THE AVRO ARROW

AVRO
CANADA

To Doug & Donette
with my best
regards.

A BOOK BY

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THOMAS B DUGELBY

CHAPTER 4

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DESIGN. 4

AREA RULE ON THE C-105.

In response to an RCAF directive in Sept./54 that Avro proceed with all alacrity on the application of Area Rule in respect to the design of the C-105 in order to investigate the drag, Avro accordingly carried this out in four phases:

(a) Investigation of numerical procedure.

(b) Model program.

(c) Application of Method to Drag Reduction of C-105 Aircraft at the Design Mach Number.

(d) Comparison with experiment.

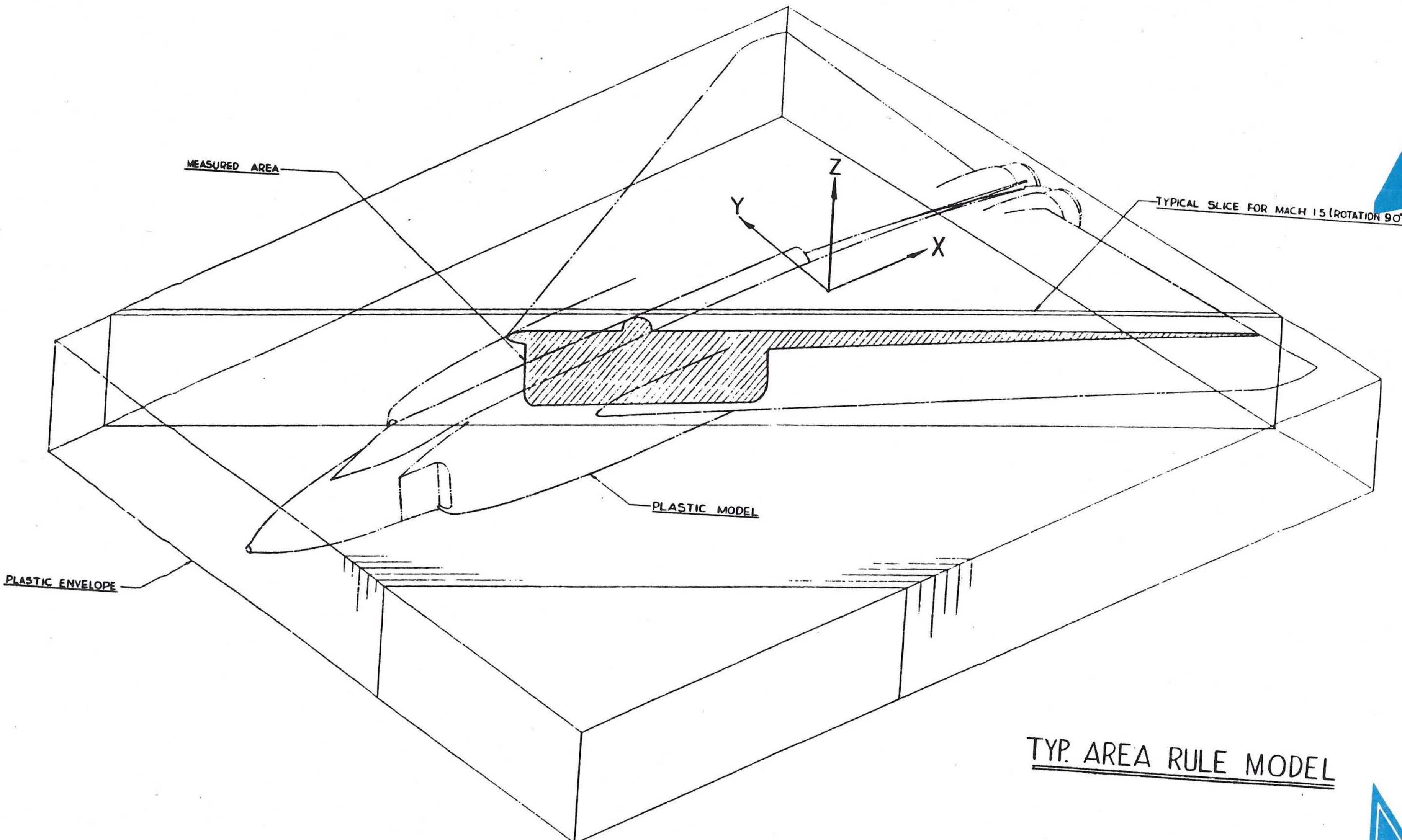
METHODS AND RESULTS.

It was found that the result depended very strongly on the methods used, both in the determination of the area distribution of the models and in the numerical calculations. Results differing by as much as a factor of 5 could be obtained using different methods on wing - body combinations having high slopes in the area distribution curves. Also, if the method of approximating the slope of the area distribution by a Fourier series is used to calculate the drag, the result depended quite strongly on the number of terms in the series as well as the degree of accuracy obtained in representing the area distribution on the model.

The wave drags calculated by this method have been used as a basis for suggesting possible changes to reduce the drag of the C-105 aircraft at the design Mach number. By means of two small modifications to the aircraft lines in the regions of the intakes and the rear nacelles, and the addition of a larger fairing between the tail pipes, a drag reduction of 37% was realized. Comparison of these results with optimum bodies of the general configuration was also carried out.

MODEL PROGRAM.

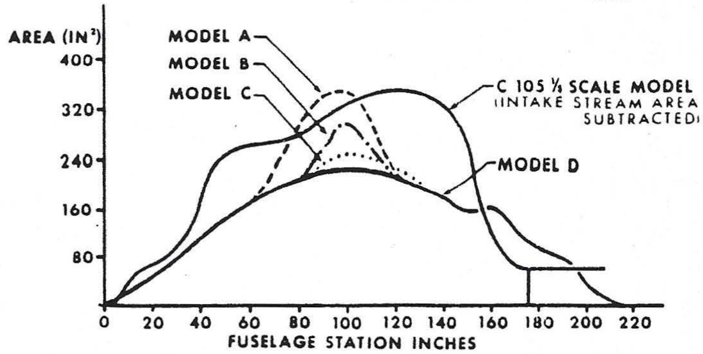
The model program consisted of taking an accurate .03 scale model of the aircraft and making plastic copies of it. These copies, painted black, were set on dowels to predetermined locations within a box and in turn encased in plastic. This resulted in a rectangular block of plastic which was conveniently handled. The cut lines desired were scribed on the surface of the block and cutting was carried out on a bandsaw



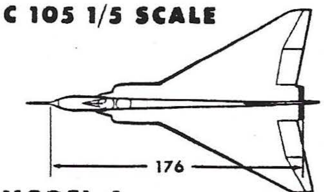
TYP. AREA RULE MODEL

METHOD OF CASTING MODEL AND MAKING CUTS

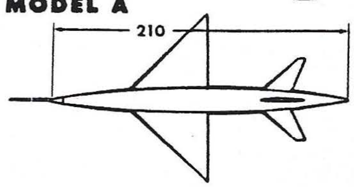
CROSS-SECTIONAL AREA DISTRIBUTIONS OF MODELS



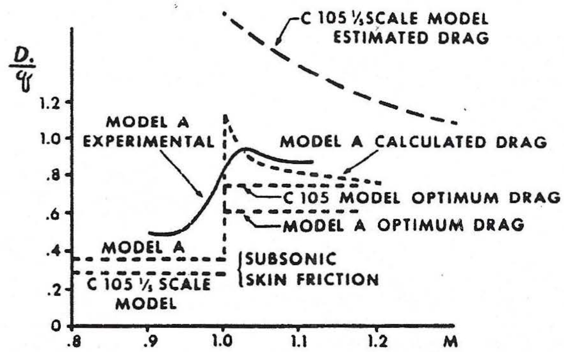
C 105 1/5 SCALE



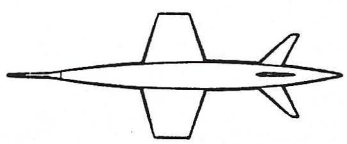
MODEL A



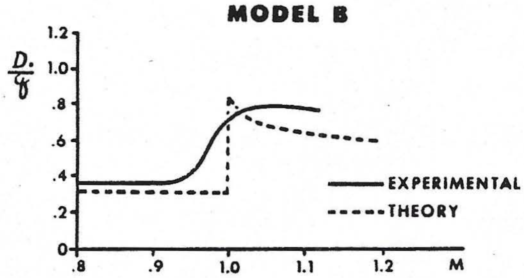
MODEL A



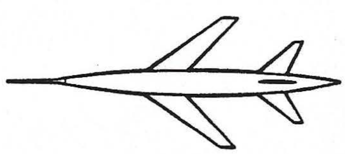
MODEL B



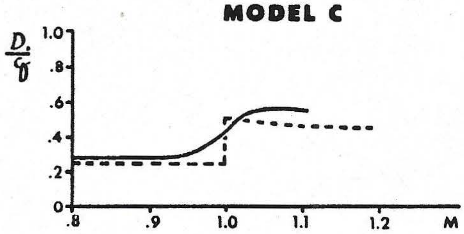
MODEL B



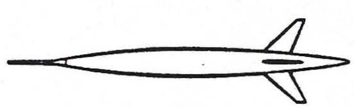
MODEL C



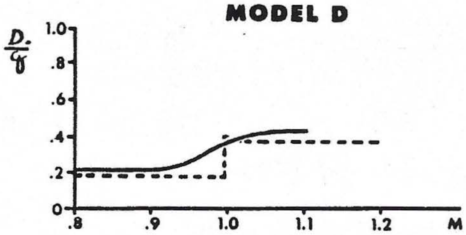
MODEL C



MODEL D



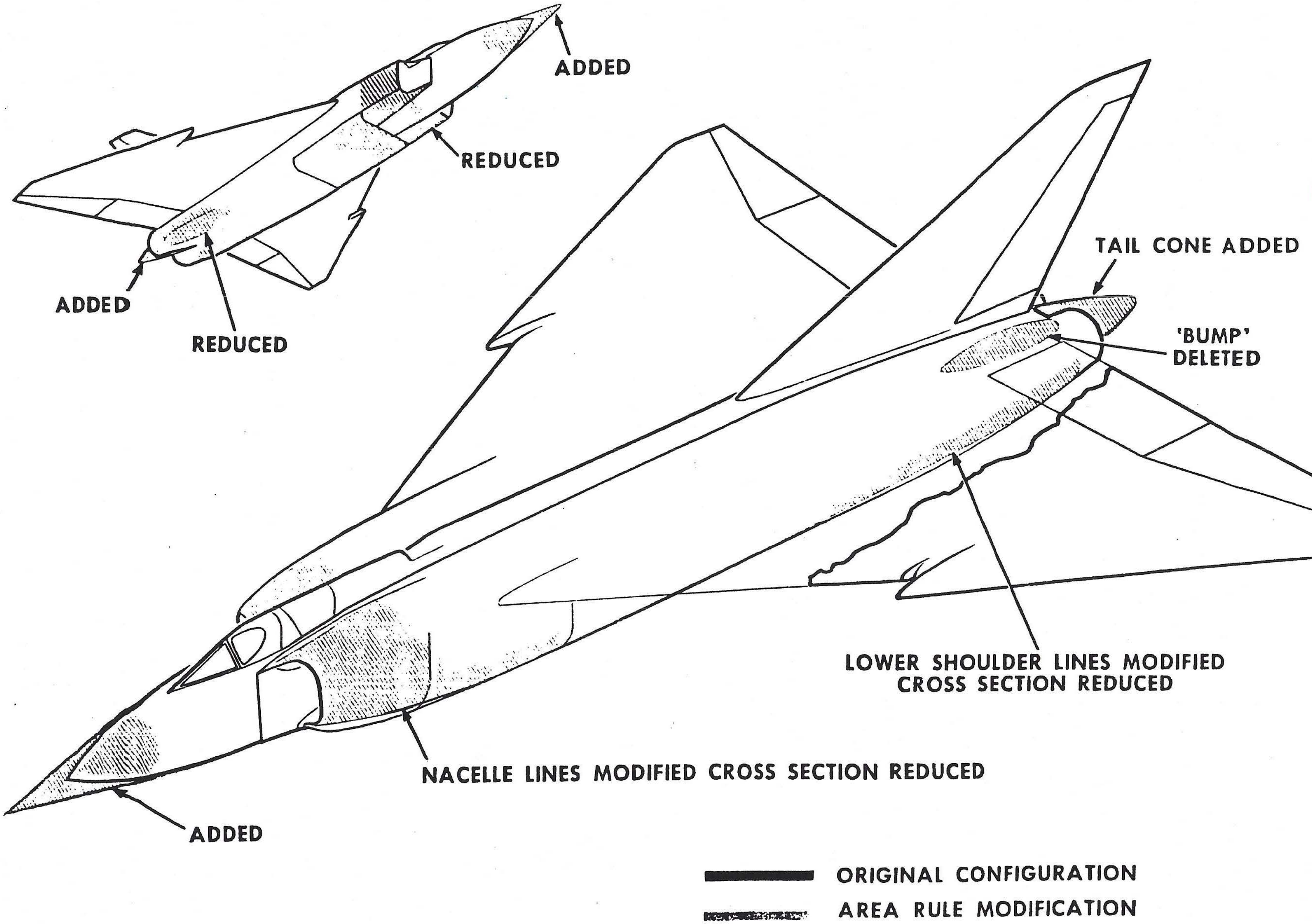
MODEL D



HOLDAWAY'S RESULTS

(N. A. C. A. RM A 53 H-17)

(NOTE:- 24 TERM FOURIER SERIES USED)



APPLICATION OF AREA RULE TO THE C-105

to these lines. After some practice very good results were obtained. The required outline of the embedded model in each slice was sharply defined by the paint line. This area was measured by means of a planimeter.

In the regions where the slope was changing very rapidly, 1/16 scale models were made and the procedures repeated. This gave much closer estimates of those critical areas. In the case of the fairing between the jet tail pipes, owing to the lack of suitable models, geometrical layouts were made to determine the sections. This latter technique was considered to be practical only as an interim phase in preliminary drag estimates, owing to the difficulty of laying out accurate oblique sections of complex shapes.

Illustrations of this work showing models, cut slices, area measurement and diagrams of the findings and modifications are shown.

APPLICATION TO C-105 AIRCRAFT.

Typical area distribution curves for the C-105 aircraft are shown for M = 1.50. The area of the stream tube entering the intakes has been added to the fuselage area in front of the ducts, and an area corresponding to the jet exhaust area has been added to the rear. These areas were calculated to be 11 sq.ft. and 22.4 sq.ft. respectively, at M = 1.50.

The curves shown have had modifications made to the intake lips and a larger fairing added between the tail pipes. The drag coefficients, based on the wing area, are given as a function of the angle of rotation both for the original and modified fuselages.

LOW TRANSONIC DRAG.

Although it is not immediately apparent, the Avro Arrow front fuselage does possess a degree of "coke bottle" shape, as defined in NACA research by Richard Whitcomb, a NACA scientist.

During the middle of the Second World War, aerodynamic engineers were becoming concerned at the lack of knowledge of the transonic region, between M = 0.8 to M = 1.2. This lack of knowledge resulted in quite some number of early jet aircraft being much slower than they should have been, and also being completely unstable, by exhibiting tendencies to roll, climb and dive and be subject to buffeting. NACA discovered that the "wiasting" or "coke bottle" effect greatly improved airflow, and tended to decrease the above objectionable tendencies.

AERODYNAMICS.

WIND TUNNEL TESTS.

Perhaps the most important aspect of the design of an aircraft, is the testing of models in various wind tunnels. To these ends, Avro conducted both high and low speed testing in the wind tunnels at NRC, Ottawa, Cornell Aeronautical Laboratories and NACA Langley, the latter two being in the United States.

THE TESTS.

The first tests were run in September of 1953, at Cornell on a .03 scale model over a Mach range of 0.5 to 1.23. This was a comparatively short program of some 215 runs constituting a preliminary check on longitudinal stability and control to prove the design and to provide basic aerodynamic data. Two wings were tested, one having a conventional 3% thick symmetrical section, on which control investigations were carried out, and the other with 0.75% negative camber. Negative camber had been shown theoretically to have a considerable advantage over zero camber in reducing up elevator angles to trim and, therefore, drag, but there was some evidence to show that the positive CMo introduced might exhibit some unacceptably large variations at transonic speeds. The tests however showed that negative camber was both feasible and desirable, and also that the aircraft had adequate longitudinal stability and control.

The next series of tests, again at Cornell, were made in April 1954. The same .03 scale model was used with minor changes, namely an increase in wing thickness from 3% to 3 1/2%, the incorporation of elevator and ailerons on the cambered wing, and the replacement of the original intake shock plates with shock ramps. A complete program of longitudinal, lateral and directional stability and control investigations were carried out.

In addition, a pressure survey of 20 taps in the fuselage was made and data obtained on fin and fuselage speed brakes and the effect of the belly tank. Again the Mach range was 0.5 to 1.23 and the tests covered some 450 runs.

From this series the fuselage brakes were found to be superior to the fin mounted brakes, having better braking action and producing less undesirable side effects, and valuable control information was obtained. The results generally were gratifying with the exception of directional stability. This proved to be unsatisfactorily low and to be peculiarly non-linear.

The third series of tests, in June 1954, was aimed primarily into finding the reasons for the poor directional stability. Faired ducts, a dorsal fin, the removal and modification of the canopy and the effects of sealed control surface gaps were all tried with no significant improvement being gained. In addition a 12 tube rake survey of internal static and dynamic pressures was made in the ducts to determine mass flow and aid in the correction of drag estimates. This series covered 252 runs.

Meanwhile directional stability was raised to an acceptable level by increasing the vertical tail area by 15%. The non-linear aspects still persisted and since the tests above had failed to find the cause it was more or less accepted as inherent in the design.

The next tests , at Cornell in July of 1954, were run in a 10ft by 12ft subsonic section at a Mach number of 0.5 only. This was mainly an investigation into stability and control at high angles of attack (up to 40 degrees). Previous tests had shown that a moderate amount of pitch up occurred at a CL of 0.7 and in an attempt to improve this, several notches were tried in the wing leading edge at the transport joint. An optimum configuration was first found and used in subsequent runs. The effect of these notches on lateral and directional was then checked. At the same time a high Reynolds number run in yaw was made in an unsuccessful final attempt to find if the Reynolds number was causing the non-linear directional stability. These tests showed no adverse characteristics at high angles of attack and resulted in a notch configuration which delayed the onset of pitch up to higher values of CL. 74 Runs were made.

At about this time information came to light that significant improvements in pitch up characteristics had been obtained on test models by extending the outboard wing leading edge. Information was meagre and a large variety of possible combinations of extensions and notches made the determination of an optimum configuration for the C-105 difficult. This was the main purpose of the fifth series of tests at Cornell in October of 1954. At low speed a variety of notches and extensions were tested and an optimum established. Most of the remainder of the test was devoted to checking this configuration through the Mach range of 0.5 to 1.23. During this period one aileron deflected runs were made, with increased balance sensitivity, to determining aileron cp.; this had been attempted in an earlier series but without conclusive results.

Several more high Reynolds number runs were also made in yaw to check the effect of a new longer nose on directional stability. This series of 216 runs established a new wing plan form, with a 10% outboard leading edge extension plus a 5% transport joint notch, which was effective in improving pitch up.

Next followed a series of armament tests. Since these required instrumented missiles a larger scale model was necessary and was built to .04 scale. The first phase of this series was begun in March of 1955 and consisted of an investigation into forces on Sparrow and Falcon missiles in up, half down and launch positions, together with the collection of data on armament bay pressures and door hinge moments. These tests were made at Mach numbers of 0.95 and 1.20 only and covered 64 runs. The second phase of 46 runs, was a study of the effects of the missiles on an aircraft. The missiles were again in the up, half down and launch positions and force data was taken on the aircraft to evaluate the effects of lowering the missiles in flight.

The third phase of 30 runs, was made to check the correlation between the .03 and .04 scale models. Stability and hinge moment data were obtained over the Mach range. During this test, an attempt was made to find values of the rather elusive CL buffet by reading pressures from two pressure taps on the upper surface of the port aileron. These showed a sudden increase in pressure at the angle of attack when separation occurred, and gave an indication of the onset of buffet.

A second series of armament tests began in April of 1955. These were to determine missile characteristics for trajectory purposes. Both Falcons and Sparrows were tested at four longitudinal positions along the fuselage, at each of which the missiles were rotated through small angles of pitch and yaw. Small strain gauges mounted inside the missiles were used to measure the forces at Mach numbers 0.95 and 1.20. The program took 120 runs.

Early in 1955 it was thought possible that the incorporation of leading edge droop could materially improve the drag due to lift. As in the case of notches and extensions a large number of configurations were possible. There were indications that the results would be sensitive to small changes in droop angle and to the combination and extent of droop inboard and outboard of the transport joint. From NACA reports it appeared that inboard droop was very beneficial but should be confined to a smaller fraction of the chord than the outboard. The plan form of the extent of the drooped leading edge was decided and a program initiated to test the effects of all possible combinations of four outboard and two inboard droop angles. This program was started in May of 1955. First the optimum configuration was chosen and once this was done a complete stability and control check was made over the Mach range. This rather lengthy program of 412 runs, had the desired result of reducing drag due to lift and led to revised stability and control data. One rather fortuitous effect was a considerable improvement in the previously non-linear directional stability. This was probably caused by improvement of the flow originating at the wing-nacelle junction due to the new inboard droop.

In November of 1955 an extensive low speed series of tests were started in the No.3, 8ft by 10ft tunnel at the NAE. These tests continued in May of 1956 and the program was completed in August of that year. Altogether 181 runs were made and covered longitudinal, lateral and directional stability and control, and investigated the effects of ground board, tank, dive brakes, undercarriage, open canopy, Reynolds Number and control interference. Instrumentation consisted of a six component main balance only.

Meanwhile to obtain supersonic data two models were tested in the 16 in. by 30 in.

high speed tunnel at the NAE. The first was a .02 scale reflection plane model and was tested in February of 1956. 177 Runs were made at Mach numbers up to 2.03 to obtain basic longitudinal stability and control data and duct pressure measurements. Results did not agree very well with Cornell data in the range 1.02 to 1.23. This was thought to be due to the fact that a half model was used; correlation of reflection plane and full model tests at NACA also showed poor agreement.

The second model, of 0.0125 scale, was a full model and sting mounted. This was tested in May and August of 1956 and gave supersonic longitudinal, lateral and directional stability and control data. The Mach range was 1.35 to 2.03 and the tests covered 177 runs.

To obtain supersonic data on a fairly large scale model, tests were proposed at RAE Bedford, and a new .03 scale model was built by Cornell. Arrangements could not be finalized but an alternative facility became available in the 4ft by 4ft supersonic tunnel at NACA Langley. 16 Runs were made there in April of 1956 at a Mach number of 1.41 giving longitudinal, lateral and directional stability and control data. These tests were later extended to Mach numbers of 1.6, 1.8 and 2.0 by testing in the 4ft by 4ft Unitary tunnel at Langley in July 1956 in a series of 97 runs.

Typical report sheets are reproduced here to illustrate the work required. Also, a list of the meanings of the model configurations is included in order togain a better understanding of the variations of the model configurations.

Example A.

SERIES 1. .03 Scale model.- September 1953.

Facility...................3ft x 4ft transonic tunnel at Cornell.
Purpose.................Longitudinal stability and control investigations including effects of camber. Runs made at Mach=0.5 in the horizontal position.
Configuration.........B1 C1 W1 W2 V1 P5
Instrumentation.......6 component main balance
1 hinge moment balance (left elevator)
1 internal static pressure tap in balance chamber.
Control deflections..Elevator 10, 0, -5, -10, -20, -30
Aileron None
Rudder None
Mach range.............0.5 to 1.23 (RN 1.23 to 1.84 x 1,000,000)
Runs.......................1 to 215

Example B.

Period 1 - Phase 1.-.04 Scale model. - March 1955.

Facility..............3ft X 4ft Transonic tunnel.

Purpose.............An investigation into forces on Sparrow and Falcon missiles, armament bay pressures and bay door hinge moments. Missiles were tested in the up, half down and fully down positions, and in the case of the Falcons, with various combinations of forward and aft missiles. Runs were all made in the horizontal position with zero yaw and at only two Mach numbers.

Configuration..........Aircraft: B5, C3, W0, N8, V3, Rs

Missiles:

A1. S-FU, S-HD, S-FD

A2. FF-FU, FF-HD, FF-FD

A3. FA-FU, FA-HD, FA-FD

Instrumentation........1. Sparrow.

Two 4 component missile balances.

Three door hinge moment balances.

14 Pressure taps in armament bay.

2. Falcons.

Four 4 component missile balances.

(Only two used at any one time).

Eight door hinge moment balances.

18 Pressure taps in armament bay.

In addition.

Two upper port aileron pressure taps.

One internal static pressure tap in balance chamber.

Two component main balance (For normal forces).

Control Deflections..None - No provision made.

Mach range.............................0.95 and 1.20 only.

Runs......................................1 to 63

The following is the meaning of the missile codes used in the above report:-

A1, A2, A3....Series of test.

S-FU.............Sparrow missile full up.

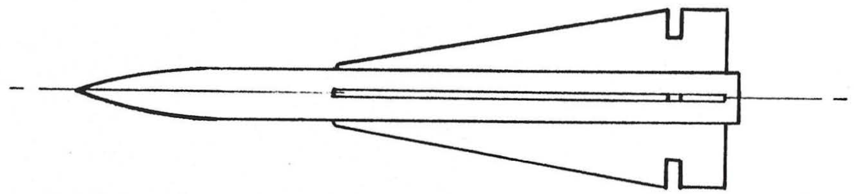
S-HD............Sparrow missile half down.

S-FD.............Sparrow missile full down.

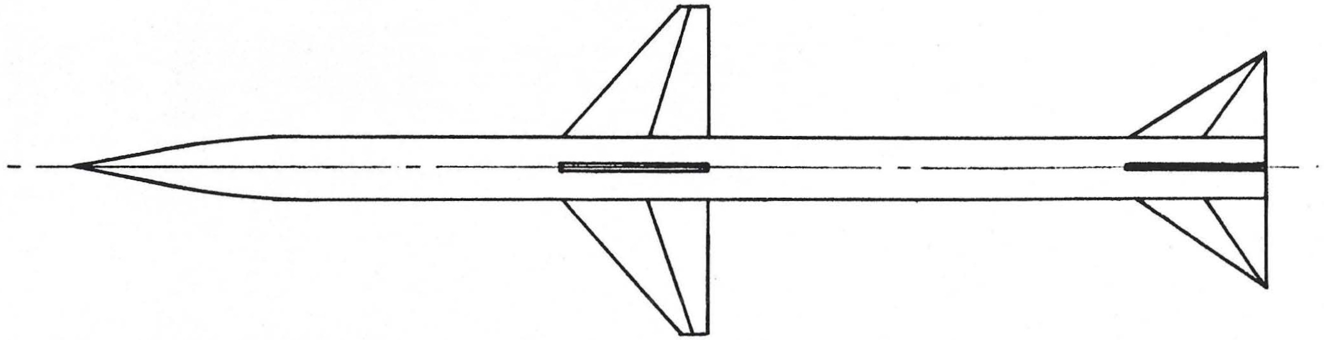
FF-FU...........Falcon missile, full forward and full up.

FF-HD..........Falcon missile, full forward and half down.

FF-FD...........Falcon missile, full forward and full down.

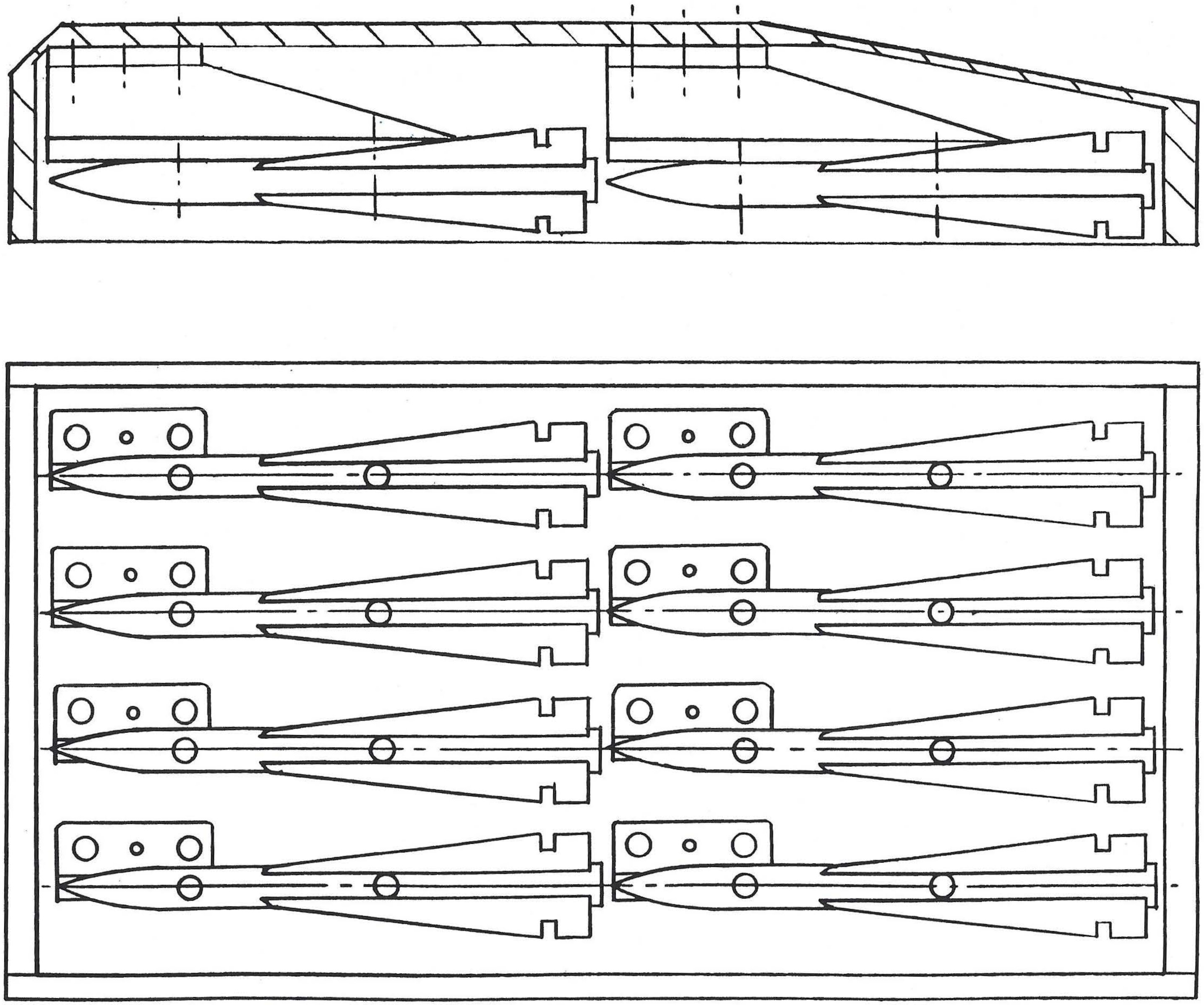


FALCON GAR-1

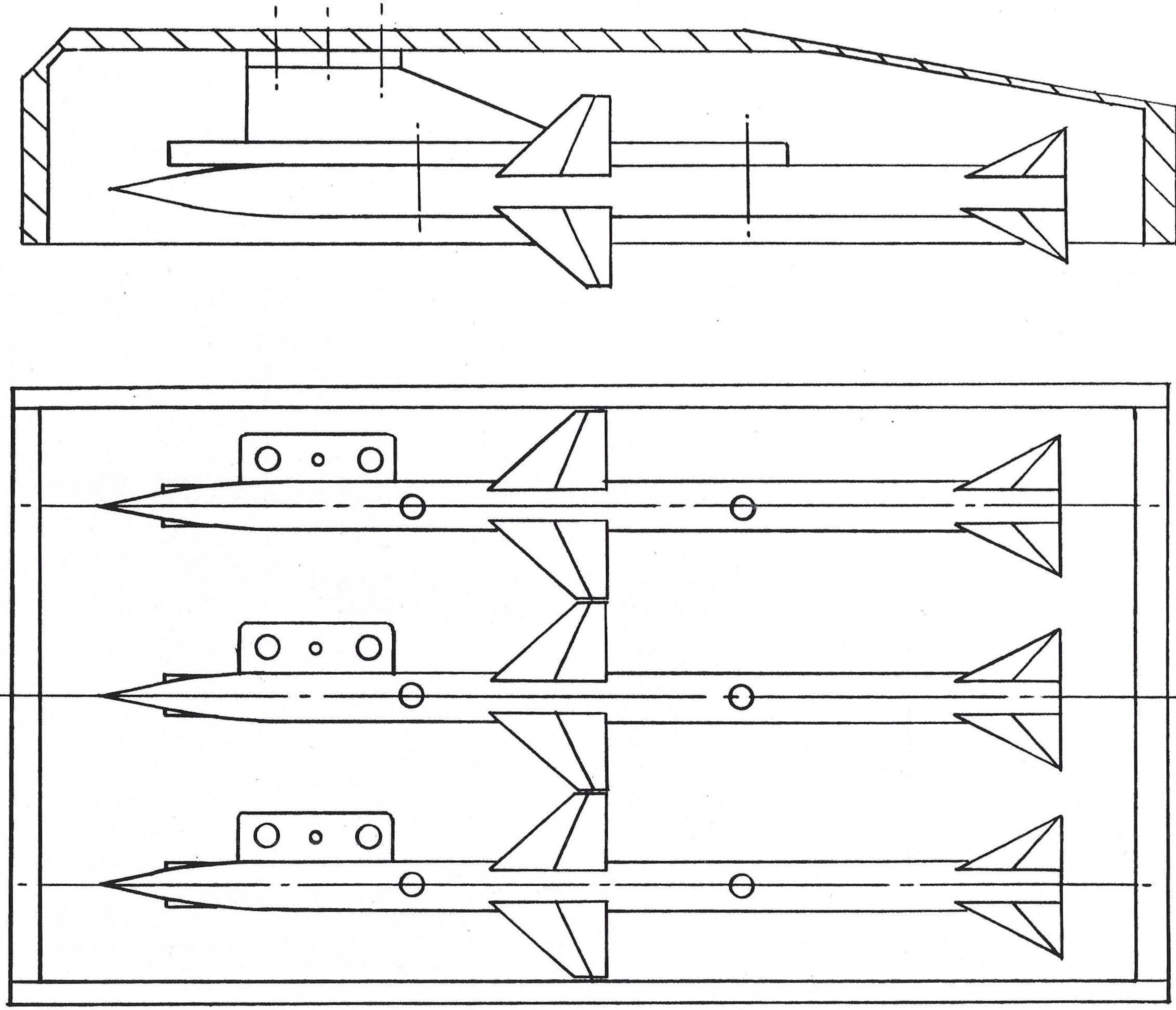


SPARROW 2

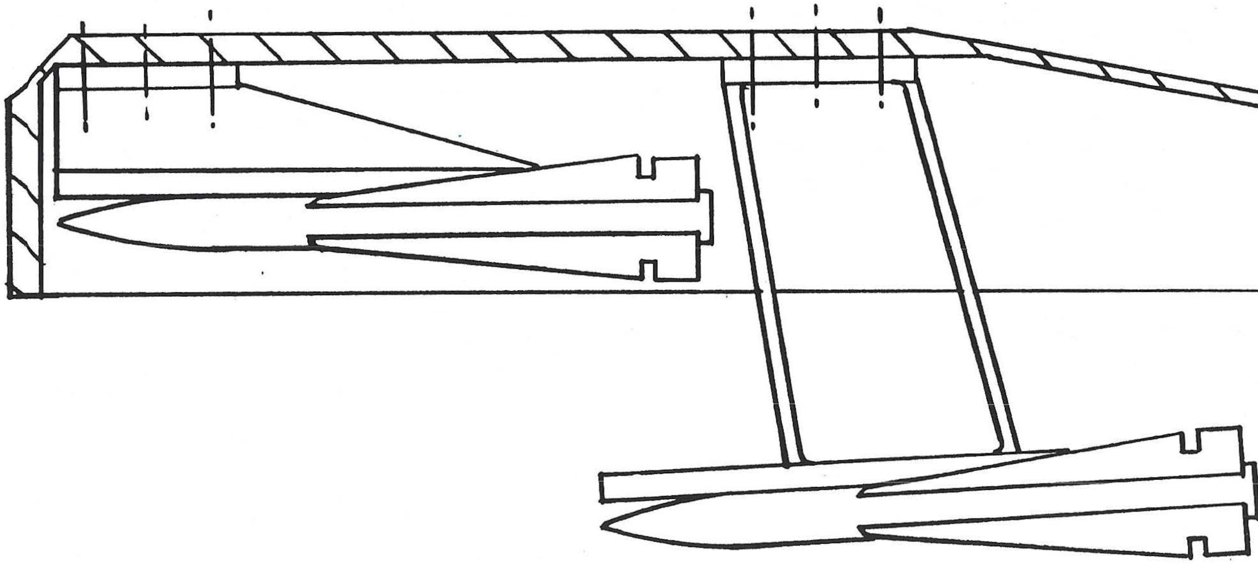
1/24 SCALE MISSILE MODELS USED IN THE WIND TUNNEL TESTS



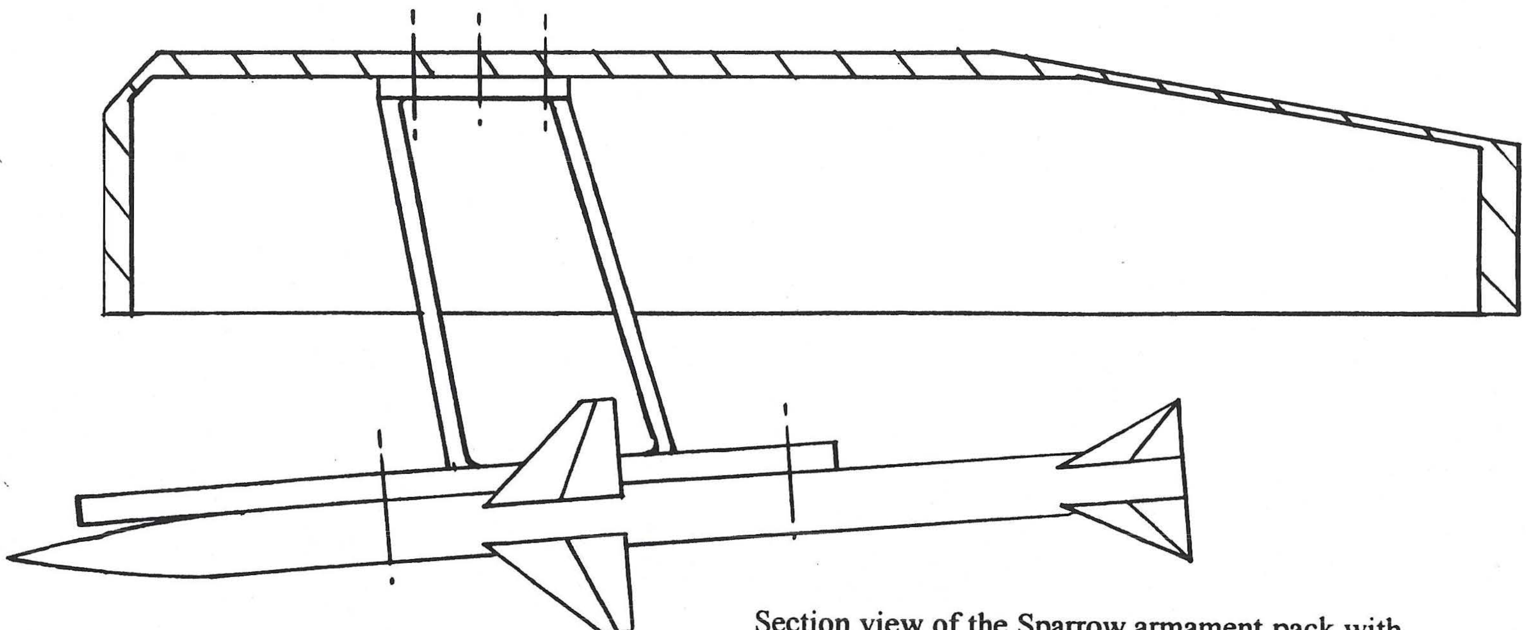
Bottom view and section showing a fully loaded armament pack with Falcon missiles in the full-up position. This pack was then bolted to the 1/24 scale model for wind tunnel testing.



Bottom view and section of a fully loaded armament pack with three Sparrow missiles in the full-up position. This pack was then bolted to the 1/24 scale model for wind tunnel testing.



Section view of the Falcon armament pack with the rear four missiles in the full-down position.



Section view of the Sparrow armament pack with the three missiles in the full-down position.

FA-FU.........Falcon missile, full aft and full up.
FA-HD.........Falcon missile, full aft and half down.
FA-FD.........Falcon missile, full aft and full down.

Example C.

Definition of configurations used before May 1955.

Body

B1....................Original body including ducts.
B2....................B1 with modified ducts.
B3....................B2 with modified rounded nose, 10" longer.
B4....................B2 with longer nose of similar shape, 5" longer.
B5....................Redesigned body.

