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AIRCRAFT DESIGN STUDIES - S68
MULTI-ROLE STRIKE AIRCRAFT

by

D. HOWE

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SUMMARY

The S68 aircraft design study was for a multi-role strike aircraft with a variable sweepback wing. The study was undertaken as the result of experience gained in a previous investigation of a naval strike aircraft, the S64 (Ref.1). The more recent design was for a land based aircraft intended to be capable of fulfilling strike, interception and search roles.

The design take off weight was 50,000 lbs and the normal maximum speed condition was Mach 2 at altitude although a short dash to M = 2.5 was catered for. An internal bay was provided for the carriage of various combinations of stores. Side by side seating was adopted for the two crew members and some difficulty was experienced with the structural layout of the fuselage and the design of the main undercarriage because of this.

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FIGURES

1. INTRODUCTION

As long ago as 1964 a design study was undertaken of a variable sweepback naval strike aircraft. This study, which was known as the S64, formed the basis of the annual design project for the Aircraft Design students of the College of Aeronautics in the 1964 to 65 academic year. Subsequent investigations were made into the mechanical and structural problems associated with variable sweepback, using the S64 design as a basis (Ref.1). As a result of this work it was decided that the lessons learnt should be applied in a further variable sweep design study. In this case a land based multi-role aircraft was selected. The design was designated the S68 and a group of some fifteen students participated in the exercise during the 1968 to 69 academic year. The components allocated to the various individuals are listed in Appendix A.

2. SPECIFICATION OF THE S68 STRIKE AIRCRAFT STUDY

Consideration was given to the multi-role versatility conferred by the use of variable sweep in proposing the specification used as the starting point for the S68 study. In certain respects the performance targets were less demanding than those specified for the earlier S64 in addition to the fact that no consideration of naval operations was required. The relaxation in performance was particularly associated with the low level dash capability. This was accepted as a result of the earlier experience with the intention of achieving a lighter and more compact design.

Three major roles were stipulated for the aircraft:-

1. Strike operations. Whilst carrying a nominal payload of 2000 lbs the aircraft was required to have the following performance:-

   a) 550 n. miles radius of operation at low level and M = 0.9
   b) 200 n. miles radius of operation at 36,000 ft altitude and M = 2.0.
   c) 1350 n. miles radius of operation at 36,000 ft altitude and M = 0.9.

   The low level dash capability required was M = 1.1
   In order to cater for a variety of weapon loads internal stowage was required in a bay some 20 ft long by 4 ft wide.

2. Interceptor operations:- Ability to carry air to air missiles or rocket pods externally below the centre wing was specified. In this role a normal Mach number of two at altitude had to be supplemented by a 30 second dash to M = 2.5 during interception.

3. Search and patrol operations. A maximum endurance of approximately seven hours at altitude and subsonic speed was required.

Two crew members were specified for the strike and search roles. Twin powerplants were required for operational safety.

3. OVERALL CONFIGURATION OF THE DESIGN

A general arrangement drawing of the project study aircraft is shown in Figure 1, whilst Figures 2 and 3 are photographs of a model. Geometric data is given in Appendix B together with a summary of the aerodynamic and inertial characteristics. Figure 4 shows the internal layout.

The two powerplants were located side by side in the rear fuselage with the air intakes positioned on the sides of the fuselage just ahead of and below the root of the high mounted wing. These were of variable geometry, semi annual type. The Rolls-Royce, Bristol Engine Division, BS 146-01R design study was used as a basis for the powerplant installed. This engine had a bypass ratio of 1.3 and a specified sea level thrust rating of 8120 lb. Use of reheat enabled this to be increased to 14,000 lb. Variable area convergent-divergent nozzles were incorporated in the design of the powerplant system.

