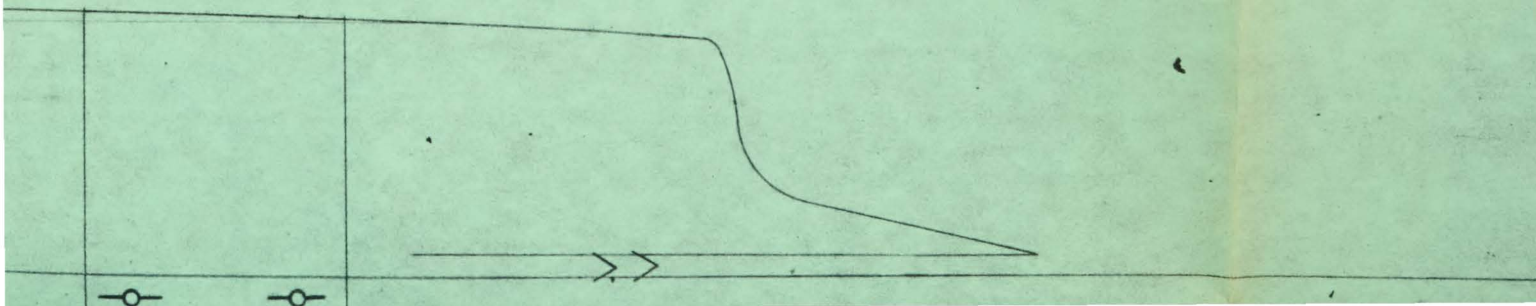
| ITEM | RANGE | ACCURACY | ACCURACY % OF FULL RANGE | SAMPLING FREQUENCY |
|---|---|---|---|---|
| Left Elevator Damper Servo Position | -0.6 to +0.6" | ± 0.01" | ± 1% | Cont. |
| Left Aileron Damper Servo Position | -0.6 to +0.6" | ± 0.01" | ± 1% | Cont. |
| Rudder Damper Servo Position | -0.5 to +0.5" | ± 0.01" | ± 1% | Cont. |
| Elevator Stick Force | -80 to +120 lb | ± 2 lb | ± 1% | Cont. |
| Limited Range | -20 to +40 lb | ± 0.6 lb | ± 1% | Cont. |
| Aileron Stick Force | -30 to +30 lb | ± 1 lb | ± 1% | Cont. |
| Elevator Jack Load | 0 to 71000 lb | ± 1400 lb | ± 2% | Cont. |
| Aileron Jack Load | 0 to 42000 lb | ± 800 lb | ± 2% | Cont. |
| Rudder Jack Load | 0 to 31000 lb | ± 600 lb | ± 2% | Cont. |

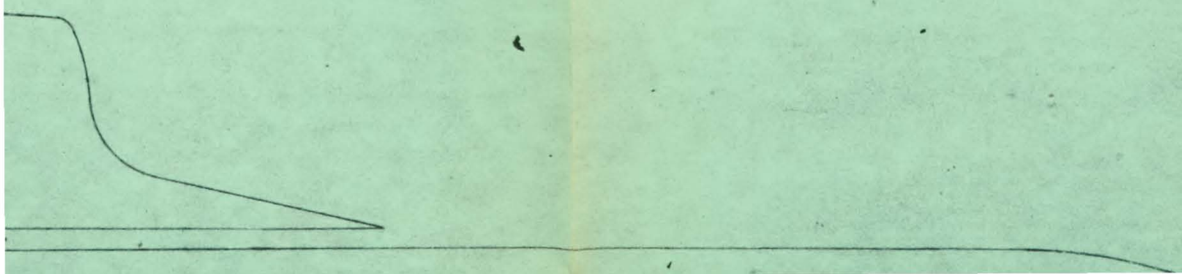
Classification cancelled / changed to Unclassified
By authority of AVRO Arrow Declassif. Board
Date 28 Jul 87
Signature B Aubrey , Co-Chairperson
Unit / Rank / Appointment DSIS 3

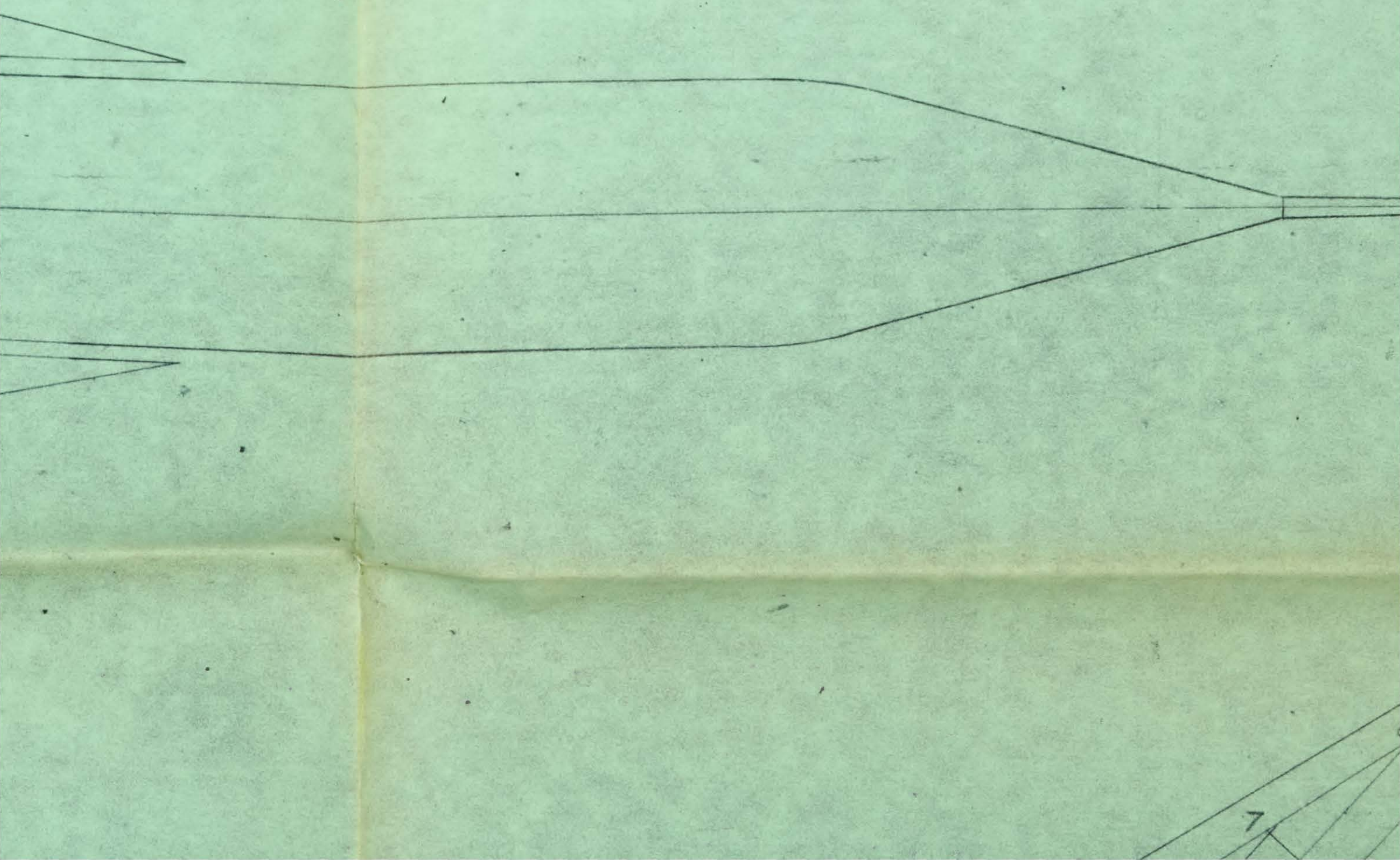








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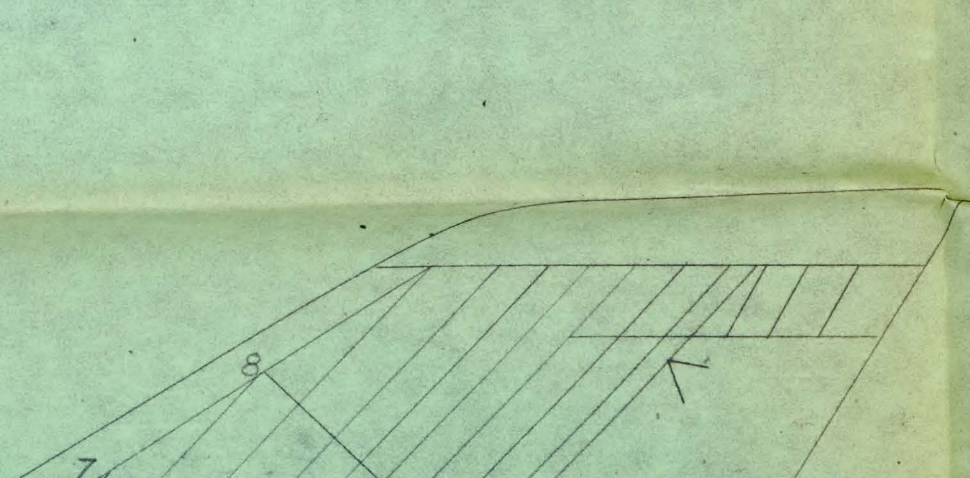

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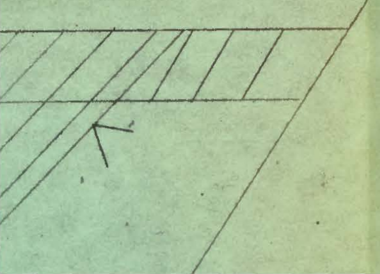