Canopy

C1....................Original canopy.
C2....................C1 in new position.
C3....................New larger canopy.

Wing

W1....................3% un-cambered wing with elevators.
W2....................3% cambered wing - no controls.
W3....................3 1/2% cambered wing with controls.
W4....................W3 plus 6 1/2% notch (A series).
W5....................W3 plus 8% notch (A series).
W6....................W3 plus 10% notch (A series).
W7....................W3 plus 5% leading edge extension.
W8....................W3 plus 8% Leading edge extension.
W9....................W3 plus 10% leading edge extension.

Note that notches on W7 W8 and W9 are indicated by N followed by the subscript A or B, denoting series, followed by the notch depth in percent. Notches tested were;-

NA5 NA6.5 NA7.5 NA8 NB7.5 NB8 NB8.5 NB9

Vertical tail

V1...................Original one-piece fin and rudder.
V2...................Fin with separate rudder - mounted on a 3 component balance.
V3...................Similar to V2 but area increased by 15%.

Miscellaneous

Ps....................Shock plates.
Rs....................Shock ramp.
T1....................Fuselage tank.
SB1..................Fuselage brakes.
SB2..................Fin brakes.

FD....................Faired ducts.
S.......................Sealed gaps.

From May 1955 onward, the following symbols and report type was used;-

Example D. 0.07 Scale model

December 1955.
Facility....................NAE No.3 low speed tunnel (6' x 10')

Purpose..................Low speed determination of elevator effectiveness and the effect of ground board..Large proportion of test period used to determine corrections to 3 point suspension.
Configuration.........Model; B2 V1 W1 E10 N5 D4-8
Tunnel; U UD I ID B BTS, G/B at .3- .4 -.7 b/2.
Instrumentation.......6 component main balance only.
Control defections...Elevator: 10, 5, 2.5, 0, -2.5, -5, -10, -15, -20, -25, -30.
Aileron: None.
Rudder: None.
Speed range...........q=70 i.e. 235 ft/sec. (RN=3.1 x 1,000,000)
Runs......................1 to 54.

The second series of symbols in use after May 1955 were as follows:-

Body.
B1...................Similar to B5 of first series symbols but with area rule applied to armament bay.
B2....................Similar to B1 but with area rule on aft nacelles (J 75 rear end).
B3...................B2 with 30 degree nose cone.

Wing.
W1.........................3 1/2% cambered wing (corresponding to W3 of first series).
E...........................Extended leading edge outboard of transport joint (subscript denotes % extension).
N...........................Transport joint notch (subscript denotes %depth).
D...........................Leading edge droop (subscript denotes angular droop in degrees; the first figure inboard, followed by outboard).

Vertical tail.
V1...................Fin with separate rudder (V3 of first series).

Miscellaneous.
IF.....................Faired intakes.
U.......................Undercarriage down. (U1 represents nose undercarriage reversed).
Co..................Open canopy. Closed canopy included in body symbols.

T....................Belly tank.
SB..................Speed brakes.

Tunnel configurations. (Applicable only to NAE No.3 tunnel)

U ....Model upright on 3-point suspension.
UD................U plus dummy struts.
I.................... .Model inverted on 3-point suspension.
ID...................I plus dummy struts.
B....................Single strut support.
BTS...............B with addition of tail sting.

WING DATA C-105.

The profiles of the wing and vertical tail do not follow the usual conical pattern having the leading and trailing edge a generator of a single cone. Wing percent lines will therefore, not be straight unless they coincide with the generatrix pattern of the local "ruled surface".

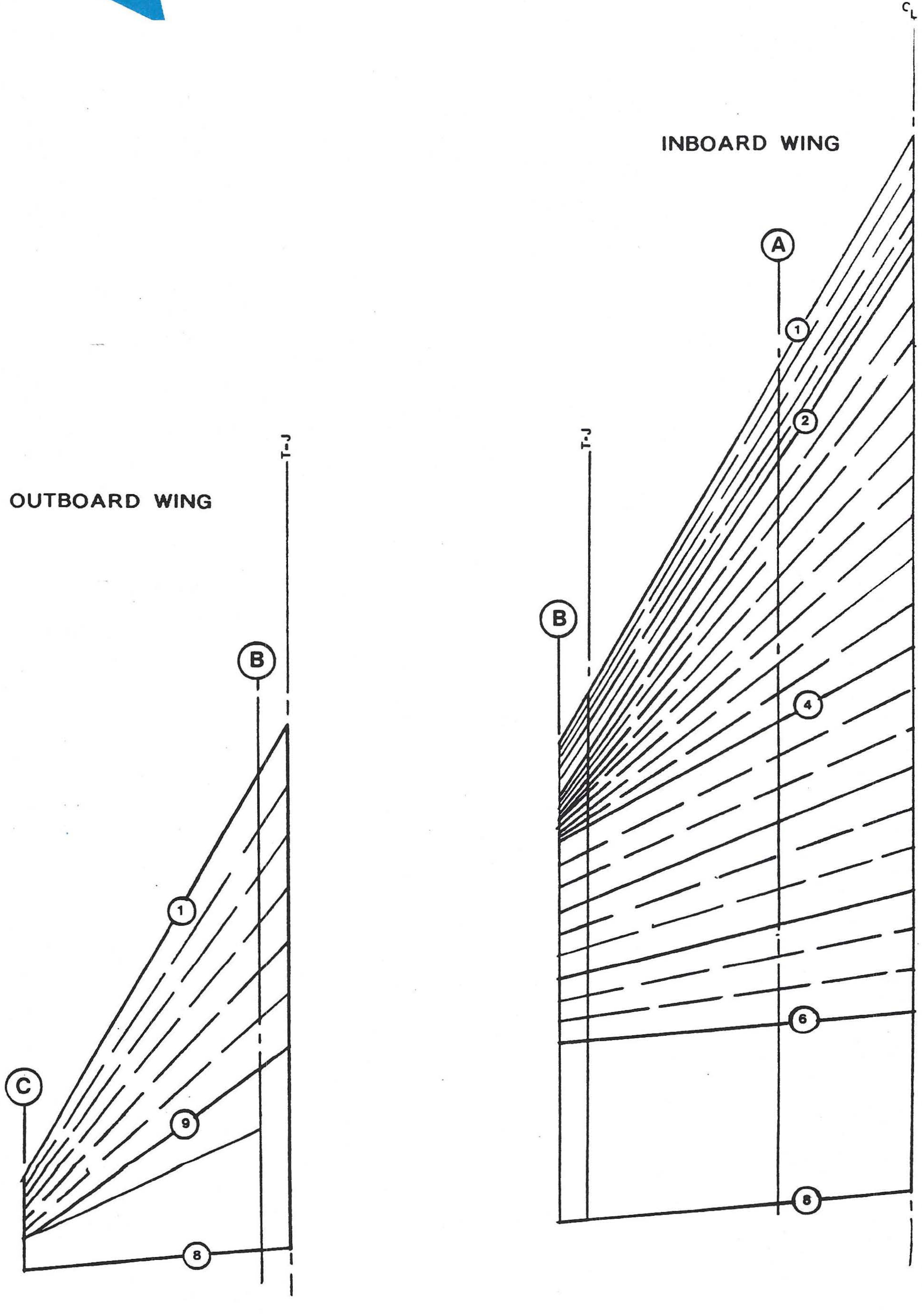
The wing will contain two separate compatible groups of "ruled surfaces" terminating at the transport joint. (This joint on the Arrow Mk1 is at 3.5 inches outboard of Directrix B). These groups are generated from a pattern of three directrix curves located span-wise at wing chord stations "A - B & C". This now implies that at the transport joint, the inner and outer wing profiles are the same up to the front spar, whereas in the original design, they were not as the outer wing panel was generated from station "C".

The directrix at chord "C" (tip), is a basic NACA 0003.8-6-3.7 section having its maximum thickness value (m) at 36.5% of the local chord. At chord "B", the (m) value has been factored to 34% of the local chord and at "A" the (m) value has been factored to 32.122% of the local chord with a basic section of NACA 0003.5-6-3.7.

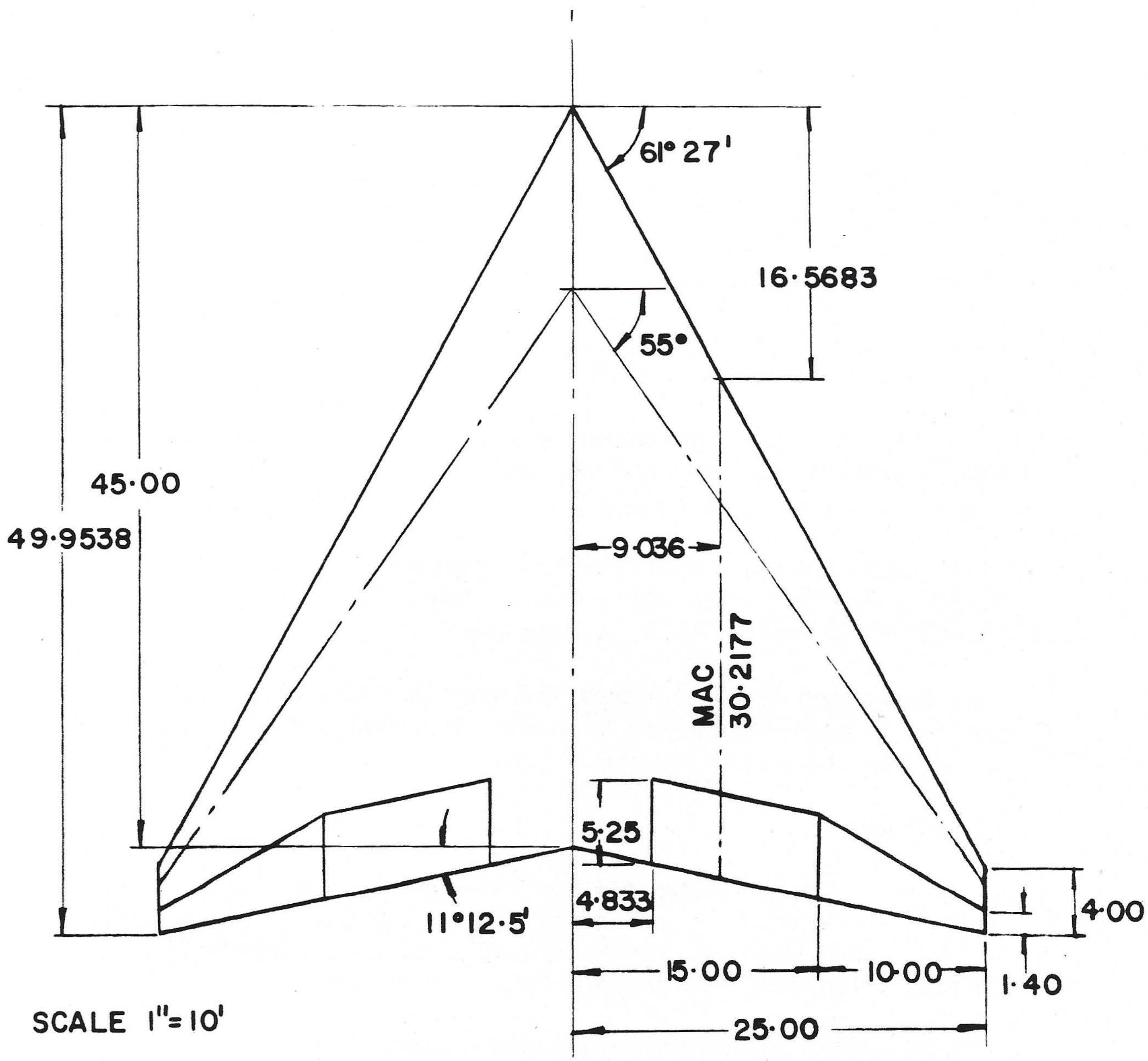
The main panel extends from the aircraft centre line to the transport joint and is made up of four separate ruled surfaces. Ruled section 2-4 will have as outer generators the front and main spars. Ruled section 4-6 will have as outer generators the main and rear spars with the centre spars being generated within the ruled section 4-6. Ruled section 6-8 will have as outer generators the rear spar and trailing edge and will be flat in profile.

The outer panel extends from the transport joint (Directrix B). Ruled section 1-9 will have as outer generators the front spar and the flat plane tangent line at 62.5% of the local chord. Ruled section 9-8 will have as outer generators the tangent line and the trailing edge and being flat, the generators are not sensitive to any pattern.

The leading edge sections of both inner and outer panels will follow their own



RULED SURFACE PATTERN. C-105 WING



C-104/u WING GEOMETRY

generators with the front spars and leading edges.

Since all three section aero-foils are different, it follows that no two generators are parallel or intersecting hence - a warped wing.

The camber 'mean line' is not sensitive to position and follows a normal pattern from root to tip. Its (m) value remains constant at 32.122% and its flat plane tangency at 62.5% of the local chord.

WING DATA - ARROW.

The ordinates at the transport joint (Arrow Directrix B), are derived from the ordinates at the elevator tip datum (Directrix B on the C-105), with the original ordinates factored to 3.5% thickness.

The Arrow front spar on the inner wing has the same origin on the Arrow Directrix B as on the C-105 front spar, therefore the generators will be the original front spar position (C-105) and the Arrow front spar position, rotated at the intersection point on Arrow Directrix B to Zu and Zl values at Directrix A.

On the Arrow, the leading edge portions of the wing are drooped ahead of the front spars. The inboard wing portion is drooped at 9 degrees and the outboard portion is drooped at 8 degrees 25 minutes of angle.

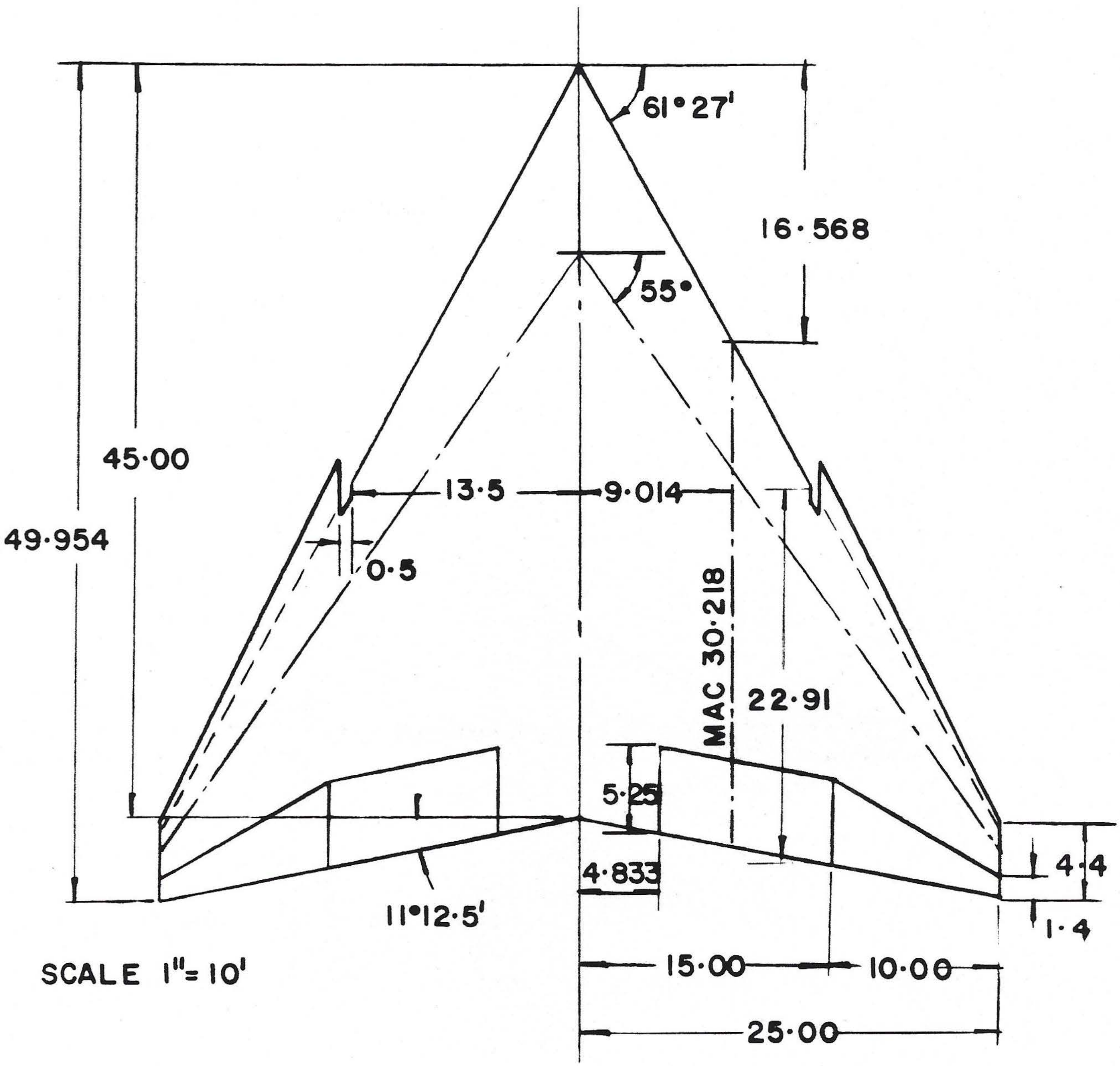
The radius of curvature of the chord line varies from 51.380" at Directrix A to 128.64" at Directrix B for the inboard wing, and from 353.868" at Arrow Directrix B to 65.495" at Directrix C for the outboard wing.

On the outboard wing, the leading edge is extended also by 10% of local chord (basic), thus giving a multiplying factor obtained by dividing the total extension ahead of the front spar by its original distance.

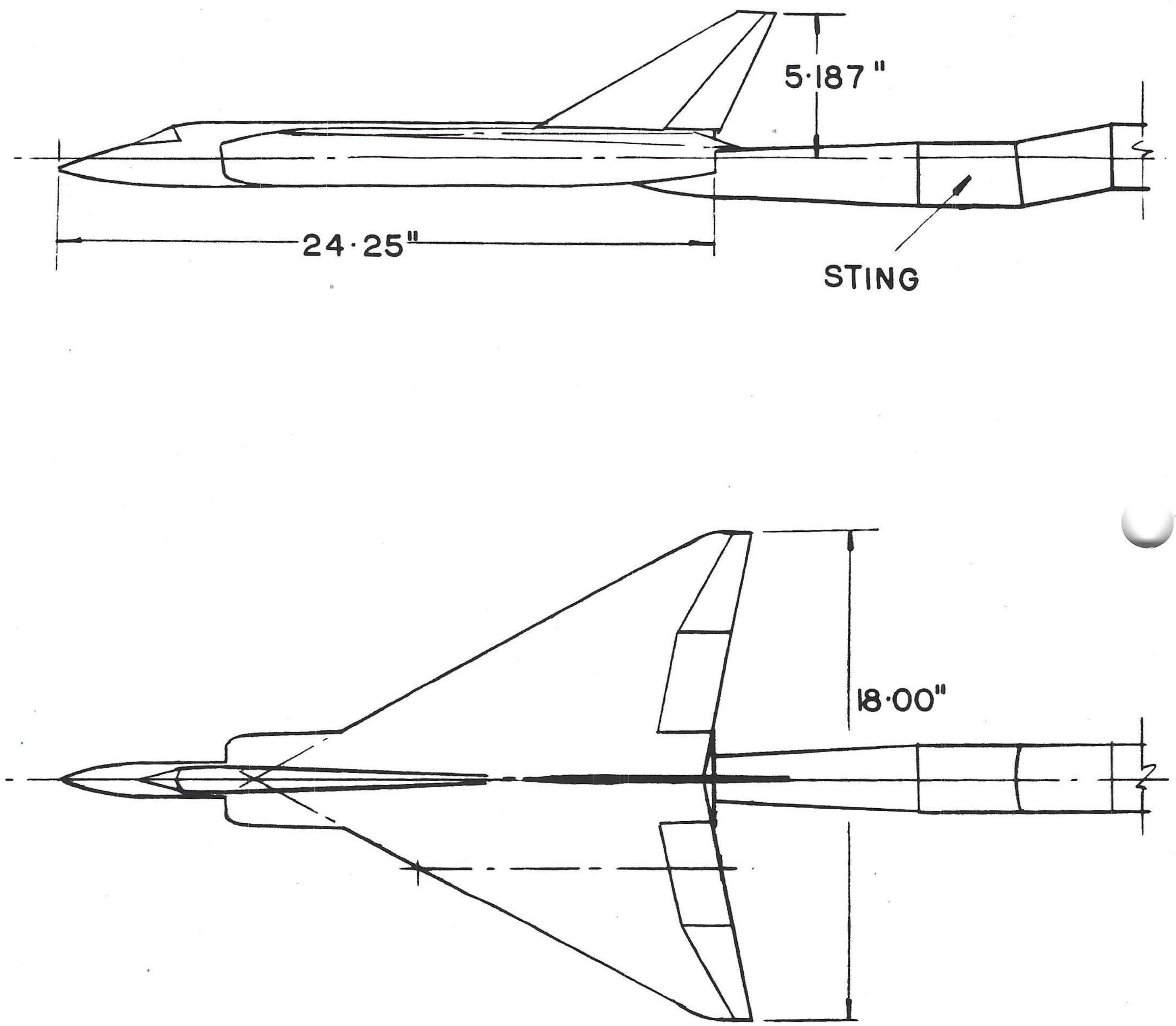
For example:

The original distance of the front spar to the leading edge at Directrix B was 30.960". The chord at Directrix B (basic) is 269.400", giving a 10% extension of 26.940". When added to the original length this gives 30.96 + 26.940 = 57.900". Dividing this by 30.960 we have 57.900/30.960 = 1.870 (factor).

The Yr ordinates up to 30.96 are multiplied by this factor. The same factor is used at Directrix C thus giving the two generating aero-foils. Once these Yr units are plotted along the new "drooped" chord line and the Zu and Zl ordinates plotted in the usual manner. The notch profile at Directrix B was derived by shortening the Yr ordinates by a factor which was obtained as follows:



C-105/CF-105 WING GEOMETRY



WIND TUNNEL MODEL MOUNTED ON A STING

(30.960 - 13.464)/30.960 = .565 (factor). Multiplying the Yr ordinates by this factor and plotting along the radius of curvature will result in the correct profile.

LEADING EDGE NOTCH, EXTENSION AND DROOP.

Once again we must use the words of Jim Floyd in his lecture to the Royal Aeronautical Society:-

"Early in the design stages, modifications were made to the original clean wing. These were the addition of leading edge droop, and a semi-span notch with outer wing chord extension. These modifications were made as a result of wind tunnel tests, carried out at Cornell Laboratories in Buffalo on a 3% complete model, sting mounted. The approximate Reynolds number used during the tests was between 1 and 2 million. These tests showed that a pitch-up or non-linearity in the CM - curve was occurring at moderate angles of attack. This phenomenon is not peculiar to delta wings, being common to all swept wing aircraft. In flight it could easily cause a tightening in the turn.

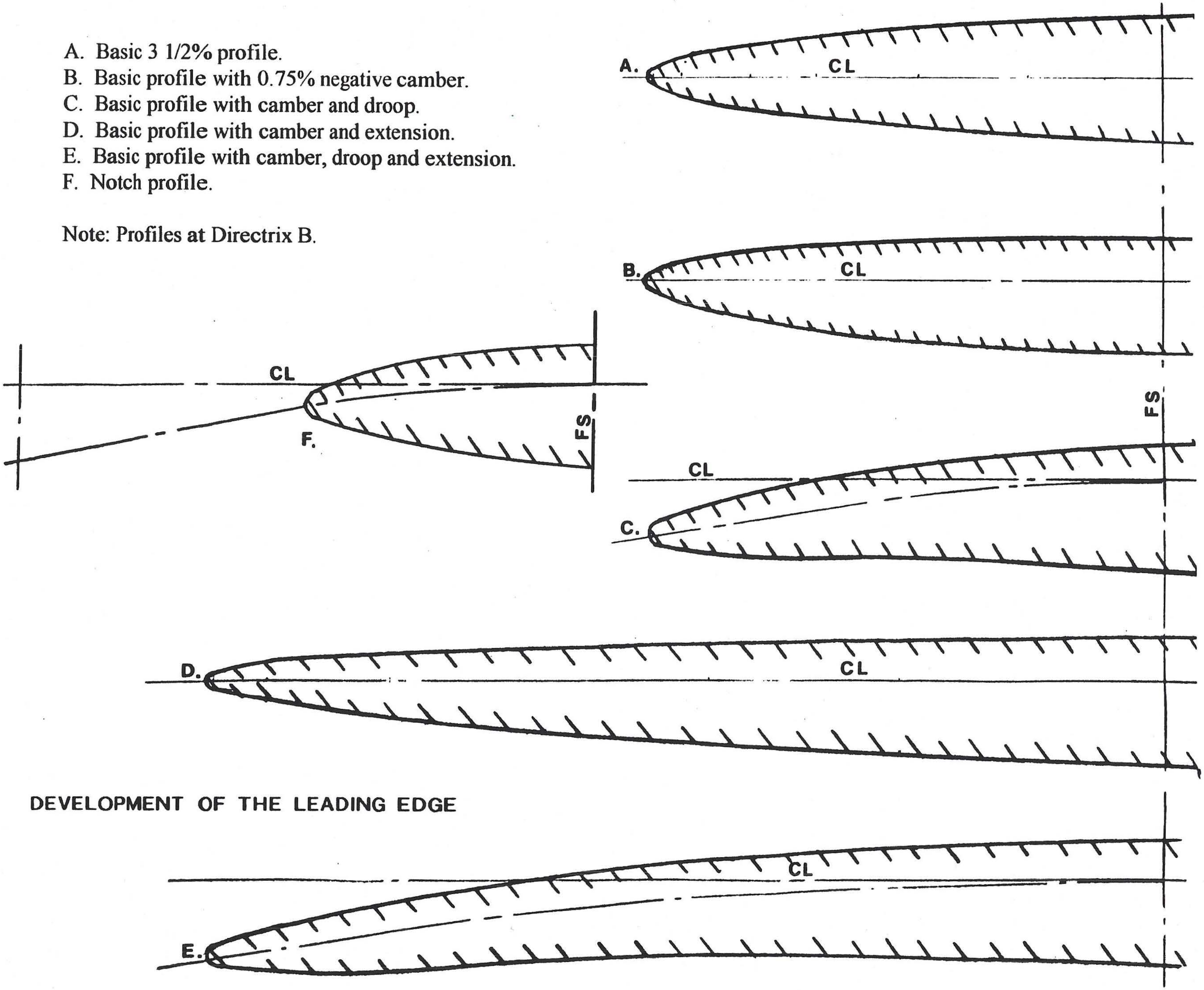
Crudely, the condition appears to be caused by vortices which start at the tip and move to the apex of the swept wing. Low pressure air is collected from the fuselage and causes a break-away outboard of the area covered by the vortex, which is mainly at the trailing edge. This is shown in the illustration and causes the effective aerodynamic centre to shift forward, giving a "pitch-up" or an abrupt change in the moment curve.

While the pitch-up appeared on test to be of small magnitude, and since very moderate amounts of pitch-up could be embarrassing to the pilot, an attempt was made to eliminate it.

Avro was aware of the work that had been done by NACA and the RAE, and the fact that a number of other aircraft which had exhibited this tendency, had used either notches in the leading edge at about mid-span, or extensions of the wing leading edge outboard, in an attempt to prevent flow separation. The notch had been used, for instance, on the English Electric F-23, and the leading edge extension had been installed on a Grumman F9F9, and a Chance-Vought aircraft. The notch has a similar effect to a fence and causes the disturbing vortices to move away from the apex of the swept wing toward the notch, which is at mid-span, and reduces the area of disturbed flow over the wing. The notch however, produces these effects by air flow rather than as a physical barrier. It was the opinion at Avro that the effects of the notch were present over the whole speed range, whereas a fence is usually only effective over smaller speed ranges, and the notch was expected to increase the drag by a lesser amount than a fence.

A. Basic 3 1/2% profile.
B. Basic profile with 0.75% negative camber.
C. Basic profile with camber and droop.
D. Basic profile with camber and extension.
E. Basic profile with camber, droop and extension.
F. Notch profile.

Note: Profiles at Directrix B.



DEVELOPMENT OF THE LEADING EDGE

In the tests however, it was found that with the notch alone, the test results were not repeatable; in other words, the same results could not be obtained in subsequent tests. When the leading edge extension was installed in addition to the notch, the results were far more repeatable. Eight different notches and three extended leading edges in various combinations were tried. The depth of the notch appeared to be the most critical parameter, and it had to be bourne in mind that too deep a notch would cause structural problems."

The illustrations show the effect of the 5% notch and 10% extension of the local chord on the outer wing, which was finally adopted, against the unmodified 3-1/2% wing at Mach 0.9, and at an elevator angle of -20 degrees.

DROOPING THE LEADING EDGE.

During the time that the tests were being carried out on the notch and leading edge extension, the work being done on the F-102 was being followed very closely with regard to a reduction in induced drag by drooping the wing leading edge, and also the work that was going on at Avro Manchester on the 707 series and Vulcan. They were drooping the leading edge to increase the buffet boundary by preventing leading edge breakaway at high angles of attack. This also influenced the choosing of the 10% increase in the outboard leading edge, to cure pitch-up. Since it was realized that, if after investigation it was found that it would be advantageous to droop the leading edge, the extension would increase the amount of effective droop.

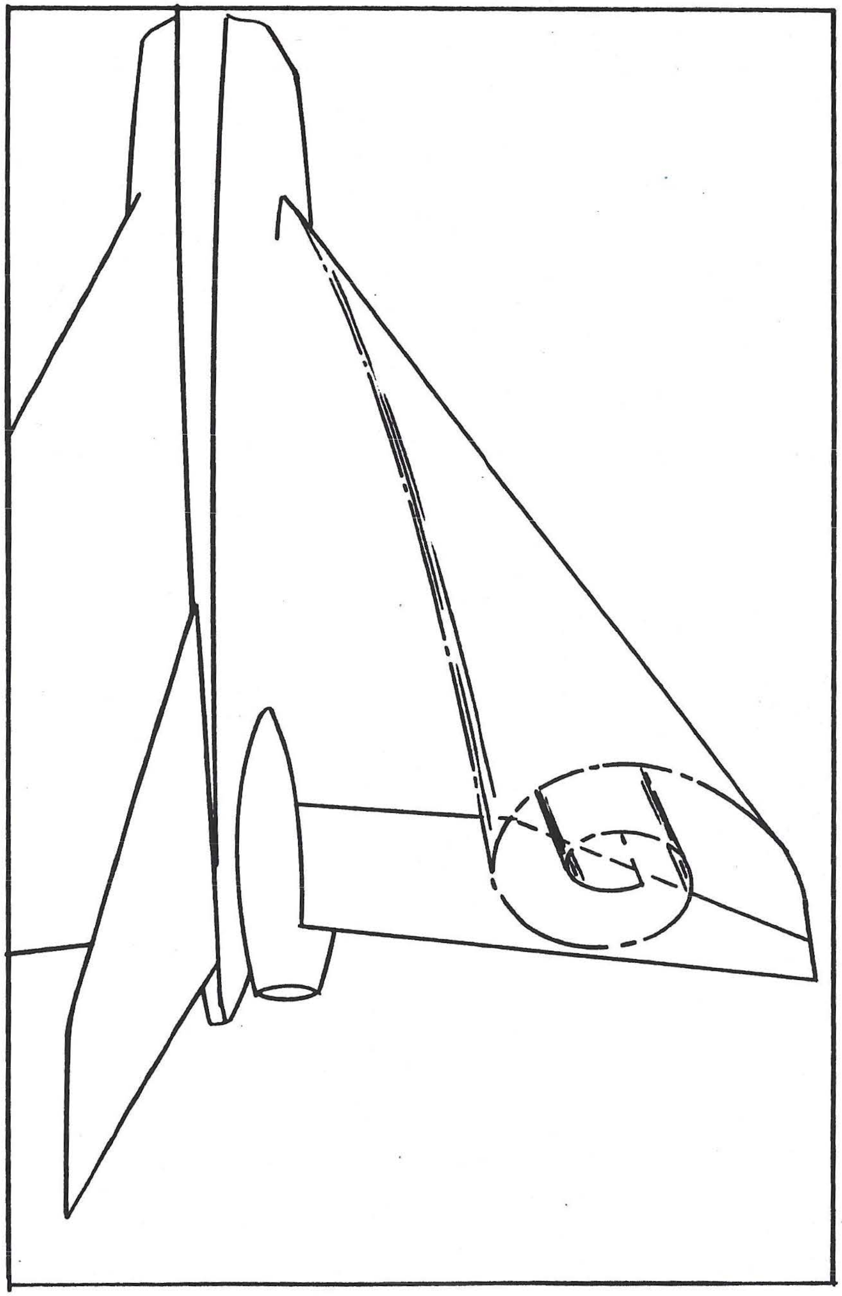
Droop was then installed on the wind tunnel model, 9 degrees inboard and 8 degrees 25 minutes outboard. This increased the buffet boundary considerably. For instance, at Mach 0.925, which is the normal subsonic cruise Mach number, the CL at which the onset of buffet was estimated, was increased from .26 with the extension alone, to .41 with the extension plus droop. The buffet, or flow separation, was indicated by pressure plots on the ailerons in the Cornell Laboratory tunnel tests. The subsequent drag did not appear to be increased appreciably". The progression of leading edge design is shown in the illustration entitled "Transition from the C-104/2 to the CF-105". It is of interest to note that the drooping, notching and extension of the leading edge was also carried out on the SAAB J29, the McDonnell Douglas F4 Phantom and the Convair F6 Delta Dart.

Explanation of airfoil specification-Inner Wing:-

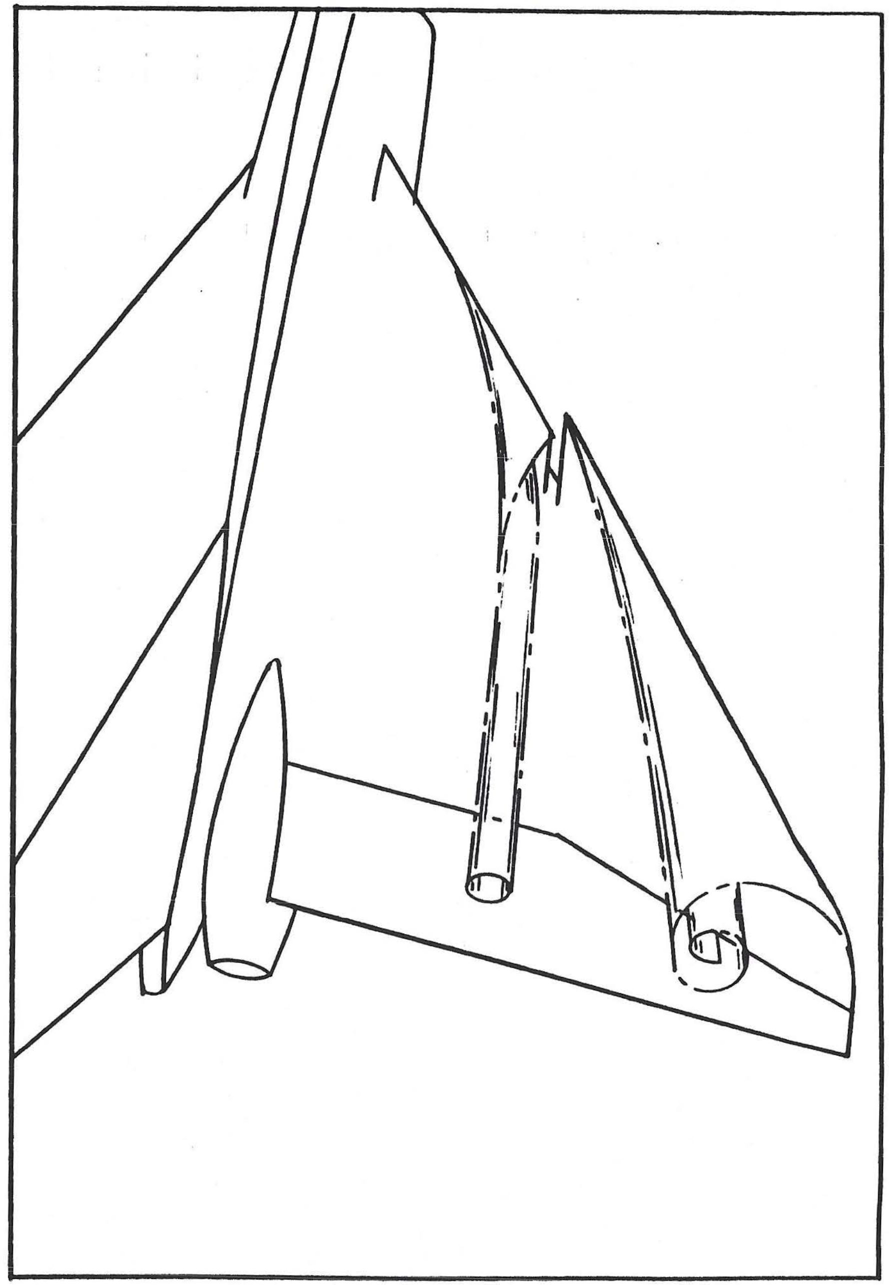
NACA 0003.5 - 6 - 3.7

Series — 3.5% thick at Dir A — L/E Rad (1.1019t C) (Normal) — 3.7% thick at Dir B

Where t = Max % thickness at section and C = Chord length.