The wing consisted of a fixed centre portion having a leading edge sweepback of 66 degreees and two moving outer panels. These were hinged at points 3.75 ft out from the aircraft centreline and well back on the local chord of the fixed centre wing. Leading edge sweepback of the outer wings could be varied between 20 degrees and 66 degrees with an intermediate position at 45 degrees for the 36,000 ft altitude, M = 0.9 strike role. The hinge position was largely determined by the geometry of the fairing structure which is shown in Figure 5. This was very similar to that proposed for the earlier S64 study and was based on the use of a cylindrical trailing edge portion which moved into the sidewall of the rear fuselage together with a discontinuity in the forward aerofoil section marked by a fence at the junction of the fixed and moving parts. The geometry of the aerofoil in the trailing edge region was arranged so that there was no change in the section entering the slot in the sidewall. However the leading edge geometry was such that the shape of the portion entering the fence varied and fitted exactly only at the two extreme positions. At intermediate positions an inflatable variable size seal was required. There was also a discontinuity in the section associated with the centre fairing thickness(aft of the hinge point). The aerofoil section geometry derived is shown in Figure 6. A further major disadvantage of the relatively aft position of the hinge was the change in chordwise location of overall aerodynamic centre associated with wing sweep.

In the fully swept position the aspect ratio was 2.24 associated with a gross wing area of 520 sq ft. With the outer wing panels in the 20 degrees swept position the aspect ratio of 5.1 corresponded with a wing area only 10 sq ft more than the high speed case. The design take off weight was 50,000 lbs and hence the take off wing loading was about 94 lb/sq ft. The whole of the movable outer wing was fitted with blown trailing edge flaps and blown leading edge devices. With the trailing edge flaps deflected to the maximum angle of 60 degrees and blowing on the estimated lift coefficient was 2.5. This enabled the approach to be made at 110 knots at the design landing weight of 40,000 lbs.

The tailplane was low mounted and of delta shape. It was designed so that each half could be moved independently and be used for both roll and pitch control. No conventional ailerons or spoilers were incorporated in the original design but low speed roll control was found to be marginal and some augmentation may have been necessary. A conventional rudder hinged to the delta shaped single fin was used for directional control. Fuselage mounted air brakes were provided. These were of the 'barn door' type and consisted of three separate units, one positioned on the lower centreline of the rear fuselage and the other two on the upper sides.

Side by side seating was used for the two crew members. This arrangement was deliberately chosen for two reasons. Firstly it was considered that it should result in a more compact crew area with better operational efficiency and secondly it could, if required be more readily adaptable to an escape capsule system than a tandem seat arrangement. However in the final design ejector seats were proposed for crew escape so the latter advantage was of no consequence. The cabin was designed for a pressure differential of 4 lb/sq in. Search radar was located in the fuselage nose.

The weapon bay was positioned below the air intake ducts and conventional clamshell bay doors were used. The fuselage volume immediately aft of the cockpit was allocated to equipment stowage and the nosewheel also retracted into this region. The main undercarriage units retracted forwards into bays located below the wing centre section alongside the intake ducts. In order to obtain sufficient track the two wheels on each unit were positioned on the outer sides of the legs. The remaining volume of the fuselage and centre wing was used as fuel tankage but no fuel was carried in the outer wing panels.

The predicted weight breakdown for the aircraft is given in Table 1 and Table 2 quotes other inertia information.

4. DESIGN REQUIREMENTS

The aircraft was designed to meet the British military requirements, Av.P.970, in so far as they were applicable.

The manoeuvre flight envelope is shown in Figure 7 and the corresponding airframe flight limitations in Figure 8. With the wing in the low speed setting the design diving speed of 620 knots E.A.S. was associated with a normal manoeuvre factor of 6.67. In the fully swept wing position the increased design diving speed of 800 knots E.A.S. corresponded with the reduced load factor of 5.0. The manoeuvre load spectrum used for the design is given in Figure 9. The intended life was 3000 hours associated with 5000 landings.

It was proposed that the wing sweep change should occur at about $M = 0.75$ to $M = 0.8$, dependent upon the sortie pattern. During wing sweep the normal acceleration factor was limited to 2.0 to cover gust and limited manoeuvre cases. The flap design speed was 200 knots E.A.S.

5. PERFORMANCE

The predicted zero lift and induced drag coefficients are given in Figures 10 and 11 respectively. The layout was area ruled for the low level dash case at M = 1.1.

Predictions indicated that the aircraft would be thrust limited at 53,000ft. altitude and M = 2.0. In this condition a theoretical radius of operation of 300 n. miles was calculated. The wing sweepback necessary to meet the 36,000 ft altitude and M = 0.9 strike case was found to be 45 degrees. The search endurance condition was achieved with the wings set in the forward position and the aircraft flying at 36,000 ft altitude and M = 0.76. A ferry range of 3800 n. miles was predicted for the case of zero payload and full internal fuel of 21,900 lbs. The best flight condition for ferry operations was found to be the same as for the search role.