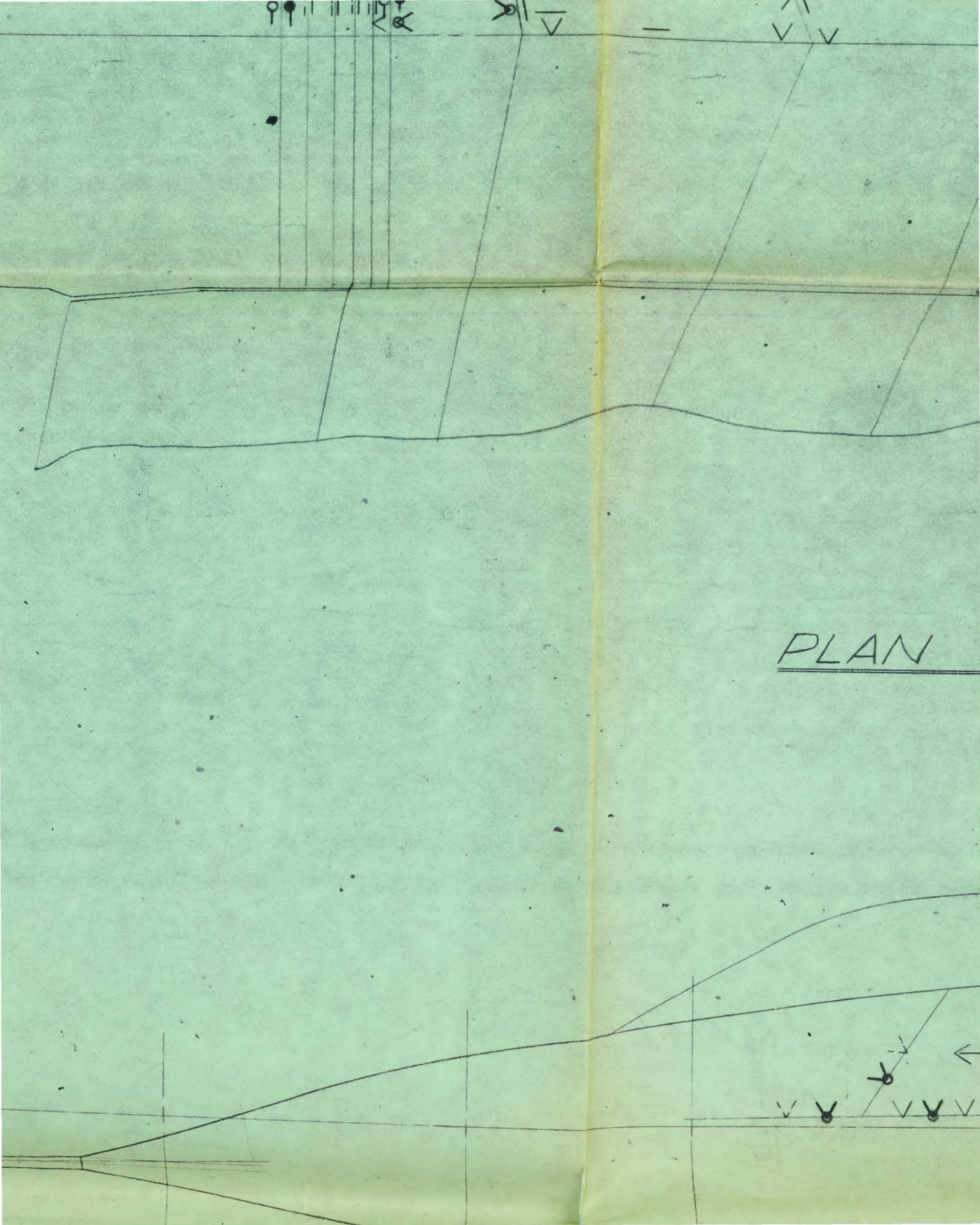
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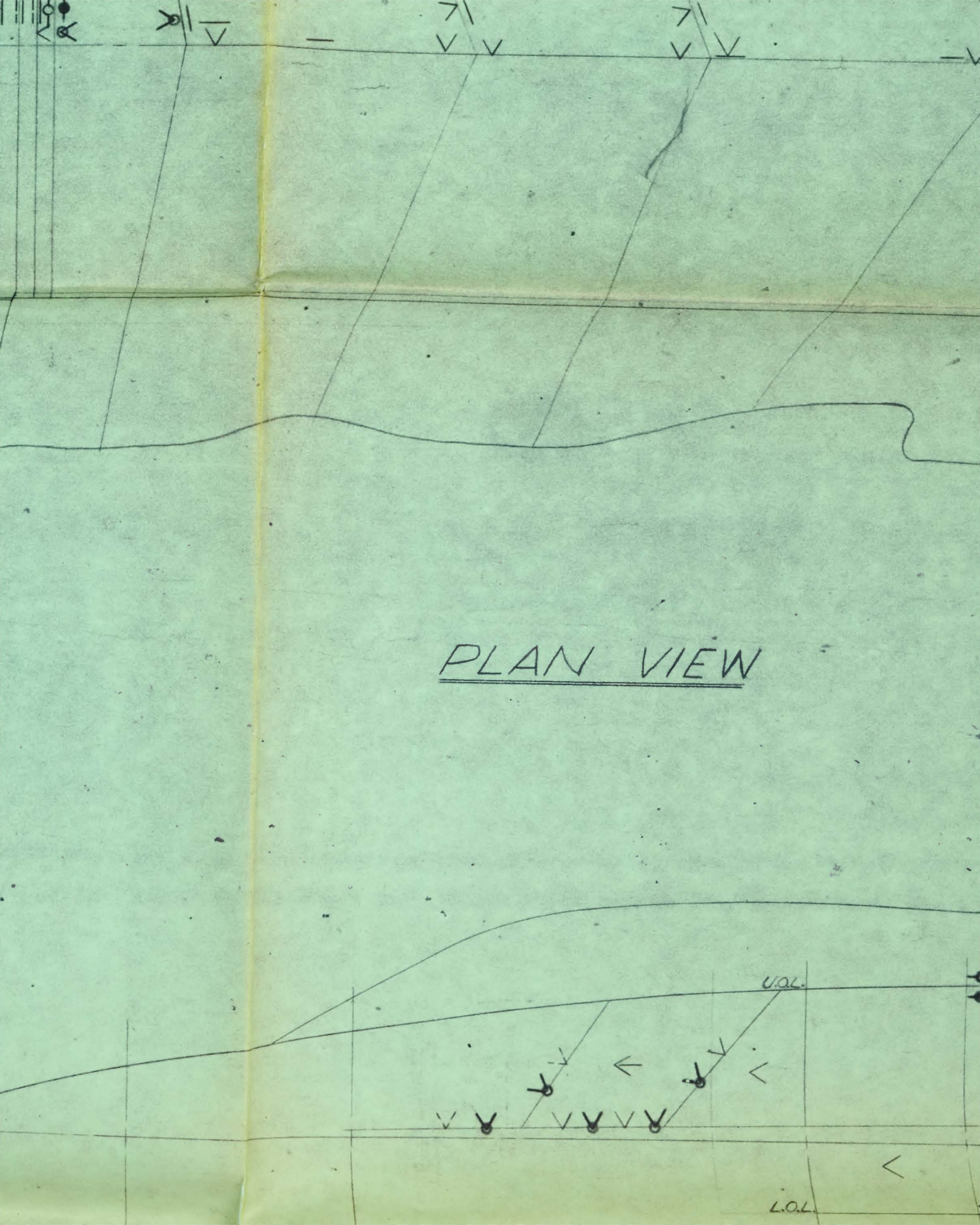
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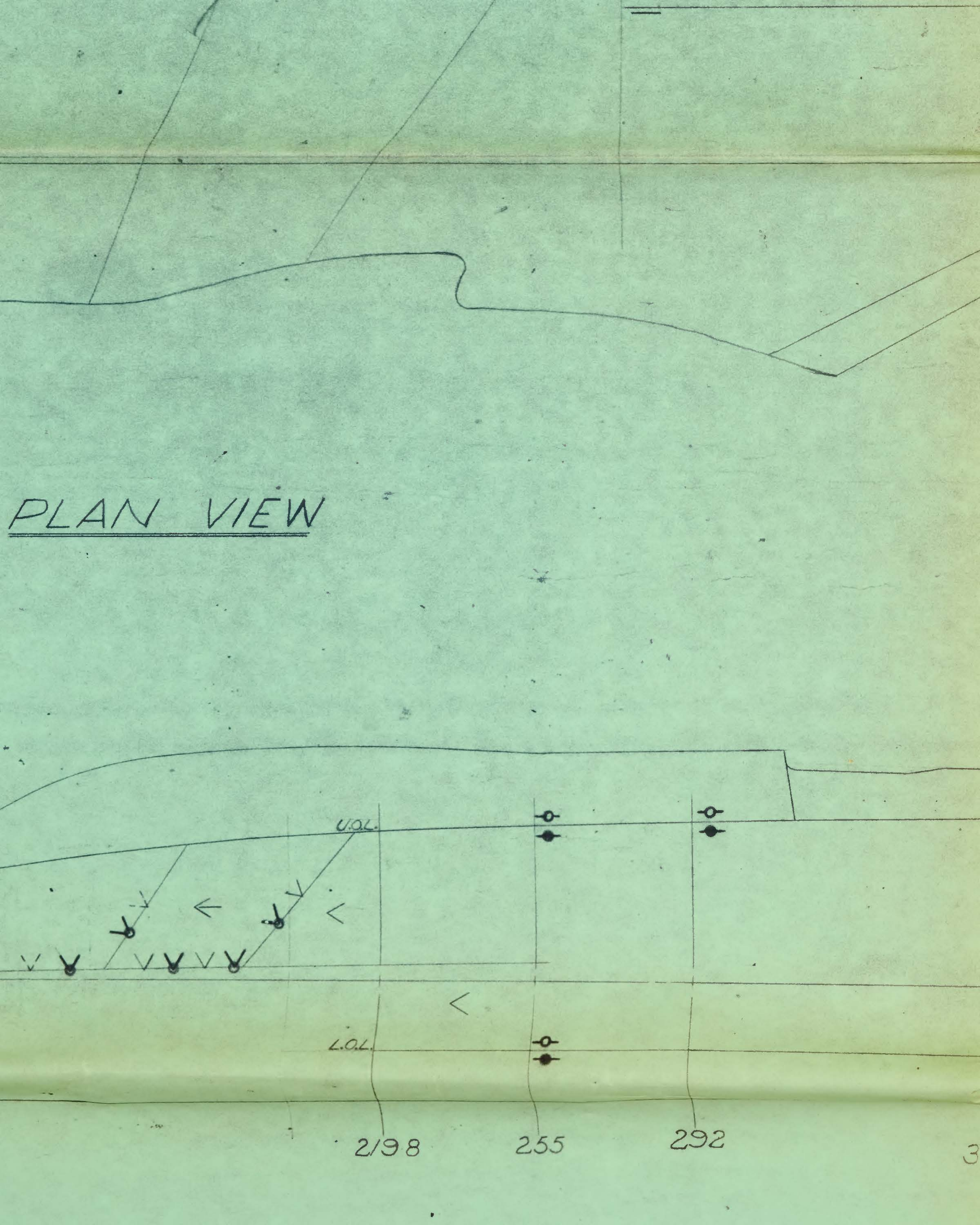