VORTEX PATTERN WITH PLAIN WING



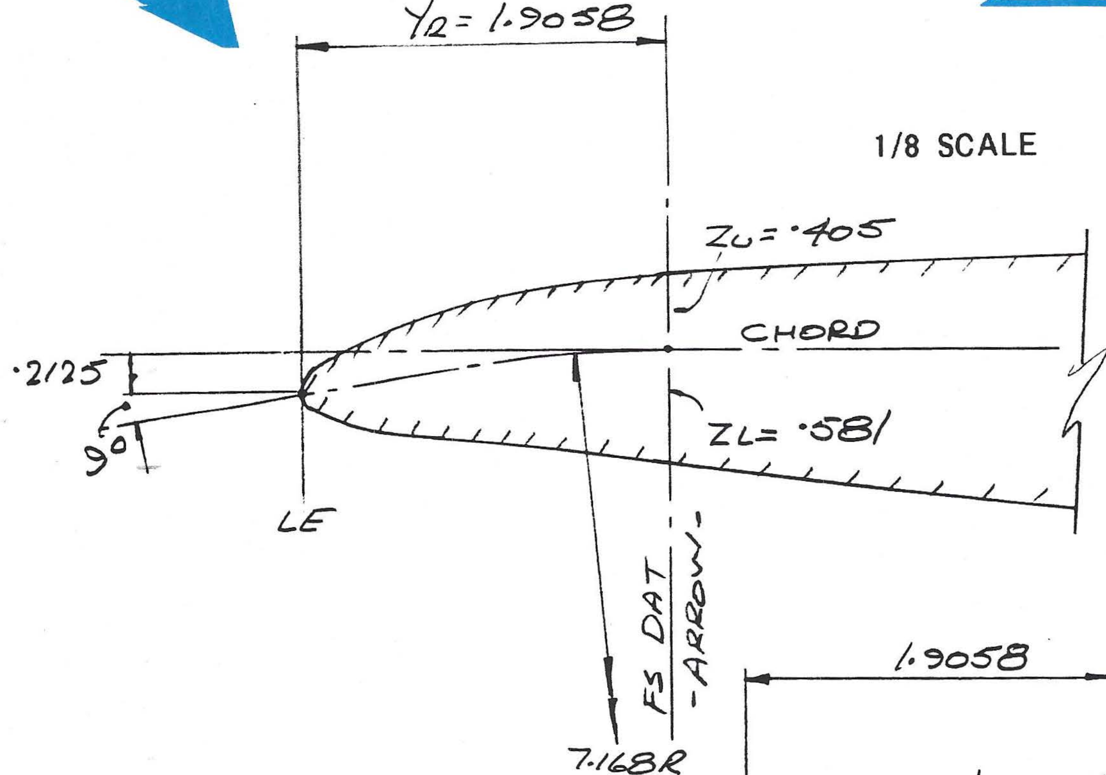
VORTEX PATTERN WITH NOTCH AND EXTENDED LE

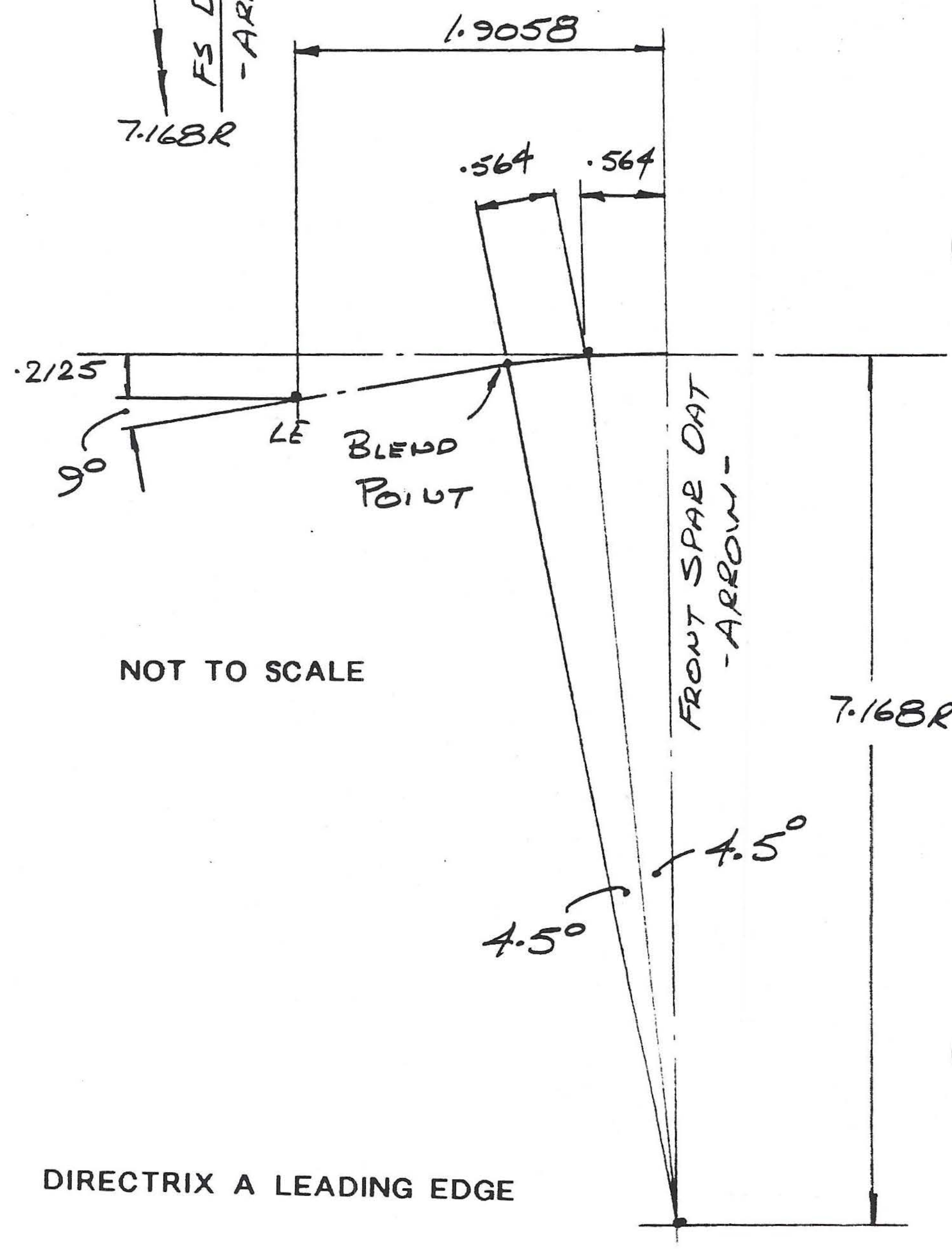
AVRO CANADA

DIRECTRIX A

Stn	Yr	Zu	Zl			Notes
1	0	0	0			PLOT ALONG
2	.071	.103	.116			9° DROOP
3	.416	.229	.273			OF
4	1.246	.353	.472			CHORD LINE
5	1.9058	.405	.581		F/S	
6	2.077	.419	.608			
7	3.323	.478	.764			
8	4.569	.513	.887			PLOT
9	6.300	.542	1.028			
10	7.148	.548	1.083			
11	8.966	.561	1.183			
12	10.782	.568	1.265			IN
13	12.600	.571	1.325			
14	14.416	.570	1.365			
15	16.232	.568	1.390			
16	18.050	.561	1.395			REGULAR
17	19.866	.553	1.389			
18	21.684	.540	1.374			
19	23.450	.526	1.347			
20	24.682	.513	1.324		M/S	
21	26.487	.493	1.282			
22	28.612	.464	1.219			
23	30.525	.438	1.151			
24	32.862	.406	1.058			FASHION
25	34.575	.379	.985			
26	36.050	.349	.930			
27	38.175	.327	.829			
28	40.300	.295	.738			
29	42.425	.263	.645			
30	44.467	.233	.556		R/S	
31	55.814	.056	.056		T/E	
				SCALE 1/8		L/E RAD = .075" NACA 0003.5 BASE

1/8 SCALE





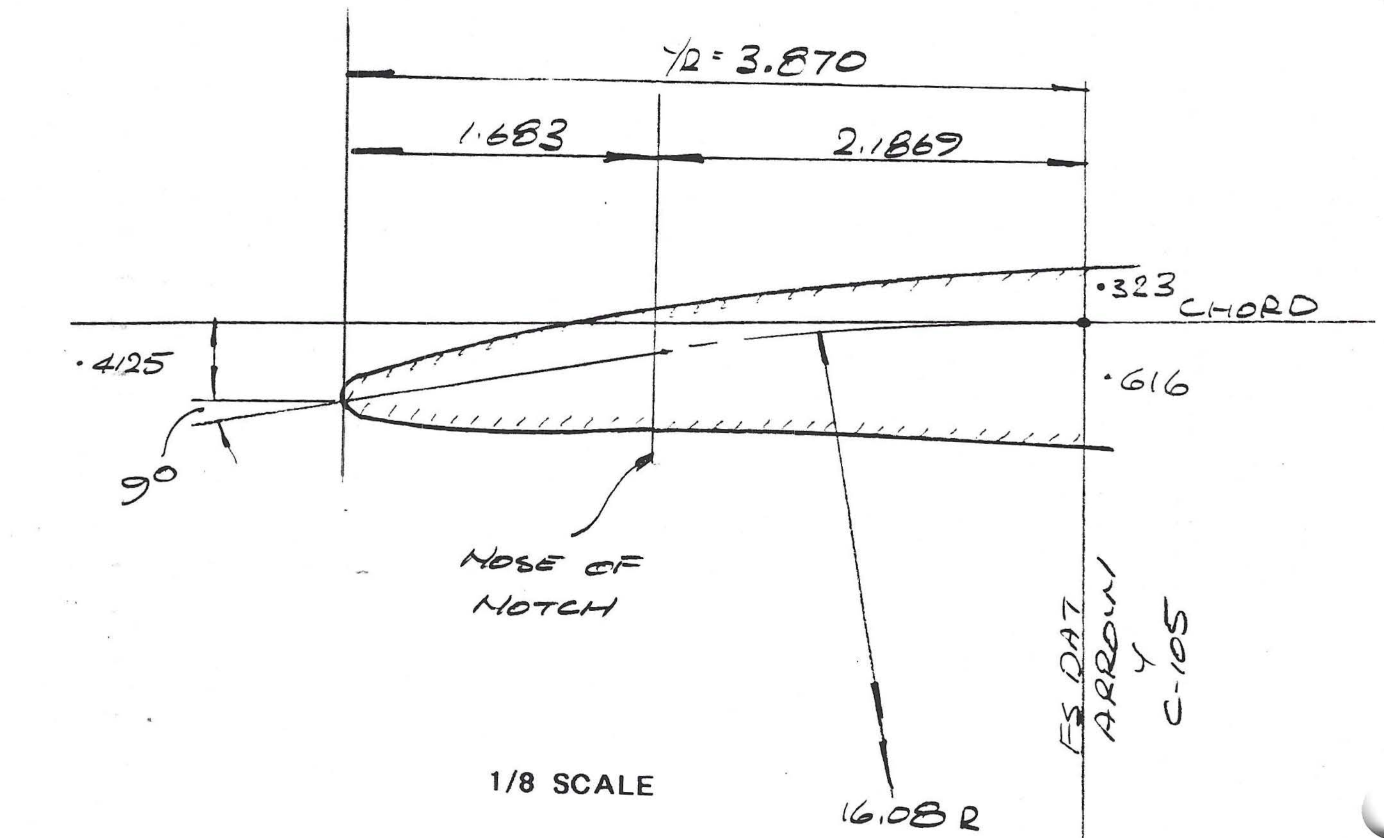
NOT TO SCALE

DIRECTRIX A LEADING EDGE

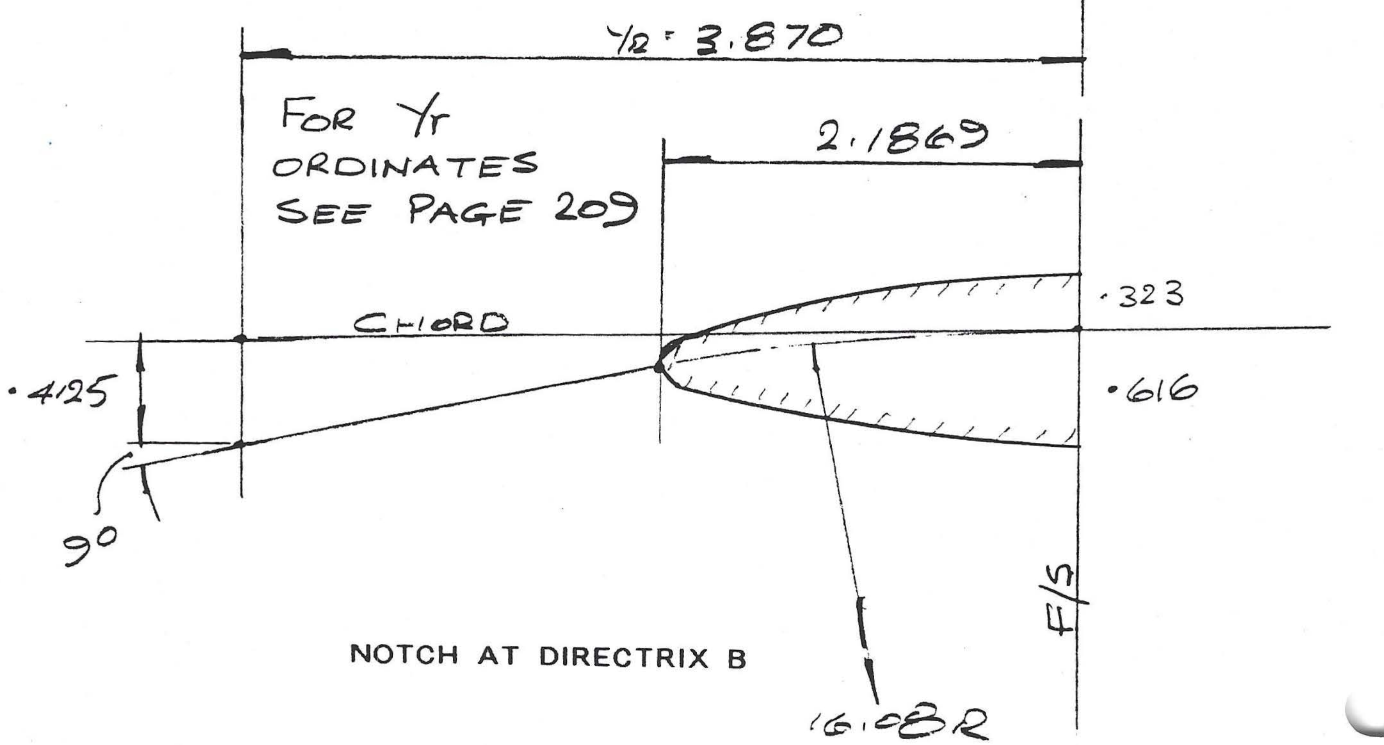
AVRO CANADA

DIRECTRIX B-BASIC

Stn	Yr	Zu	Zl			Notes
1	0	0	0			PLOT ALONG
2	·044	·048	·054			9° DROOP
3	·255	·136	·163			OF
4	·765	·206	·281			CHORD LINE
5	1.276	·246	·362			
6	2.041	·280	·458			
7	2.807	·305	·531			
8	3.870	·323	·616		F/S	
9	4.114	·325	·632			
10	4.637	·329	·664			
11	5.159	·333	·692			PLOT
12	5.682	·336	·717			
13	6.204	·340	·737			
14	6.727	·341	·754			
15	7.250	·340	·769			IN
16	7.772	·340	·780			
17	8.295	·340	·788			
18	8.803	·338	·794			
19	9.157	·337	·797		M/S	REGULAR
20	10.359	·337	·814			
21	11.773	·330	·820			
22	13.047	·325	·811			
23	14.602	·316	·793			
24	15.734	·309	·771			FASHION
25	16.724	·299	·753			
26	18.139	·289	·713			
27	19.553	·271	·667			
28	20.968	·259	·606			
29	22.327	·233	·556		R/S	
30	33.675	·056	·056		T/E	
				SCALE 1/8		L/E RAD = ·051" NACA 0003.7 BASE



1/8 SCALE

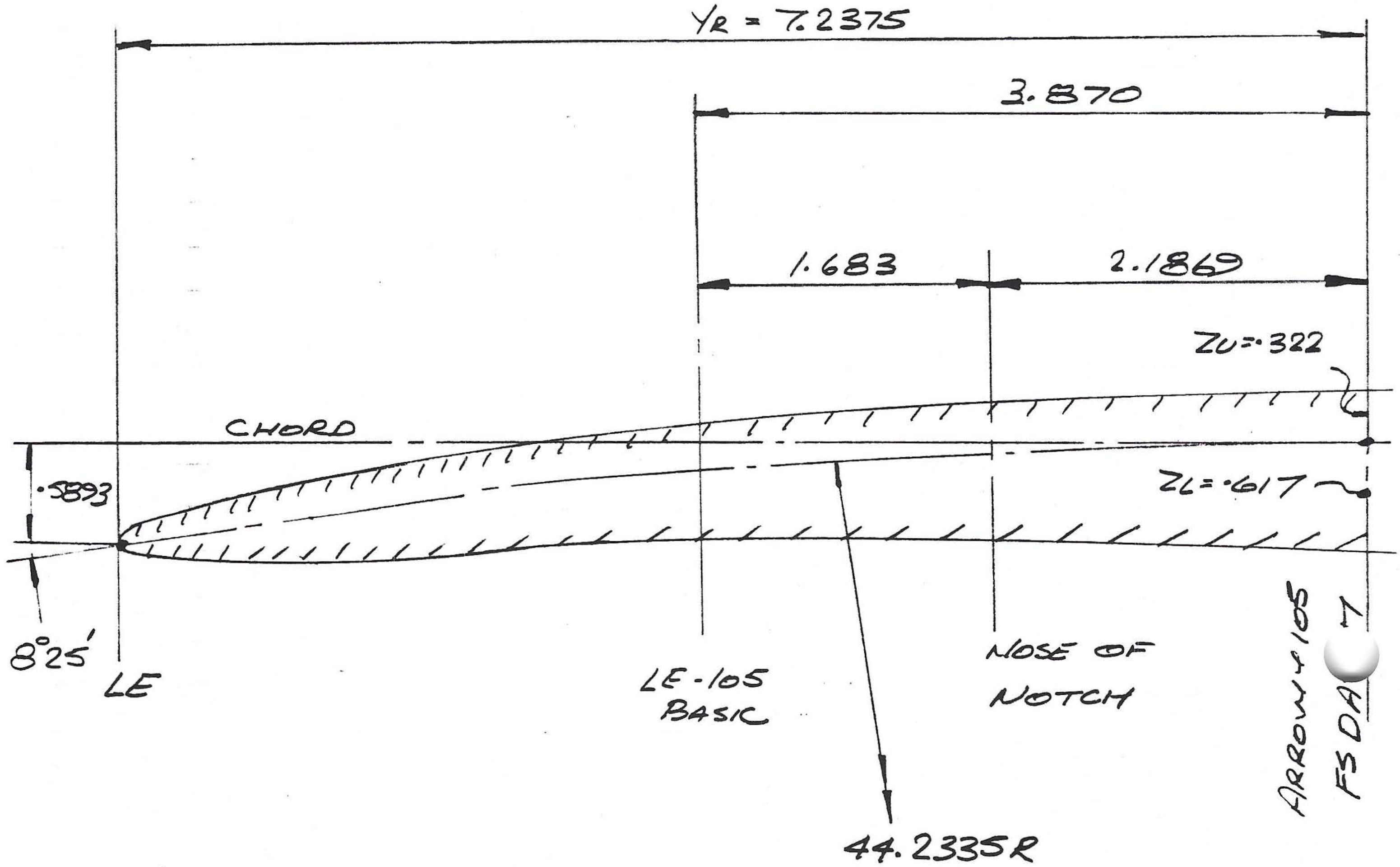


NOTCH AT DIRECTRIX B

DIRECTRIX B-BASIC

DIRECTRIX B-EXTENDED

Stn	Yr	Zu	Zl			Notes
1	0	0	0			PLOT ALONG
2	·082	·048	·054			8°25' DROOP
3	·477	·136	·163			OF
4	1.43	·206	·281			CHORD LINE
5	2.386	·246	·362			
6	3.817	·280	·458			
7	5.249	·305	·531			
8	7.237	·323	·616		F/S	
9	7.481	·325	·632			
10	8.004	·329	·664			PLOT
11	8.526	·333	·692			
12	9.049	·336	·717			
13	9.571	·340	·737			
14	10.094	·341	·754			IN
15	10.617	·340	·769			
16	11.139	·340	·780			
17	11.662	·340	·788			
18	12.170	·338	·794			REGULAR
19	12.524	·337	·797		M/S	
20	13.726	·336	·814			
21	15.140	·330	·820			
22	16.414	·325	·811			
23	17.969	·316	·793			
24	19.101	·309	·7716			MANNER
25	20.091	·299	·753			
26	21.506	·289	·713			
27	22.920	·271	·667			
28	24.335	·259	·606			
29	25.694	·233	·556		R/S	
30	37.042	·056	·056		T/E	
				SCALE 1/8		L/E RAD = ·051" NACA 0003-7 BASE

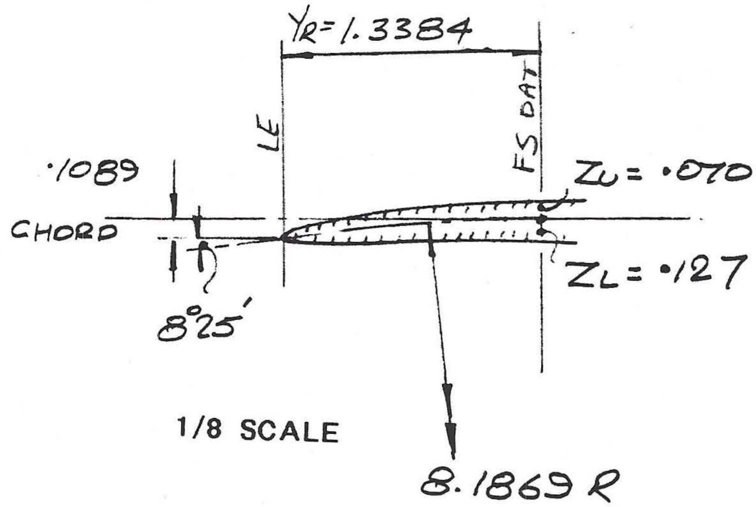


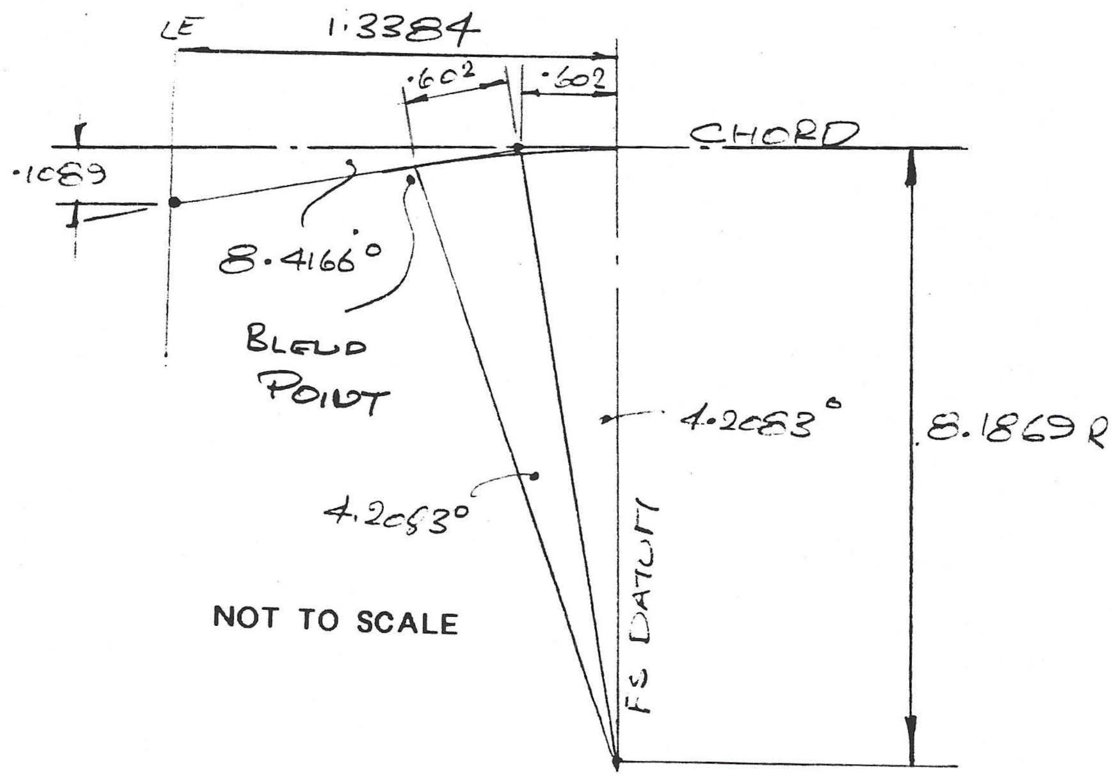
1/8 SCALE

DIRECTRIX B (ARROW) EXTENDED LEADING EDGE

DIRECTRIX C-EXTENDED

Stn	Yr	Zu	Zl			Notes
1	0	0	0			PLOT ALONG
2	·087	·020	·023			8°25' DROOP
3	·179	·037	·046			OF
4	·391	·050	·070			CHORD LINE
5	·731	·059	·095			
6	1·174	·067	·118			
7	1·339	·070	·127		F/S	
8	1·482	·074	·134			
9	1·703	·077	·147			
10	1·917	·078	·155			PLOT
11	2.125	·079	·163			
12	2.325	·079	·167			
13	2.357	·079	·167			
14	2.519	·079	·170			IN
15	2.707	·079	·170			
16	2.889	·079	·170			
17	3.066	·079	·167			
18	3.239	·079	·165			REGULAR
19	3.405	·077	·163			
20	3.567	·076	·161			
21	3.603	·076	·159			
22	3.725	·076	·157			
23	3.876	·074	·150			FASHION
24	4.026	·072	·147			
25	4.171	·070	·140			
26	4.312	·070	·136			
27	4.448	·067	·130			
28	4.523	·067	·125			
29	6.623	·023	·023			
				SCALE 1/8		L/E RAD =·010" NACA 0003.8 BASE





DIRECTRIX C (ARROW) EXTENDED LEADING EDGE

FIN BASE CHORD

Stn	Yr	Zu	Zl			Notes
1	0	0	0			NACA 0004
2	·171	·128	·128			BASIC
3	·228	·142	·142			
4	·356	·180	·180			
5	·712	·248	·248			
6	1.425	·338	·338			
7	2.137	·399	·399			
8	2.850	·445	·445			
9	4.275	·508	·508			
10	5.700	·545	·545			
11	7.125	·564	·564			
12	8.550	·570	·570			
13	11.400	·551	·551			
14	14.250	·503	·503			
15	17.100	·433	·433			
16	19.950					
17	22.800					
18	25.650					
19	27.075					
20	28.500	·056	·056			
				SCALE 1/8		

AVRO CANADA

FIN TIP CHORD

Stn	Yr	Zu	Zl			Notes
1	0	0	0			NACA 0004
2	·051	·038	·038			BASIC
3	·068	·042	·042			
4	·106	·054	·054			
5	·212	·074	·074			
6	·425	·101	·101			
7	·637	·119	·119			
8	·850	·133	·133			
9	1·275	·151	·151			
10	1·700	·162	·162			
11	2·125	·168	·168			
12	2·550	·170	·170			
13	3·400	·164	·164			
14	4·250	·150	·150			
15	5·100	·129	·129			
16	5·950					
17	6·800					
18	7·650					
19	8·075					
20	8·500	·025	·025			
				SCALE 1/8		

LOFTING

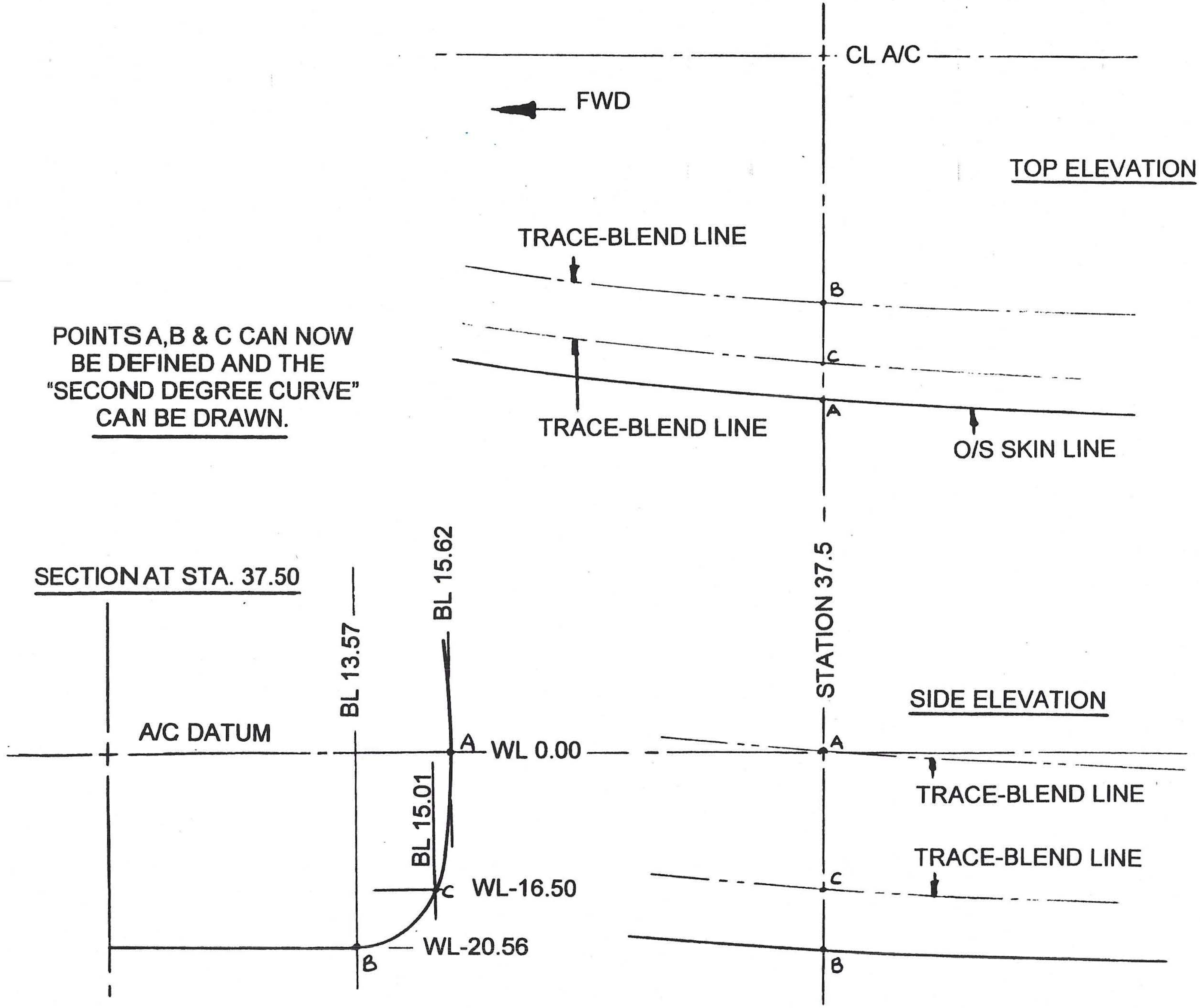
The *"Lofting"* procedure as used at AVRO was a very exact method of drawing contoured parts to very tight tolerances. These drawings were drawn on Class Cloth or Mylar, both of which are very inert materials and are capable of holding the exact shape or profile of whatever is traced or drawn on them. AVRO was one of the first in Canada to introduce and apply this process and it was one of the reasons why the ARROW tooling was completed in a minimum time span and that it was so accurate, a definite requirement in order to control interchangeability. This process of *"Lofting",* was not new, as shipbuilders had used it for hundreds of years in order to lay out the lines of a ship, and the term comes from the fact that in order to do this, a large building or LOFT as it was called, was required.

The term *"Lofting",* is the drafting procedure used to develop a curved surface such that any cutting plane intersecting it will produce a smooth curved line. *"Lofting"* also is used to establish intersections of curved surfaces with each other. In view of the numerous curved surfaces in an airplane, it is evident that *"Lofting"* is an important function of the engineering department.

The process involves the drawing of the basic lines of the aircraft on aluminum sheets with the aid of a very hard pencil. Say a 9H, or some similar material, and using aids such as ship curves, splines and ducks for laying out faired lines. The ducks are simply lead weights fitted with metal prongs which rest on the splines, holding them in place. The splines are lengths of hardwood or in some cases, plastic, and are used to connect the basic points of a curve under construction. A duck is used at each point, and when each duck can be lifted in turn and the spline does not move, then the curve can be said to be *"Faired in".* To loft a set of lines, plan and elevation views are laid out in their proper sequence. *"Station lines"* are then drawn vertically in the elevation view to form vertical transfer reference planes, and *"Buttock lines"* are then drawn parallel to the airplane centre line in the plan view to form vertical longitudinal reference planes, and finally, *"Water lines"* are drawn perpendicular to the station lines in the elevation view to form horizontal reference planes. Since three dimensions are required to locate a point in space, a point could be thus located as follows:-

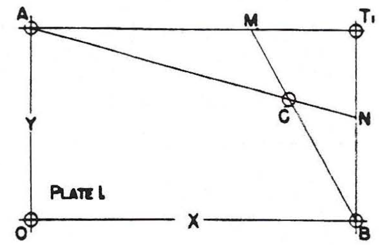
STATION 37.5
WATERLINE 20.56
PORT BUTTOCK LINE 13.57

On the ARROW, a decision was made early in the design that all curves should be of a *"Second degree"* nature. This was desirable as it had been shown in previous practice that curved surfaces could be faired in easier by using the limits from the master lines loft. Examples of the AVRO sheets on their construction are shown, points A, B and C and P1, P2, and P3 being taken from the master lines.



A SIMPLE ILLUSTRATION SHOWING ORIGIN OF POINTS A,B&C

3.3 Construction of a Second-Degree Curve



This method of construction of a second degree curve is applicable to all conic sections and if carefully done will produce a curve which will satisfy the mathematical equation for any point on the curve.

Given: Two points of tangency A and B; tangents normal to the x and y axis; point C on the required curve.

Construction:

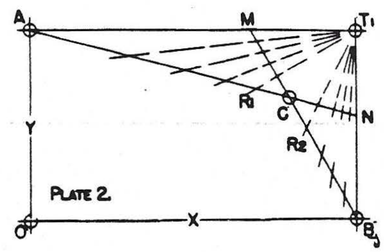


Plate 1: Extend tangents A and B to meet at tangent intersection T_1.

Draw construction lines AN and BM to pass through point C, the point on the required curve, which is within the boundaries of the tangent lines A and B.

Plate 2: With centre T_1 draw ray lines cutting construction lines AN and BM at a suitable number of places.

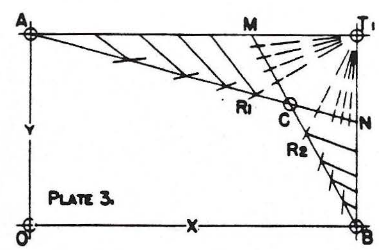
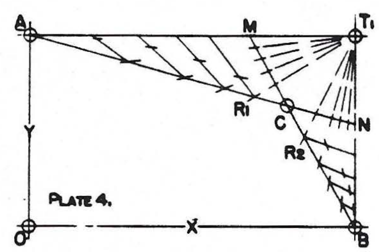


Plate 3: From centre A draw rays through ray intersections on BC. From centre B draw rays through ray intersections on AC.

Plate 4: Draw lines from A to ray intersections on CM, beginning at point M. Where these lines cut the rays from B will be points on the required curve.

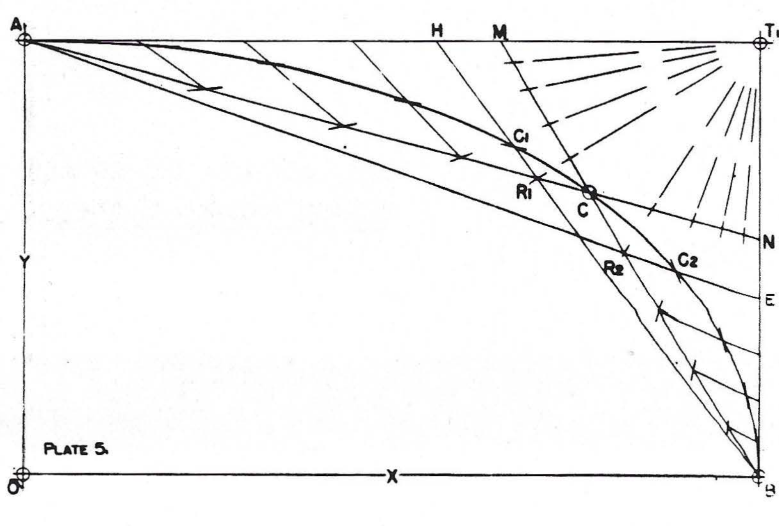


Similarly, draw lines from B to ray intersections on CN, beginning at point N. Where these lines cut the rays from A will be further points on the required curve.

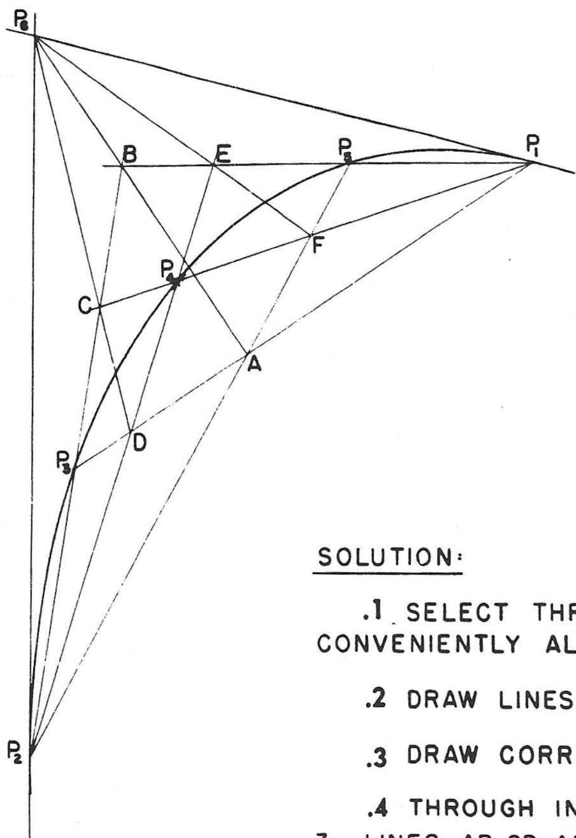
Plate 5: This is the final construction of a second degree curve. The points are joined to give the required curve through point C.

As examples, C_1 and C_2 are points on the required curve. A ray T_1R_1 (Plate 2) is drawn to cut AN at R_1. BH is then drawn through R_1 (Plates 3 and 5). Point C_1 is the intersection of BH and a line from A to the point where T_1R_1 crosses CM.

Similarly C_2 is found by drawing T_1R_2 to BM, drawing AE, and then a line from B to CN.



3.4 Construction of a Tangent to a Second-Degree Curve at Two Given Points



REQUIRED:

TO DRAW TANGENTS TO THE GIVEN POINTS P_1 AND P_2

SOLUTION:

.1 SELECT THREE POINTS (P_3, P_4, P_5) SPACED CONVENIENTLY ALONG THE CURVE.

.2 DRAW LINES P_1P_5 EXTENDED, P_1P_4 EXTENDED AND P_1P_3.

.3 DRAW CORRESPONDING LINES P_2P_3, P_2P_4 AND P_2P_5.

.4 THROUGH INTERSECTION POINTS A, B, C, D, E & F, DRAW 3 LINES AB, CD AND EF WHICH INTERSECT AT THE COMMON POINT P_6.

.5 LINES P_1P_6 AND P_2P_6 ARE THE REQUIRED TANGENTS.

NOTE: THE METHOD DESCRIBED ABOVE IS ACCURATE FOR SECOND-DEGREE CURVES ONLY.

CF-105 STRUCTURAL PLASTIC AND ANTENNA RESEARCH MODEL PROGRAMS.

STRUCTURAL PLASTIC MODEL PROGRAM.

1/5 SCALE OF 3% T/C WITH PORTION OF WING.

This model was used to check deflections and stresses with the results obtained by Stress Analysis. It was completed at Avro on Sept.15/54 and tested there in Jan./55.

1/5.25 SCALE FRONT FUSELAGE WITH AIR DUCTS AND FUEL TANKS.

This model was used to check deflections and stresses for applied unit load cases. It was completed at Avro on Feb.1/55 and tested there in Apr./55.

1/5.25 SCALE SEGMENT OF FRONT FUSELAGE STRUCTURE.

The model was used to check the effect of stiffness of ducts on deflection of the front fuselage. It was completed at Avro on Apr.7/55 and tested there in Apr./55.

1/5.25 SCALE CENTRE WING PORTION WITH FIN, FRONT AND REAR FUSELAGE STRUCTURE.

This model was used for checking deflections and stresses due to loads applied to the fin. It was completed at Avro on June 15/55 and was scheduled for test between June and Sept./55 but tests were delayed pending cost and program review.

1/5.25 SCALE COMPLETE STRUCTURAL MODEL OF THE AIRCRAFT.

This model was used for checking deflections and stresses due to different loading cases. It was completed at Avro on Aug.31.55 and was also intended to serve for the static test of the full sized aircraft between Oct. and Dec./55 but tests were delayed pending cost and program review.

Note:- All the above models were designed, manufactured and tested by Avro.

ANTENNA RESEARCH MODELS.

1/48 SCALE COMPLETE MODEL IN SHEET METAL.

This model was a free-flight model for antenna research. It was completed in Jan./55 by Sinclair Radio Lab. and tested there in Jan./55.

1/48 SCALE MODEL (MODIFIED).

This model was used for low frequency radio compass research. It was completed in June/55 by Sinclair Radio Labs. and tested there between June/55 and Sept./55.

1/18 SCALE. COMPLETE MODEL CAST ALUMINUM.

This model was used for UHF and L-Band Antenna research. It was completed in Apr./55 by Sinclair Radio Lab. and tested there in Apr./55.

1/8 SCALE COMPLETE MODEL IN SHEET COPPER.

This model was used for Exp. UHF and L-Band Antenna research. It was completed in July/54 by Sinclair Radio Lab. and tested there in Aug./54.

FULL SCALE BELLY MOCK-UP - 2 MODELS.

These were used for UHF and L-Band Antenna research. They were completed in Oct./55 by Sinclair Radio Lab. and tested there in oct./55 over an extensive period.

FULL SCALE FIN MOCK-UP.

This model was used for Fin Cap Antenna and X-Band Antenna research. It was completed in June/55 by Sinclair Radio Lab. and tested there in June-Sept./55.

Note;_ All the above antenna models were designed, manufactured and tested by Sinclair Radio Laboratories Ltd.

CF-105 WIND TUNNEL PROGRAM.

3/100 COMPLETE MODEL-STING MOUNTED.

This model was used for Subsonic and Transonic 3 Axis Stability and Control. It was designed, manufactured and tested at Cornell in Buffalo in Sept./53 and tested there as follows in the 3ft. x 4ft. Transonic and 10ft. x 12ft. Subsonic tunnels as follows;-

Stage 1. Sept./53 - Longitudinal Stability with and without Camber. T/C = 3%. M = 0.5 - 1.23.

Stage 2. Apr./54. - Longitudinal and Lateral Stability and Control with Camber. T/C = 3.5%. M = 0.5 - 1.23.

Stage 3. Jun./54. - Longitudinal Stability. Check Directional Stability and Control, New Nose and New Canopy. M = 0.5 - 1.23.

Stage 4. Jul./54. - Notch Investigation, Complete Test with Optimum Notch, Low Speed, High Angle of Attack. M = 0.5.

Stage 5. Oct./54. - Notch Investigation at all Speeds. Longitudinal and Directional Stability, High Reynolds Number, New Nose, Leading Edge Extension and Notch. M = 0.5 - 1.23.

4/100 COMPLETE MODEL STING MOUNTED.

This model was used for Transonic Armament tests for the Falcon and Sparrow Missiles, in conjunction with Longitudinal, Directional Stability and Control. It was designed, manufactured and tested at Cornell in Buffalo in Mar./55 and tested there as follows in the 3ft. x 4ft. transonic tunnel:-

Stage 1. Mar./55. - Longitudinal and Directional Stability Comparison between .03 and .04 scale models. M = 0.5 - 1.23.