6. DETAIL DESIGN FEATURES

6.1 Structural Design

The layout of the major structural members is shown in Figure 12. Light alloys of the RR58 type were used as far as was possible. The main exceptions were the use of titanium alloys for parts of the blown flap systems and rear fuselage and steel in the undercarriage and wing hinge structures.

6.1.1 Wing

Although several loading cases designed various parts of the wing structure the most severe case for much of the component, including the hinge, was the 6.67g manoeuvre at high subsonic speed. It is convenient to describe the wing structure under seven headings.

a) Centre Wing Box

The centre wing structure consisted of a rectangular box some 2.5 ft wide by 1.5 ft deep which terminated at either end in the female lugs for the basically single pin hinges. The distance between the hinge points was 7.5 ft and the overall span of the box about 9.3 ft. The whole component was assembled by bolting and welding together individual forgings in stainless steel, specification FV520. This 80 ton material was preferred to higher strength steels of the maraging type on the assumption of better crack sensitivity properties. Titanium alloys were not considered due to the relative sparseness of material data available. Each of the coverskins of the box was forged in one piece, inclusive of the hinge lugs. The box section proper occupied only the central 4 ft of the span and in this region the forging was machined to give a stiffened skin 0.11 ins thick. This thickness was increased to 0.35 ins over the tapering planform out towards the lugs themselves. Integrally machined webs and ribs were bolted onto the covers to complete the box. These included fore and aft webs which closed the tapered portions as close to the hinge pins as was possible. These webs were braced back to the end rib of the uniform section by seven full depth welded webs. In addition an annular shear member was used to join the upper and lower lugs around the hinge assembly.

b) Hinge

The general layout of the hinge system is shown in Figures 13 and 14. The structural design philosophy for the hinge was that of a basic safe life backed up whenever possible with fail safe features. Thus all the lugs were effectively duplicated and designed such that in the event of any one of them failing the remaining component could withstand 67% of the ultimate design load. Considerable thought was given to the provision of access for ease of inspection. Wing moments were transmitted through the vertical surfaces of the pair of main bearings. Each of these had a nominal diameter of 13 ins and depth of 2 ins and they were spaced 14 ins apart. The bearing material was of the reinforced PTFE type, known as Fibreslip, and was 0.012 ins thick on a steel backing bush. It was stepped for ease of assembly and incorporated a 0.25 ins wide horizontal annulus for reacting vertical hinge loads. The bearing bush was bolted to the paired male lugs on the outer wing box. There was no complete hinge pin as such but rather two hinge bushes which mated into the bearings and were located in place by being tied together by 8 long bolts. These hinge bushes were attached to the female lugs of the centre wing box. No provision was made for adjustment of the horizontal, load bearing surfaces. In the event of excessive wear occurring the complete bearing assembly would be replaced.

c) Outer Wing

The outer wing lugs were of somewhat different design to those used in the centre wing. The major reasons for this were the requirements to duplicate the male lugs and change from the FV 520 lug material to the light alloy used for the greater part of the outer wing construction. Each of the halves of the male lugs were bolted together round their periphery and to the FV520 inboard skin of the outer wing box. The upper and lower lug assemblies were joined by an FV520 shear wall which was curved in plan to follow round the outside of the hinge assembly. The lugs and the curved shear wall were connected to the front and rear FV520 webs to complete the tapered box section.

The major change in construction material was made at the spanwise station coincident with the fence and leading edge kink in the 20 degrees swept condition. At this section the FV520 skins were joined to the integrally machined RR58 main covers. The skin joint was made across the top of what may be regarded as the root rib of the outer wing. The steel was recessed onto the much thickened light alloy and bolted to it. The substantial moments which resulted from the large offset of the lines of action of the cover end loads were reacted in the rib across the depth of the box. This rib consisted of two machined light alloy channels spaced apart to give the necessary bending strength.