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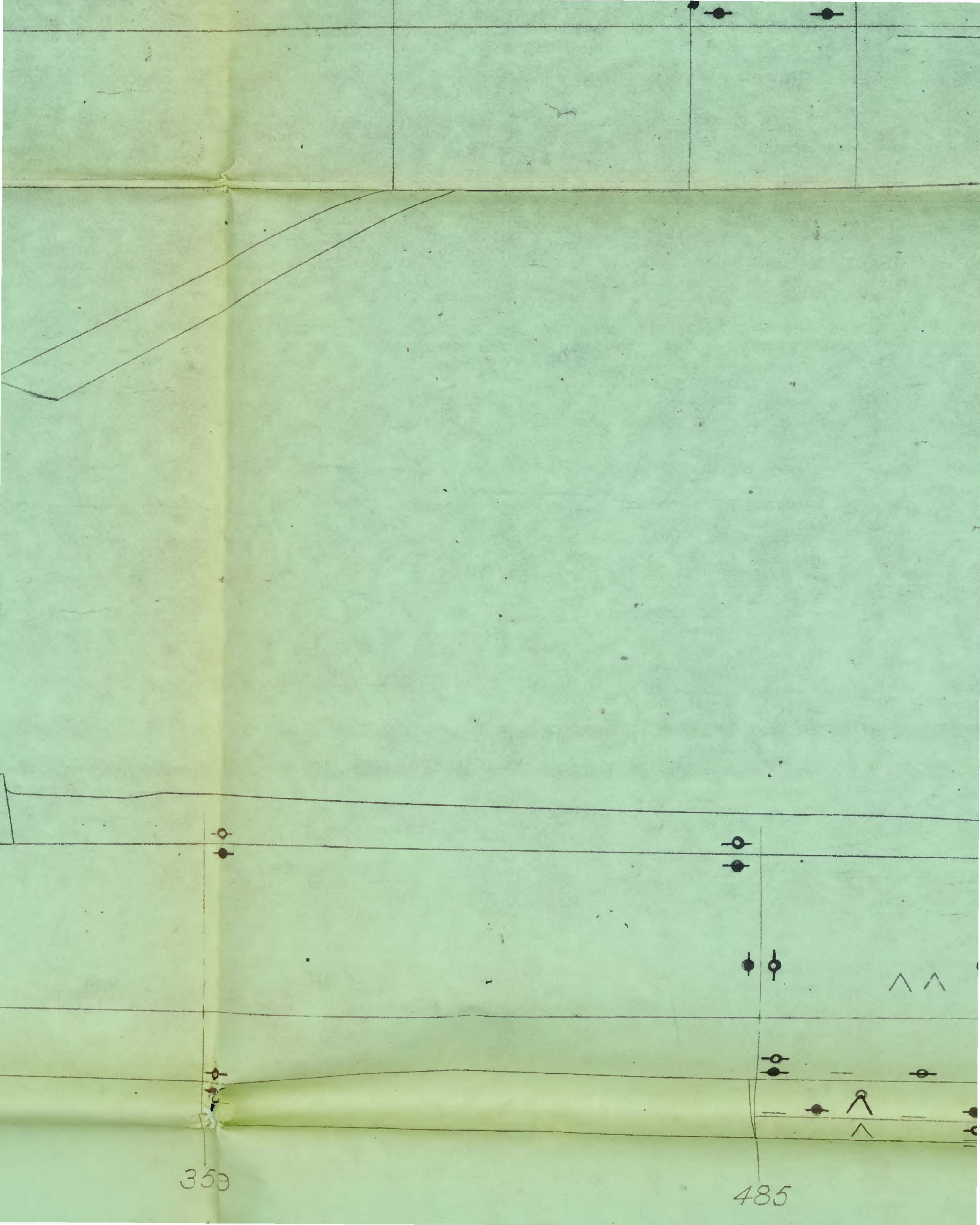
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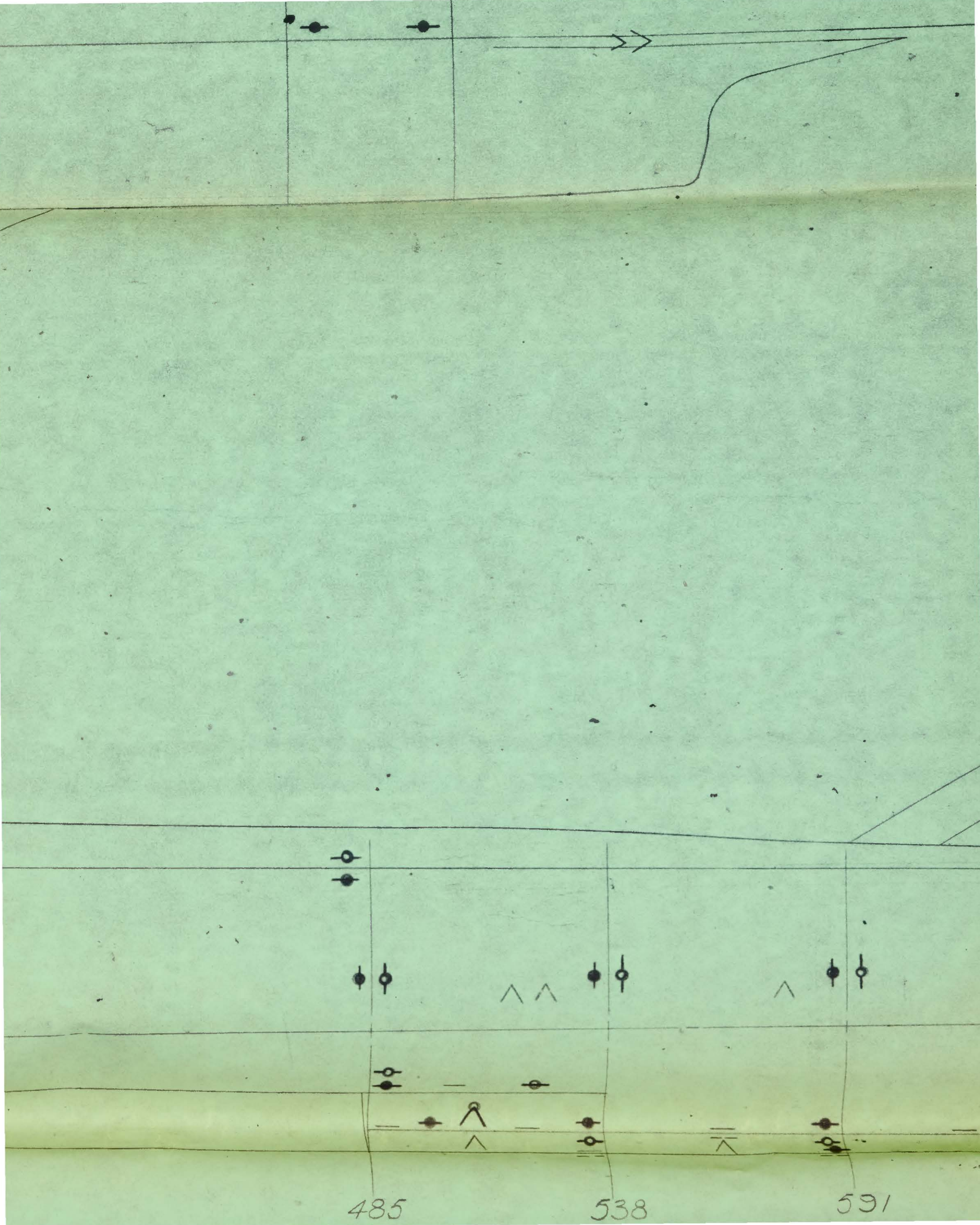
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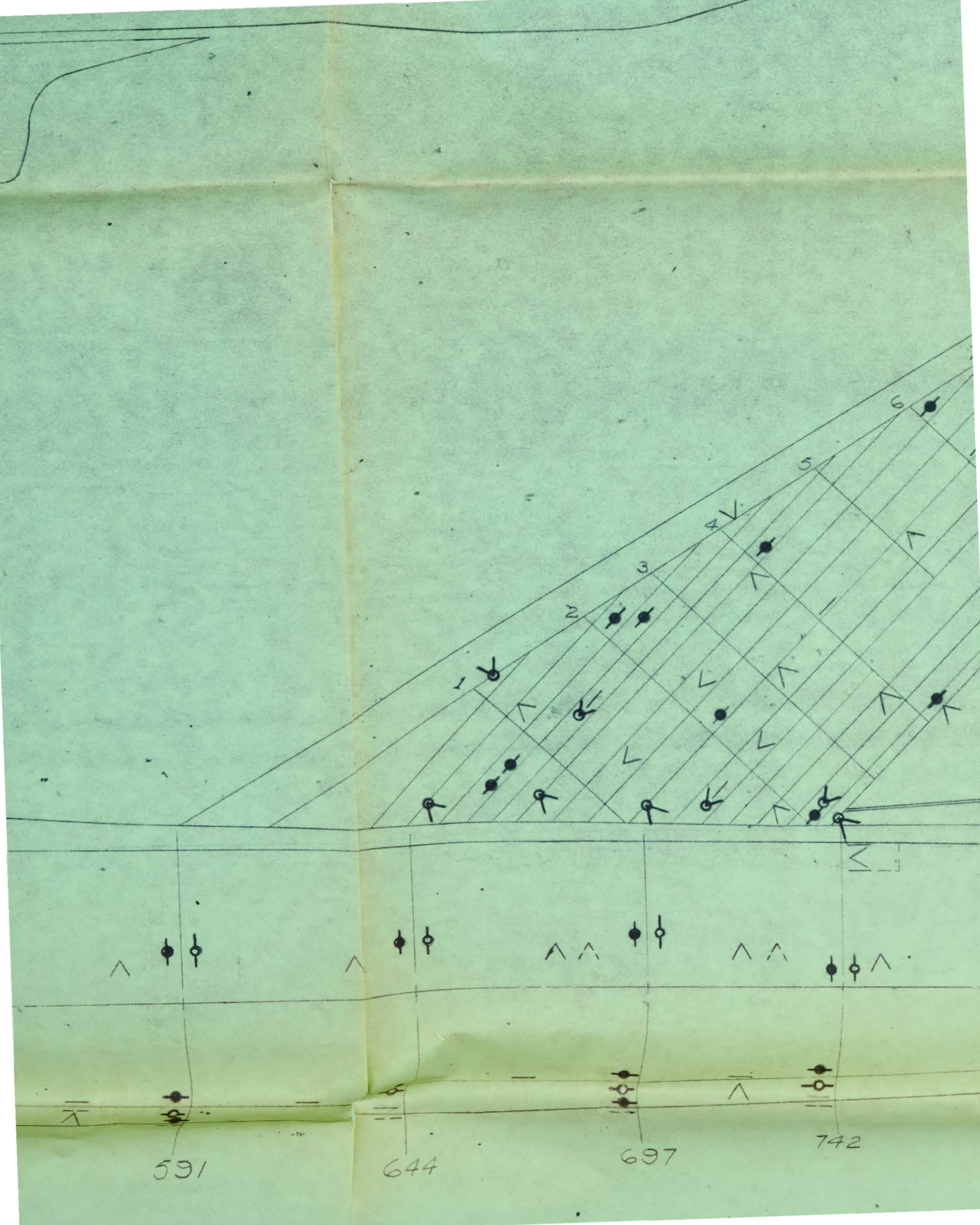
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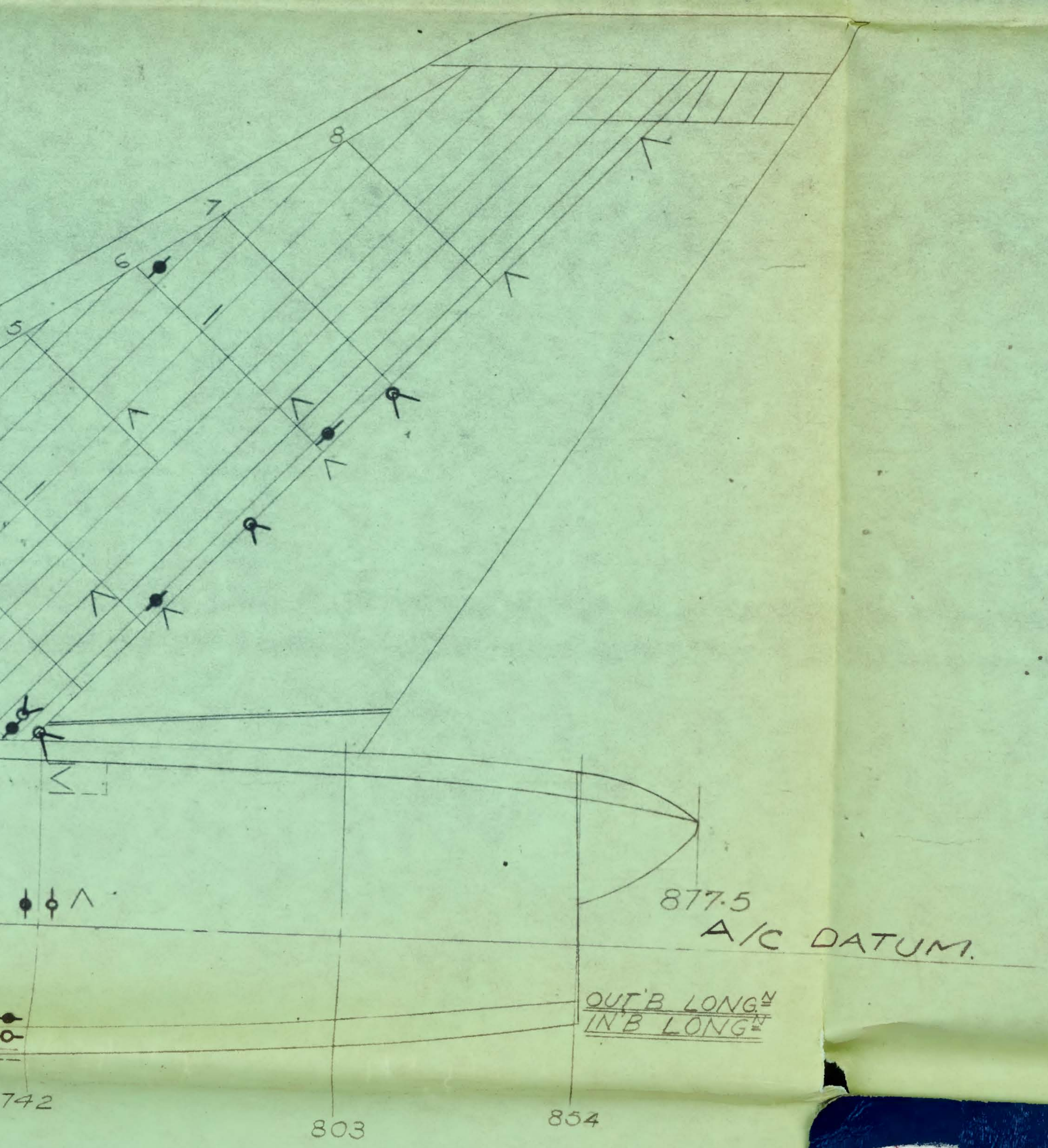