Stage 2. Mar./55. - Transonic force tests on missiles, armament bay pressures, bay door hinge moments. M = 0.9 - 1.2.

Stage 3. Mar./55. - Transonic tests for missile effect on aircraft. M = 0.95 - 1.2.

Stage 4. Apr./55. - Transonic force tests on missile for trajectory analysis. M = 0.95 - 1.2.

Stage 5. May./55. - Longitudinal Stability. Investigate leading edge droop. M = 0.5 - 1.2.

Stage 6. May./55. - Complete longitudinal and direct stability and control tests with optimum droop. M = 0.5 - 1.2.

Stage 7. May./55. - Investigation at high Reynolds Number and high angle of attack. M=0.5.

1/10 SCALE REFLECTION PLANE WING.

This model was designed, manufactured and tested by the NAE in Ottawa in Jan./55. It was used for Subsonic, Preliminary Study of Icing Conditions on Longitudinal and Lateral Control, and was tested in the 10ft. x 5.7ft. low speed tunnel at the NAE in Jan./55. This test was an extension to the NAE icing research program, and the model was approximate only.

1/8 SCALE REFLECTION PLANE WING.

This model was designed and manufactured by Avro in Mar./55 and was used for Subsonic, more advanced study of Icing Conditions with Notch and Leading Edge Extensions included. It was tested at the NAE low speed 10ft. x 5.7ft. tunnel in May/55.

7/100 SCALE COMPLETE MODEL.

This model was designed, manufactured and tested by both Avro and the NAE in Apr./55 and was used for Subsonic Canopy and Missile Jettison Tests and Ground Effects. It was tested at the NAE 10ft.x 5.7ft. low speed tunnel in Ottawa in May/55.

One run was completed at high incidence at the end of May. Further testing scheduled June-July/55 but suspended due to model re-work for Notch, Leading Edge Extension and Droop. Resumed Nov./55.

1/80 SCALE COMPLETE MODEL STING MOUNTED.

This model was designed and manufactured at Avro in Apr./55 and was used for Supersonic, Lateral and Directional Stability and Control. It was tested by the NAE in the 16in. x 30 in. Supersonic tunnel in July/55. Due to the balance not being ready the tests started in Mid-July/55. M = 1.23, 1.36, 1.56, 1.8 and 2.0. The model was returned to Avro for the addition of the Notch, Leading Edge Extension and Droop. Tests resumed in Nov./55.

1/40 SCALE FUSELAGE INTAKE MODEL.

This model was designed and manufactured at Avro in Apr./55 and was used for Supersonic Study of Air Flow through Intakes. It was tested in the Supersonic 10in. x 10in. tunnel at the NAE in Ottawa in Mid-June.55. Preliminary tests were completed by Mid-June/55 at M = 1.4, 1.8 and 2.0. Further tests continued.

1/50 SCALE REFLECTION PLANE MODEL.

This model was designed, manufactured and tested by the NAE in Ottawa in Sept./55. It was used for Supersonic Longitudinal Stability and Control, also Lateral Control. It was tested in the 16in. x 30in. Supersonic Tunnel in Oct./55 with M = 1.23, 1.36, 1.56, 1.8 and 2.0.

1/24 SCALE COMPLETE MODEL.

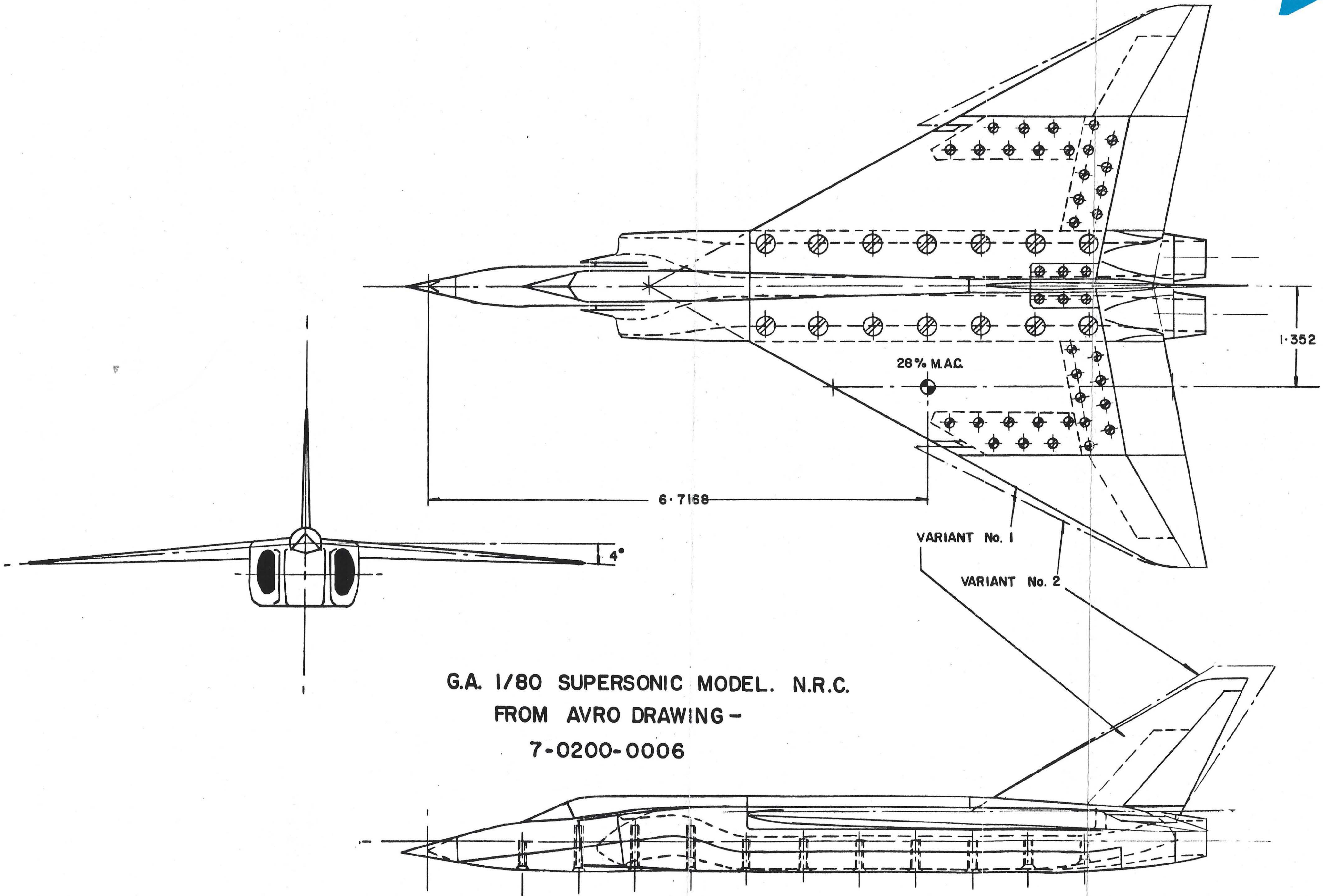
This model was designed and manufactured by the NAE in Ottawa in June/55, and was used for Subsonic Spin Characteristics and Recovery. It was tested in the NAE Spinning tunnel without final results.

1/6 SCALE FUSELAGE INTAKE MODEL.

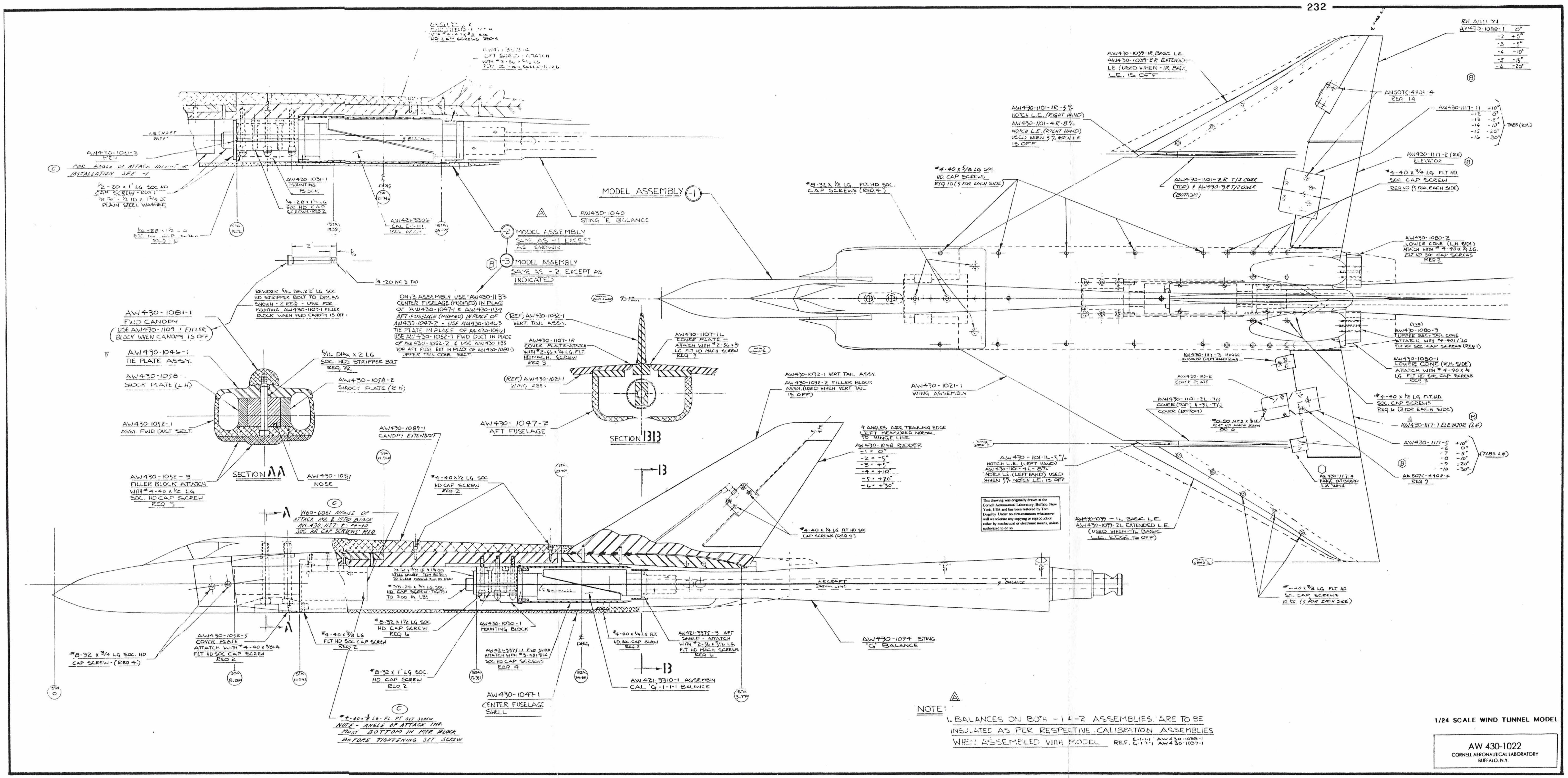
This model was designed and manufactured by Avro in Oct./55 and was used in the Supersonic Study of Air Flow through the Intakes. It was delivered to the NACA Lewis Laboratory in Cleveland for testing in the 8ft. x 6ft. Supersonic tunnel in Nov./55 where it was instrumented by Lewis Labs.

3/100 SCALE COMPLETE MODEL.

This model was designed and manufactured by Cornell in Buffalo in Oct./55 and was used for Supersonic Directional Stability at high angles of attack. It was scheduled to be tested in the 3ft. x 3ft. Supersonic tunnel at Bedford, England, but due to both it and the Langley 4ft. Supersonic and Unitary Plan 4ft. Supersonic being heavily booked, testing was delayed. Probability of the Unitary Plan tunnel being used later for speeds between M = 1.4 and 2.0.



G.A. 1/80 SUPERSONIC MODEL. N.R.C.
FROM AVRO DRAWING -
7-0200-0006


MODEL ASSEMBLY -1
MODEL ASSEMBLY -2 SAME AS -1 EXCEPT AS SHOWN
MODEL ASSEMBLY -3 SAME AS -2 EXCEPT AS INDICATED
AW430-1040 STING 'E' BALANCE
AW430-1081-1 FWD CANOPY USE AW430-1109-1 FILLER BLOCK WHEN CANOPY IS OFF
AW430-1046-1 TIE PLATE ASSY.
AW430-1058 SHOCK PLATE (LH)
AW430-1058-2 SHOCK PLATE (RH)
5/16 DIA. x 2 LG SOC. HD. STRIPPER BOLT REQ. 2
AW430-1052-1 ASSY FWD DUCT SHELL
AW430-1052-3 FILLER BLOCK ATTACH WITH #4-40 x 1/2 LG SOC. HD CAP SCREW REQ 5
SECTION AA
AW430-1051 NOSE
AW430-1089-1 CANOPY EXTENSION
AW430-1047-2 AFT FUSELAGE
SECTION BB
AW430-1032-1 VERT. TAIL ASSY
AW430-1021-1 WING ASSEMBLY
AW430-1047-1 CENTER FUSELAGE SHELL
AW430-1030-1 MOUNTING BLOCK
AW430-1034 STING 'G' BALANCE
AW421-3310-1 ASSEMBLY CAL 'G'-1-1-1 BALANCE
AW430-1052-5 COVER PLATE ATTACH WITH #4-40 x 3/8 LG FLT HD SOC CAP SCREW REQ 2
#8-32 x 3/4 LG SOC. HD CAP SCREW - (REQ 4)
#8-32 x 1/2 LG SOC HD CAP SCREW REQ 6
#8-32 x 1" LG SOC HD CAP SCREW REQ 2
#4-40 x 1/2 LG SOC HD CAP SCREW REQ 2
NOTE - ANGLE OF ATTACK INS. MUST BOTTOM IN MTG BLOCK BEFORE TIGHTENING SET SCREW
+ ANGLES ARE TRAILING EDGE LEFT MEASURED NORMAL TO HINGE LINE
AW430-1098 RUDDER
AW430-1080-2 LOWER CONE (LH SIDE)
AW430-1080-1 LOWER CONE (RH SIDE)
AW430-1117-1 ELEVATOR (LH)
AW430-1117-2 (RH) ELEVATOR
AW430-1099-1L BASIC L.E.
AW430-1099-2L EXTENDED L.E. (USED WHEN -1L BASIC L.E. EDGE IS OFF)
AW430-1099-1R BASIC L.E.
NOTE:
1. BALANCES ON BOTH -1 & -2 ASSEMBLIES ARE TO BE INSULATED AS PER RESPECTIVE CALIBRATION ASSEMBLIES WHEN ASSEMBLED WITH MODEL
1/24 SCALE WIND TUNNEL MODEL
AW 430-1022
CORNELL AERONAUTICAL LABORATORY
BUFFALO N.Y.

1/50 SCALE CANOPY MODEL WITH DORSAL AND NOSE FUSELAGE.

This model was designed and manufactured by Avro in May/55 and was used for High Subsonic and Rake Survey of Canopy and Dorsal investigations. It was tested in the 10 in. x 10 in. Supersonic tunnel at the NAE in Ottawa in June/55. Rake Surveys with original canopy and canopy modified in water tunnel. M = 0.71 and 0.88.

CF-105 WATER TUNNEL PROGRAM.

3/100 SCALE CANOPY MODEL WITH DORSAL AND NOSE FUSELAGE.

This model was designed and manufactured by Avro in May/55 and was used for the Water Tunnel Test with Visual Flow Check on Canopy/ Dorsal Combination. It was tested by the NAE, Ottawa in the 9.84in. x 13.11in. water tunnel in May/55. The test was to determine whether loss of fin effectiveness might be caused by flow breakaway around the canopy. Canopy modified for optimum flow.

FUNCTIONAL TEST RIGS AND SPECIMENS.

MECHANICAL TEST RIG.

This rig was used to check on the functioning, operating times and reliability of all systems on a complete set of aircraft mechanical equipment.

FUEL SYSTEMS RIG.

This rig was used to check on fuel transfer and flow rates, line losses etc., at all aircraft attitudes on a complete set of aircraft fuel system components.

PRESSURIZATION AND AIR CONDITIONING RIG.

This rig was used to check on the functioning of pressurization and cooling systems under all temperature conditions on complete aircraft nose, equipment bays, air conditioning and refrigeration systems.

ENGINE INSTALLATION RIG.

This rig was for the testing of engine installation and operation, engine cooling, afterburner operation, intake performance and starting. The rig incorporated the engine and afterburner, associated fuel system, fuselage intake approach surface and cowlings.

FLYING COCKPIT RIG.

This consisted of a CF-105 aircraft nose and cockpit mounted on a Mk2 CF-100 airframe, and was used for the suitability of control and equipment layout, take-off and landing evaluation of the cockpit.

AIRBORNE ARMAMENT.

This was to investigate and evaluate the aircraft missile launching systems when mounted on a Mk4 CF-100 for flight trials.

FIRE CONTROL EQUIPMENT INSTALLATION RIG.

This rig was used to check on the installation and functioning of fire control equipment in the radar nose and electronic bay.

STRUCTURAL TEST RIGS AND TEST SPECIMENS.

COMPLETE AIRCRAFT STRUCTURAL TEST.

A complete wing and fuselage structure was used in compliance with R-1803-9A as called for in ARDCM 80-1. Chapter 3.6.

CONTROL SURFACE LIMIT LOAD AND STIFFNESS TESTS.

To test the strength and stiffness of the control surfaces, mounting and trailing edge structures under limit loads.

COCKPIT PRESSURIZATION PROOF TESTS.

To test the strength of a nose structure when the cockpit is under proof pressure.

CANOPY AND WINDSCREEN PRESSURIZATION TEST.

This rig mounts the actual components for proof and ultimate loads under varying temperature conditions.

GROUND RESONANCE TEST.

This uses the Mechanical Test Rig together with the first model airplane in accordance with contractual requirements of ANC 12 as called for in ARDCM 80 Chapter 4.4.

ARMAMENT MOUNTING STRENGTH TEST.

To test the missile and rocket pack lowering devices with weighted dummy stores to proof and ultimate loads.

UNDERCARRIAGE AND TAIL SKID DROP TESTS.

To test the energy absorbing qualities of the landing units under proof and ultimate loads.

OUTER WING COMPRESSION TEST.

A 3 spar compression box was supplied to represent outer wing construction to test the compression strength.

INNER WING PANEL TESTS.

To check allowable stresses on representative skin and stringer panels.

TRANSPORT JOINT TEST.

To check stress distribution on sections of the transport joint.

EFFECT OF ELEVATED TEMPERATURES.

To determine heat flow, transient heat conditions and the effect of elevated temperatures on material properties in the aircraft structure.

FATIGUE TESTING.

To determine the fatigue life of details subjected to loads.

MOCK-UPS - CONTRACTUAL REQUIREMENTS ARDCM 80-1 CH 0-24.

COCKPIT AND ADJACENT STRUCTURE.

To represent crew stations, inclosure, windshield controls, seats, armament control, and instruments including radar and radio.

ARMAMENT INSTALLATION.

To establish space, layout and accessability of weapon installations.

CONTROL SURFACE AND SYSTEMS OPERATION.

To establish space, layout and accessability check of control surface mounting and operation.

MAIN UNDERCARRIAGE INSTALLATION.

To establish space, layout and accessability check of undercarriage installation.

RADAR AND FIRE CONTROL SYSTEMS.

To establish space, layout and accessability check of the radar and fire control system.

MISCELLANEOUS TESTS.

INTEGRAL FUEL TANK SPECIMEN.

To investigate the method of construction and sealing of the tank when under load using a sample box of the wing tank construction and sealing, and the suitability of sealants under high temperature conditions.

SIMULATED FUEL CELL.

To check on the ability of the pressure fuel system and valves to pick up fuel from shallow wing tanks by using a fibre-glass tank with observation windows.

SCALE MODEL FUEL SYSTEM.

Use of a 1/8 scale model to determine fuel levels under various fuel loadings and airplane attitudes.

BEARING TEST RIG.

To determine the deterioration of bearings under repeated loads at a high level. Bearings by various Manufacturers were used.

CONTROL SURFACE HINGE TEST.

To check on the deterioration of piano hinges under repeated loads at a high level. Sample hinges were used on the rig.

MOBILE COCKPIT RIG.

To investigate cockpit vision up to take-off attitude by using a vehicle mounted wooden cockpit mock-up. Undesirable reflections were eliminated by using this rig.

NON-FLYING PROTOTYPE.

This was a metal mock-up used to try installations and to train production crews in sequence manufacture.

WOODEN MOCK-UP.

Full size model of the airplane to check on installations.

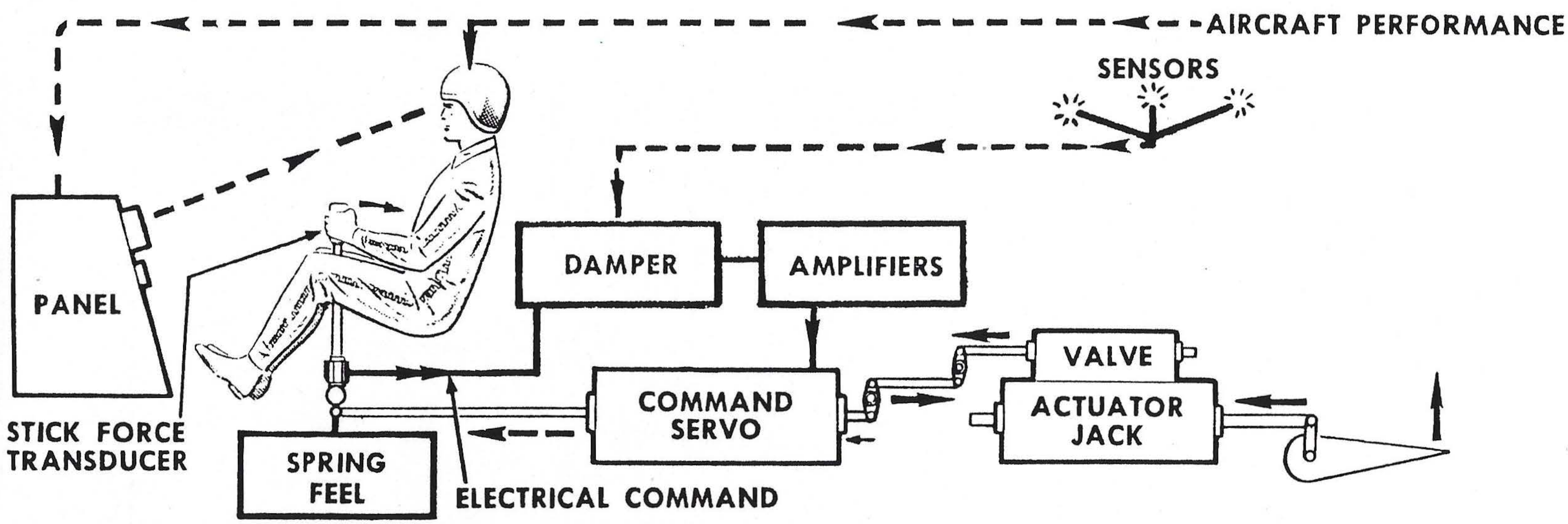
FLYING CONTROL SYSTEM.

A fully powered flying control system was selected for the CF-105 because:-

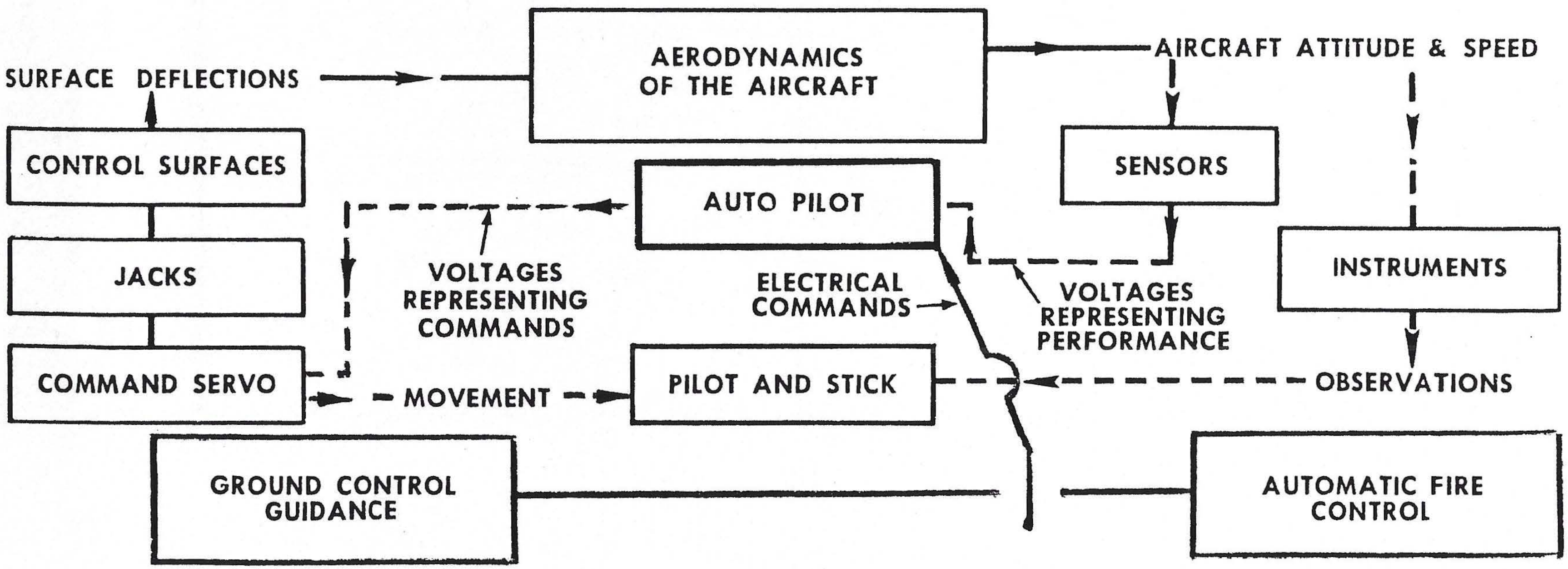
1. It appeared to be the only practical solution to the operation of such large chord and highly loaded control surfaces that were employed on the airplane.

2. Any other non-reversible type of system would require that the control surfaces be mass balanced, and in view of the large chord surfaces used, this would involve a prohibitive weight penalty.

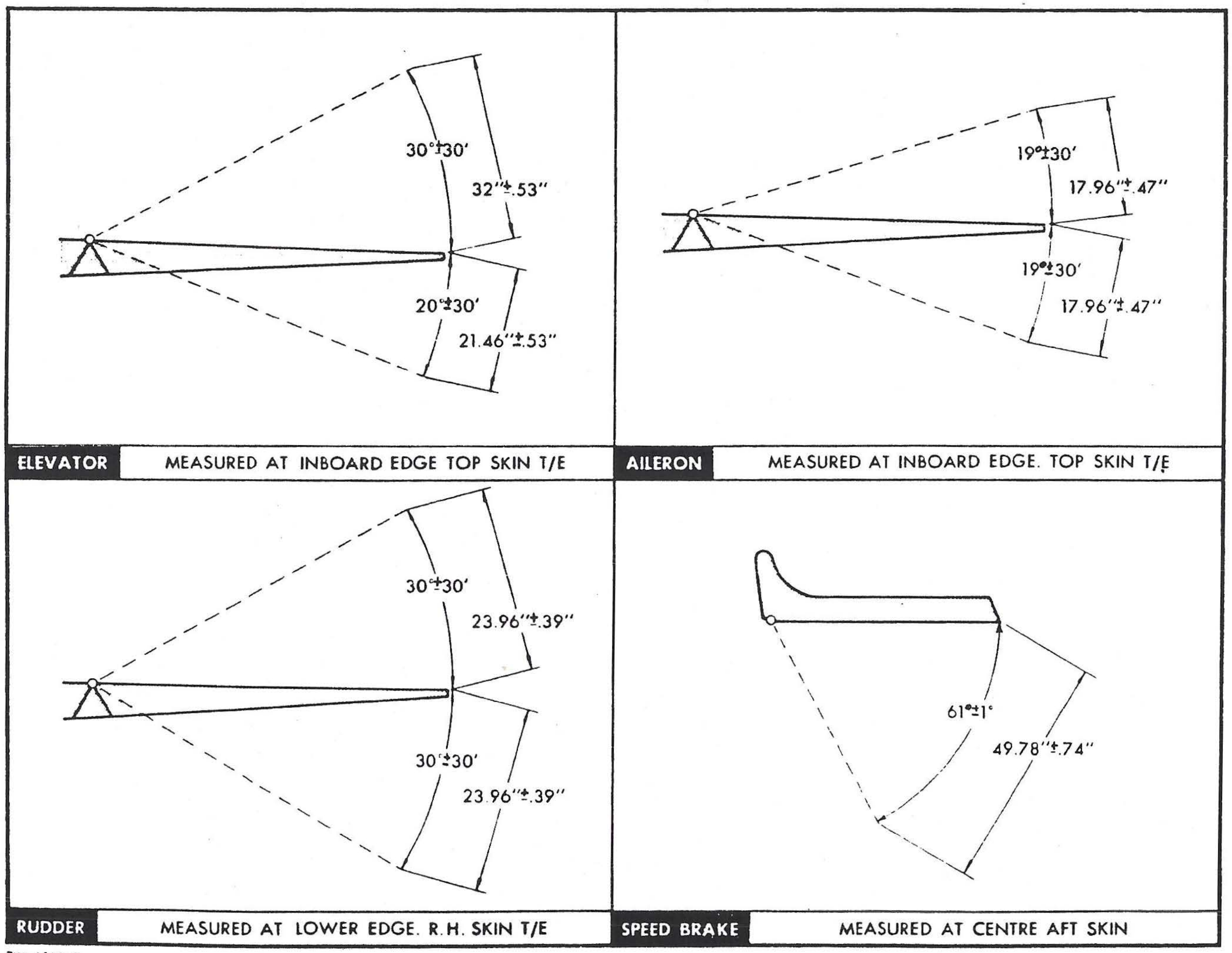
3. Any type of aerodynamically assisted control system employing geared tabs etc., was ruled out because of the unpredictable and variable nature of supersonic airflow as far aft on the wing chord. Having chosen a fully powered control system, it was necessary that a standard of reliability comparable to a mechanical system had to be achieved. To this end, a system employing a minimum number of reliable actuators was selected, and complete duplication of power systems was provided for.



FLYING CONTROL SYSTEM - MANUAL MODE (ELEVATOR)



BLOCK DIAGRAM OF FLYING CONTROL SYSTEM - AUTOMATIC MODE



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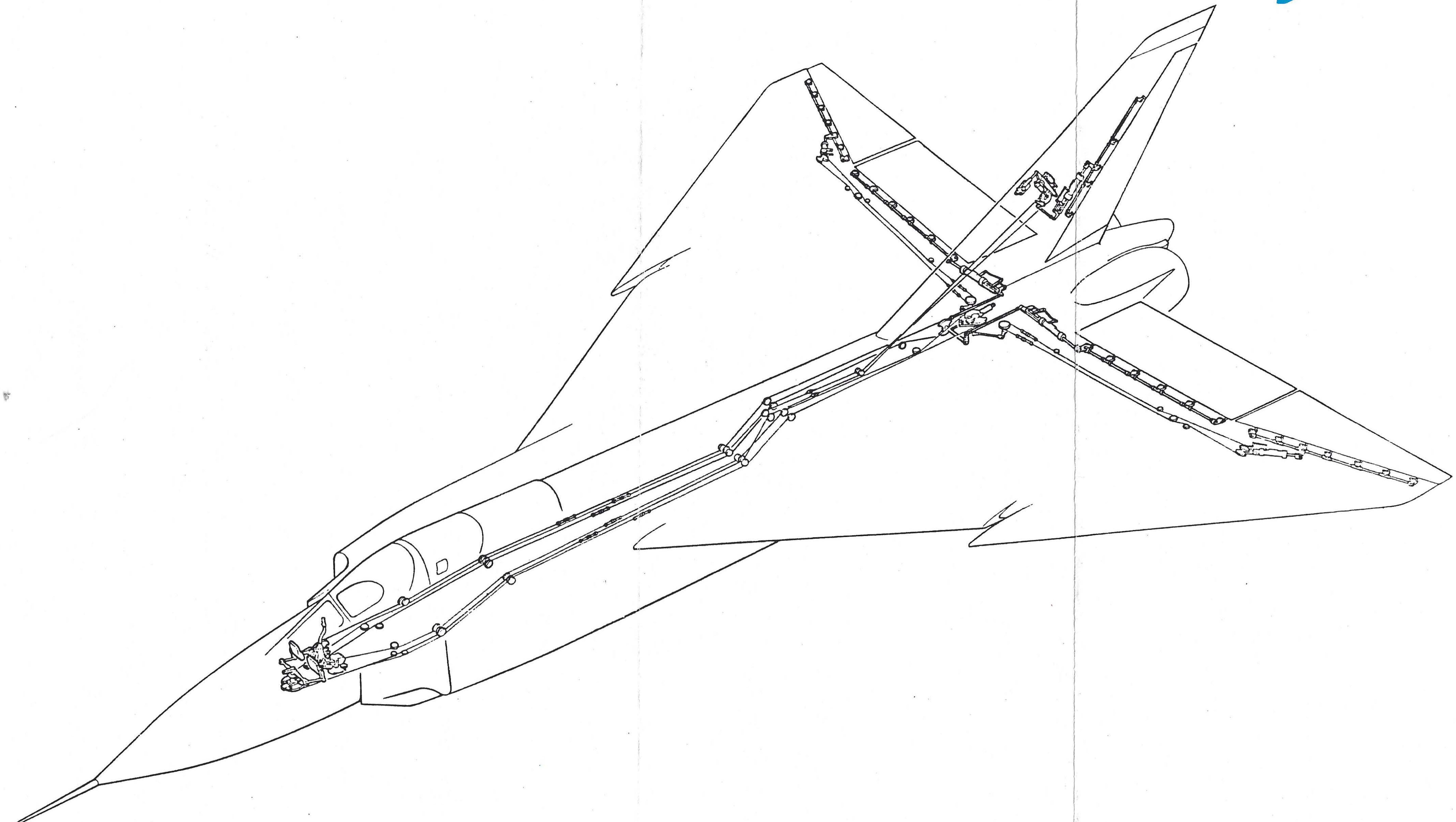
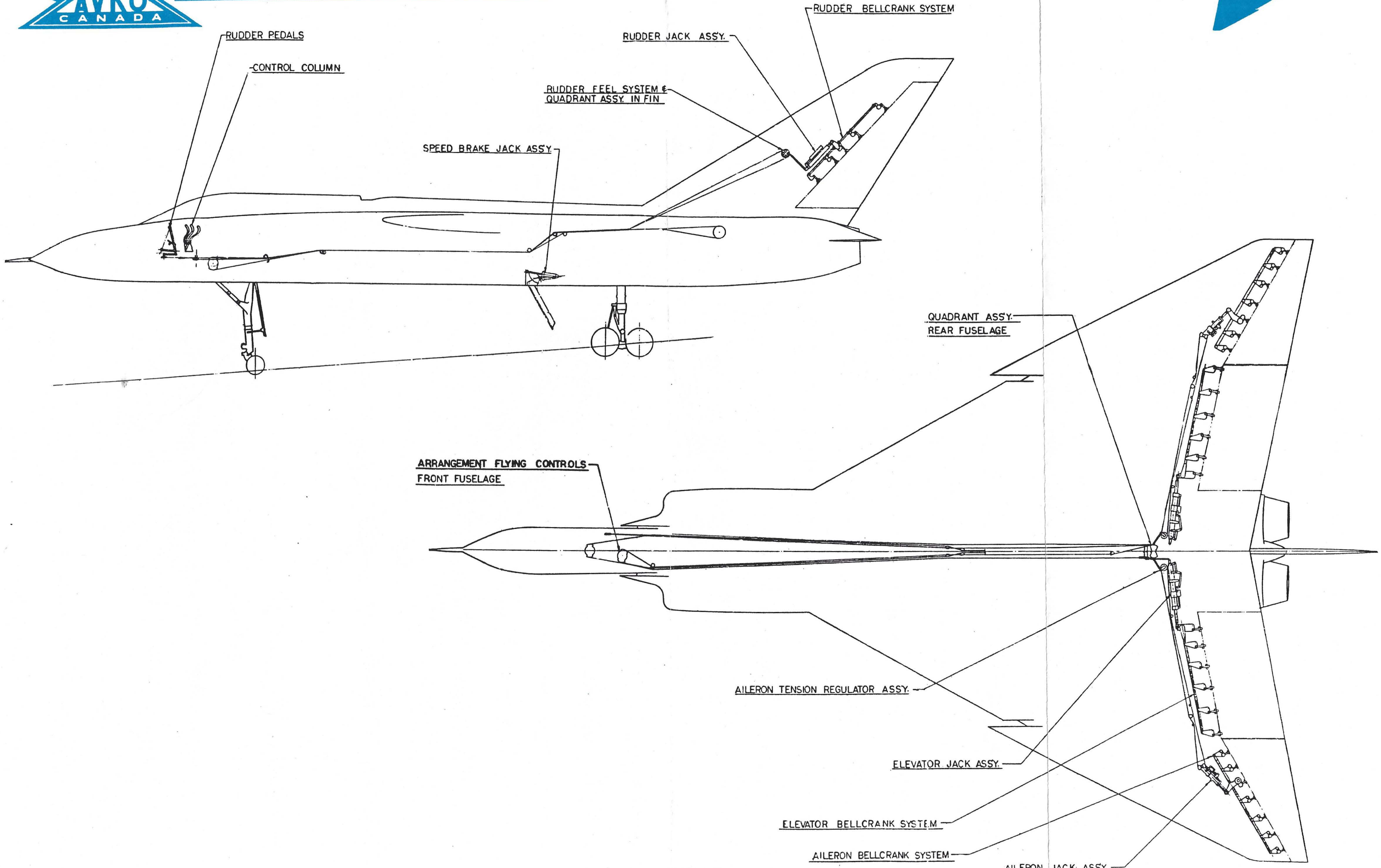
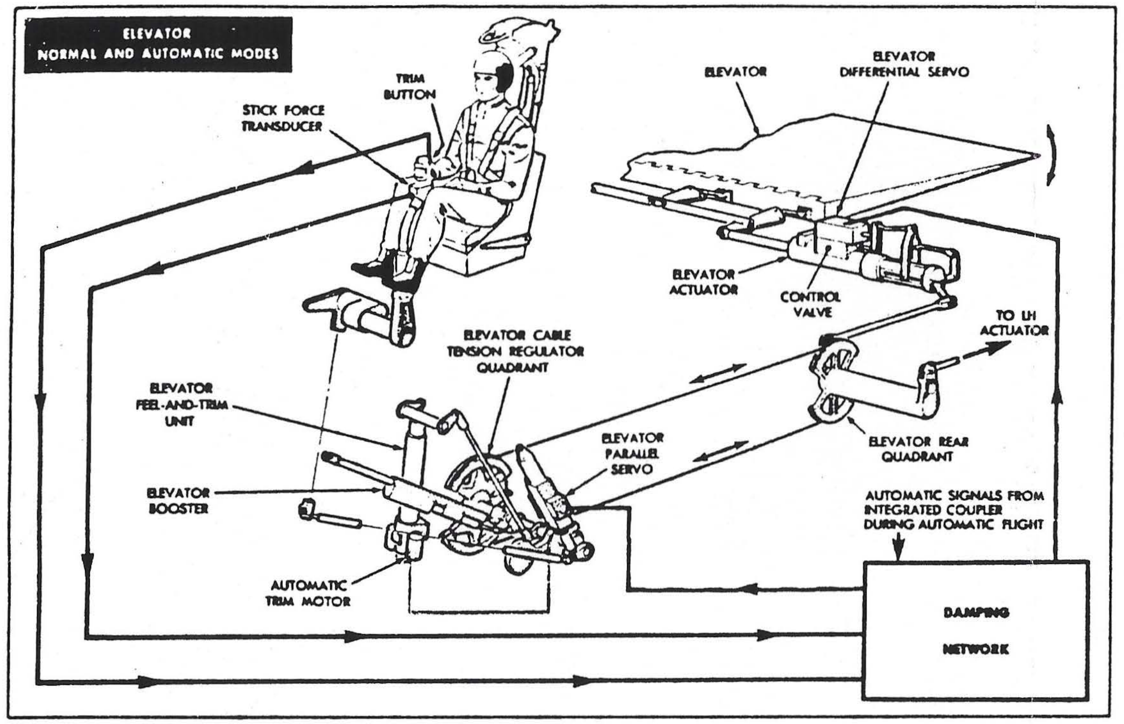
LIMITING ANGLES OF CONTROL SURFACE MOVEMENTS