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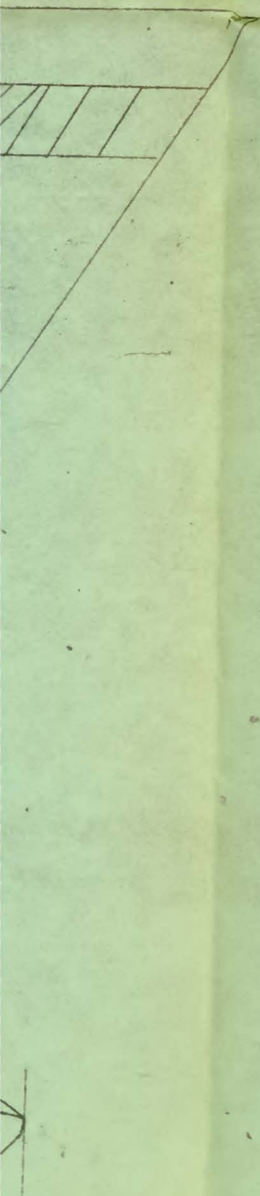

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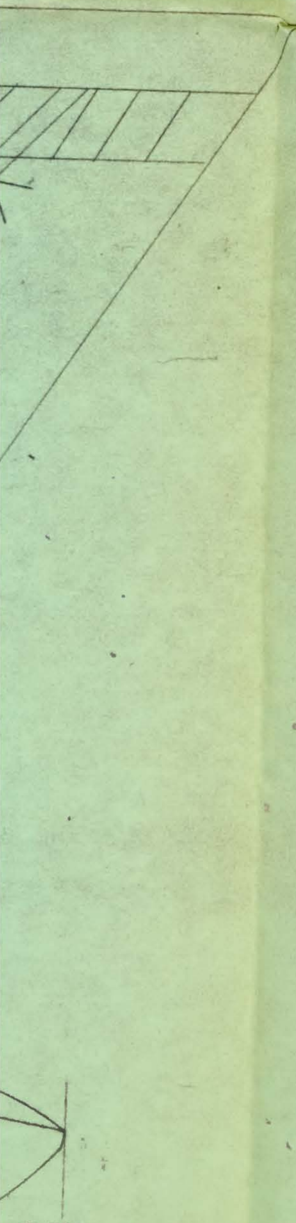

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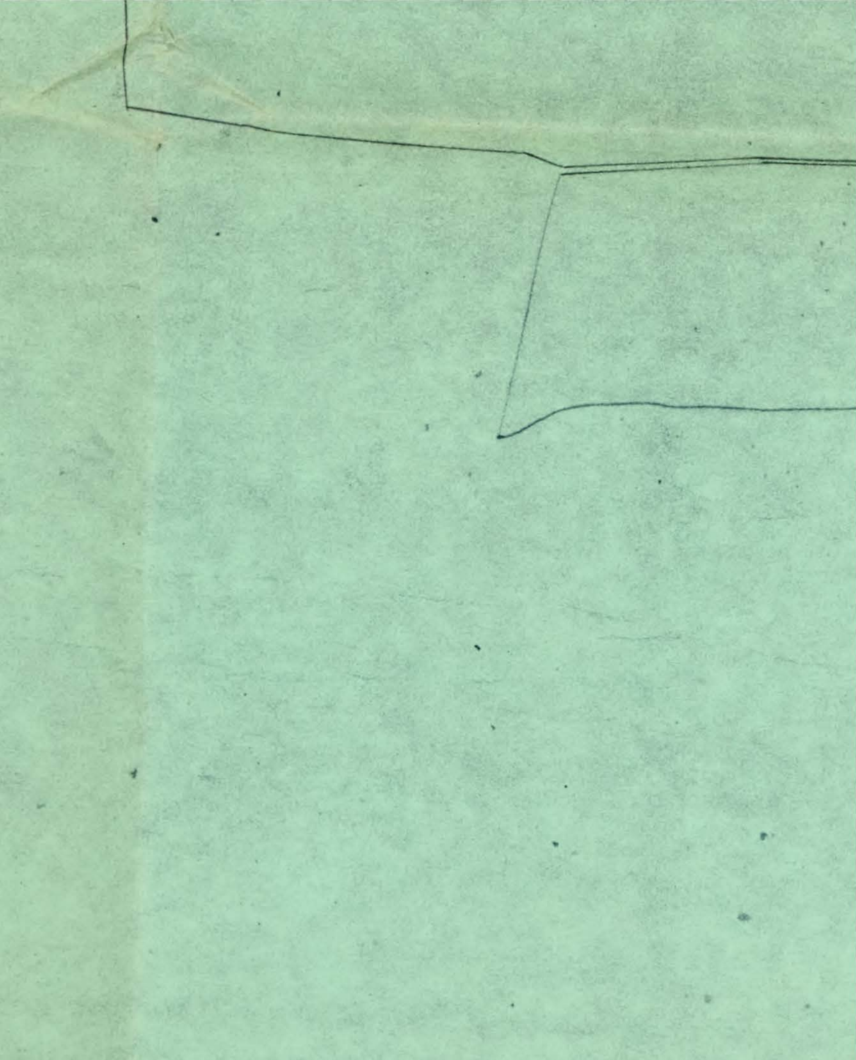
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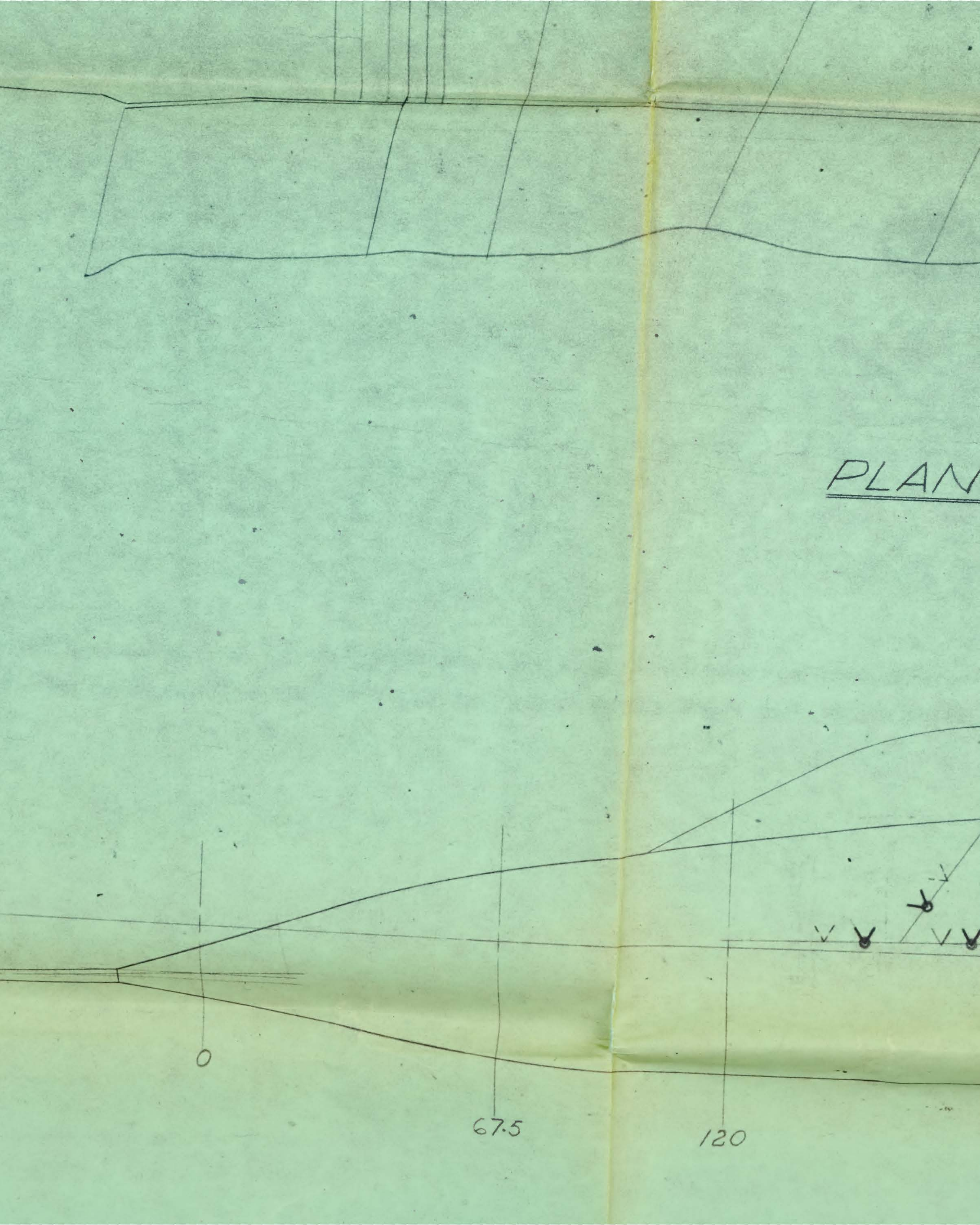
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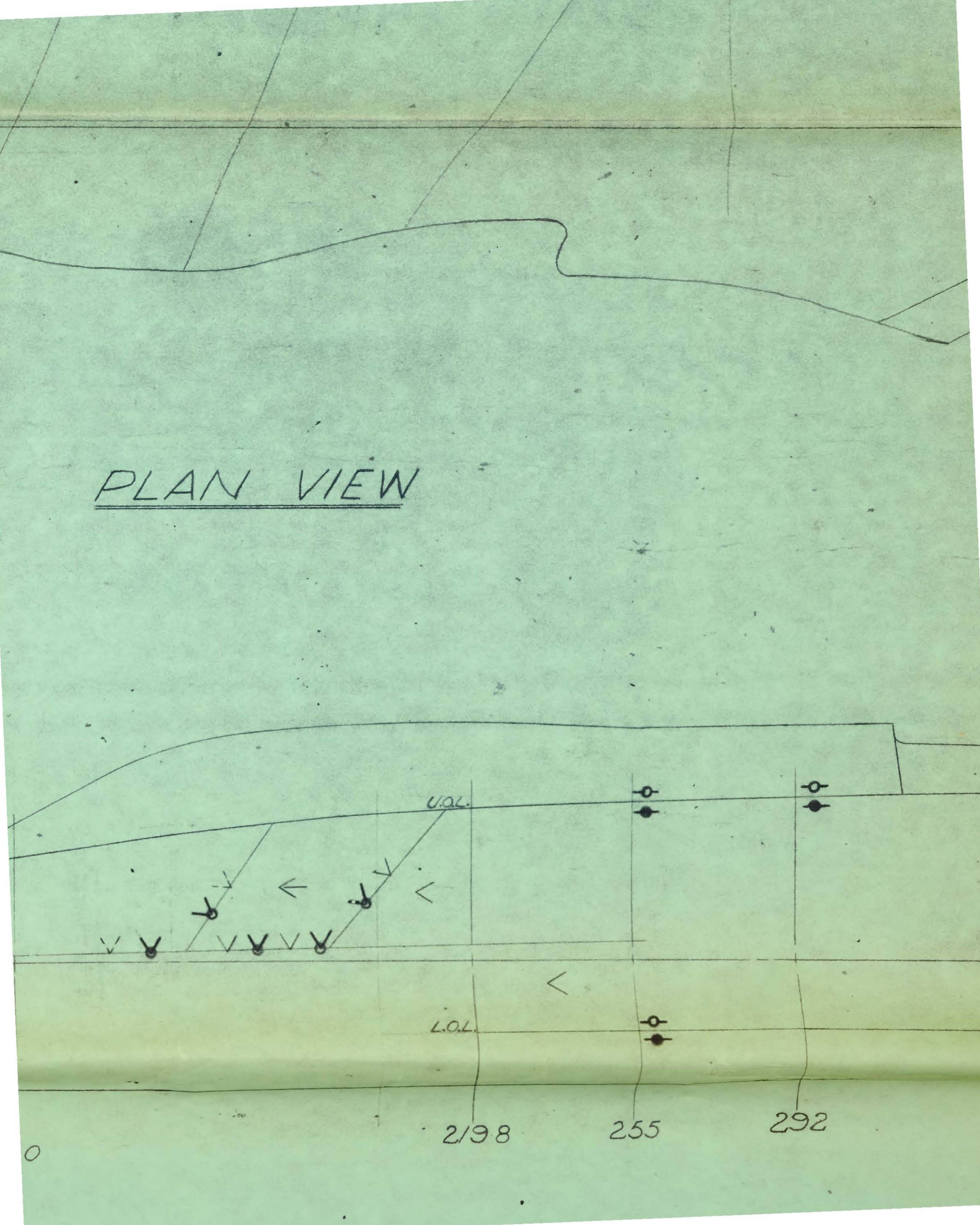
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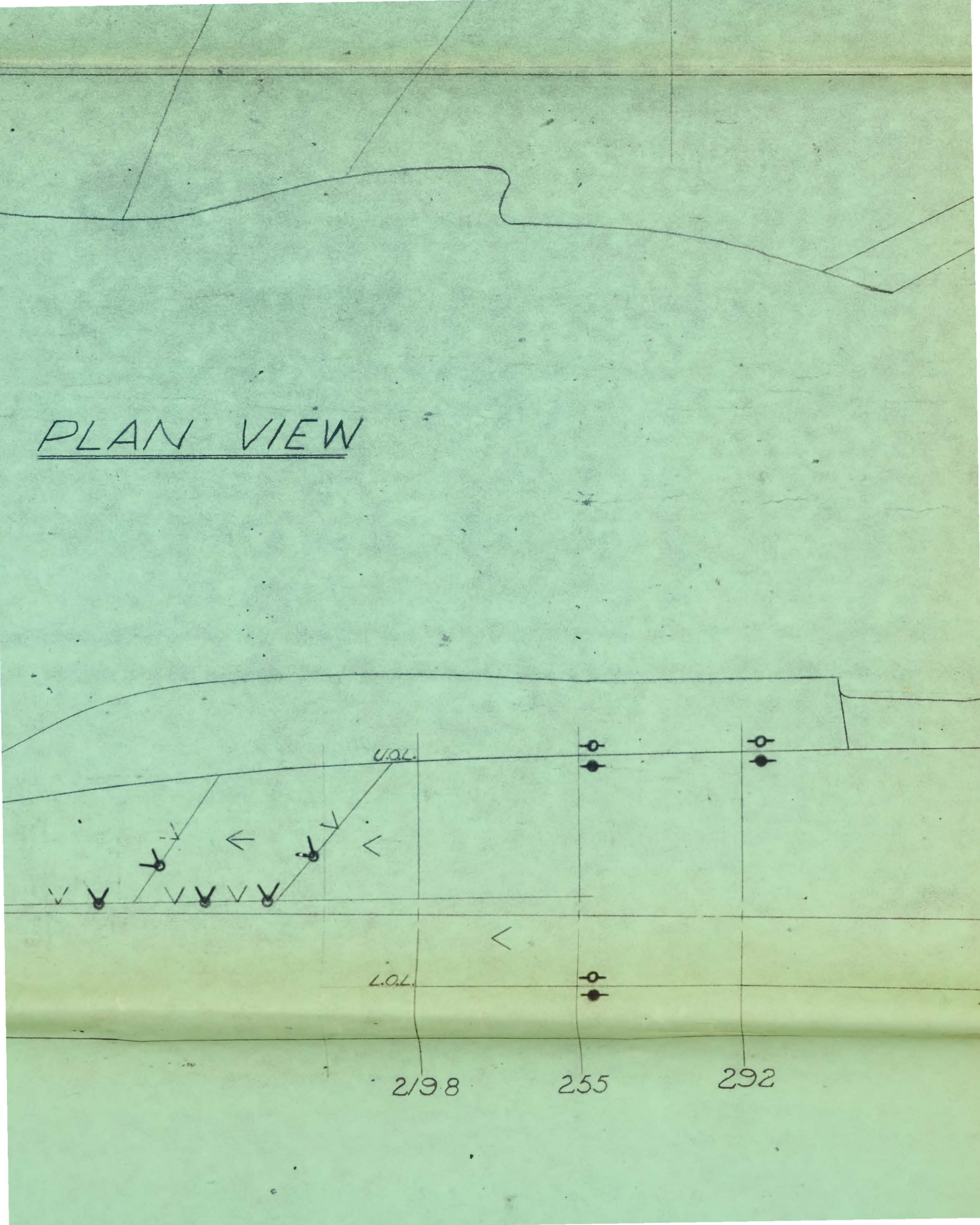
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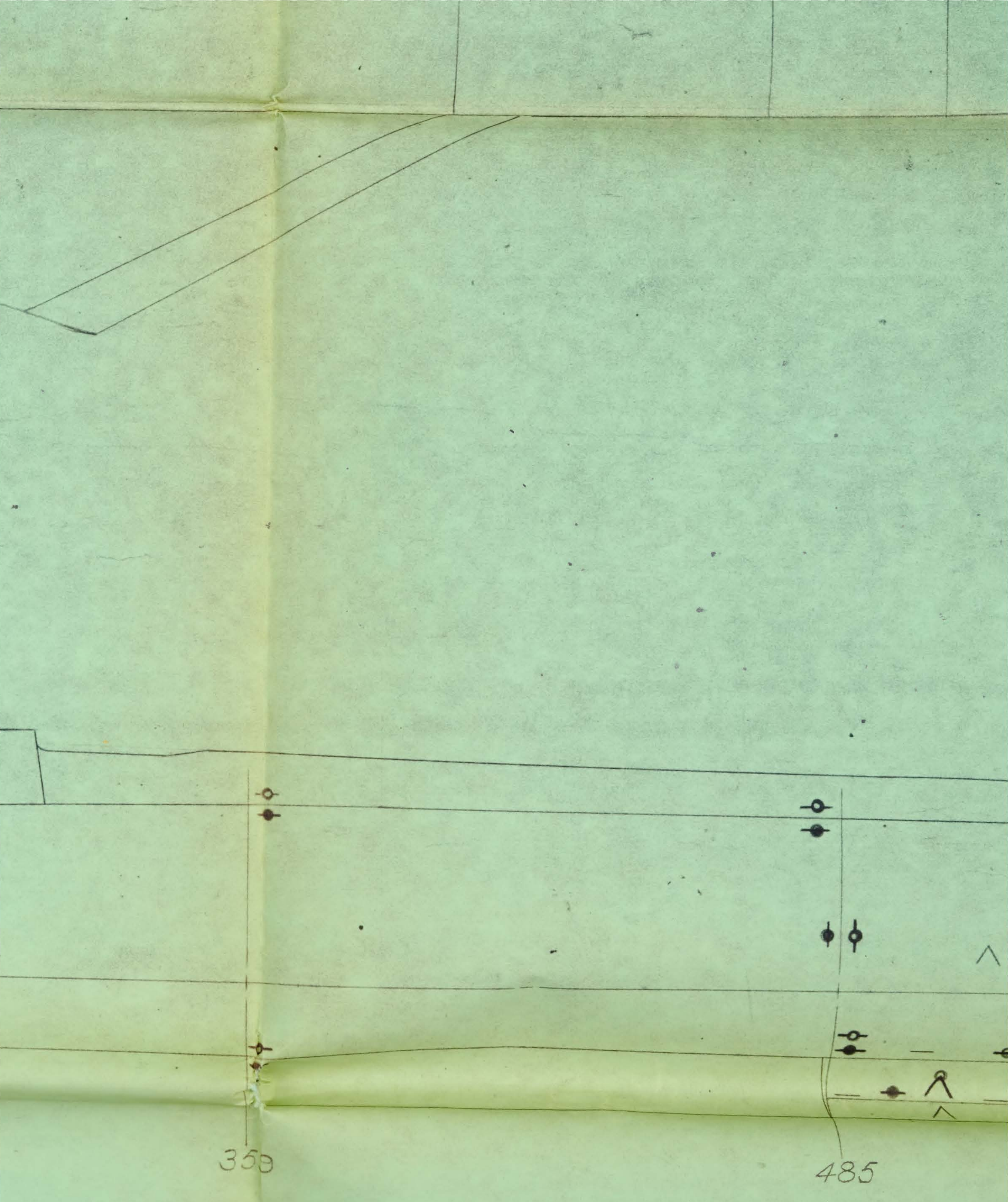