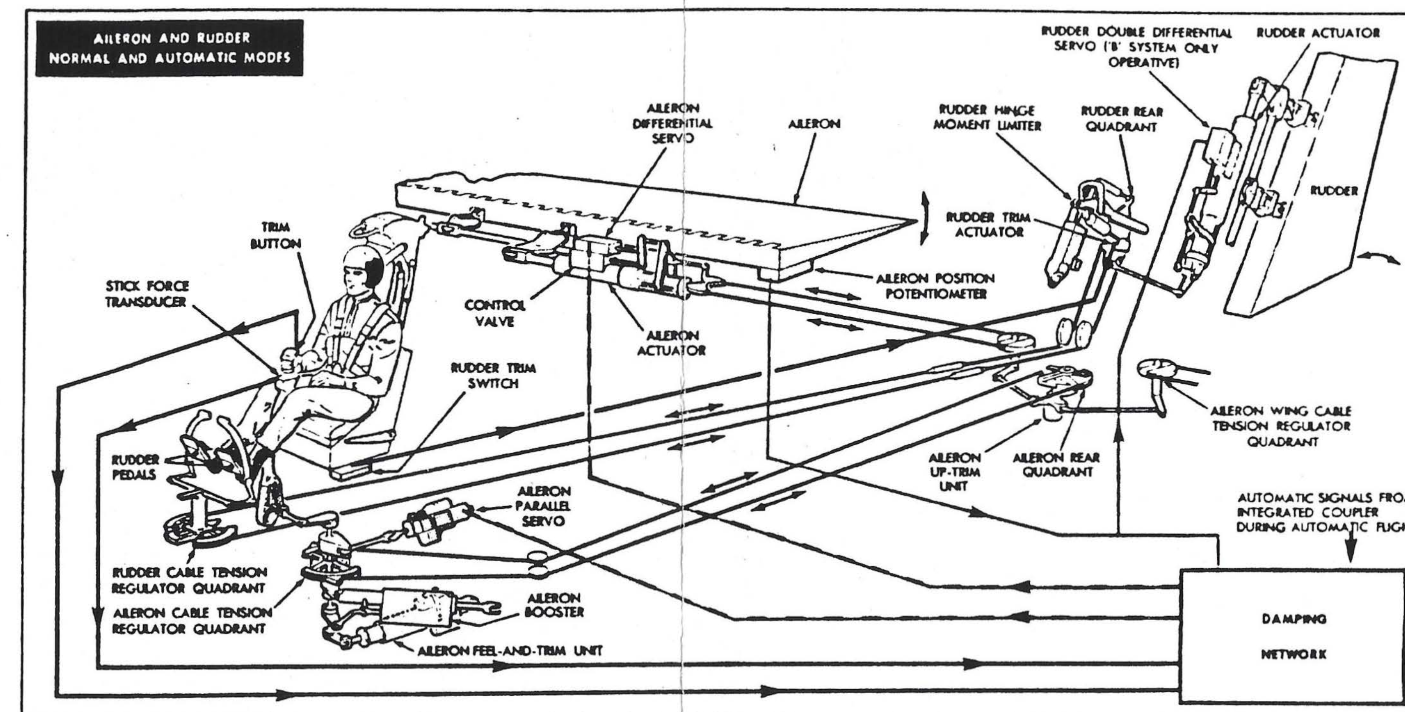


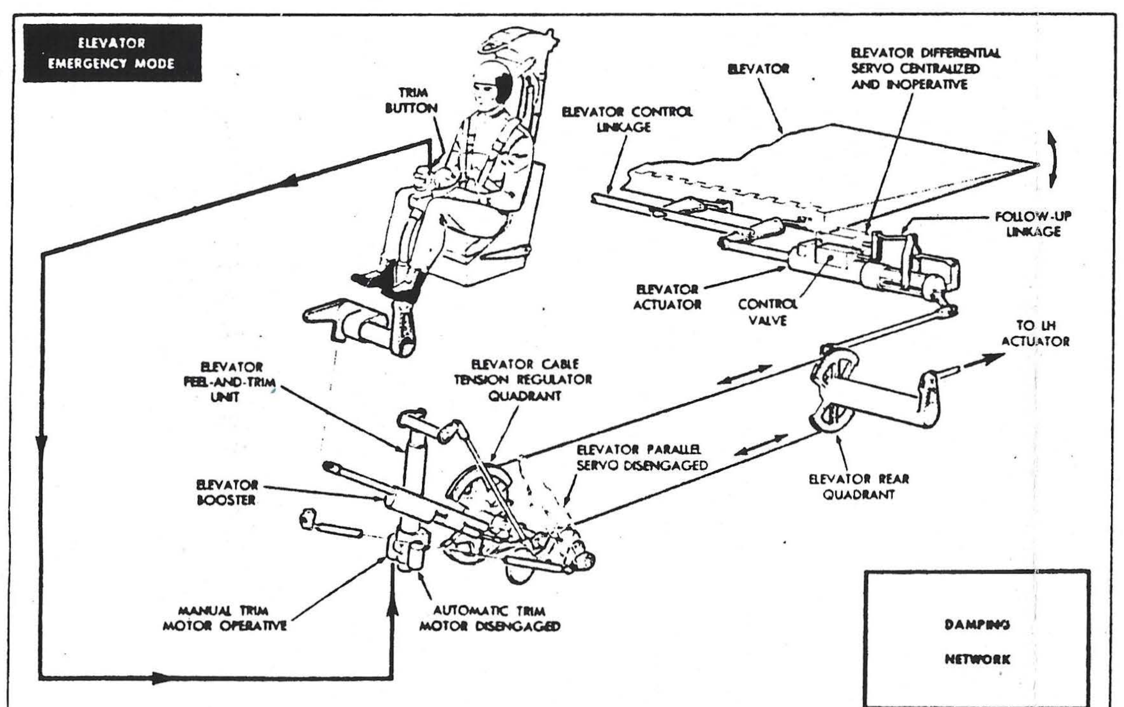
FLYING CONTROLS

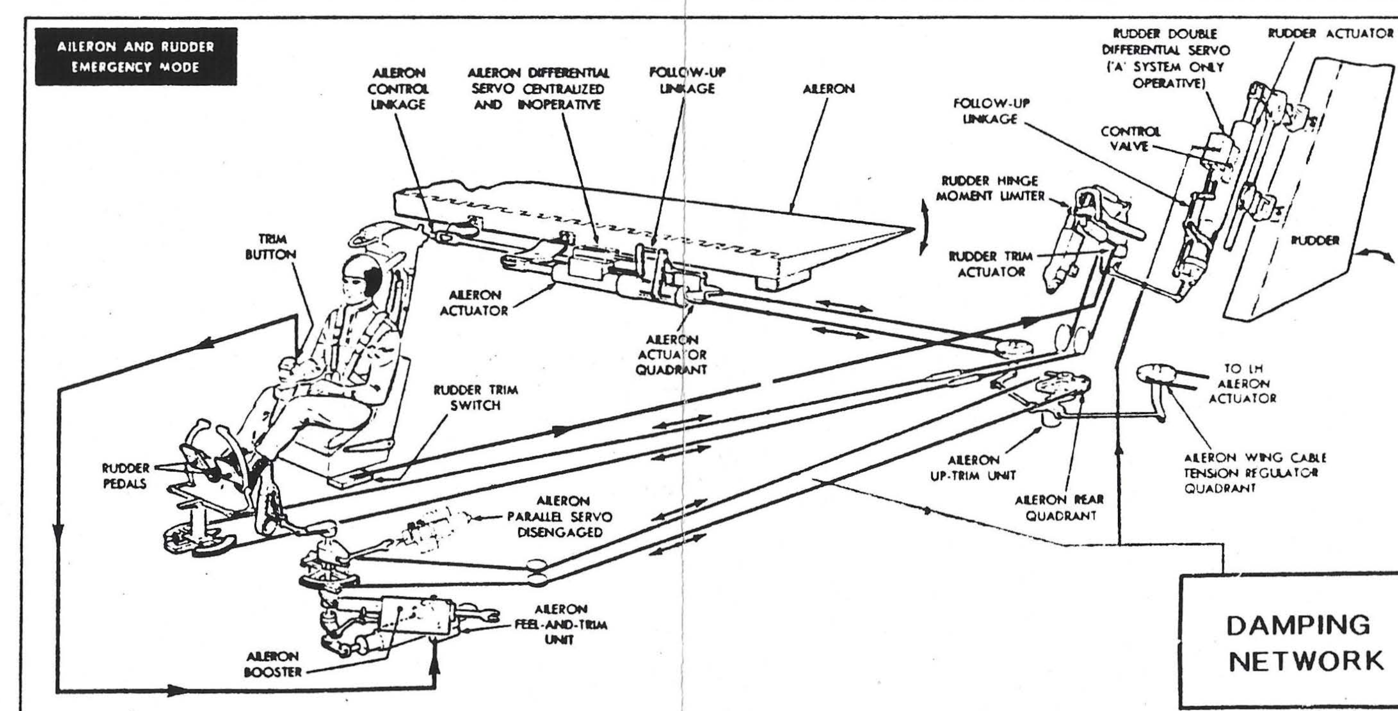


FLYING CONTROLS









COMMAND SIGNALS — DAMPING AND TURN CO-ORDINATION SIGNALS — COMBINATION OF COMMAND AND DAMPING SIGNALS — AUTOMATIC FEEL TRIM SIGNALS

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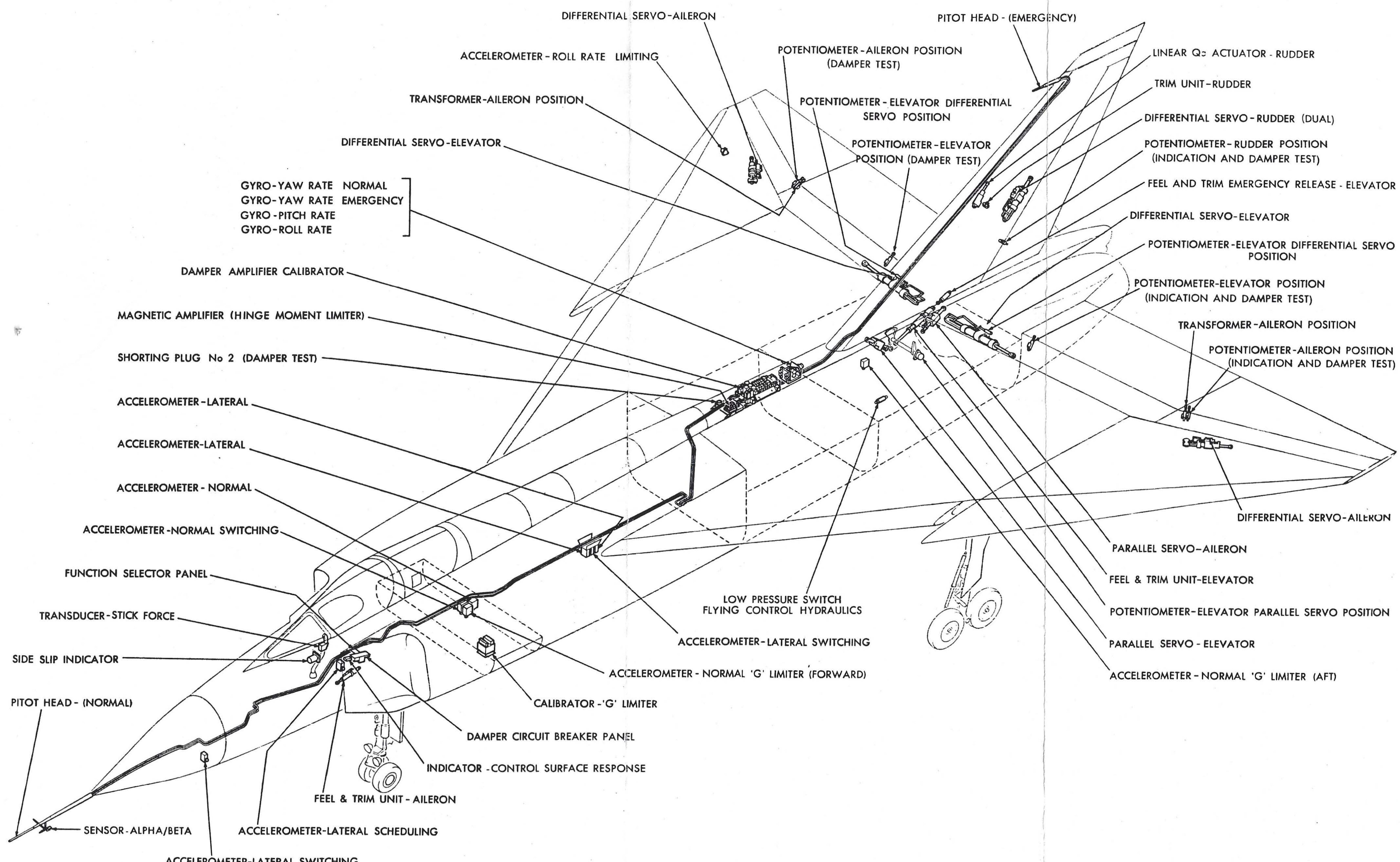


DAMPER AND ASSOCIATED EQUIPMENT LOCATION

An hydraulic power system was selected because of its reliability, actuator compactness, system response and performance under extreme environmental pressure and temperature conditions.

Because of the large power requirements for control of the aircraft, the hydraulic power was derived from engine driven pumps, one half of each flying control hydraulic system being driven from each engine. This arrangement provided for full control forces at half the maximum rate during single engine operation, and thereby permitted complete control during the most extreme asymmetric thrust conditions.

There were three modes of control of the aircraft, namely:-

1. THE MANUAL MODE.

This primary mode of operation used signals from the pilot input force transducers to control, through magnetic amplifier circuits, elevator and aileron command servos. These electro -hydraulic servos then operated the actuator cylinder control valves through short mechanical linkage systems.

During the command servo operation, turn co-ordination was provided by the aircraft damping system and the associated electronic coupling networks.

Pilot "feel" during this mode could be produced in any form found desirable, as it was comparatively simple to design a network which would produce as elaborate a feel system as was desired, taking into account stick force/g, dynamic pressure on the control surfaces, stick position, stick rate, etc.

2. THE AUTOMATIC FLIGHT MODE.

In this mode, signals were received from either ground guidance stations or the aircraft fire control system, and were fed by the automatic flight control system as signals to the command servos, to be transformed to control surface movements.

3. THE EMERGENCY MODE.

This mode was provided for use in the event of failure of the primary stick force mode, and involved a conventional cable and mechanical linkage system to directly control the surface actuator valves. To provide the pilot with adequate control system "feel" during this mode, stick forces were produced by positional feel springs and an elevator stick force per "g" bob-weight. Rudder pedal forces were developed through a similar positional feel spring, the effective spring constant of which was variable with qc. This latter feature was a requirement to prevent the pilot from inadvertently applying high side loads to the fin during supersonic flight.

4. ARTIFICIAL DAMPING SYSTEMS.

Artificial damping was provided about all three axis by a rate gyro system, feeding through a scheduling network to differential servos which supplied signals to the actuator control valves by differential movement of the pilot's input linkage. This system was operational only during the two normal modes and therefore there was no force feedback to the stick to make the pilot aware of the damping system operation. In the event of system failure, both automatic and manual means of shutting the system off were provided.

For all three control surfaces, bell cranks were selected as the coupling mechanism because of their simplicity and reliability, plus the fact that several bell cranks could be used for each surface thereby lowering the stress on each one. (The force on one elevator alone was about 60,000lbs.

5. SPEED BRAKES.

Speed brakes were supplied for subsonic use and were controlled manually by a selector switch and operated by two hydraulic jacks. In the interest of reliability of the Flying Control System, the brakes were powered by the Utility Hydraulic System. (See C-105. Dive Brake Performance.)

EVOLUTION OF THE CF-105 AND NOSE SECTION.

Much thought went into the design of the nose section, due not only to aerodynamic problems but the vacillating requirements of the RCAF concerning various diameters of radar dish. Several configurations were under consideration and in Dec./54 longer nosed and faired windscreen sketches were prepared. Mr Ken Barnes, a Senior Design Engineer in the Project Design Office, was kind enough to loan the Author originals of the following sketches.

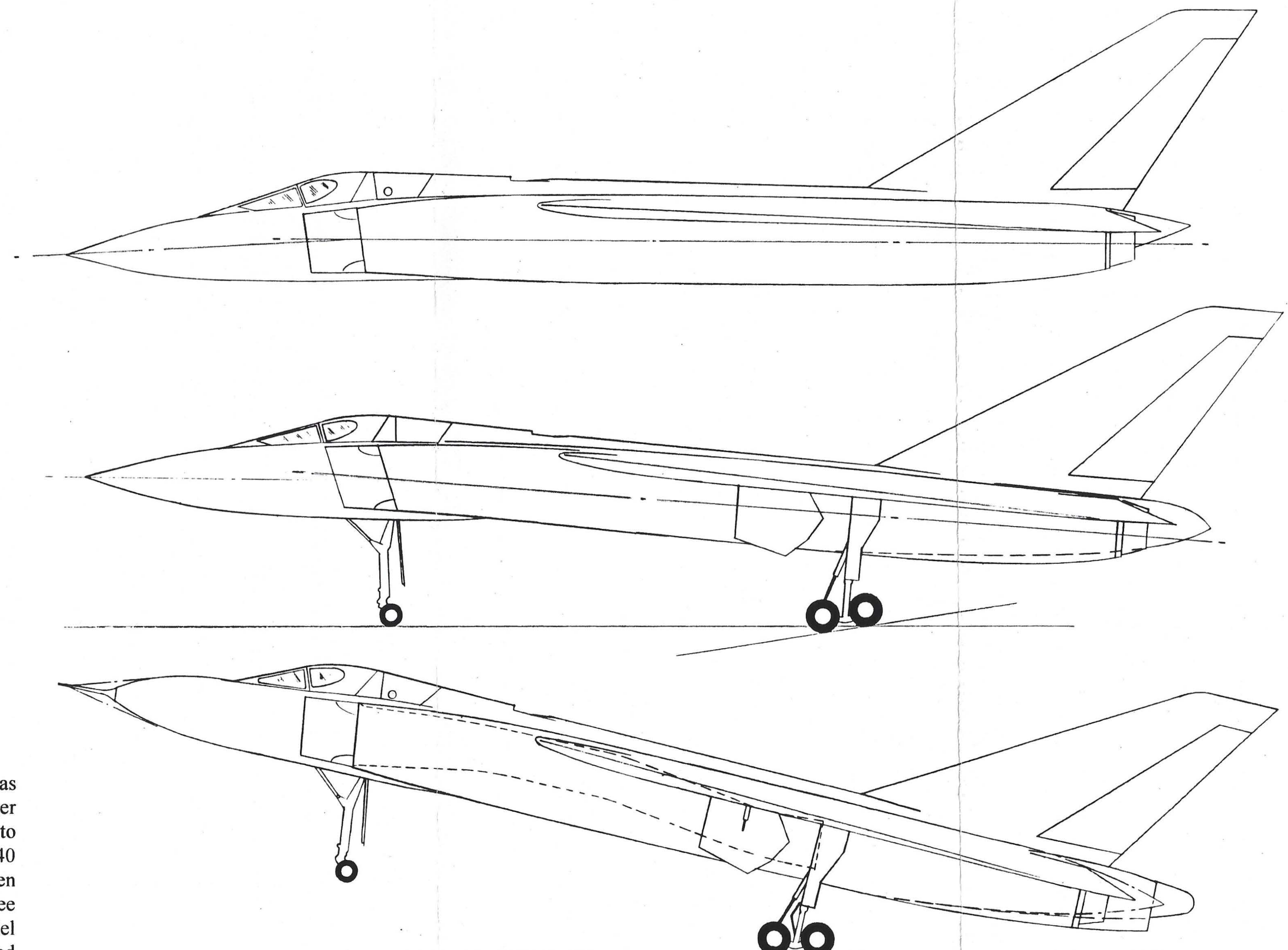
The version settled upon was the longer nosed 30 degree radome as seen on the Arrow Mk1. It should be noted here that according to the Arrow Mk 2 Weights and Measures report 7-0400-34, Iss. 16 and 25, sheet 2, the projected production Arrow Mk2 radome would have been about 12" shorter than that of the Mk1. Results of the wind tunnel tests and area rule investigations were also becoming available and the development of the tail cone and fuselage refinements were beginning to show.

FREE FLIGHT MODEL PROGRAM.

AIM OF FREE FLIGHT TESTS.

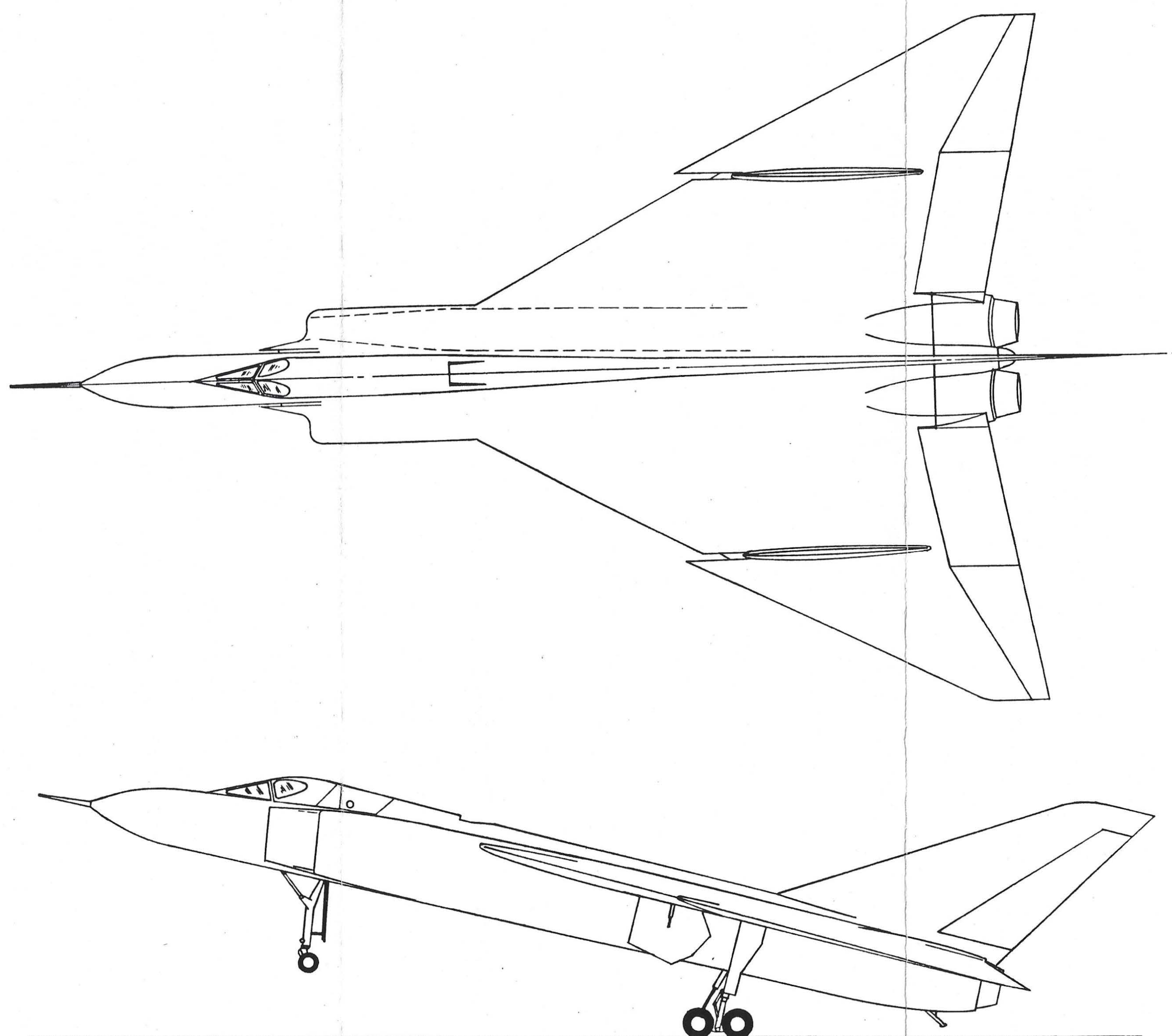
The original purposes of the Free-flight Model Tests were to provide dynamic stability and control data for the C-105 in the pitching plane, and dynamic stability data in the yawing and rolling planes. Model speed would cover most of the C-105 supersonic and transonic speed range, while the model, with dimensions and inertia to scale, would be free to move in all planes.

Design and manufacturing difficulties in the elevator operating system delayed the completion of the longitudinal stability models, while the mechanism to produce the



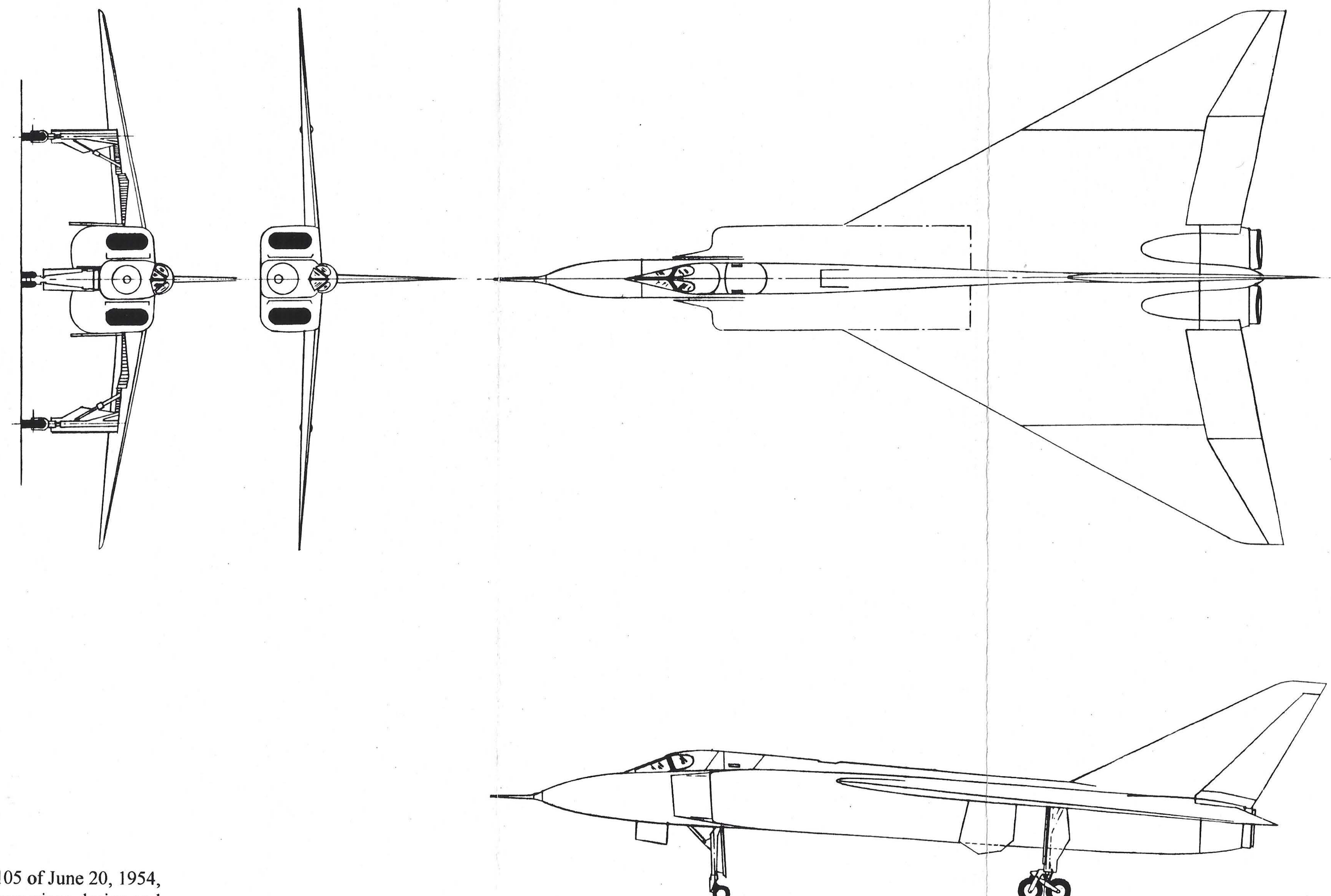
On Dec. 21, 1954, the C-105 as contemplated with a longer and larger nose. The front fuselage also was shown to be wider in order to accommodate a 40 inch diameter radar dish. This can be seen to better advantage by consulting the Free Flight 1/8 scale model drawing of model number 7. The tail designs were introduced for converting to the J-75 engine, together with nacelle and tail cone modifications.

C-105 J75 ENGINE



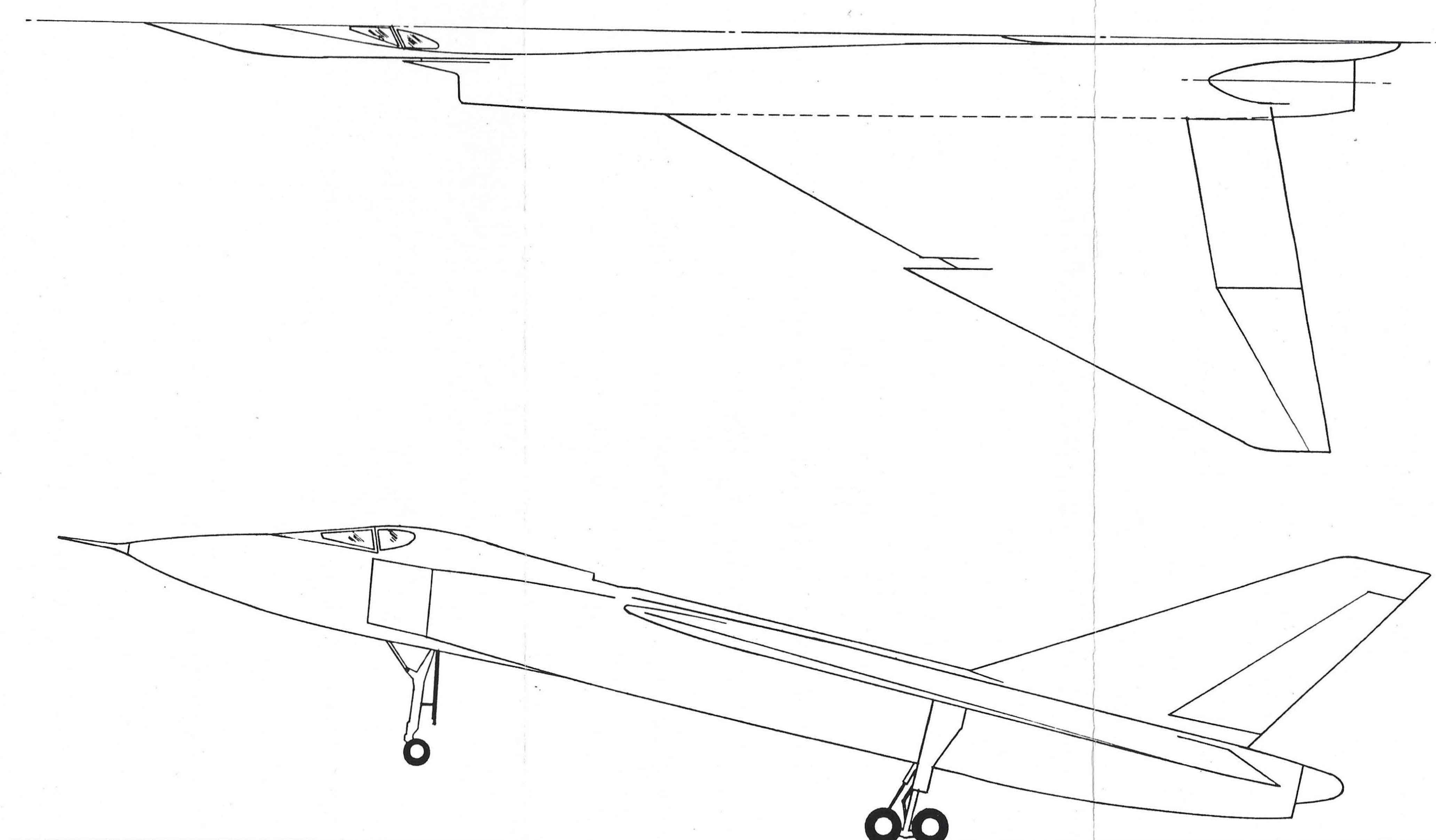
The C-105 as it would have appeared as of Nov. 26, 1954, now with the J-67 engine. Intake ramps and refined undercarriage are also shown.

C-105 J67 ENGINE

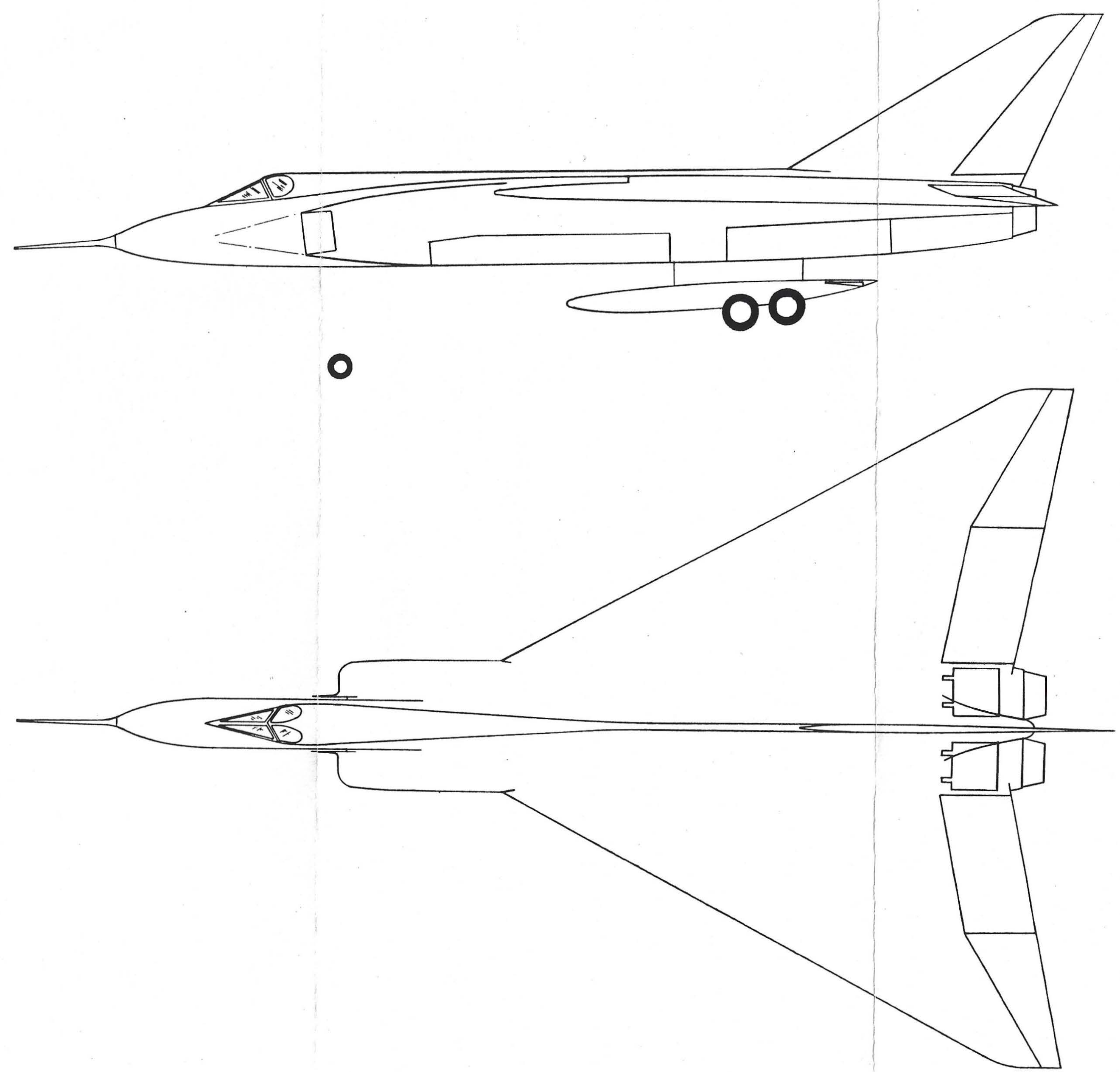


The now named C-105 of June 20, 1954, showed the early undercarriage design and refining of the intakes is apparent. The RB-106 engine was still contemplated.

C-105 JUNE 20, 1954



The now named CF-105 with the J-75 engine, refined tail cone together with slotted and extended outer wings. The cockpit at this stage still has the extended profile.



C-104 version 'U'. This displayed the sweep-back of the trailing edge of the wing and an increase in the wing span of 2 ft. from the 48 ft. Of the C-104/2.

C-104 VERSION "U"

yawing disturbance in the directional stability models had yet to be proven in a crude model. In the meantime it was decided to go ahead with another phase of the program, that of determining aircraft drag from free-flight model tests.

Up to this time the only experimental data for the C-105 was from Wind Tunnel Tests, with the models, in both .03 and .04 scale, mounted on a "sting". There were several possible causes of inaccuracy in tunnel measurement of drag; the effect of the "sting", relatively low Reynolds Number of test, and the difficulty of making an accurate strain gauge drag balance free from interaction of the other components.

A more accurate assessment of C-105 drag was possible from free-flight tests, because of freedom from interference, much higher Reynolds Number and more reliable means of drag measurement. The effects upon aircraft drag of two "Area Rule" modifications to the fuselage and canopy contours, were also investigated in this series of free-flight tests.

The decision to embark on a series of free-flight tests using C-105 models was made in the middle of 1953. A ground launch method was chosen, in which the model was accelerated up to flight speed buy a booster rocket before separation of the booster. While in free-flight, subsequent behavior of the model was determined from data radioed, or telemetered, down to a ground station from equipment contained in the model.

Choice of the ground launch technique was made in preference to other methods, such as air launch from an aircraft, or testing in a ballistic range. In ballistic tests, an elegantly simple system of obtaining early design data, a very small scale model of the aircraft was fired from a large calibre gun; however, the model usually carries no instrumentation, accuracy is limited and speed range restricted. Air launching utilizes gravity force to accelerate the model, so that maximum speed is usually limited.

Even if the model was rocket-boosted, control and measurement of trajectory and speed is difficult. Using the ground launch, speed and trajectory may be carefully controlled and measured, while accurate telemetry measurements are made easier.

The first four models fired were "crude" models and the last three were representative or "drag" models. Subsequent models were disturbed while in free-flight, in the directional and pitch planes, to ascertain the stability both laterally and longitudinally.

After an assessment of the data to be telemetered from the model while in flight, and the internal space therefore required for the appropriate instrumentation and electronics, and also in order to obtain the greatest test Reynolds Number, a model scale of one-eighth full size was decided upon.

Free-flight models were equipped with an FM/FM telemetering system utilizing standard R.D.B. channels. Various booster motors and combinations of booster motors were considered, the one being chosen was the "Nike" booster (JATO XM5) of approximately 45,000 lbs thrust and 150,000 lb. sec. impulse.

During model construction, considerable difficulty was experienced in the manufacture of accurately profiled wings for the scale models for the drag and stability tests. Initial efforts to cast them in aluminum alloy were unsuccessful owing to warping of the castings, and efforts to correct the warp mechanically, failed. Machining the wing from cast billets of magnesium alloy also proved unsatisfactory, and the model wings were finally machined from rolled billets of magnesium alloy. As and interim measure, for model #5, a composite fabricated wing was used.

During the program, an attempt was made to keep up-to-date with design changes. A description of each of the first seven models together with their launch dates and location follows. All models were to 1/8th scale with the exception of the fins which were made oversize to ensure model stability, and the wings of models 5,6 & 7 had 0.75% negative camber as on full scale. Models 8 & 9 were lateral stability models and numbers 10 & 11 were longitudinal stability models. Models 1 to 5 and 8 to 11 were fired at the Point Petre Range of the CARDE near Picton, Ontaio, and models 6 & 7 were fired at the Wallops Island Range of the NACA Division of the PARD in Virginia, USA. All models were launched from mobile "zero-length" launchers, placed on a concrete firing ramp.

CF-105 FREE FLIGHT MODEL PROGRAM.

1/8 SCALE CRUDE MODELS. (Models 1 & 2).

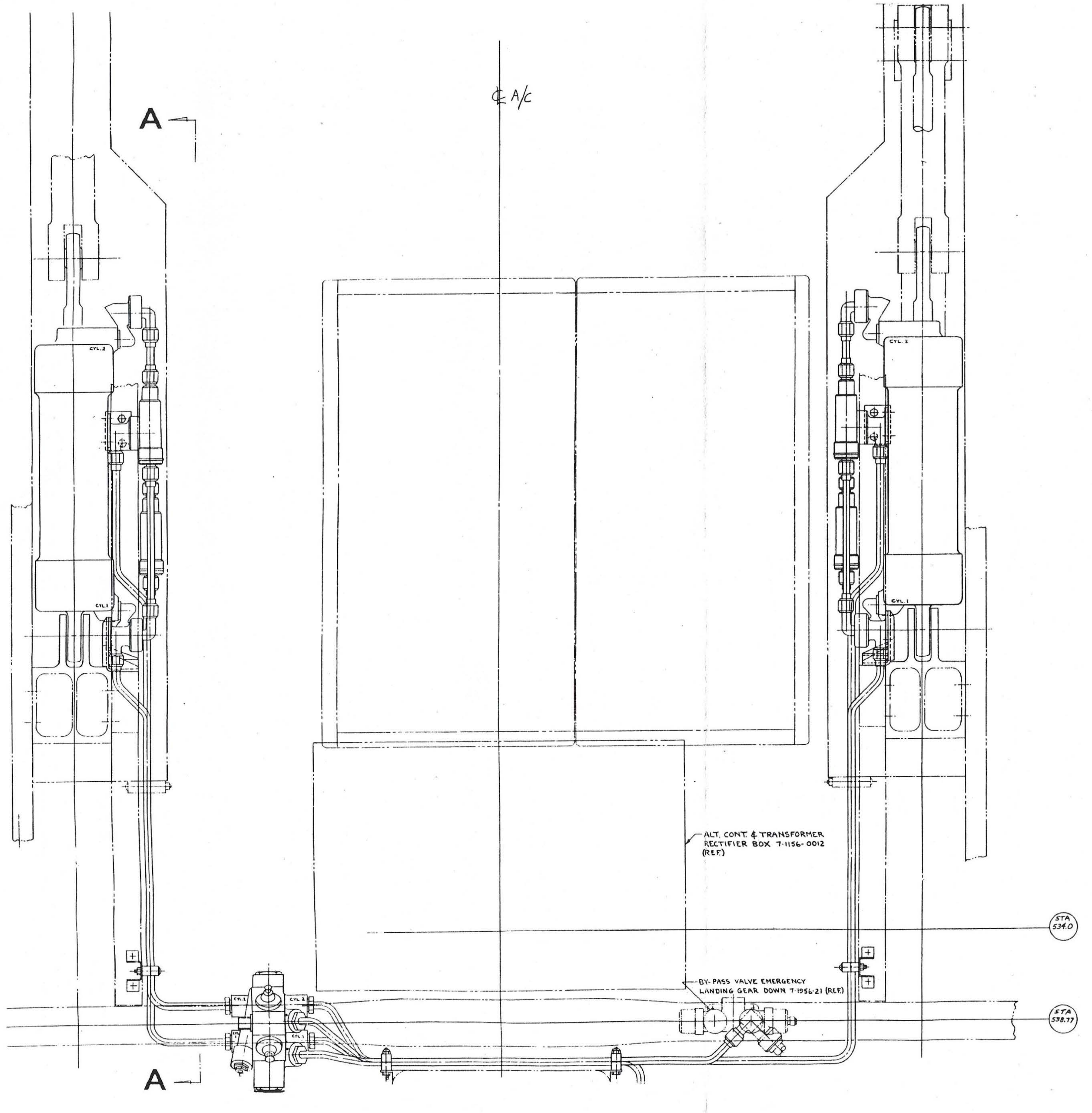
Designed and manufactured by Avro in Dec./54, these two models were for checking the firing technique, Telemetering and Tracking. They were attached to NIKE boosters and fired out over Lake Ontario at the CARDE range at Picton. The test date was Dec. 15/54.

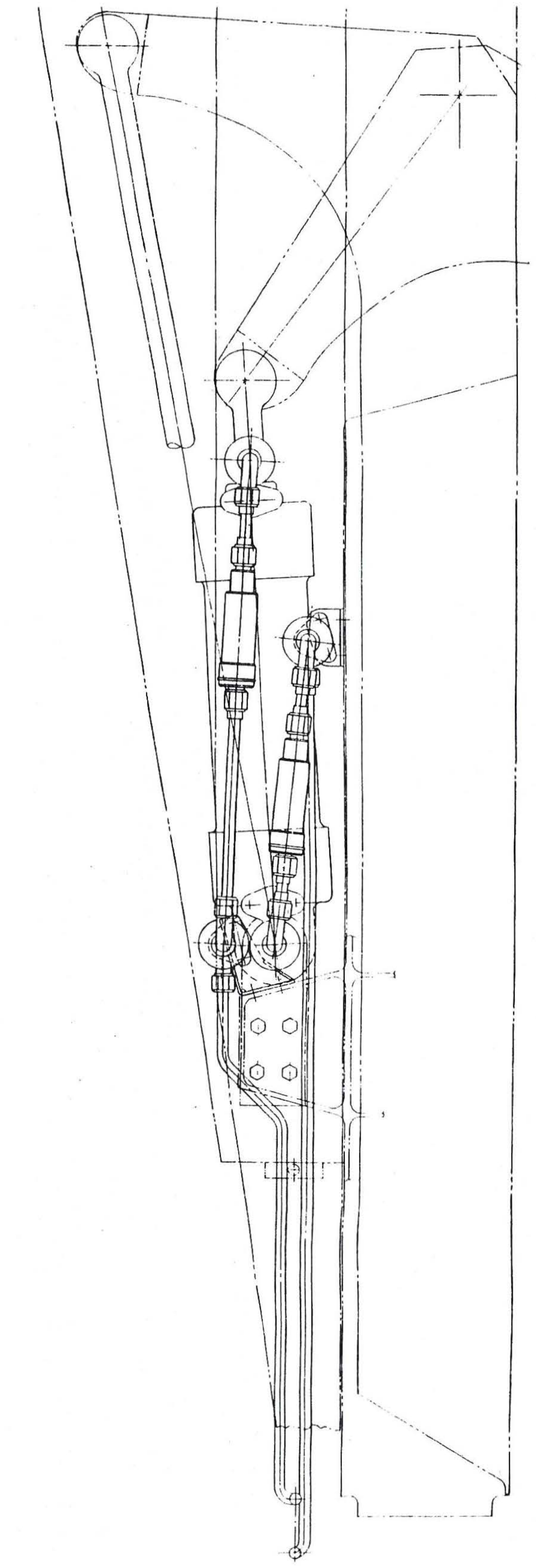
1/8 SCALE CRUDE MODEL. (Models 3 & 4).

Two additional crude models were made, again at Avro, in Apr./55 for the purposes of checking the functioning of Yaw Impulses. They were fired at the CARDE range at Picton, Ontario on May 1/55.

1/8 SCALE DRAG MODEL. (Model #5).

This was the first drag model, and was finished in Apr./55 and used as a Telemetry System Check and Preliminary Drag Check, including Flow through Air Intakes and Ducts. The C of G was at .25 MAC with 8% notch with no extensions. The model was equipped with J-67 intakes and duct and J-75 rear fuselage. The control surfaces were fixed. It was fired at the CARDE range at Picton, Ontario on June 15/55.





SECTION A-A

ARRANGEMENT OF DIVE BRAKES

1/8 SCALE DRAG MODELS (2) INCLUDING EXTENDED LEADING EDGE, NOTCH AND DROOP. (ONE MODEL TO INCLUDE AREA RULE MODIFICATIONS.)

These two models were completed in June and July/55 and were used to check Drag with two Different Air Intakes and Ducts. They were intended to be fired at the CARDE range in Picton Ontario, on Aug. 26/55 and the second on Sept. 30/55., but Doppler Tracking delayed the launching until they were finally fired at the NACA Station, Wallops Island, Virginia, USA on May 9 and 15/56.

Model #6.

This model was the second drag model. The C of G was at .25 MAC with a drooped leading edge, 5% notch, 10% extension (outboard of notch) and a 30 degree conical radome. Intake duct and rear fuselage were for the J-75 engine, with pressure rakes in the duct. Partial area ruling was employed with fixed control surfaces.

Model #7.

This was the third drag model. The C of G was at .25 MAC with a drooped leading edge, 5# notch, 10% extension (outboard of notch), and a 30 Degree radome. Intake duct and rear fuselage were for the J-75 engine, with pressure rakes in the duct. Special area ruling was applied with fixed control surfaces.

1/8 SCALE YAW STABILITY MODELS (2) INCLUDING EXTENDED LEADING EDGE, NOTCH AND DROOP PLUS PARTIAL AREA RULE MODIFICATIONS.

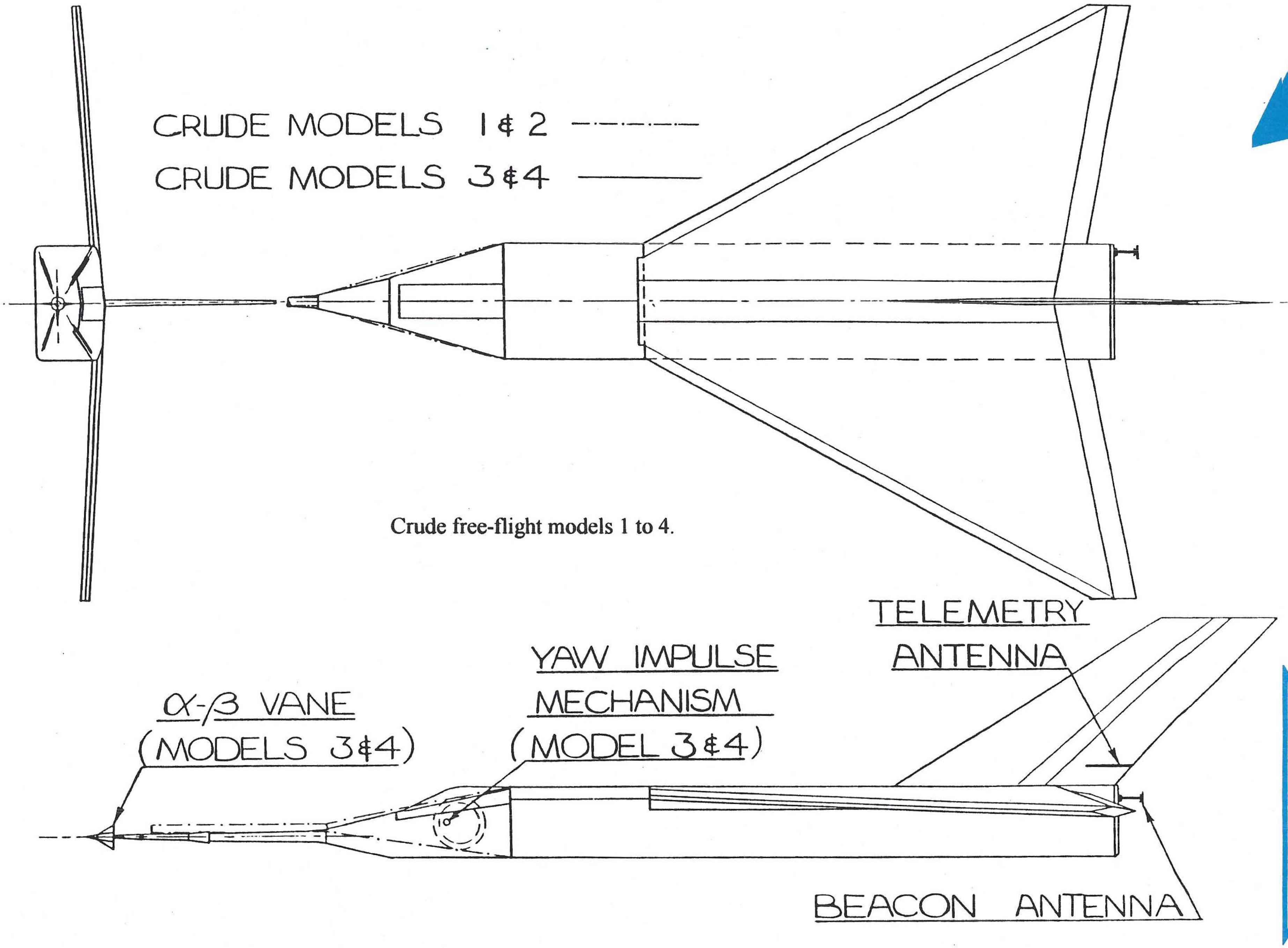
These two models were completed at Avro in Sept./55 and were used to check Directional Stability. They were scheduled to be fired at the CARDE Range in Picton, Ontario, on Oct. 31/55 but this was delayed by a cost investigation. They were however, fired at a later date.

Model #8.

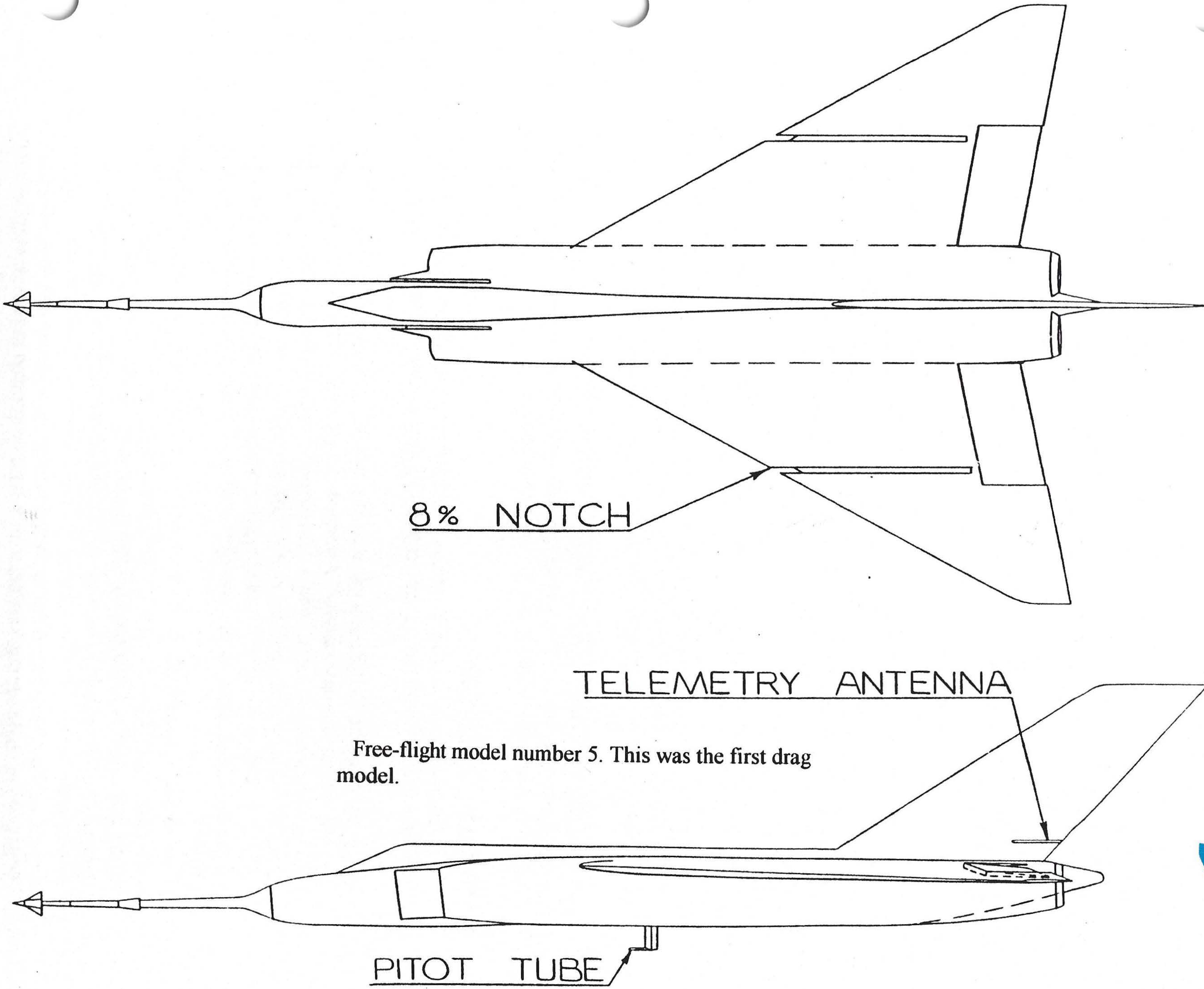
This was the first lateral stability model with the C of G at .25 MAC, with drooped leading edge, 5% notch, 10% extension (outboard of notch) and 30 degree radome. Final J-75 intakes were applied with 30 degree conical radome. The model had partial area ruling and fixed control surfaces, with yaw impulse mechanism fitted, and the model ballasted to "raise" the principal axis to a more representative position.

Model #9.

This was identical to model #8 except that it had boundary layer ejectors.



Crude free-flight models 1 to 4.



Free-flight model number 5. This was the first drag model.

1/8 SCALE MODELS WITH MOVEABLE ELEVATORS (2) INCLUDING EXTENDED LEADING EDGE, NOTCH AND DROOP PLUS ALL AREA RULE MODIFICATIONS.

These two models were completed at Avro in Oct./55 and were used to check Lateral Stability. They were scheduled to be fired at the CARDE Range in Picton, Ontario, on Dec. 15/55 but this was delayed by a cost investigation. They were however, fired at a later date.

Model #10.

The C of G was established at .20 MAC and droops, notches and extensions together with a 30 Degree nose were applied. Final J-75 intakes and rear fuselage were provided together with moveable elevators (hydraulic operation) and the model was ballasted to adjust principal axis. All area rule modifications were applied.

Model #11.

This was the same as model #10 except that the C of G was at .27 MAC.

FFM'S #6 to #9 has a static pressure probe in front of the vane: on #10 and #11 this was removed and reasonably good readings were obtained, as on model #5. Outsize fins were provided on all models for stability.

1/8 SCALE SPARE MODEL WITH 5 BOOSTERS.

This spares program was to have been decided after the results from the preceding tests were finalized, however, it was not proceeded with.

INTAKE DESIGN.

The design of the intake ducts for the C-105 aircraft, due to their long nature and structural integrity with the rest of the fuselage structure, are worthy of note.

The intakes of the C-105 are located on the side of the fuselage about 14 ft. from the nose. The are approximately "D" shaped and external compression is achieved by a two-dimensional ramp with a 12 degree wedge attached to the side of the fuselage. The duct from inlet to engine diffuses from 5.6 sq.ft. at the inlet to 7.0 sq.ft. at 9 ft. from the inlet - it then has a constant diameter circular section for a distance of 22 ft. back to the compressor face. The duct area variation curve is shown.

Immediately upstream of the compressor face is located a flush intake of variable area which completely encircles the duct. This intake opens into the engine bay and its purpose is to bleed air from the main intake duct into the bay, the air then being dumped through a suitable exit at the rear. It was the intention at the time for reasons of simplicity, to have the flush 'bypass' intake fully open at Mach numbers greater than 1.5 and to close it to a fixed intermediate setting at speeds less than Mach 1.5. At this intermediate setting, sufficient air would be allowed to pass through to cool the engine etc. The maximum area of the bypass was chosen so that the main intake

FREE FLIGHT MODEL NUMBER 6
(DRAG MODEL NUMBER 2)

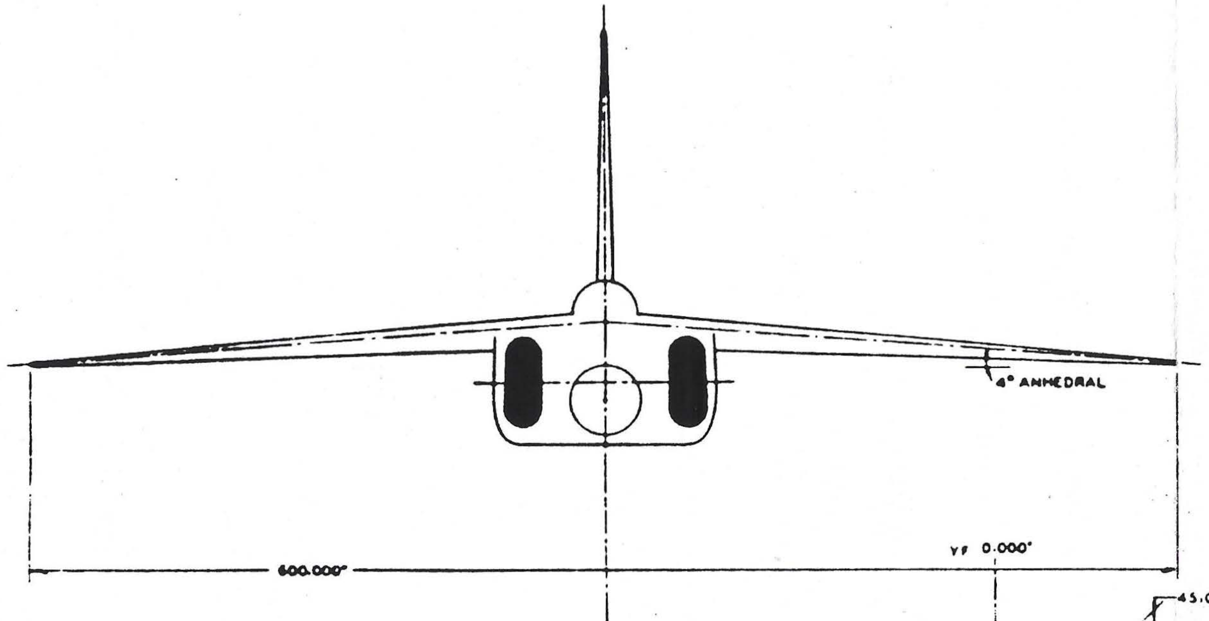
FIRING DATE...........MAY 9. 1956

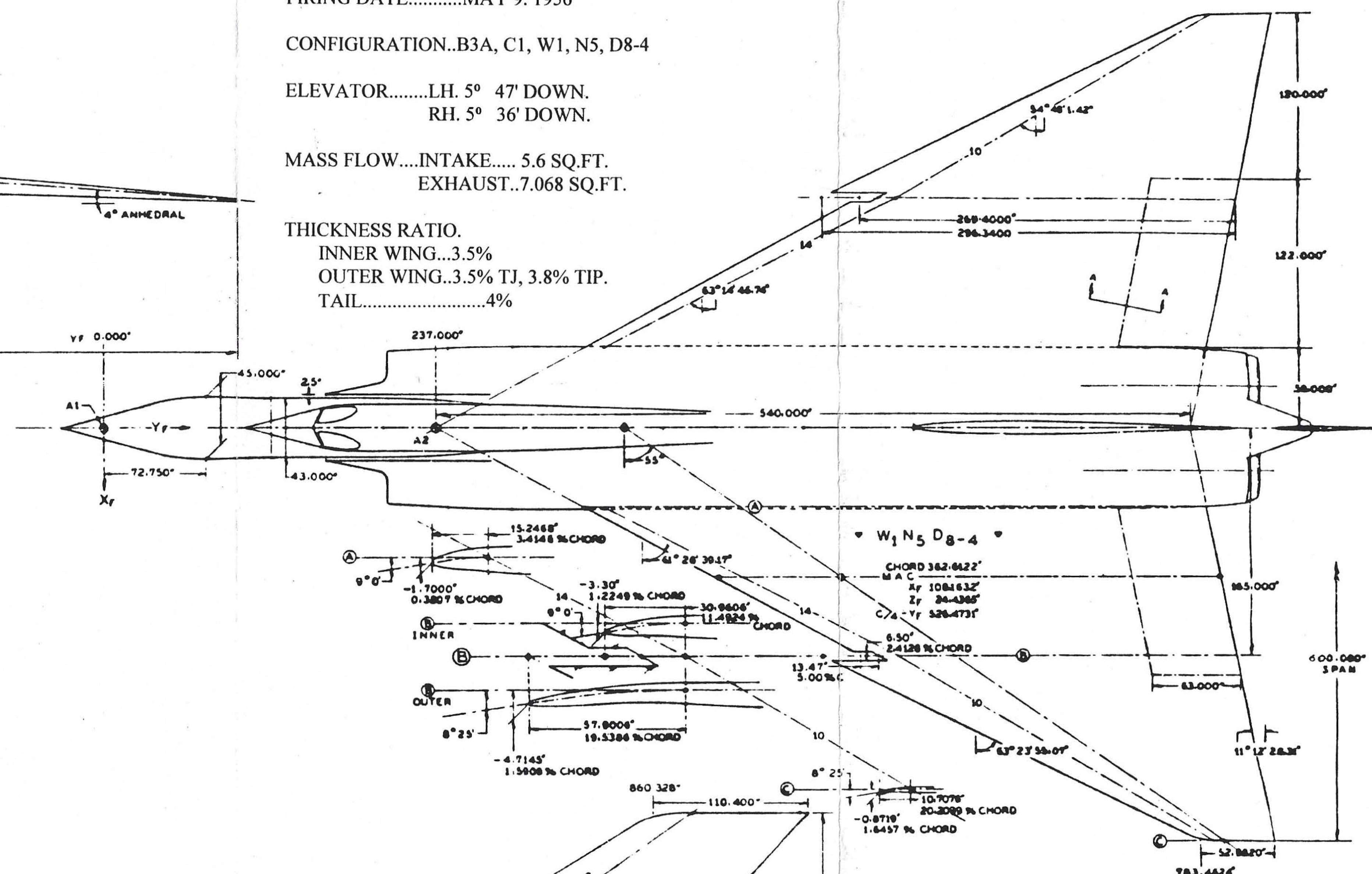
CONFIGURATION..B3A, C1, W1, N5, D8-4

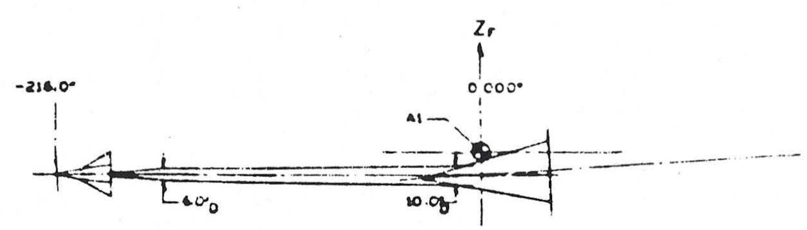
ELEVATOR........LH. 5° 47' DOWN.
RH. 5° 36' DOWN.

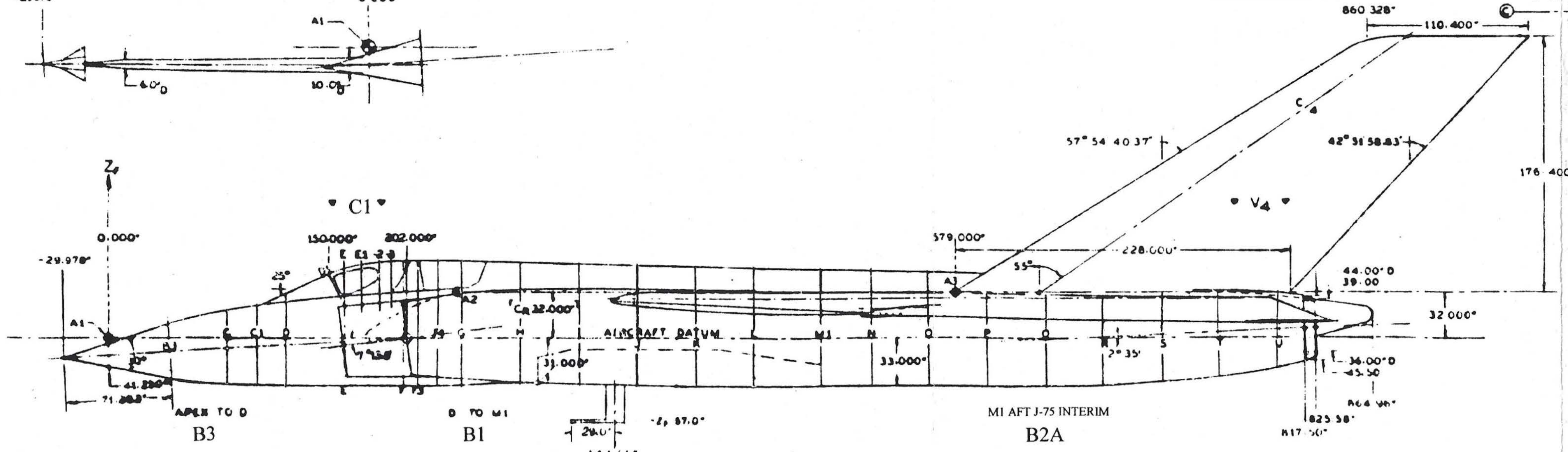
MASS FLOW....INTAKE..... 5.6 SQ.FT.
EXHAUST..7.068 SQ.FT.

THICKNESS RATIO.
INNER WING...3.5%
OUTER WING..3.5% TJ, 3.8% TIP.
TAIL.........................4%









FUSELAGE CONFIGURATION, B3A

FREE FLIGHT MODEL NUMBER 7
(DRAG MODEL NUMBER 3)

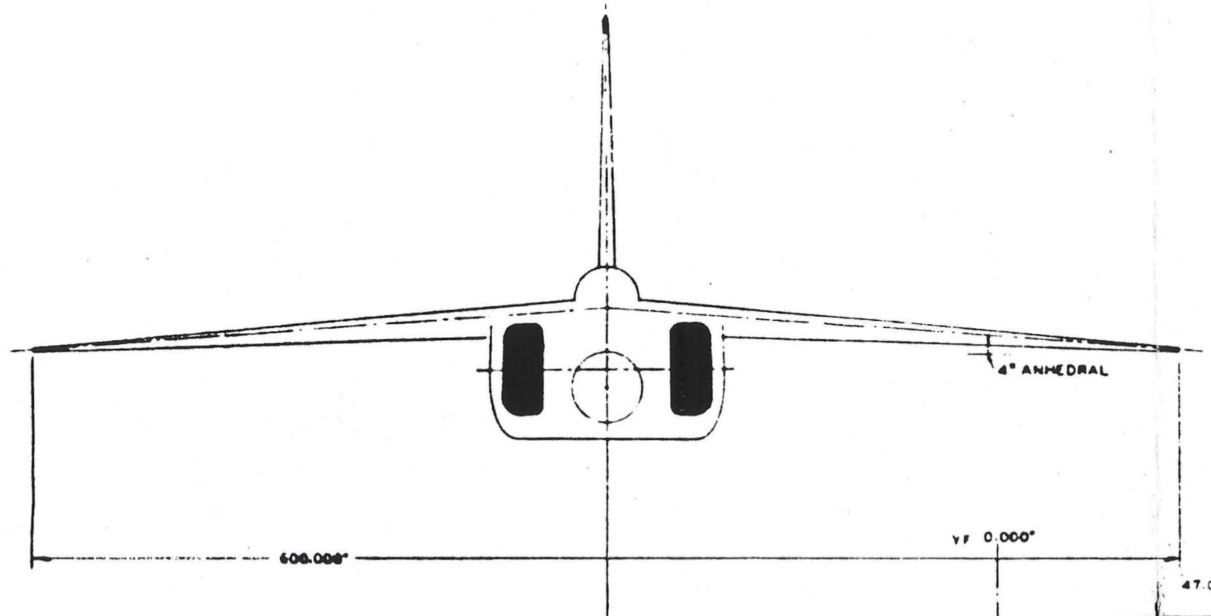
FIRING DATE...........MAY 15. 1956

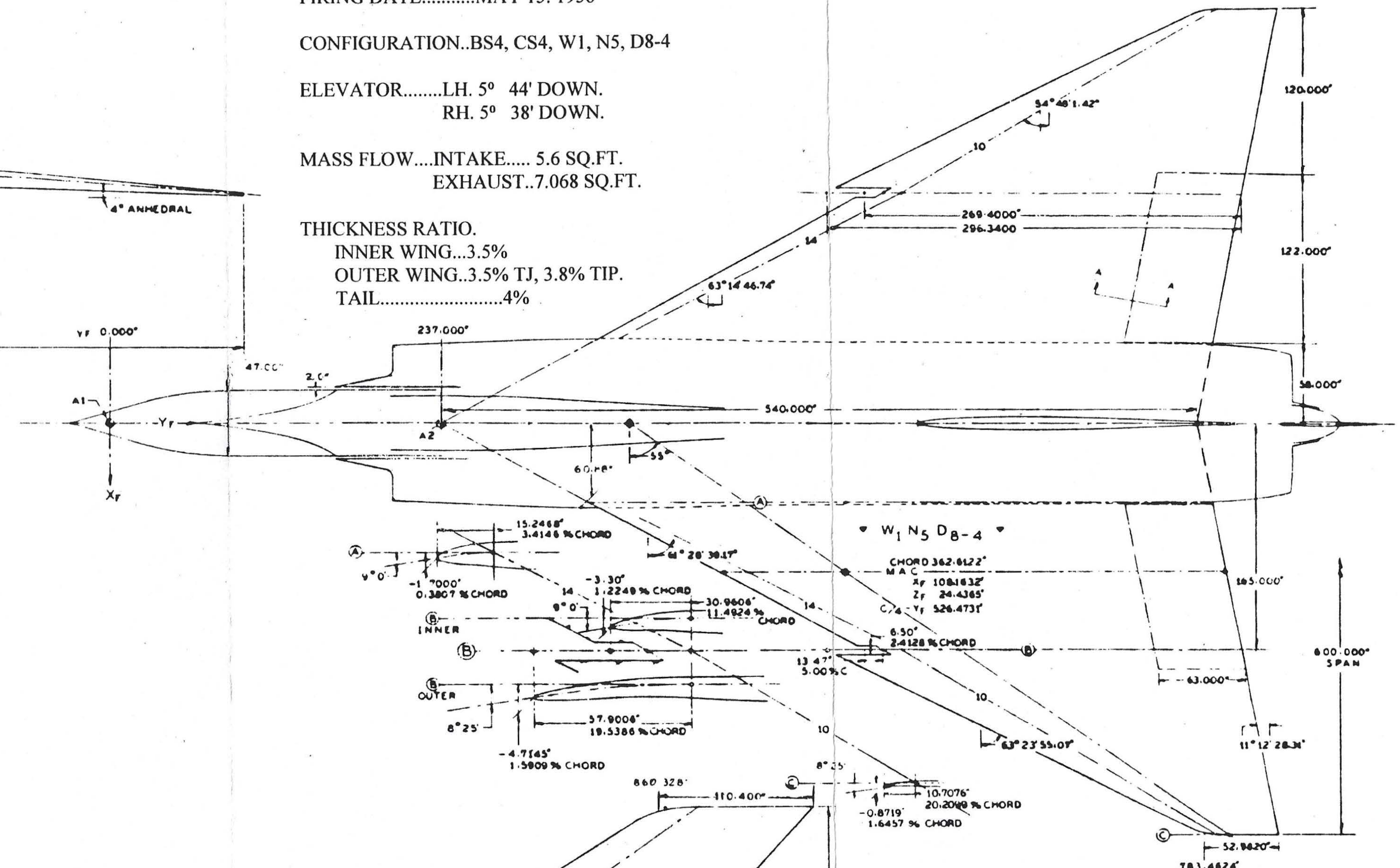
CONFIGURATION..BS4, CS4, W1, N5, D8-4

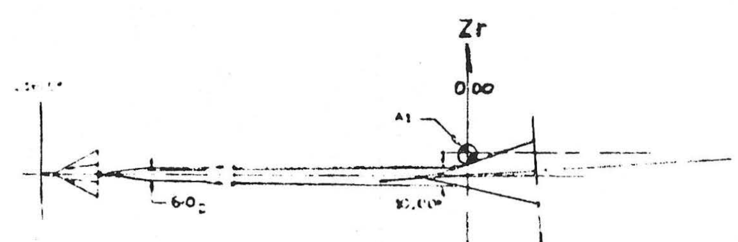
ELEVATOR........LH. 5° 44' DOWN.
RH. 5° 38' DOWN.

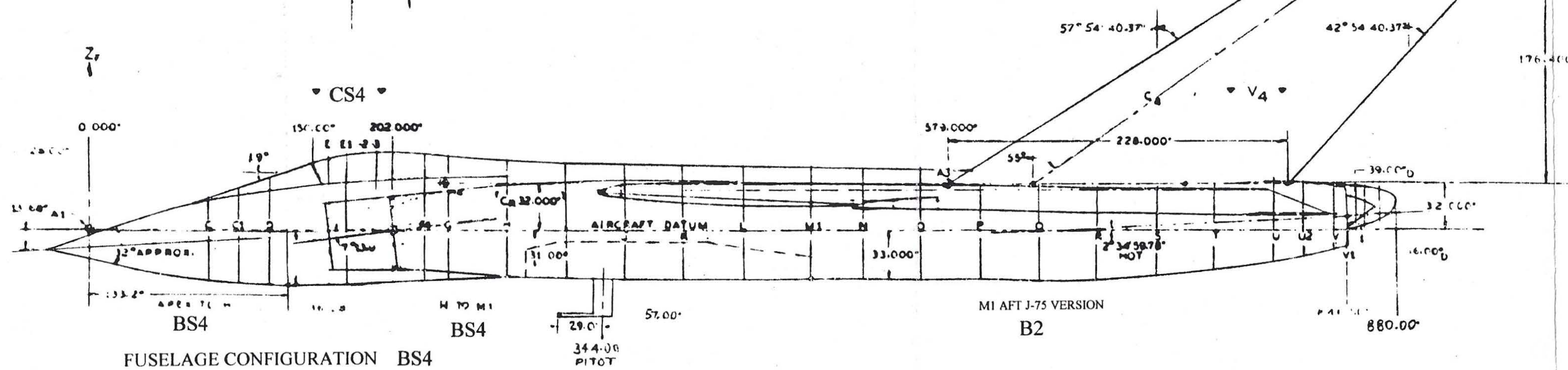
MASS FLOW....INTAKE..... 5.6 SQ.FT.
EXHAUST..7.068 SQ.FT.

THICKNESS RATIO.
INNER WING...3.5%
OUTER WING..3.5% TJ, 3.8% TIP.
TAIL........................4%









FUSELAGE CONFIGURATION BS4

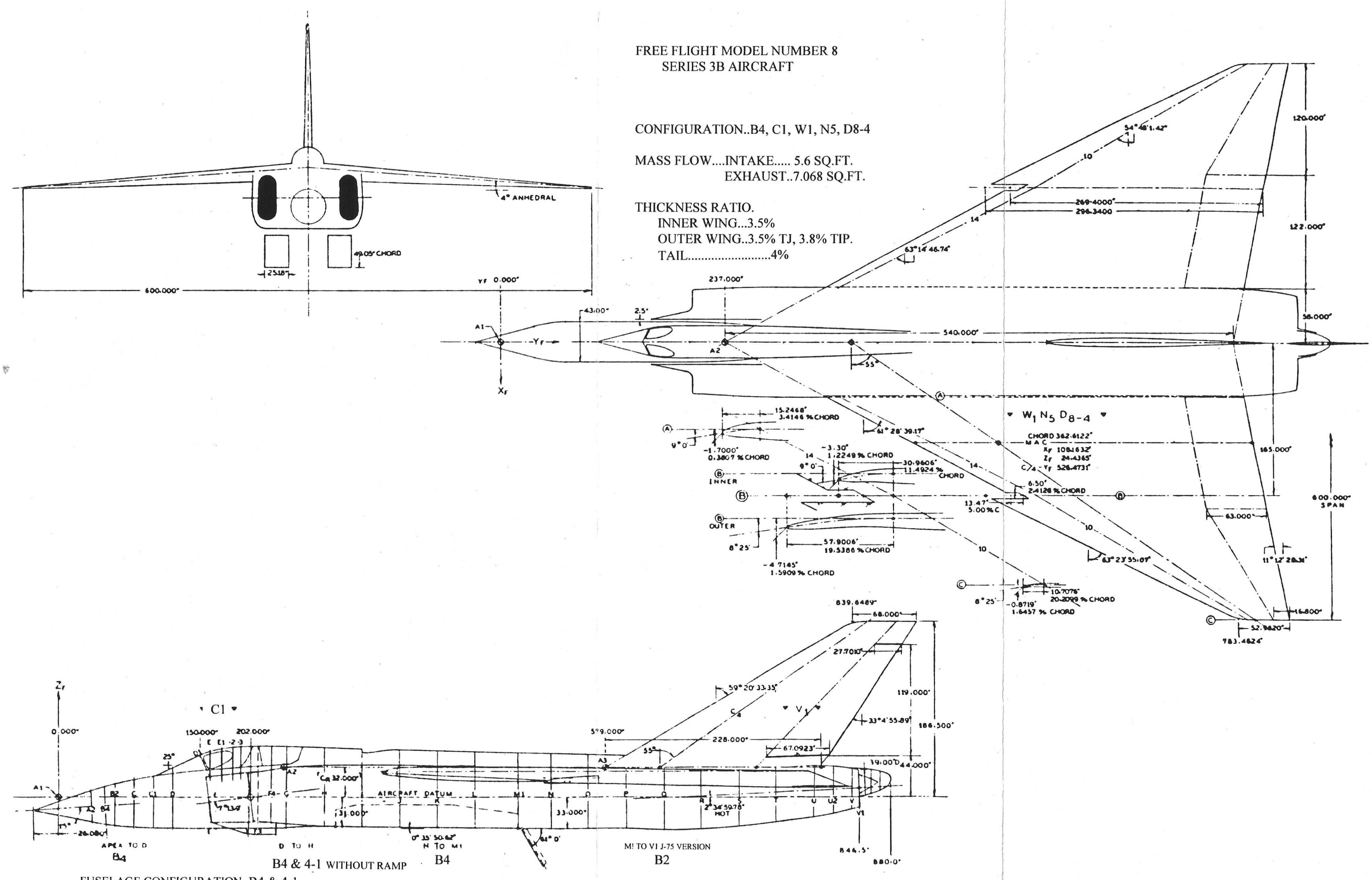
AVRO CANADA

FREE FLIGHT MODEL NUMBER 8
SERIES 3B AIRCRAFT

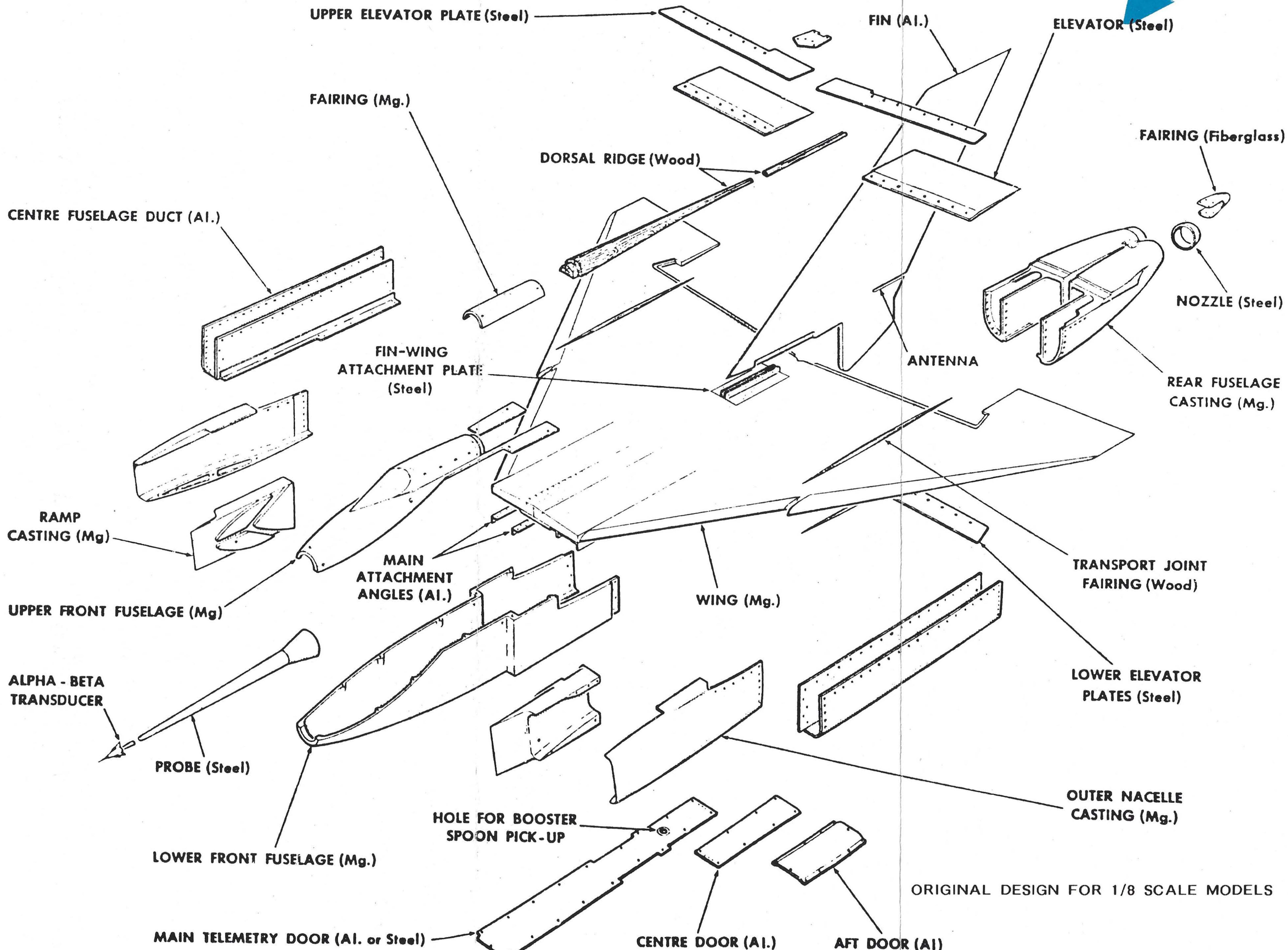
CONFIGURATION..B4, C1, W1, N5, D8-4

MASS FLOW....INTAKE..... 5.6 SQ.FT.
EXHAUST..7.068 SQ.FT.