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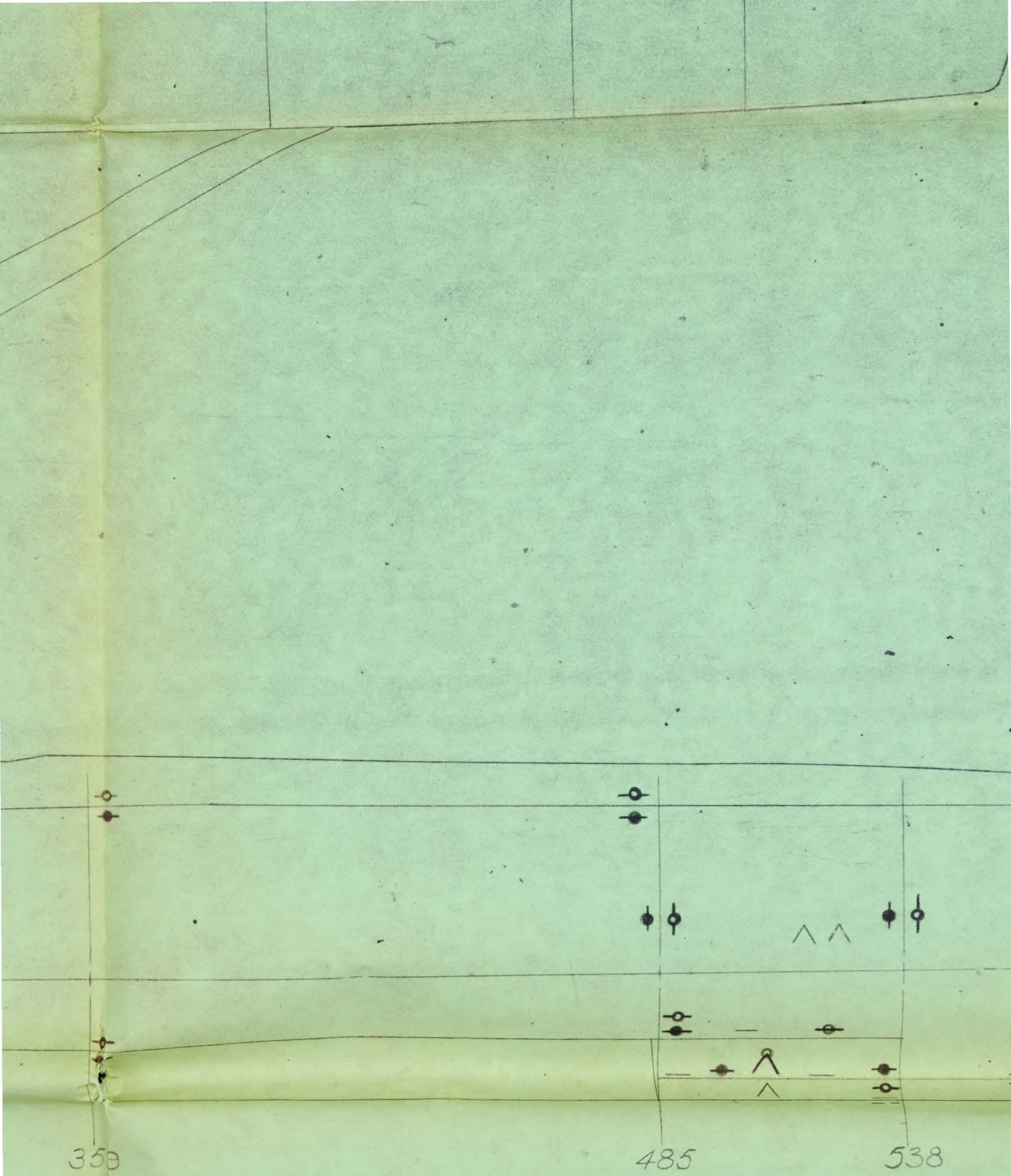
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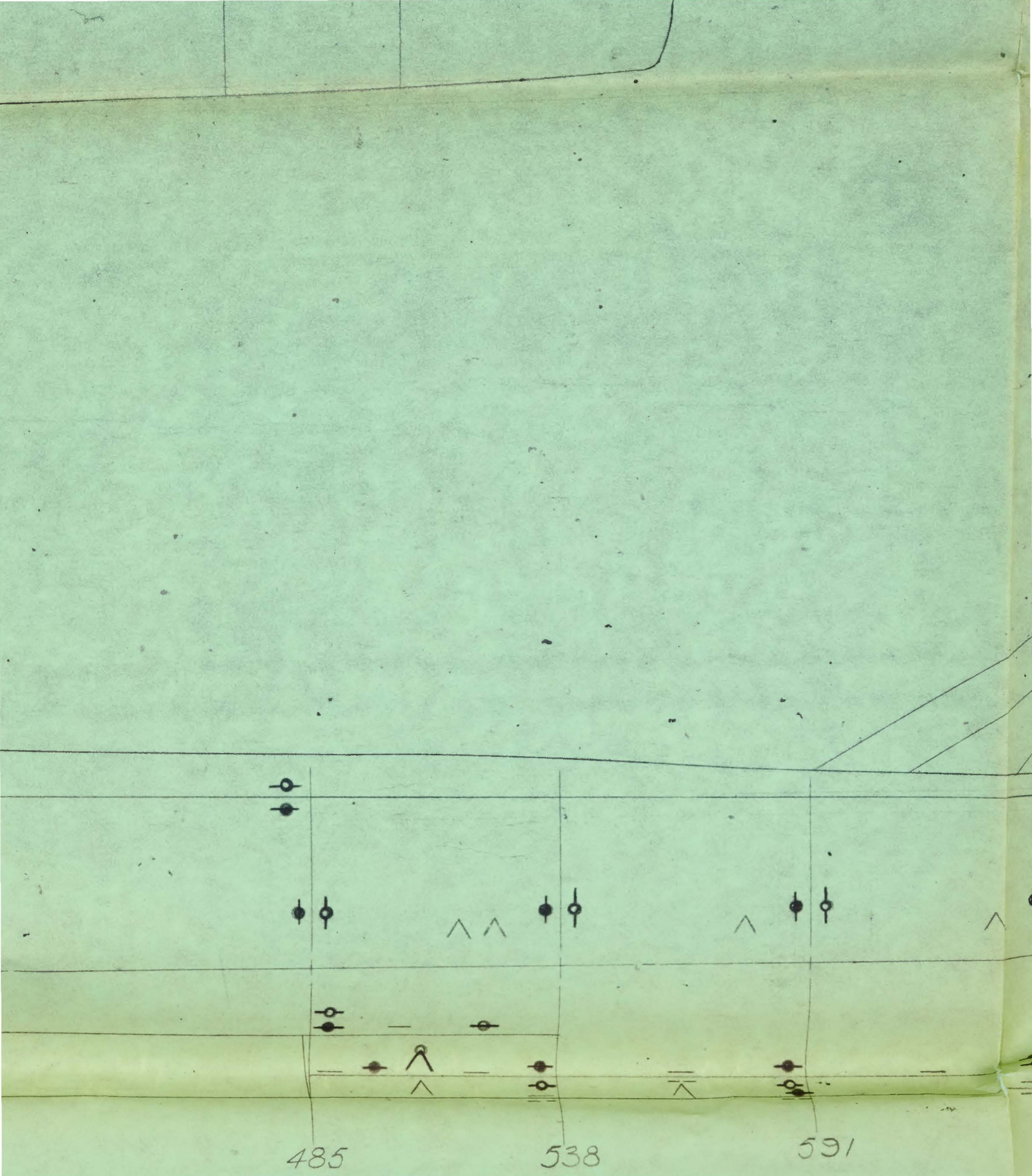
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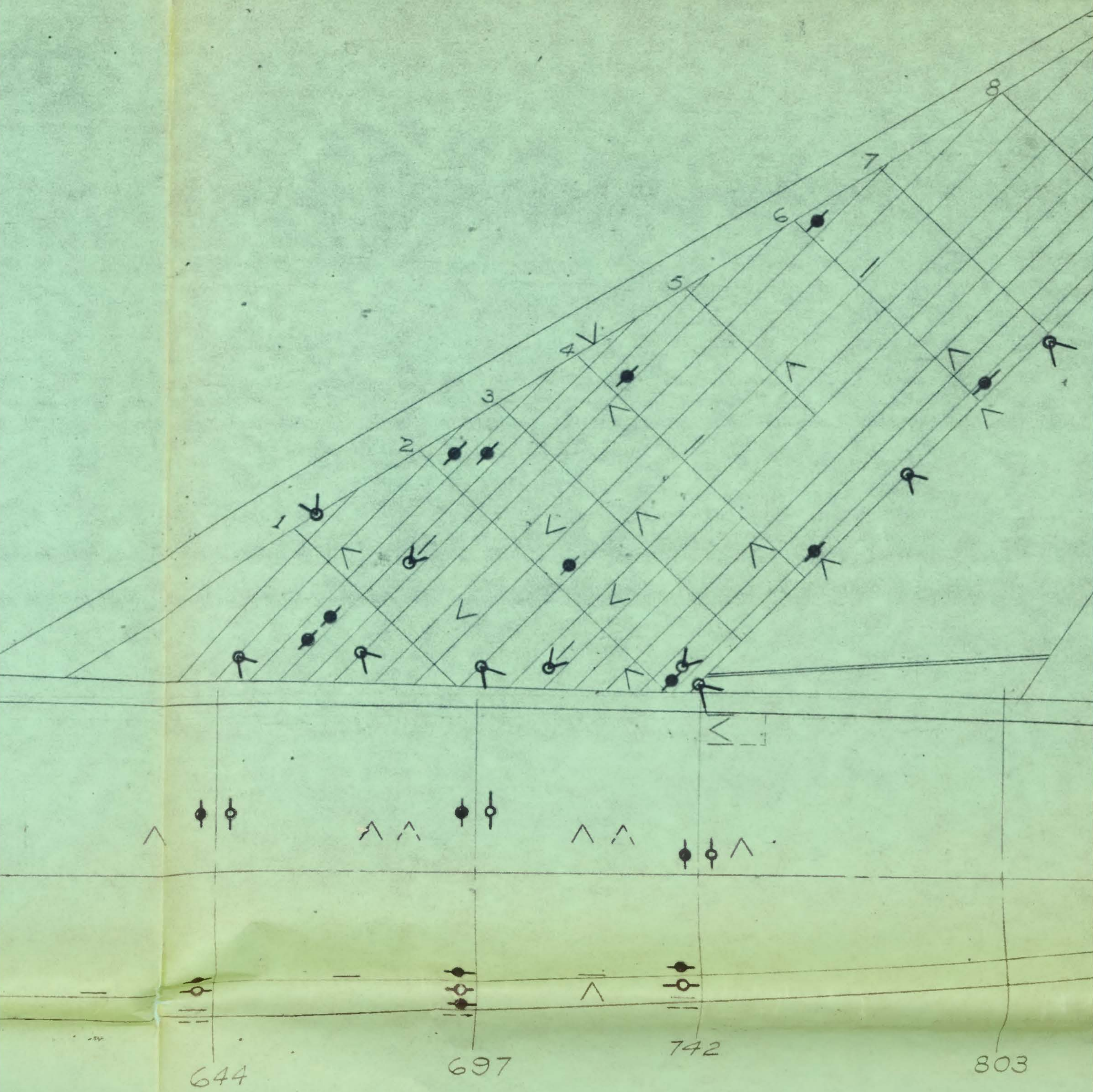
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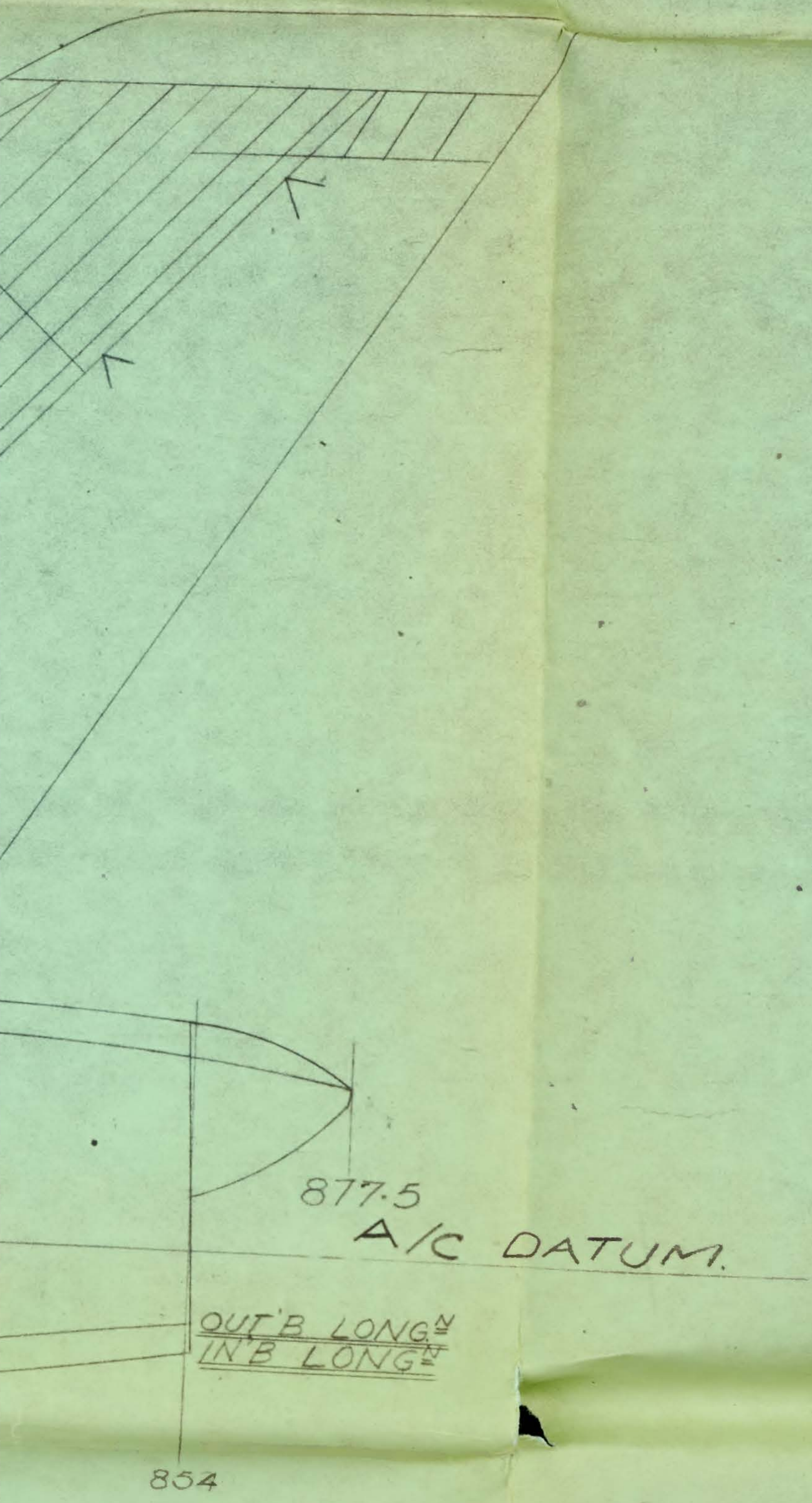