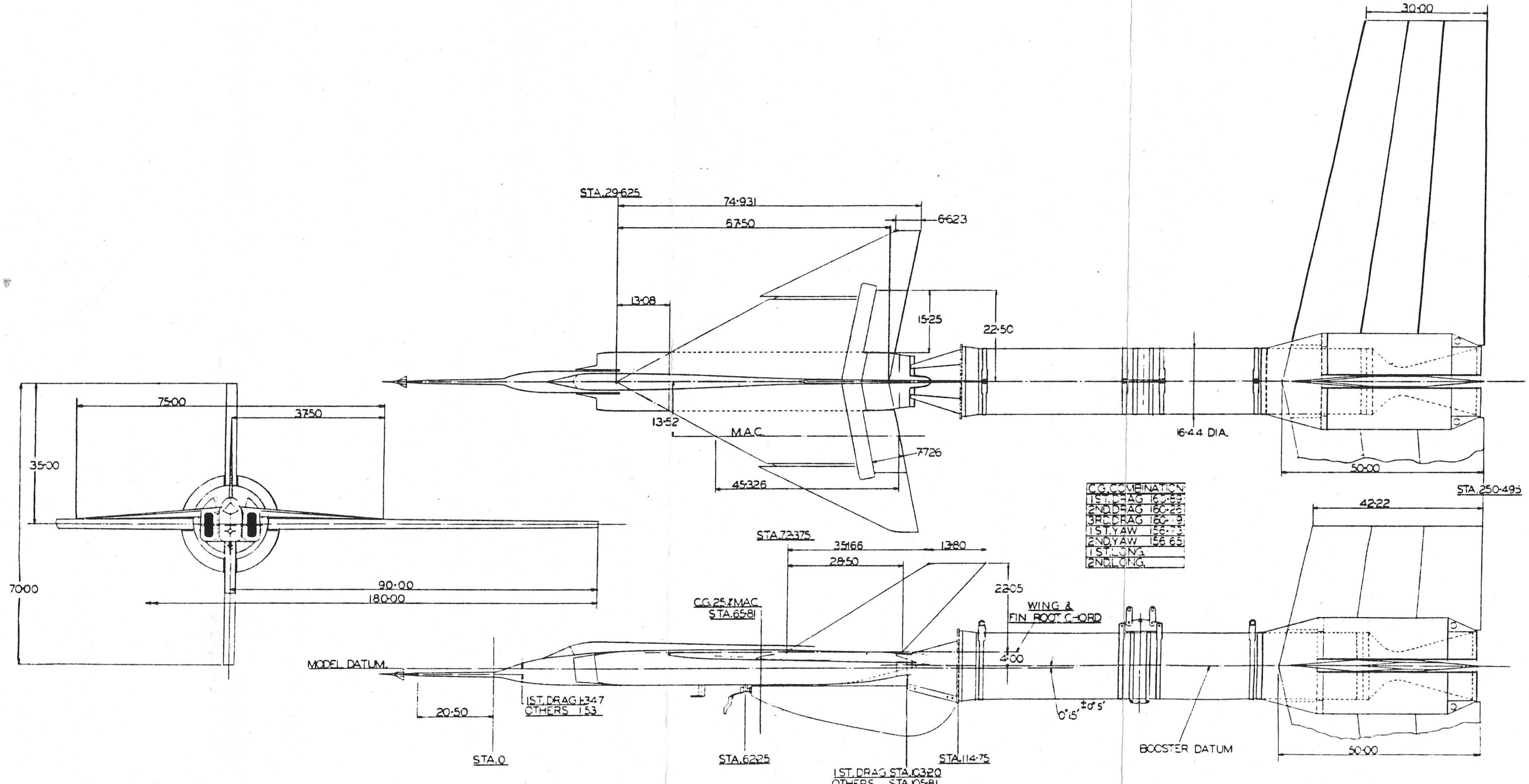
THICKNESS RATIO.
INNER WING...3.5%
OUTER WING..3.5% TJ, 3.8% TIP.
TAIL........................4%



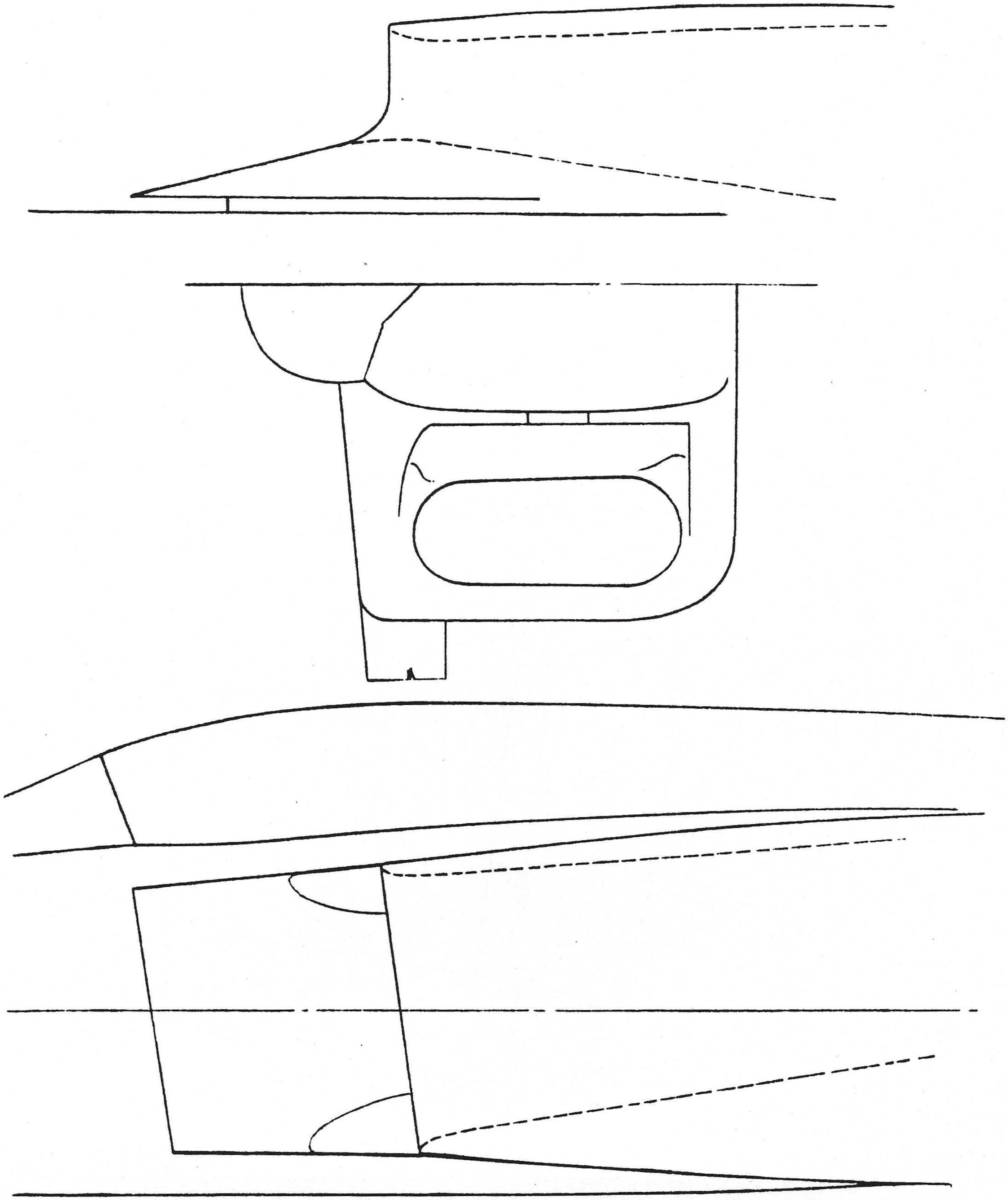
MODEL NUMBER 8 OF SERIES 3B



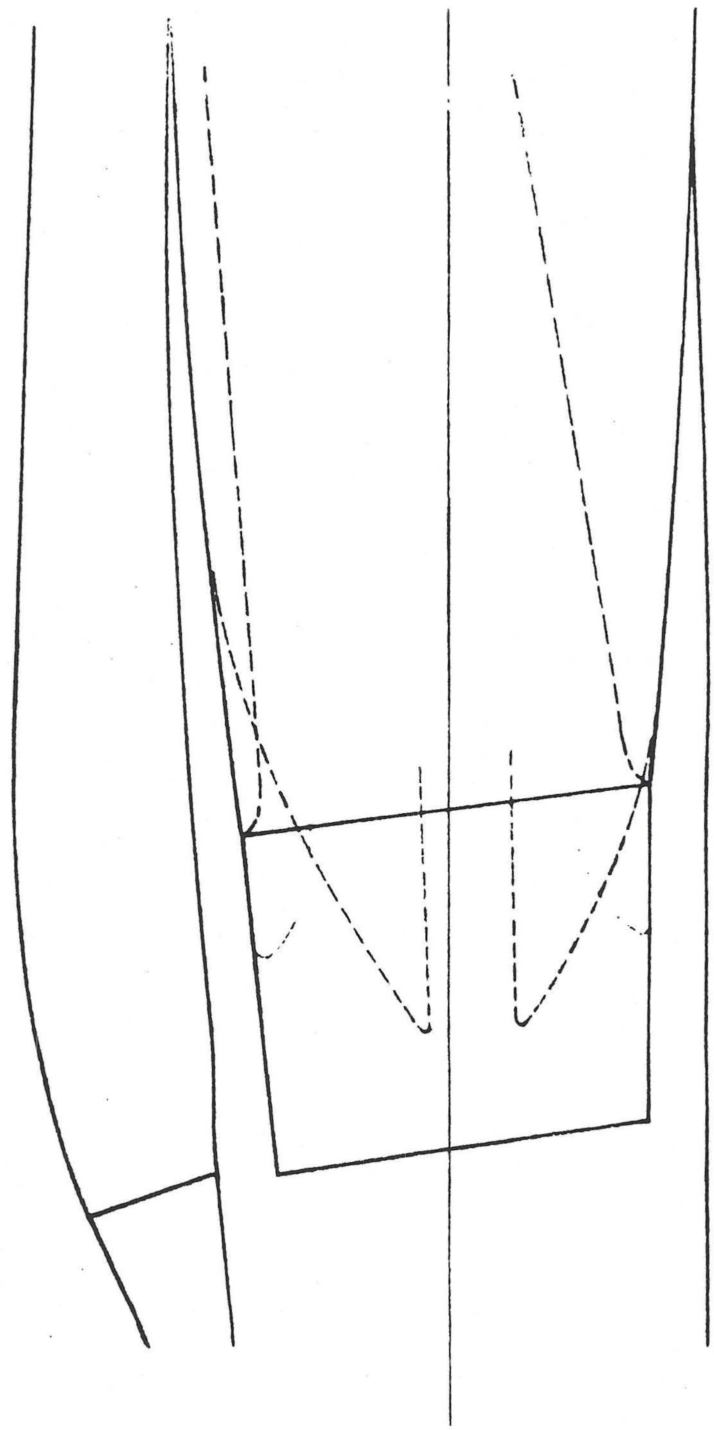
ORIGINAL DESIGN FOR 1/8 SCALE MODELS



TYPICAL MODEL SHOWN ATTACHED TO A NIKE BOOSTER ROCKET

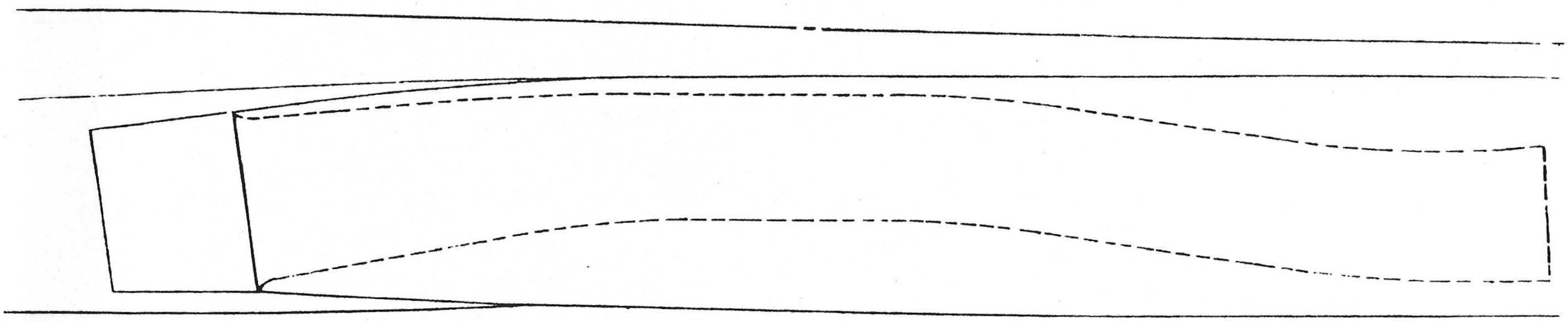


INTAKE LAYOUT FOR C-105

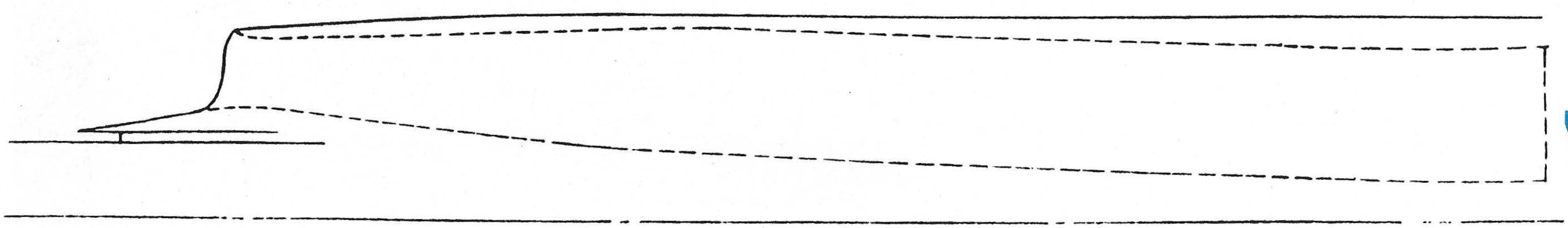


AIR CONDITIONING INTAKE C-105

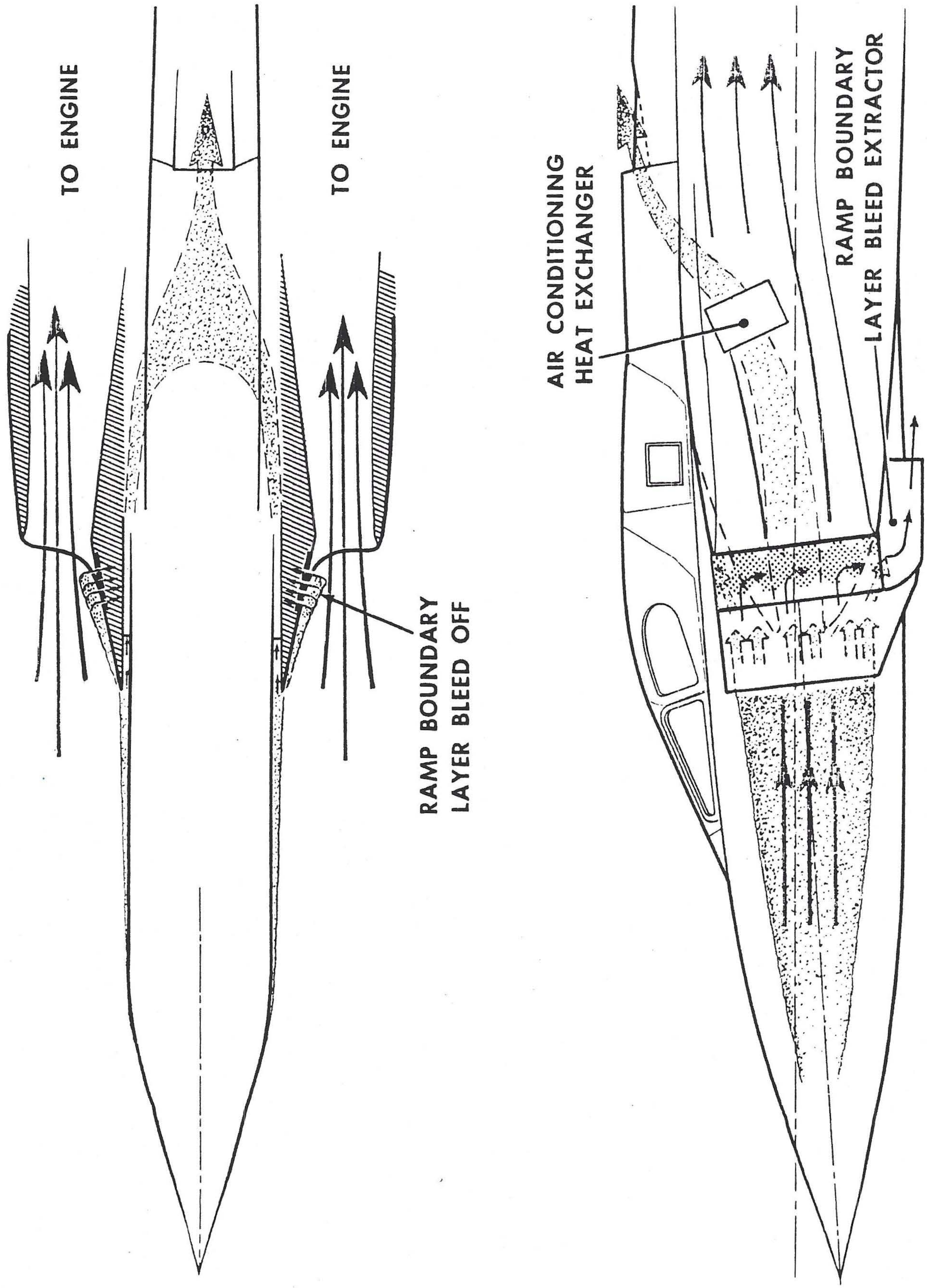
SIDE VIEW



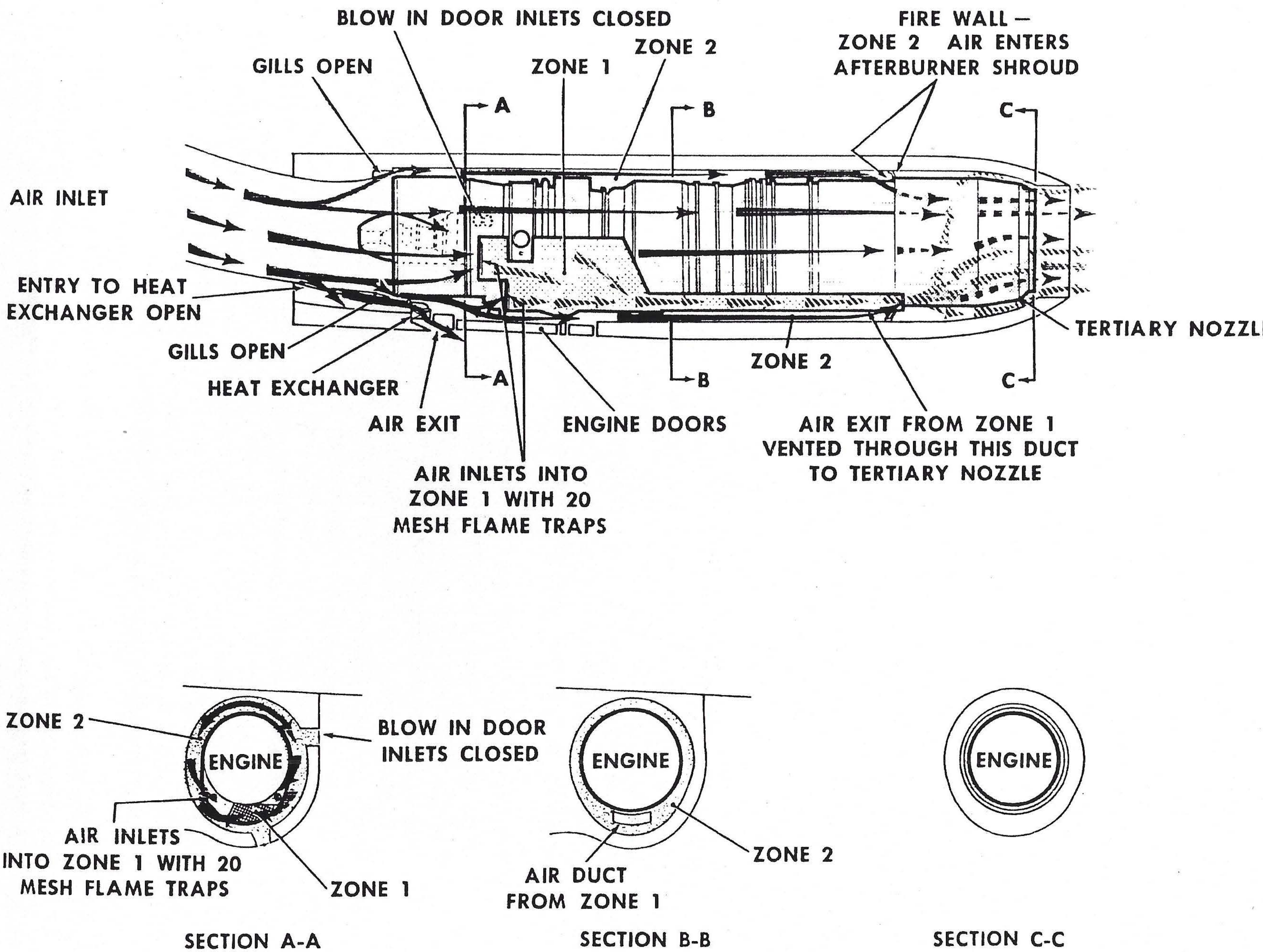
TOP VIEW



C-105 INTAKE DUCT LAYOUT



AIR FLOW AT INTAKES



ENGINE COOLING AIR FLOW

was just choked at Mach 1.5 in the stratosphere.

One of the main reasons for incorporating a bypass in this installation was to increase the minimum mass flow ratio at which the intake would have to operate at M > 1.5 in an effort to avoid "buzz". An appreciable increase in performance was apparent however, due to (1) elimination of duct pressure losses (all the duct boundary layer was bled through the bypass) and (2) reduction in spillage drag.

It was generally agreed that variable intake geometry was not required up to a Mach number of around 1.5 and it was estimated that with this particular intake-bypass arrangement that only a small performance gain would result if a variable ramp intake was chosen. The distance between the inner surface of the external compression ramp and the fuselage side was 2.5 inches, this distance being estimated to be sufficient to enable all the fuselage boundary layer ahead of the intake to flow under the ramp, In this way, the hope was to avoid a reduction in pressure recovery due to fuselage boundary air entering the intake and possibly complicating the "buzz" problem.

The air flowing under the ramp passes over a boundary layer splitter and in the centre of the splitter was located an intake which was called the "splitter bleed". This provided the charge air for the air conditioning system and this so charged air circuit was designed so that the bleed operated just off the choking mass flow ratio throughout the high speed range (M > 0.8). In this way, the prevention of any shock waves due to the splitter from moving out in front of the ramp were effected and the boundary layer entering the engine intake was thickened up.

As the flight speed is increased the pressure rise across the normal shock in front of the intake increases and eventually the flow at the foot of the shock separates. This separation occurs when the Mach number ahead of the shock exceeds 1.33 - with a 12 degree ramp this corresponds to a free stream Mach number of 1.73. It was not certain whether this separation was a particularly bad thing or not. Tests on an intake similar to that of the C-105 did not indicate anything worse than a slight decrease in pressure recovery whilst on others it seemed to have precipitated 'buzz.' It was decided therefore, as an insurance policy, to suck a portion of the boundary layer off the ramp through a porous strip running parallel to the shock. The boundary air was sucked away by a fan in the air conditioning system - this fan absorbed the load from the expansion turbine. The big question was 'How much air had to sucked away to prevent separation ?'. As far as was known, no tests had been made on a similar arrangement, and so the decision was made to suck twice the amount of air contained inside the displacement thickness of the boundary layer.

Originally, it was the intention to dispose of this air in the air-conditioning system, but due to pressure and volume requirements, it was finally decided to dispose of it through plenums, situated on the undersides of the wedges.

The choice of intake area for the J 75 installation was influenced by two primary considerations, one being to obtain optimum performance at high altitudes at about M = 1.5 and the other was to keep structural modifications to a minimum.

A critical structural case occurred in the whole of the intake duct when the engine was run at Military Rating on the ground. The variation of suction in the duct with inlet area is shown.

The variation of installation thrust losses with intake area at the tropopause is shown. The installation thrust losses are defined as the loss of thrust due to shock and duct skin friction losses plus spillage drag plus momentum loss in the bypass. It can be seen that the installation losses with the C-105 scheme were fairly insensitive to intake area and the area was therefore, chosen to comply with the structural requirement at S.L.Static. This chosen area was 5.6 sq.ft.

The variation of the choking mass flow with Mach number through a 5.6 sq.ft. intake at tropopause is shown. Also shown is the variation of the required engine flow at Military and idling rating plus cooling and bypass flow.

The sort of wind tunnel test program that Avro would have desired was similar to that carried out by N.A.C.A. on the F-102, with additional tests at M = 2.0.

C-105 DIVE BRAKE PERFORMANCE.

SUMMARY.

Dive brakes with an area of 14 sq.ft. were provided near the middle of the under-surface of the fuselage of the C-105. These gave no change of trim at subsonic speeds. However, their use was not permitted at supersonic speeds due to a change of trim that rapidly became prohibitive at speeds in excess of M = 1.0.

INTRODUCTION.

The provision of effecting dive brakes always presents a difficult problem. This received extensive study in the case of the C-105 and it was concluded that the only possible location for brakes was on the under surface of the fuselage.

LOCATION.

The reason for this decision lay in the fact that the wing was very thin and was filled with fuel tanks, undercarriage and control surface operating mechanisms. Even the fin did not offer any space for brakes. However, if it were possible to find space, it would be very difficult to design brakes in the wing that would not interfere with the elevators or ailerons. To overcome this difficulty, it would be necessary to use very large gaps between the brakes and the wing, which in turn would have required a mechanically complex device which would have been exceptionally difficult to stow. Even locations on the fuselage sides or top did not appear favorable when elevator and rudder interference was considered. Accordingly, the under-fuselage location was chosen after all other positions had been eliminated.

SIZE OF BRAKES.

The brakes on the under surface of the fuselage had an area of 14 sq.ft. This compares with the profile drag area of 11 sq.ft. for the whole aircraft at subsonic speeds. It is thus evident that their drag was substantial relative to a clean aircraft. In the approach and landing configuration, the L/D is quite low due to the high span loading and poor induced drag efficiency. This reduced the need for high drag flaps, so that it was felt that those provided were more than adequate. This point was verified by experience with the Avro 707 aircraft and the F-102 which landed without brakes because the lower brake would foul the ground.

TRIM CHANGES.

There was virtually no change of trim at subsonic speeds when the speed brakes were deployed, while the elevator angle to trim became excessive at speeds in excess of M = 1.0. For this reason, the use of brakes at supersonic speeds was prohibited. A study of the tactical situation at supersonic speeds failed to reveal any need for brakes on a missile carrying aircraft.

POWER PLANT.

POWER PLANT GROUND SUPPORT EQUIPMENT.

Together with the design of the aircraft, ground handling equipment also had to be designed, built and tested that would operate under any conditions. This equipment consisted of engine handling units, starters, standby power, electrical and hydraulic power for ground checkout and trouble shooting of various aircraft components aboard the aircraft, refueling, towing etc.

POWER PLANT HANDLING.

In 1956, it was decided that the CF-105 aircraft allotted to the aircraft development program were to be fitted with Pratt and Whitney J75 power plants, with the exception of aircraft No. 6 which was reserved for flight development of the Orenda "Iroquois" engine. A total of 31 J75 engines had been ordered for the development program and it was planned that aircraft No. 11 and all subsequent aircraft were to be fitted with "Iroquois" engines. The design of the handling equipment for the J75 engine was completed in 1956 and manufacture of the units by the Company's Manufacturing Division commenced upon receipt of DDP authorization with their being ready in March of 1957.

The critical dimensions for the Iroquois engine differed considerably from those of the J75, which resulted that new handling equipment had to be designed. Due to the speed and urgency of the requirement being completed by November, 1957, the Company contacted Air Logistics Corporation of Pasedena, California, who specialized in the design and manufacture of prototype engine handling equipment, and invited them to submit a proposal for such equipment for the Iroquois installation.

In addition, stands had to be designed for engine build-up as well as installation. The equipment so designed and used for both the J75 and Iroquois engines, together with engine mounting differences, are shown in following illustrations.

GROUND SUPPORT EQUIPMENT.

Due to wide experience in this field, the Consolidated Diesel Electric Corp. of Canada Ltd., worked together with Avro to design and produce two new vehicles for the ground servicing of the CF-105. Also due to its American parent organization, which had produced many such vehicles for the USAF and the USN, Consolidated Diesel were soon able to produce the required items for the CF-105, by using many proven standard parts. One vehicle was on a modified MA-2 chassis, built at Consolidated's plant in Stamford, Connecticut. This was used for electronics and fire control servicing. The other was from on a modified Jeep chassis. This contained the starting gear etc.

STRUCTURAL TESTING.

INTRODUCTION.

In order to understand the complexities in the design of such a sophisticated aircraft as the Avro Arrow, we must have an insight as to the background of design and the work of materials and component testing before the final designs can be created. To this end, we offer the following:-

This report broadly deals with the static and fatigue test program, the general philosophies behind the program and the results obtained.

The test program was divided into five phases as follows:-

PHASE 1. PRELIMINARY DESIGN TESTING.

This testing, although not a contractual obligation, was required by the Company to establish the design. The main aspect of this testing was the time element, and had to be carried out early in the design stages of the aircraft. As a result, the test specimens are simplified and usually differ in some respects from the final article.

PHASE 2. PROOF OF COMPLIANCE.

This series of tests were conducted almost entirely on the static test aircraft and was engineered to meet the requirements of Specification MIL-S-5710.

PHASE 3. POSSIBLE COMPONENT FATIGUE TESTING.

In the early stages, no definite plans were made along these lines as it was assumed that specimen fatigue testing along with a static test specimen which had been well strain gauged, would have sufficed. Later experience would provide actual production component testing.

PHASE 4. FAIL-SAFE TESTING.

This type of testing is not an alternative to component fatigue testing but it does in some respects reduce the need for component fatigue testing. It is intended to use the remains of the static test article.

PHASE 5. ELEVATED TEMPERATURE TESTING.

The growing importance of heat in aircraft structures required more testing and development of a research nature. Creep and transient temperatures causing induced thermal stress were and still are important problems requiring testing coupled with theoretical analysis.

APPLICATION.

PHASE 1. PRELIMINARY DESIGN TESTING.

Both static and specimen fatigue testing was done during this phase. Where the problem was primarily one of stability or buckling, static tests were used, and where the problem was due to high stresses and stress concentrations, fatigue testing was employed. In some cases the first test specimens proved to be quite satisfactory. However, in many of these tests development work was required and the final test specimen differed considerably from the original. In the case of fatigue testing, if the original specimen was considered inadequate, new and redesigned specimens were ordered, built and tested. All specimen changes were then incorporated into the aircraft design. For ease of assessment this phase of testing was broken down into aircraft components as follows:- Fuselage, Centre Section, W.ings, Fin and Control Surfaces. Certain of these test reports will be discussed here:-

FUSELAGE - INTAKE DUCT - (INCLUDING PHASE 4 FAIL-SAFE TEST).

A series of tests were inaugurated on the round portion of the .032 aluminum alloy duct. The test specimen was a 13 1/2 ft. section of the intake duct built to production standards by the Production Department. The intake duct is subject to high pressures and depressions, pressures in flight and depressions during engine ground run-up. The problem of high depressions in the long intake ducts during engine run-up were a particularly difficult problem facing aircraft designers using large jet engines. The weight of such large diameter ducts is very large, and a very close assessment, both analytically and test wise was considered essential.

Two identical specimen were ordered and tested. The point of initial buckling was the prime objective of the test and it was necessary to establish the difference between a duct that had been pressurized to 10 psi., and one that had come straight from manufacturing. Initial pressurizing blows the duct round and removes the worst of the flats and manufacturing discrepancies.

It was proven that initial pressurization of the duct to 10 psi., did in fact raise the point of initial buckling to a satisfactory level. The significant aspect here is in regard to panel flutter. If the panels were allowed to buckle at too low a point, they would almost certainly come apart due to panel flutter.

As a result of these tests it was decided to pressurize each production duct to 10 psi., prior to engine run-ups. It may have been possible to discard this process in favor of engine programming during run-ups and take-offs on later production aircraft.

Other objects of the duct tests were as follows:-

1. Leak rate.

2. To substantiate the strength of the duct under limit pressure.

3. To substantiate the strength of the duct under ultimate pressure.

4. The fail safe characteristics of the duct under pressure loads.

The results and conclusions were as follows:-

a. The leak rate using normal riveting techniques were unsatisfactory. As a result, all riveted joints were glued on production and static test aircraft.

b. The duct satisfactorily withstood limit pressures. A factor was included to take into account the effect of temperature.

c. The duct satisfactorily withstood ultimate pressures with a factor included for temperatures.

d. Fail Safe Test. The only test conducted up till 1956 of a fail safe nature, was on a section of the intake duct that had been used for pressure and depression tests. This consisted of pressurizing the duct to limit pressure and firing .50 calibre bullets through it at strategic points. The pressure in the duct receded in a satisfactory way and no disastrous failure of the duct occurred.The design of the C-105 aircraft employed the fail safe concept as much as was possible. The basic concept was to design a structure in such a way that damage would be localized.

Redundant structures are generally good fail safe structures. In pressure vessels, small aspect ratio panels will tend to localize cracks thus preventing disastrous "rips".

This concept was so new at the time that very little experience was available. It was planned to use the "remnants" of the static test article to conduct testing of this type and to gain much needed experience along this line, as well as to improve the fail safe characteristics of the C-105 aircraft.

PHASE 2. PROOF OF COMPLIANCE.

Most of these tests were carried out using the full static test specimen. The purpose of these test were as follows:-

1. To comply with the requirements of MIL-S-5710.

2. To substantiate load distribution.

3. To confirm stiffness and distorted shapes.

4. To confirm limit load requirements.

5. To confirm ultimate load requirements.

The C-105 aircraft had to be tested in the proper environment, that is, it had to be a complete aircraft. The interaction effects and the integral nature of fuselage, centre section , wings and fin absolutely dictated this policy. There could be no satisfactory component testing as was able to be done on the CF-100 aircraft. A possible exception was the dive brake.

Although these tests were primarily to prove the structural integrity in line with the requirements, the initial tests were simplified in order that some testing would be completed before the flight trials. These initial tests as described in the following text were planned to clear the aircraft for initial flight testing only.

MAIN UNDERCARRIAGE SPRING BACK STATIC TEST.

A large area of the wing around the undercarriage cut-out is critical for the undercarriage spring back case as well as the landing gear itself. Although in importance this test was secondary to the rolling pull-out case, it was definitely required before flight. Sine the aircraft rigging for the R.P.O. case would support the aircraft for the landing case, this test did not seriously disrupt the R.P.O. case. A six week set back of the R.P.O. case estimated, was considered satisfactory.

Approximately limit load was applied, strain gauge readings, wing and undercarriage deflections were recorded. Complete proof of compliance tests were conducted at a later stage in the program. (These later tests showed that the application of loads up to 60% of the limit load were satisfactory, but during subsequent tests to 70% limit load, the right-hand main landing gear back stay failed at 60%. Improved design was then undertaken by Dowty in England.)

PHASE 3. COMPONENT FATIGUE TESTING.

The undercarriage was the only item under the initial program to be fatigue tested. This was due to the complexity of detail and the use of super high heat treated steel. A single gear was tested, and development tests were necessary depending upon the results. Further component fatigue testing on the aircraft structure was carried out after the static and flight tests. A case in point was the landing accident to RL 25201 in which the gear on the port leg failed to rotate properly and the aircraft collapsed onto its belly after leaving the runway.

PHASE 5. ELEVATED TEMPERATURE TESTING..

This phase of testing continued to be more important as the speed range of the C-105 aircraft was extended. A considerable amount of testing was necessary to clear the aircraft to Mach 2, but a much larger amount of research testing would be required to clear the aircraft for higher Mach. numbers. To effect the creep and induced stresses due to transient temperatures required close attention.

TRANSIENT HEATING OF WING FUEL TANK MODEL.

This was the first of this type of testing. The torque box tested was similar to the static test inner wing boxes. The objective was to determine the heat flow through the box and the resulting thermal effects. Heat was applied under very close time control and temperatures and stresses measured by the use of continuous trace thermocouples and strain gauges. The foregoing are but a very small fraction of the work that went into the design of the CF-105, and , as the reader will no doubt conclude , Avro Aircraft were conducting research which ultimately enabled other organizations to benefit in their quests for higher goals - namely the conquest of space.

ACCIDENT TO ARROW RL25201.

The accident occurred at Malton Airport at 15.29 on June 11, 1958 when landing at the completion of flight #11. The accident was due to the left-hand landing gear not being fully extended when it was locked down. As a result the wheel bogie was not parallel to the aircraft's line of flight.