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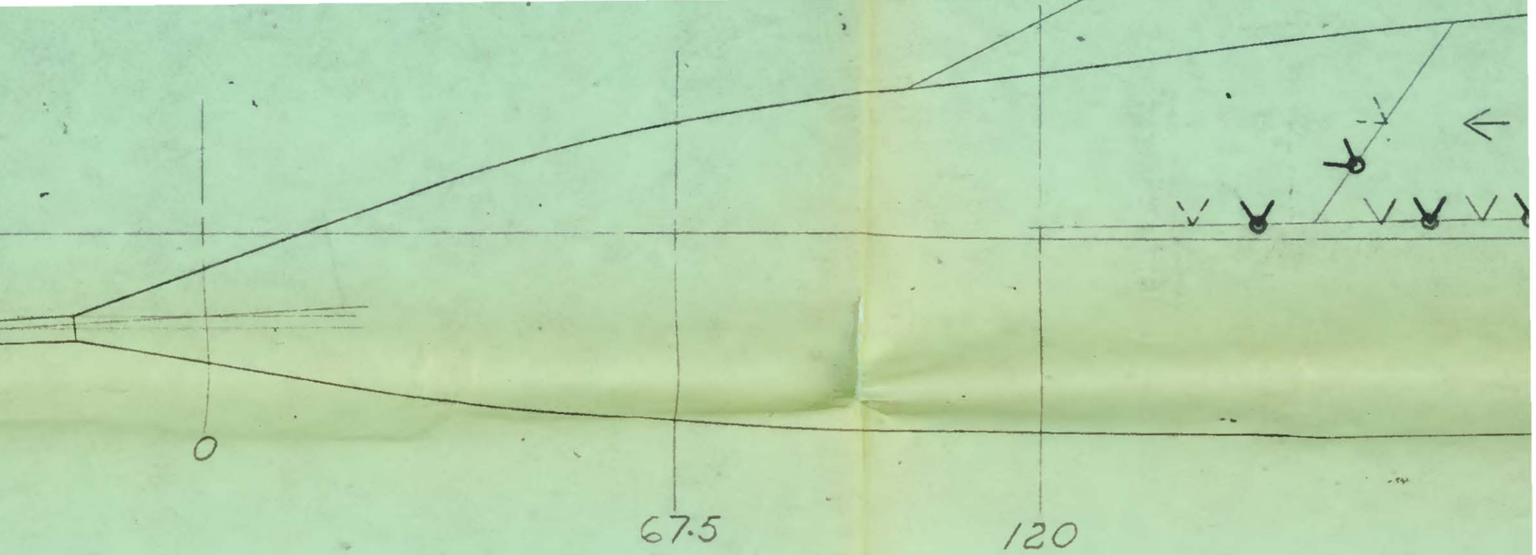


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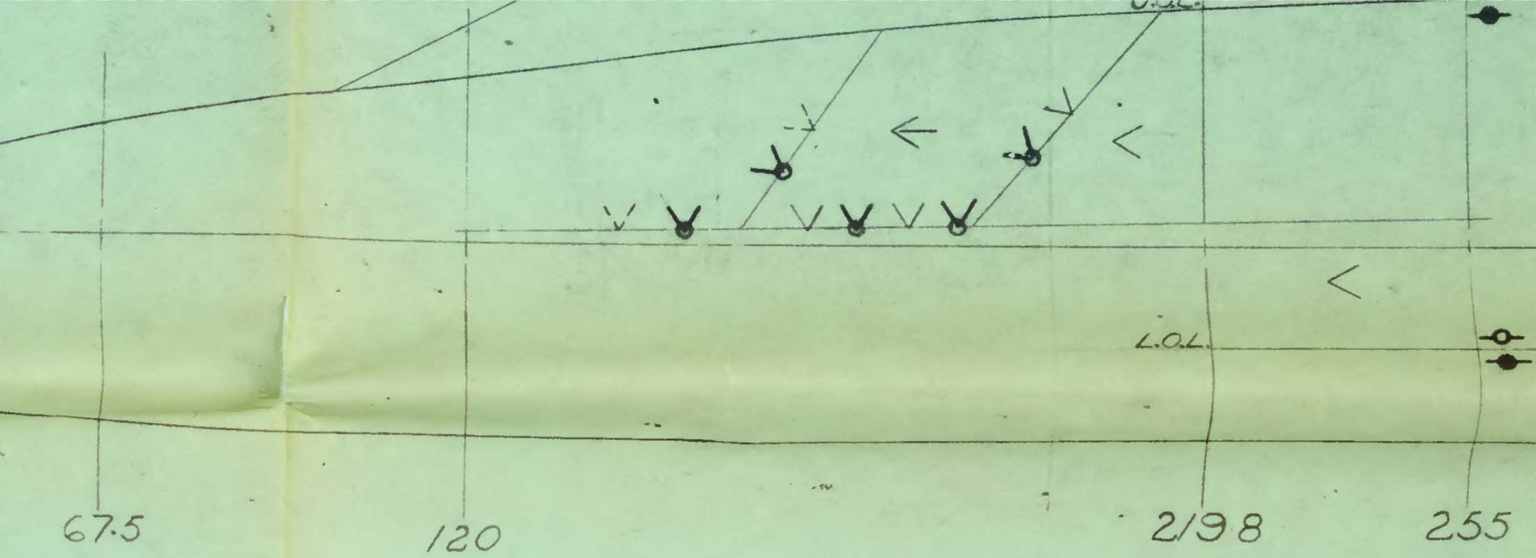
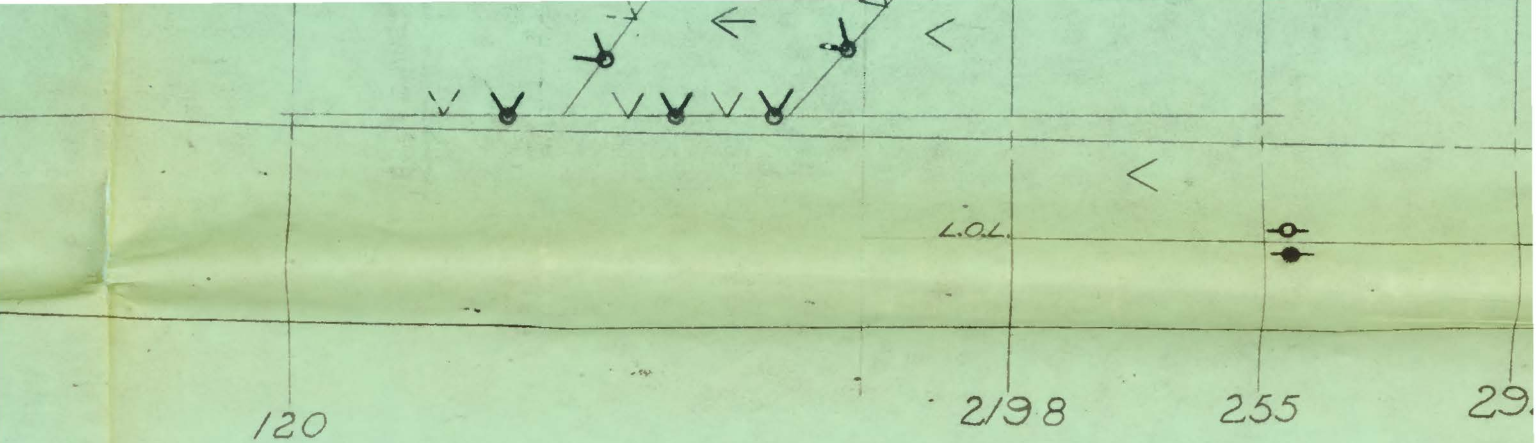
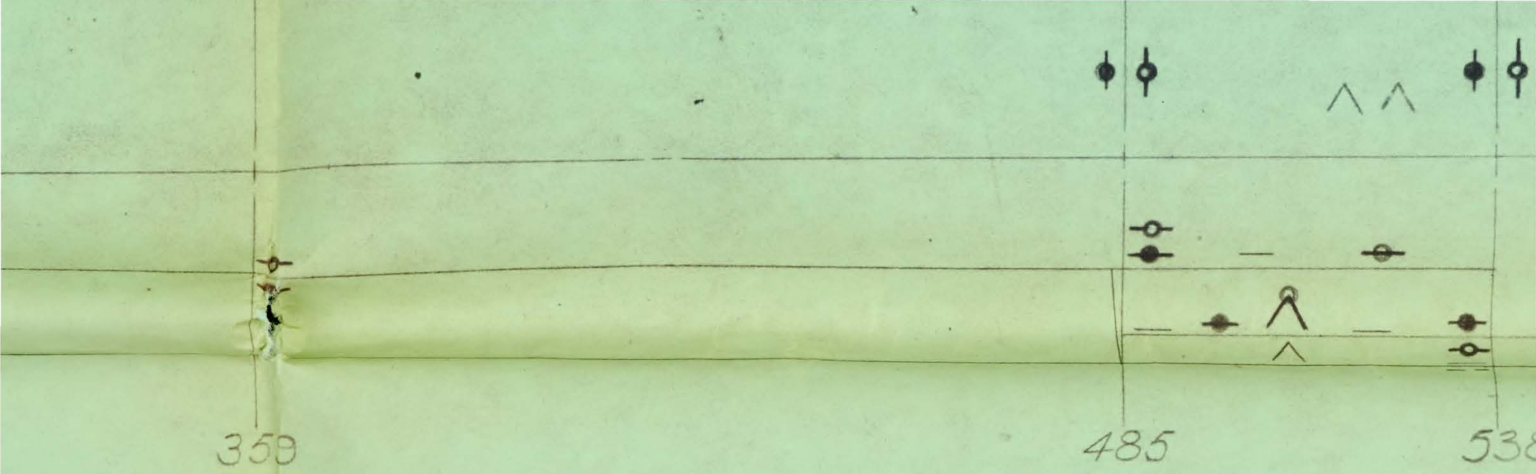