The pilot was unaware of the landing gear malfunction during his approach, as the cockpit indicators showed the landing gear DOWN and LOCKED. Observers who were in radio contact with the aircraft were unable to see that the final extension and turning of the leg had not been completed. The Sabre chase plane had returned to base prior to the accident, due to fuel shortage.

HISTORY OF THE AIRCRAFT.

AIRCRAFT.

Serial number....................25201
Number of flights..............11
Number of flying hours......11 hours, 30 minutes.
Number of flying hours......3 hours, 5 minutes.
since last periodic
inspection.
Nose wheel steering...........Not fitted.

LANDING GEAR.

Manufacturer.......................Dowty Equipment of Canada Ltd.
Type...................................Tandem Bogie.
Number of landings............10
prior to accident.
Number of landing gear.......155
functions.
Last ground function............Prior to flight 11 - Eight ground check.
functions on June 9, 1958.
Last strip examination.........Prior to flight 10.
of landing gear.

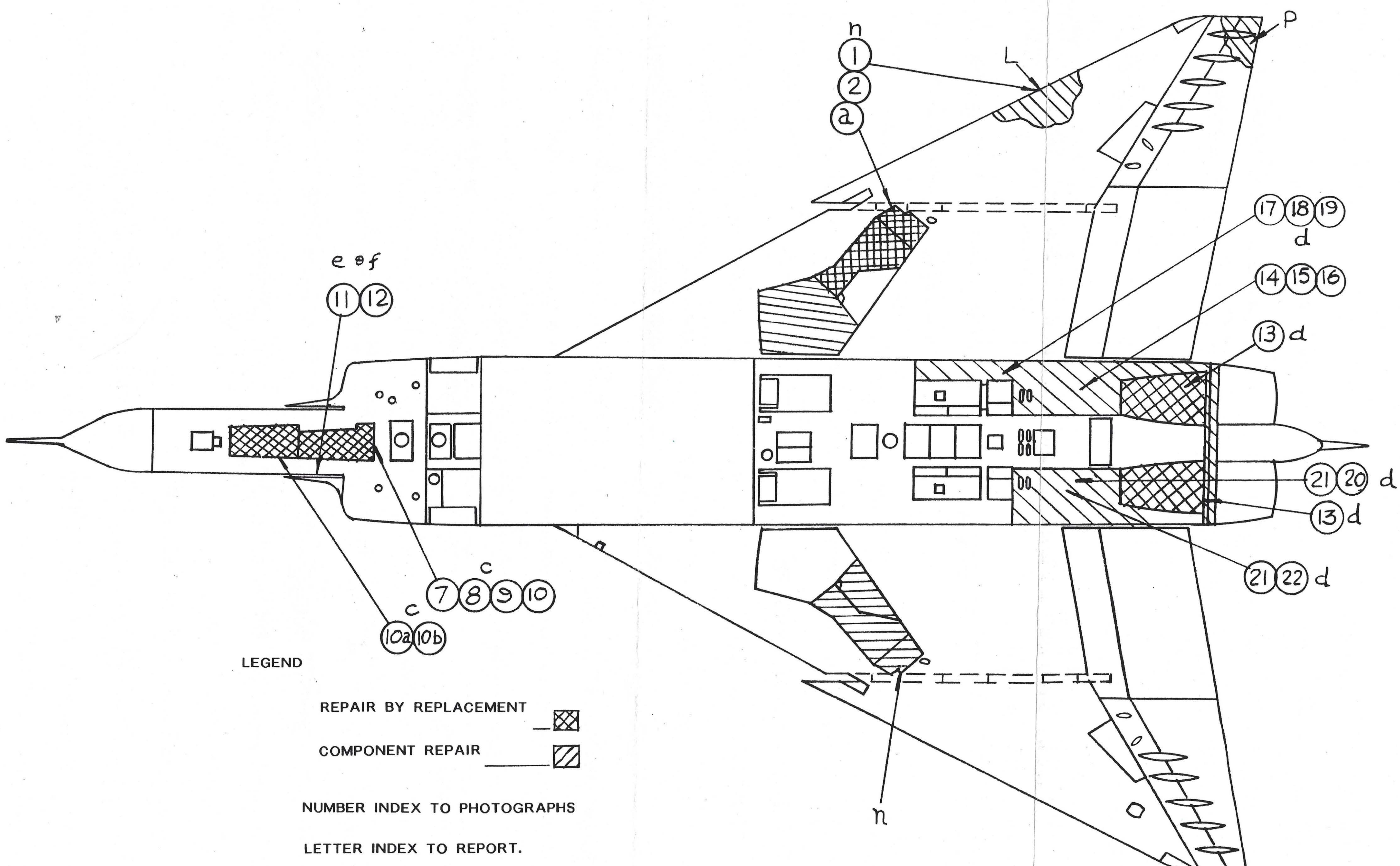
New brake pads were fitted prior to flight 10. These were the revised 1" thick type compared with the 3/4" of the normal brakes, giving a higher kinetic absorbtion.

ACCIDENT DETAILS.

The pilot selected landing gear when in the circuit for landing and as stated the cockpit indicators showed DOWN and LOCKED. The aircraft touched down on the end of runway 32. The photographs show that the left-hand leg extension and turning had not been completed prior to touchdown. The partial extension can be seen and that the wheels were not in line with the aircraft's longitudinal axis.

Immediately after touchdown, the aircraft's weight caused the left-hand landing gear to turn further and assume the position which it would normally occupy when stowed in the landing gear bay. The drag chute was then deployed. In the photographs, smoke is shown coming from the lefthand tires, due to their misalignment with the aircraft's path. The aircraft continued to swing towards the left-hand side and corrective brake action had no effect in arresting the swing. The pilot, suspecting a heavy cross wind, then considered that the drag chute may be causing the swing and jettisoned it. The aircraft continued to swing to the left and eventually left the runway. When the left-hand wheels struck the soft ground, the aircraft swung violently to the left, causing the landing gear to collapse due to the excessive loads imposed upon it.

ARROW 1-25201 LANDING ACCIDENT



LEGEND

REPAIR BY REPLACEMENT

COMPONENT REPAIR

NUMBER INDEX TO PHOTOGRAPHS

LETTER INDEX TO REPORT.

ANALYSIS OF THE ACCIDENT.

Photographs taken of the left-hand landing gear shortly after the accident, show the retracting chain broken off, and protruding from the dust cover. Instrumentation records showed that the touchdown speed was 170 kts. IAS, the rate of descent was 5 to 6 ft/sec. and that the drag chute was streamed at 150 kts.

From photographs taken during examination subsequent to the accident, it became apparent that the lowering chain had become folded and jammed at the end cap due to the sliding member not being able to extend fully, which in turn was due to tolerance problems, and the leg continuing to fold down to the extended position. It was found that with this particular leg, the parts had never been honed together and consequently, when they were slow to overcome friction, the chain looped and became jammed thus prohibiting further extension and twisting of the bogie.

Photographs taken of the underside of the aircraft when being examined in flight test show the extent of damage, all of which was repaired or replaced and the aircraft was restored to its flying duties.

SURVEY OF THE DAMAGE

The initial survey revealed the following damage:-

a. LH main landing gear torn out of its housing.
b. RH main landing gear broken at top of wheel bogie.
c. Nose landing gear torn out of attachment fittings.
d. Rear fuselage buckled and torn mostly on LH side.
e. LH air-intake ramp buckled at bottom edge.
f. LH ai-intake scoop buckled inside.
g. V-struts broken at Station 742.0.
h. Vertical strut buckled at Station 717.36.
k. LH main landing gear door buckled.
l. LH outer wing leading edge buckled.
m. Engine bay bottom formers and skin buckled from Station 697.0 to Station 742.0.
n. Inner wing skins over main landing gear pivot housings damaged due to salvage operation.
p. LH aileron damaged due to wing striking ground.

The reader is referred to the attached sketches and photographs showing the attitude of the aircraft on the ground after the accident and the resulting damage.

CONCLUSION.

Remedial actions were taken pending the design of the Mk 2 undercarriage. The main extension spring was increased in strength and an adjusting screw was added to the chain mechanism.

PER ARDUA AD "ASTRA".

ENTER THE "ASTRA" SYSTEM.

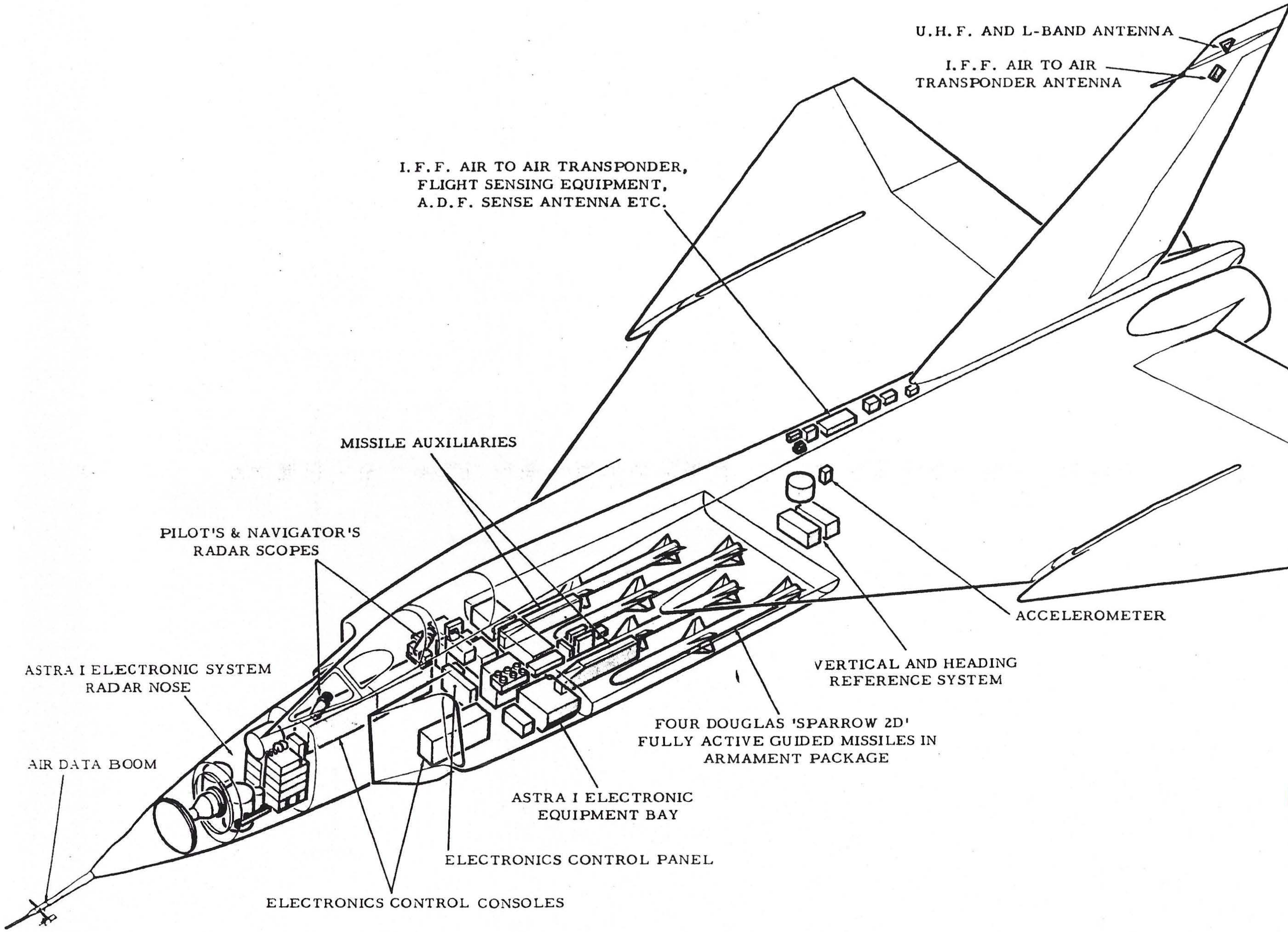
The fire control and weapons system as the RCAF required it, were the real undoing of the entire Arrow program. The RCAF, going against all advice, stated that it wanted the largest radar dish possible, namely 40" diameter. Not only would this have necessitated another complete redesign of the front fuselage and radome, but the Hughes Company could not supply such a thing in time due to USAF pressure on their own wants. In spite of the fact that Avro advised the RCAF that such a disc was not required, the RCAF then awarded a contract for design and production of a system to be known as ASTRA, which would embody automatic flight, fire control, navigation and communications all in one. The company chosen for this miracle, was RCA with Minneapolis-Honeywell as its associate. This was an intolerable situation as RCA did not have any experience in the design of such a sophisticated system. Constant changes originating from RCA to Avro resulted in change after change in the structure design to accommodate these. It is ironic that Hughes did later develop such a system for the F-106. If this was not enough, the weapons system for the Arrow was also plagued with problems.

Back in 1947, the Defence Research Board initiated work on studying the design of air-to-air missiles, and in 1950 approval was issued for the design and manufacture of such a missile, intended for use by the RCAF. This missile was to become the Velvet Glove, a first generation, semi-active missile, developed in part by Dr Gerald Bull at the CARDE. The program was undertaken however with the understanding that if similar missiles of superior quality were to be developed in the US or Europe, then it would be canceled in favor of these missiles.
Other missiles were of course being developed and one in particular, the Sparrow 2, a missile being developed jointly by the US Navy and Douglas, attracted the Canadian Government to cancel the Velvet Glove in 1956, rather than up-grade it. The Sparrow 2, a short time later was itself canceled in 1957 and the Canadian Government took over its continuation and awarded a contract to Canadair of Montreal for its development. By 1958, the costs had spiraled astronomically and the whole project was canceled by the Government. The RCAF hastily revised their demands to reinstating the Falcon missile together with the MB-1 un-guided missile that possessed nuclear capability. This blundering about and lack of decisive thought and action placed the cost in the region of an extra $200,000,000 above the cost of development of the Arrow aircraft itself, in fact, it was joked about at Avro that Astra meant "astronomically expensive".

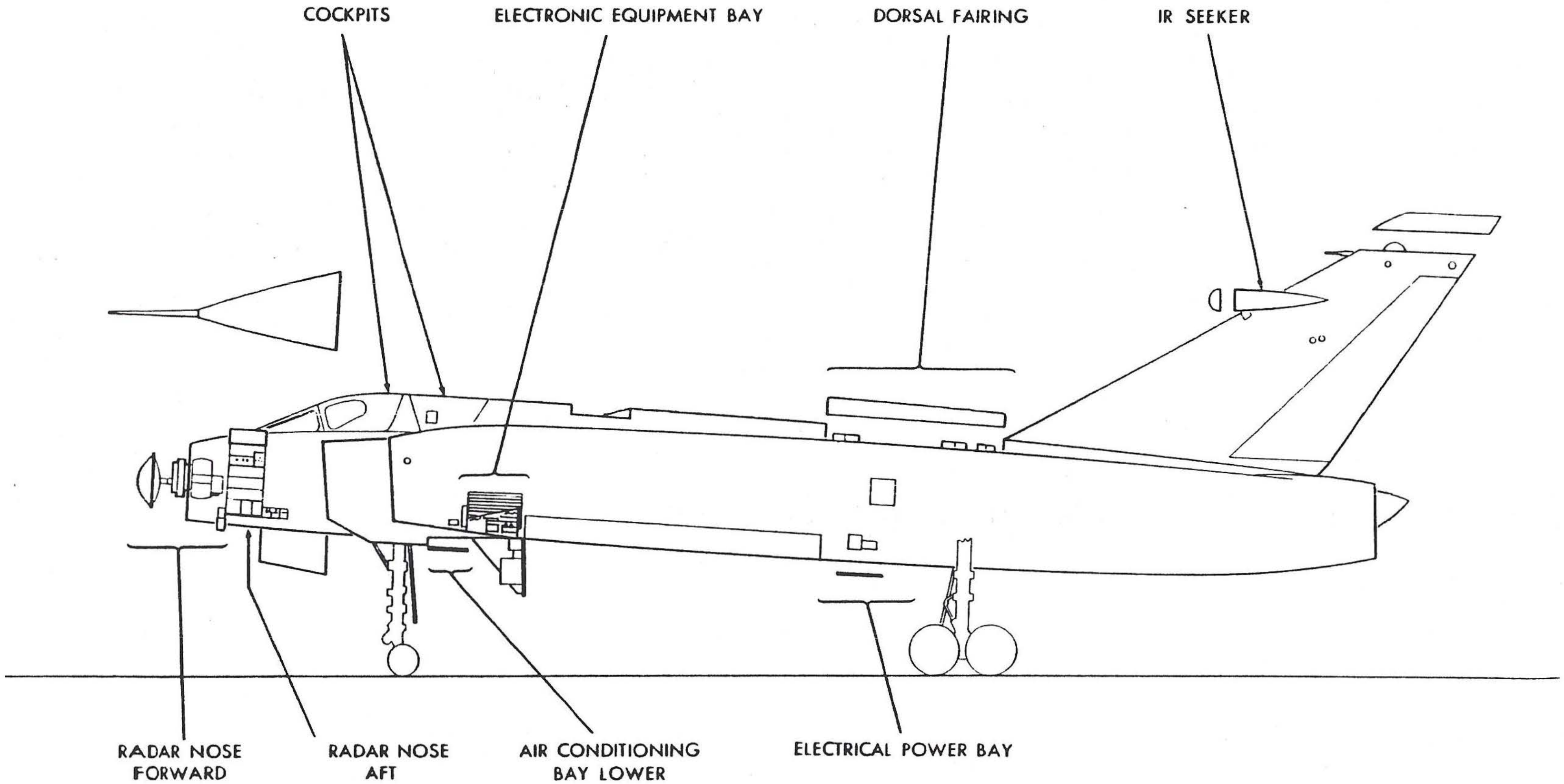
WHAT WAS "ASTRA".

INTRODUCTION.

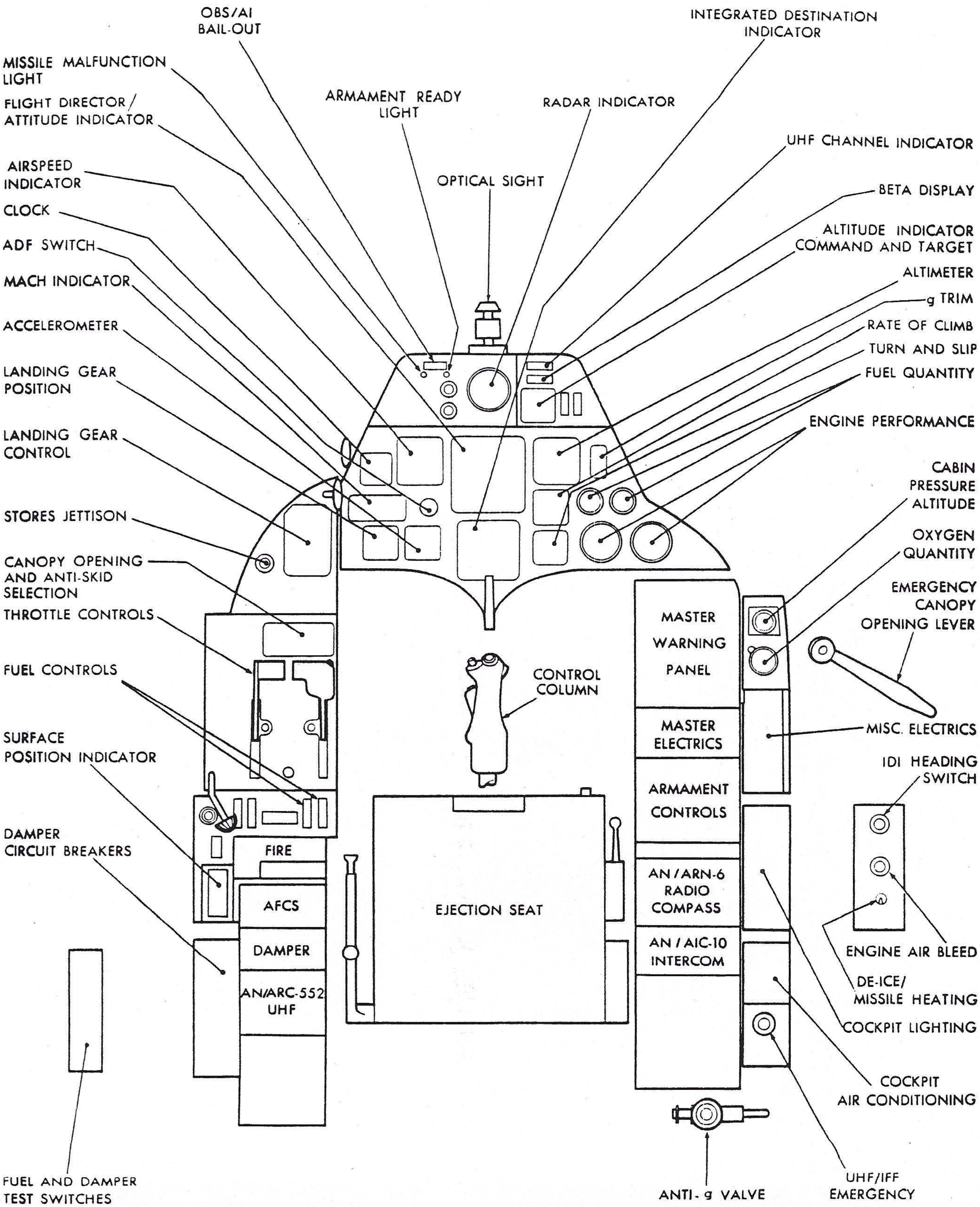
The following is an overall view of the ASTRA system, a detailed explanation of which will follow.



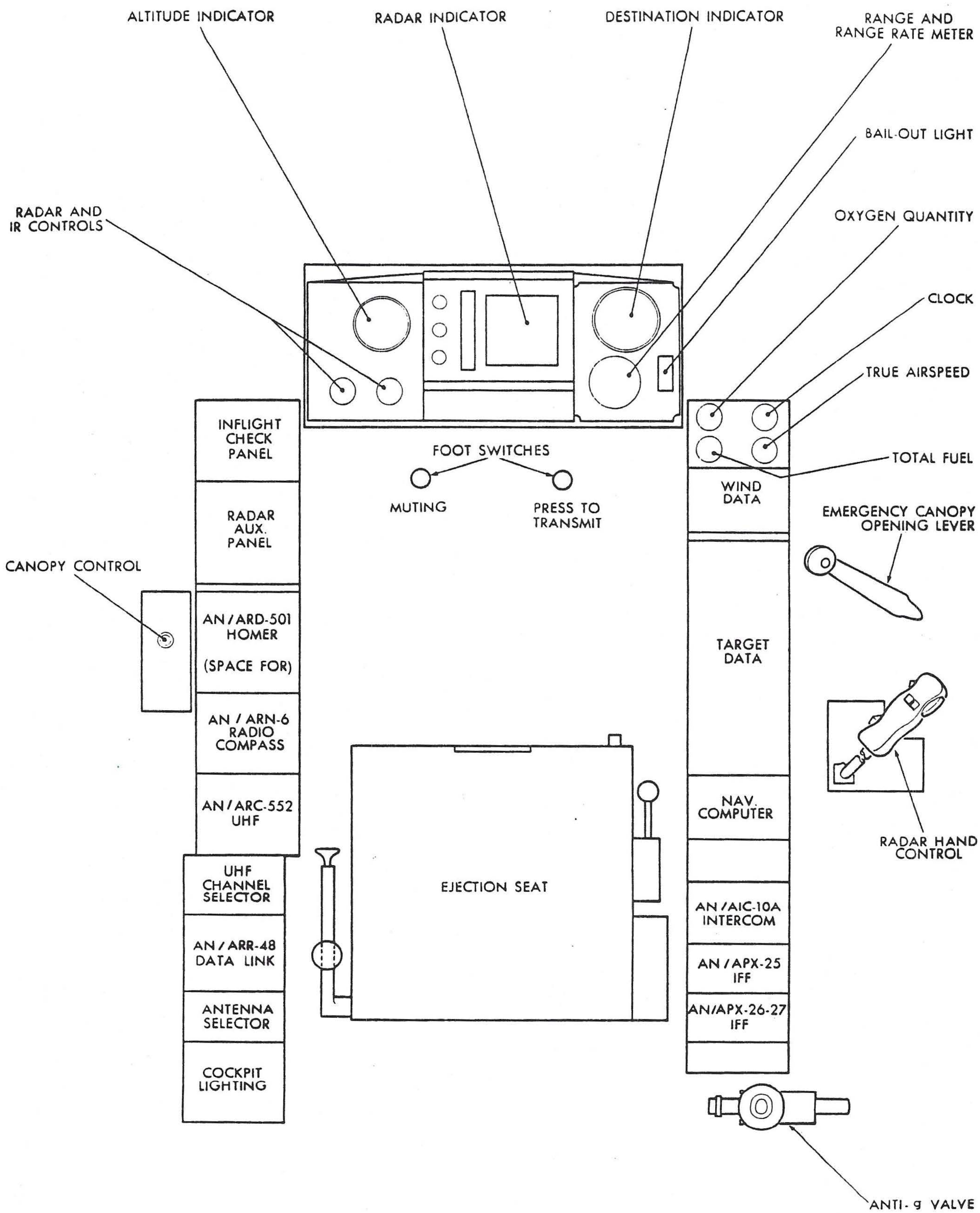
ASTRA 1 INSTALLATION IN ARROW 2



ELECTRONIC EQUIPMENT LOCATIONS IN ARROW 2



PILOT'S COCKPIT



OBSERVER'S COCKPIT

The complete electronic system for the ARROW 2 comprised those major electronic sub-systems necessary to the performance of the Arrow interceptor mission, the several phases of which indicate the various functions the electronics system must perform. These mission phases were:

(a) Navigation to a fixed or moving target and missile firing.
(b) Detection, interception and tracking of the target.
(c) Return to base.

These functions were to be accomplished by the four electronic sub-systems: fire control, navigation, telecommunications and automatic flight control. The various stages of ASTRA development in the ARROW 2 were as follows:

(a) Partial ASTRA - This system included the minimum necessary communication, navigation, IFF, flight instruments and air data equipment, required to fly the aircraft during the research and development parts of the program.

(b) Developmental ASTRA - This would eventually have included all the sub-systems of the complete ASTRA system. Variations in systems however, would have occurred as the result of system development.

(c) Pre-production ASTRA - This would have been manufactured with production tooling, but would not have been subjected to full quality control and inspection.

SUB-SYSTEMS.

FIRE CONTROL SUB-SYSTEM.

The fire control sub-system included airborne interception radar, ballistics, kinematics, computer, missile auxiliaries, optical sight and associated equipment. The primary functions of the sub-system were to acquire and track the target, provide the correct steering signal for manually or automatically steering the interceptor during attack, to automatically fire the interceptor's armament at the correct time and to provide a visual indication for breakaway. The system was designed to operate with Sparrow 2D guided missiles and long range unguided rockets.

The radar, in addition to target detection and tracking, also provided navigational aids in the form of radar beacon interrogation and ground mapping facilities. Passive homing on X-band jamming signals was to be provided as an added counter-counter-measure capability. The fire control sub-system was also designed to operate in conjunction with an infra-red system for passive detection and tracking.

TELECOMMUNICATIONS SUB-SYSTEM.

The telecommunications sub-system provided two-way UHF communication, voice coded data link reception, UHF homing, LF/MF radio compass, homing on L-band jamming signals (ECM homer), air-to-air and ground-to-air IFF coverage (interrogator and transponder units), and crew interphone.

The telecommunications sub-system was composed of the following equipment:

(a) UHF communications....................AN/ARC-552
(b) UHF homer...................................AN/ARA-25
(c) Radio compass..............................AN/ARN-6
(d) Interphone.....................................AN/AIC-10A
(e) Data link.......................................AN/ARR-48 type
(f) Radar homer..................................AN/ARD-501
(Space provision only)
(g) Air-to-ground IFF transponder.......AN/APX-25A
(h) Air-to-air IFF interrogator.............AN/APX-26
(i) Air-to-air IFF transponder..............AN/APX-27

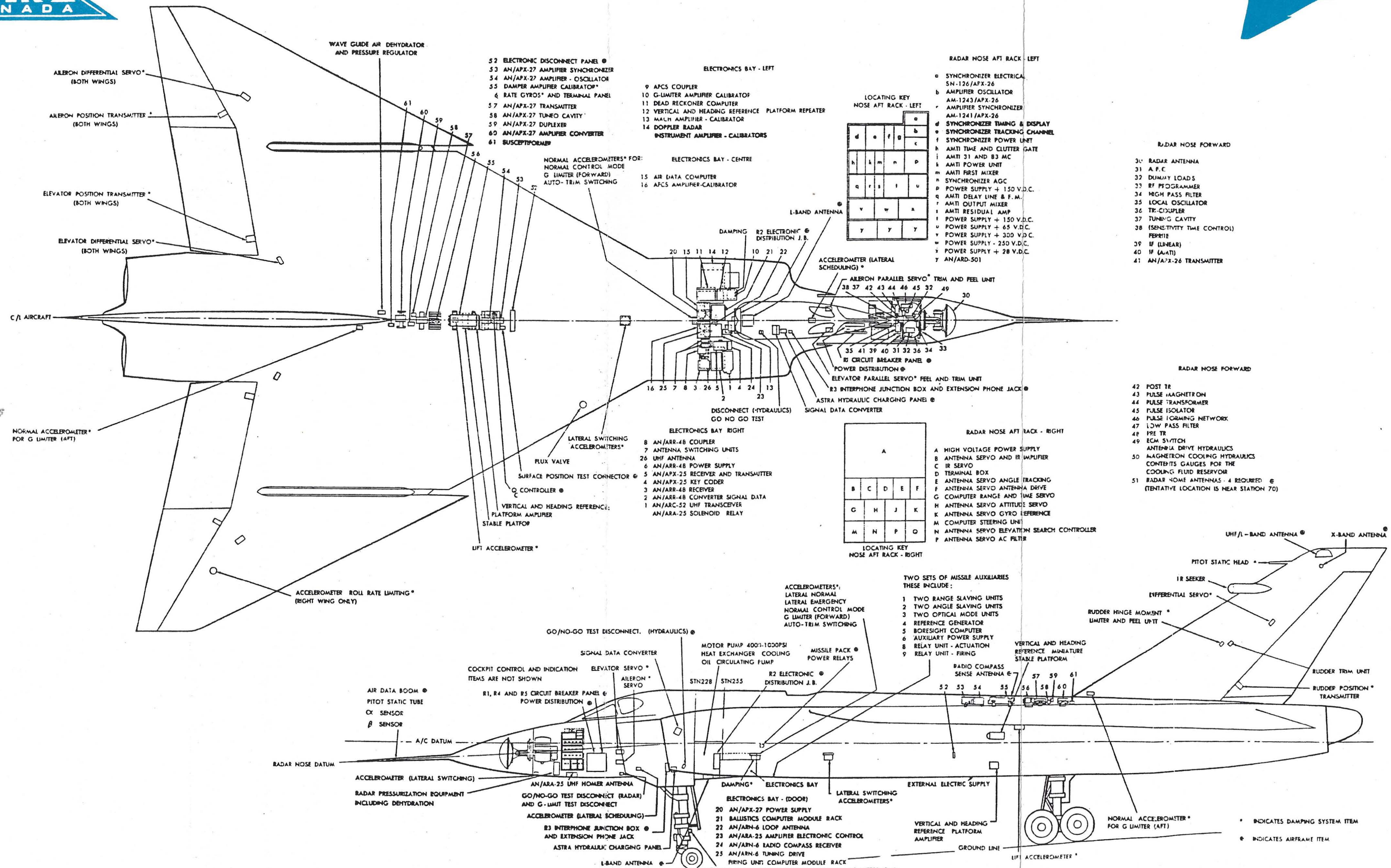
NAVIGATION SUB-SYSTEM.

The navigation sub-system consisted of a dead reckoning navigation computer, a doppler radar and an integrated destination indicator. The sub-system was self-contained and was capable of indicating the actual geographic location of the aircraft (with bearing and distance to target or base) at all times. The navigation computer received inputs from telecommunications, radar and air data equipment and supplied outputs to the fire control radar, automatic flight control system and the navigation display indicator. Display data included the present position of the interceptor, interceptor track, range and bearing to the target and radio compass and UHF homer azimuth data.

AUTOMATIC FLIGHT CONTROL SUB-SYSTEM.

The automatic flight control sub-system consisted of the vertical and heading reference unit, the air data computer and the AFCS couplers. The function of the system was to automatically control the flight path vector and orientation of the aircraft. The AFCS commanded the control surface motions through the damping system by means of the couplers. The flight path method of control was accomplished by reducing all outer loop commands (those received via telecommunications) to the common denominator of flight path direction in space, compatible with basic inner loop commands. The AFCS provided vertical and heading reference information for all other sub-systems, was the central source of air data information and provided the outer loop control of the aircraft. The inner loop control was supplied by the aircraft damping system which although separate from the electronics system, was designed to operate as a manoeuvring, all-attitude stability augmenter in conjunction with the AFCS.

The AFCS enabled the aircraft to operate in the following modes:
(a) Automatic attack.



ELECTRONIC EQUIPMENT LOCATIONS IN ARROW 2

(b) Automatic navigation.
(c) Automatic approach for landing.
(d) Manual manoeuvring.
(e) Pilot assisted functions, consisting of:

1. Pitch attitude hold.
2. Heading hold.
3. Bank hold.
4. Altitude hold.
5. Mach hold.

DETAILED DESCRIPTION OF ASTRA.

The Mk1A System was termed the "Minimum ASTRA System, the basic sub-systems of which were:-

1. Aircraft instruments (Attitude indicator and Mach meter).

2. Navigation (Dead-reckoning computer, low and medium frequency automatic direction finder (ARN-6) UHF automatic direction finder (ARA25).

3. A1 radar (Antenna, transmitter, receiver, synchronizer, power supplies, search computer and programmer, AMTI signal processor, range and angle track).

4. Fire control computer and Sparrow 2 missile auxiliaries.

5. Automatic flight control (Hold modes).

6. Air data computer.

7. Vertical and heading reference (Vertical gyro (GG48), directional gyro (LDG-1), flux valve and 3-axis repeater).

8. Identification (Ground-to-air transponder (APX 25A)).

9. Communications (UHF command set (ARC 552), intercom (AIC10A).

10. Ground support equipment.

The Mk1B system, or "Full System", differs from the Mk1A by the addition of the following:-

1. Doppler navigation radar.

2. Data link receiver and coupler.

3. Infra red seeker.

4. Air-to-air IFF interrogator.

5. Air-to-air IFF transponder.

6. AFCS integrated coupler.

7. Increased power magnetron.

The following modes were not available in the Mk1A due to the lack of a portion of the equipment required for the mode.

1. Beacon reception (Since the a-a IFF interrogator receiver was used in this function).

2. Quasi-passive ranging (Since this mode depended on IR for angle tracking).

The capability of the Mk1A system was different from the Mk1B in the following basic areas due to lack of equipment, data or flight evaluation time:-

1. Manual insertion of GCI data, rather than automatic.

2. Manual control of the aircraft, rather than automatic.

3. Decreased detection range by 10%.

4. Manual insertion of meteorological wind data, rather than automatic Doppler radar inputs.

5. IR and QPR ECM capabilities.

6. Air-to-air identification.

7. Genie capability.

It is worthy to note here the functions that were to be performed by Ground Support. The maintenance concept was based on semi-automatic flight line test sets to minimize aircraft down time and in-aircraft repairs, complemented by second-line test, alignment, and repair consoles in the maintenance shops.

The semi-automatic fire control test set generated a target return which was detected and tracked by utilizing 85% of the fire control system, and checking the system for miss resulting from alignment errors of failures.

The navigation system was also checked by the insertion of a typical problem which in this case was done by the manual entry controls.

The communication system was checked by means of a test set which operated through the ARC 552, AIC 10A and the ARA 25.

The AFCS, air data computer, attitude indicator, and Mach indicator were checked by the flight control semi-automatic test set.

EMPLOYMENT OF DEVELOPMENT AIRCRAFT.

In the RCAF report No. S1038CN-180, which was issued after the cancellation of the Sparrow missile and the ASTRA system, the aircraft allocated to the development of the armament system and the Iroquois engine were as follows:-

25201. To be used for the development of aircraft systems, trial installations and as a target aircraft from February 1958 to March of 1961.

25202. To have the MA-1C fire control system installed together with the MA-1C/MB-1 weapon at Malton, then to be used as a target aircraft from June 1958 till March 1961.

25203. To be used for stability and control together with structural integrity, from September 1958 until February 1960.

25204. To be used for performance and handling trials, and the installation of flight test gear from October 1958 until February 1960.

25205. To be used for the installation of the MB-1, flight test of the MB-1 and MB-1/GAR Falcon pack development, from December 1958 until January 1961.

25206. To be used for aircraft systems reliability, fly and fix programs, from March 1959 until January 1961.25207. To be used for the Orenda test program, from May 1959 until January 1961.

25208. To be used for airframe reliability and structural integrity, from July 1959 until January 1961.

25209. To be used for Iroquois back-up and Arrow 2 intensive flying, from September 1959 until January 1961.

25210 - 25213. To be used for the MA-1C/MB-1/GAR program, from January 1960 until March 1961.

25214. To be used for performance handling trials, from April 1960 until March 1961.

25215. To be used for Avro trial installations, from July 1960 until March 1961.

25216. To be used for intensive flying in cold weather, from July 1960 until March 1961.

25217 - 25220. To be used for probability of placement and development of tactics, from August 1960 until March 1961.

25223. Onwards. Delivery to squadrons.

Aircraft	Role	Airframe sensing instrumentation supplied by AVRO	Recording Instrumentation		Pack Type (Note: Weapon packs will be available for all ARROW 2s in addition to instrument and weapon-instrument packs listed)	Remarks
			Specified and supplied by	Installation design and Installation by		
25201	AVRO airframe development	Yes	AVRO	AVRO	Instrument	
25202	(a) AVRO airframe development (b) RCAF Phase 2 test (c) AFCS development (d) Antenna pattern testing	Yes	AVRO	AVRO	Instrument	Special Instrumentation for RCAF Phase 2 Testing (See Appendix 13)
25203	(a) AVRO airframe development (b) Weapon pack development	Yes	AVRO	AVRO	(a) Instrument (b) Weapon Pack with radar nose instrumentation	Canadair CTV program operated on this aircraft
25204 and 25205	(RCA ASTRA I (development (	Yes	RCA	AVRO	Instrument	See also Appendix 14
25206	(a) AVRO airframe development (b) RCAF Phase 2 test	Yes	AVRO	AVRO	Instrument	Special instrumentation for RCAF Phase 2 testing (See Appendix 13)
25207	Orenda Iroquois development	Yes	Orenda	AVRO	Instrument	See Appendix 15
25208	AVRO airframe development	Yes	AVRO	AVRO	Instrument	
25209	RCA ASTRA I development	Yes	RCA	AVRO	Combination weapon-instrument	

INSTRUMENT PACK. ARROW 1 & 2

Aircraft	Role	Airframe Sensing instrumentation supplied by AVRO	Recording Instrumentation		Pack Type (Note: Weapon packs will be available for all ARROW 2s in addition to instrument and weapon-instrument packs listed)	Remarks
			Specified and supplied by	Installation design and Installation by		
25210	(a) AVRO electronic system installation (b) Canadair Sparrow 2 Mk 1 testing	Yes	AVRO	AVRO	Combination weapon-instrument	
25211 and 25212	(Canadair Sparrow 2 (Mk 1 development (	Yes	Canadair	AVRO	Combination weapon-instrument	
25213	(a) AVRO development (b) Weapon system demonstration	Yes	AVRO	AVRO	Combination weapon-instrument and weapon	
25214	Co-ordinating Contractor Weapon System demonstration	Yes	AVRO	AVRO	Combination weapon-instrument and weapon	
25215	AVRO structural integrity	Yes	AVRO	AVRO	Instrument	
25216 and 25217	(RCAF Phase 4 (evaluation (	Yes	AVRO	AVRO	Weapon	Special radar nose instrumentation
25218 and 25219	(RCAF Phase 5 (evaluation	Yes	AVRO	AVRO	Combination weapon-instrument	

INSTRUMENT PACK. ARROW 1 & 2

Air-craft	Role	Airframe sensing instrument-ation supplied by AVRO	Recording Instrumentation: Specified and supplied by	Recording Instrumentation: Installation design and installation by	Pack Type (Note: Weapon packs will be available for all ARROW 2s in addit-ion to instrument and weapon-instrument packs listed	Remarks
25220 through 25225	(RCAF Phase 6 (evaluation (	Yes	AVRO (if required)	AVRO (if required)	Weapon	
25226 and 25227	(RCAF Phase 7 (evaluation (	Yes	AVRO	AVRO	Combination weapon-instrument	
25228 through 25233	(RCAF Phase 8 (evaluation (	Yes	AVRO (if required)	AVRO (if required)	Weapon	
25234 through 25237	(Attrition ((	Yes	AVRO (if required	AVRO (if required)	Weapon	

INSTRUMENT PACK. ARROW 1 & 2