SIDE ELEVATION

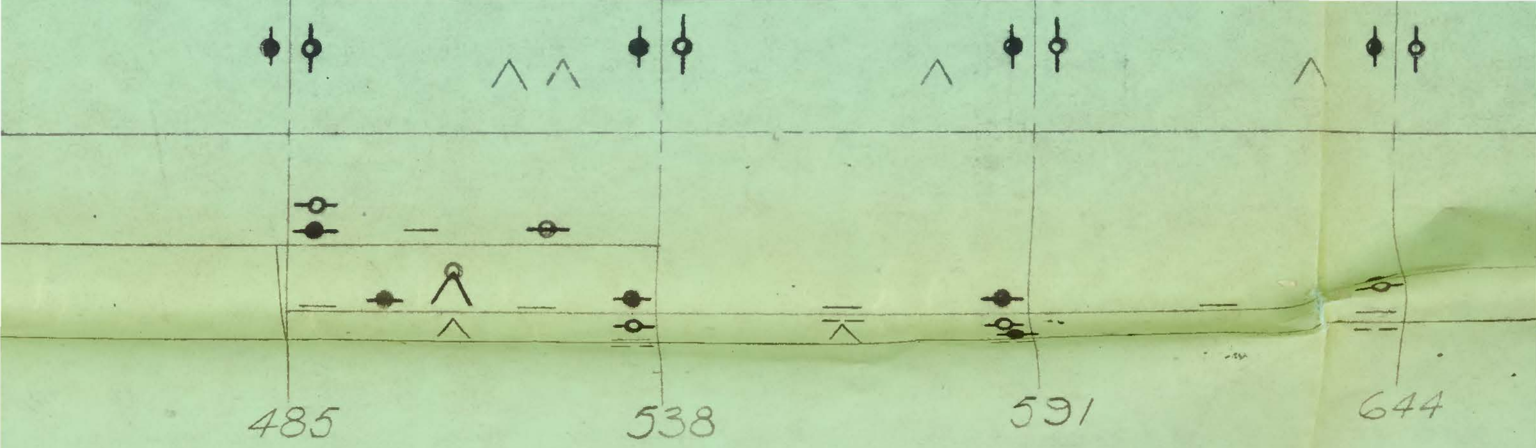



## SIDE ELEVATION


| METHOD 'A'. | COMPONEN |
|---|---|
| | F'WD FUSELA |
| 400 GAUGES | AFT FUSELA |
| BASIC SYSTEM + BACK-UP SYSTEM | INNER WIN |
| UPPER SKIN / LOWER SKIN | INNER WING |
| LOWER SKIN | OUTER WING |
| | OUTER WING |
| UPPER SKIN | FIN. |
| | TOTAL GAU |




| METHOD 'A'. | COMPONENT | GAUGES. |
|---|---|---|
| | FWD FUSELAGE. | 40 |
| 400 GAUGES | AFT FUSELAGE. | 60 |
| BASIC SYSTEM + BACK-UP SYSTEM | INNER WING L.H. | 160 |
| UPPER SKIN<br>LOWER SKIN | INNER WING R.H. | 40 |
| LOWER SKIN | OUTER WING L.H. | 45 |
| | OUTER WING R.H. | 15 |
| UPPER SKIN | FIN. | 40 |
| | TOTAL GAUGES. | 400 |




| | |
|---|---|
| ONENT | GAUGES. |
| USELAGE. | 40 |
| USELAGE. | 60 |
| WING L.H. | 160 |
| WING R.H. | 40 |
| WING L.H. | 45 |
| WING R.H. | 15 |
| | 40 |
| L GAUGES. | 400 |

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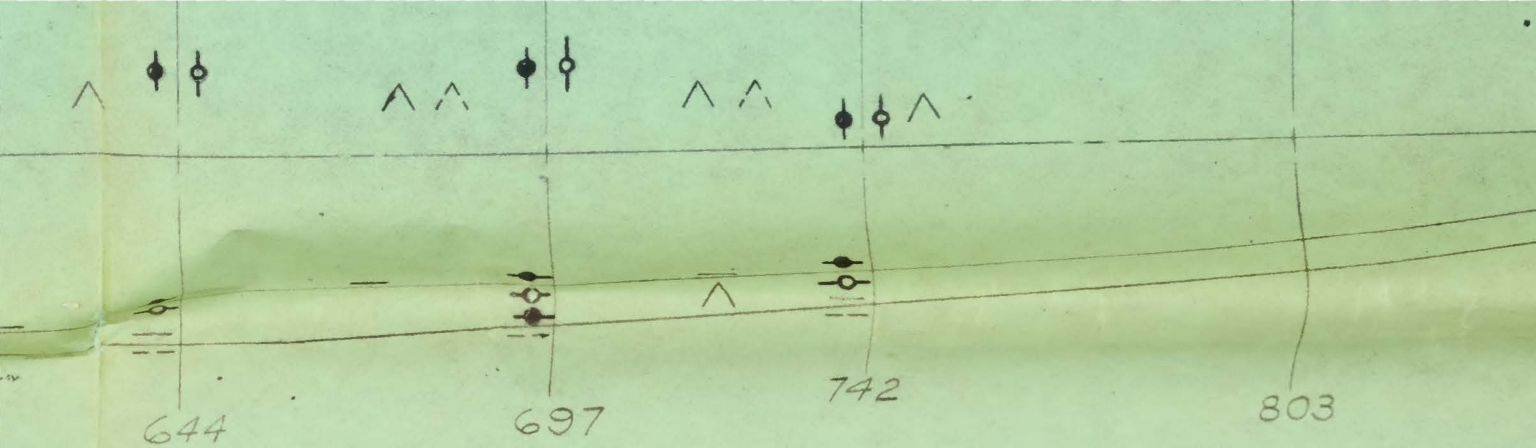
| ISSUE No. | 1 | |
| DRAWN BY | D.ROY | |
| DATE | 5 FEB 58 | |

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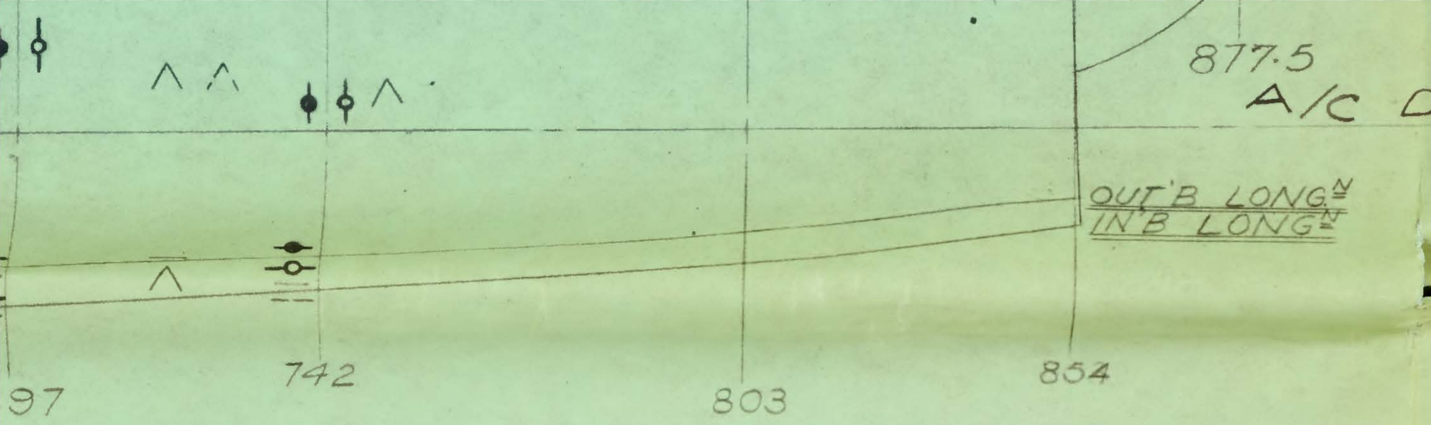


A/C - 25

ARROW 2. STRU

| REFERENCE DRAWINGS | |
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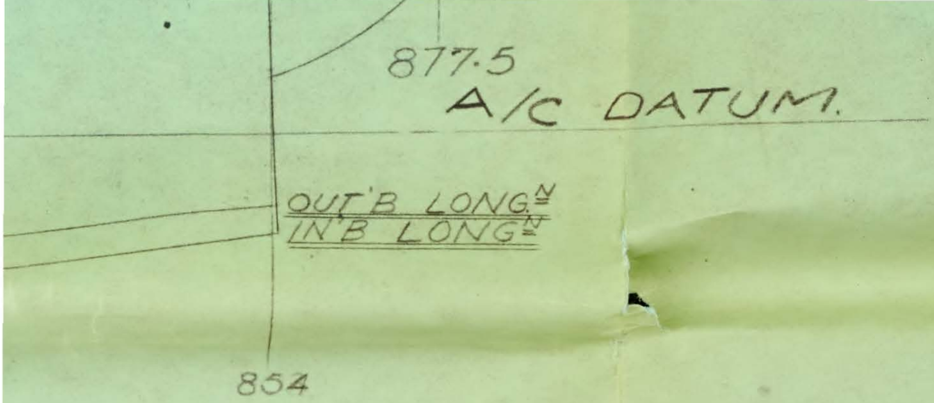


# A/C - 25211.

## ARROW 2. STRUCTURAL INTEGRITY

| REFERENCE DRAWINGS | |
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| DWG. NO. | DESCRIPTION |
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AVRO AIRCRA
MALTON

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|---|---|
| CLASS | MISCELLANEOUS R |
| DESCRIPTION | FLIGHT TEST STRAIN G |
| GROUP | |
| COMPONENT | AIRCRAFT COMPLET |

TEGRITY PROGAM.

| | | | |
|---|---|---|---|
| RO AIRCRAFT LIMITED<br>TON ONTARIO | | | |
| LANEOUS REF. DWG. | SCALE 1/24 | ARROW. 2<br>TYPE | NEXT ASSEM |
| TEST STRAIN GAUGING | FINISH TO DWG. | DRAWING NO. | |
| | LIMITS UNLESS NOTED | 7-4400-15007 | |
| FT COMPLETE. | | | |

~~RESTRICTED~~