The two spar layout was continued out to the tip, with the front spar being positioned on the 19% chord line and the rear spar on the 65% chord line of the wing in the forward setting. At the inner end the integrally machined covers had a thickness of 0.75 ins with 1.5 ins deep by 0.375 ins thick zed section stiffeners placed at about 5.0 ins pitch across the chord. The cover thickness decreased to 0.14 ins near to the tip and the stiffener section to 1.0 ins deep by 0.15 ins thick. The web of the machined channel section front spar tapered from 0.52 ins thickness at the inner end to 0.2 nearer to the tip. The rear spar was also of channel section but was built up of extruded angle edge members with a 0.1 ins thick plate web. The machined channel section main box ribs were placed at a mean pitch of 28 ins along the rear spar. The leading and trailing edge fairings used both corrugated and zed section reinforced skins which were chemically etched for minimum weight. The ribs were pressed.

d) Inner Wing Fairing Structure

A large part of the plan area of the inner wing was covered by essentially local fairing structure. The design of this was complicated by the need to allow clearance for the fairing extensions and the leading and trailing edges of the moving outer wing panels. Extensive use was made of light alloy honeycomb sandwich panels for the aerodynamic surfaces. The main problem was that of providing adequate support for the panels in those regions where clearance was required. Full depth light alloy webs were used where this was allowable and elsewhere shallow steel support beams were employed. These were attached to both the full depth parts of the fairings and to the hinge lugs. It was found necessary to taper the panels in depth over those parts of the fairings outboard of the hinge centrelines, from a maximum of 1.5 ins to 0.6 ins at the outer extremity. The honeycomb sandwich faceplate thickness was standardised at 0.064 ins and extruded light alloy edge members used throughout.

Full depth spanwise ribs were used over the forward portion of the fairing. These were placed at about 7 ins pitch and supported 0.125 ins thick skins. The fence members at the outer end of the fairings were built up from pairs of contour machined components. Each pair was joined along the horizontal centreline and bolted to the skins and sandwich panels.

e) Leading Edge Flaps

The leading edge flaps were of the simple droop type and extended along the whole leading edge of the outer wing. Each one was manufactured as a single unit and was hinged at two points across the span. Construction was based on an I section main spar with the blowing duct located behind it. The spar, duct and rear upper skin over which the air was ejected were all in titanium alloy DTD5083. The nose portion of the flap, forward of the spar used full depth honeycomb to stabilise the light alloy skins. Machined light alloy ribs were placed at the hinge points. The blowing nozzles were located along the knee.

The flaps were operated by means of screw jacks located at the hinges and driven from a single power unit located mid way between the hinges through cross shafts. The power unit consisted of duplex hydraulic motors driving a gearbox.

f) Leading Edge Flaps

Each of the single piece trailing edge flaps extended out to the tip from the nominal root of the outer wing. Support was provided at three points, the centre one of which gave lateral restraint. The single main spar was built up from a plate web with extruded tee section booms. A spanwise channel member was placed adjacent to the trailing edge. The three hinge ribs employed a built up construction but apart from these the closely spaced ribs were channel section pressings. The blowing duct was placed in the nose of the flap, forward of the spar. This, together with the 0.064 ins thick upper skin, was in titanium alloy. The lower skin was 0.1 ins thick light alloy. The blowing nozzles were positioned over the spar.

An hydraulically driven ball screw actuator was used to operate each flap. It was located at the centre hinge.

g) Wing Actuation and Services

The movement of the outer wing panels was accomplished by means of ball screw actuators. Their positions are shown in Figure 13. Each one was connected across from the inner wing box to the rear face of the outer wing lugs. The assemblies incorporated both vertical and horizontal pivots at the attachment points to cater for distortion as well as changes of alignment during wing sweep. Each actuator was driven by a shaft from a central gearbox and the pair were also directly interconnected. This interconnection provided both for synchronisation and back up drive in the event of a primary shaft failure. The gearbox was driven by duplex hydraulic motors supplied from independent sources.

Electrical leads to the outer wing were passed along the forward face of the hinge assembly and a loop was incorporated to cater for the change in length during wing sweep. The loop was attached to a strip of flexible berylium copper to provide vertical restraint. Hydraulic supplies to the flaps were passed along the lower surface of the inner wing to the centre of the hinge where swivel banjo joints were positioned. From there they passed along the lower surface of the outer lugs and into the box beyond the lugs. The blowing air supply for the flaps could only be used with the wings in the forward setting. The supply terminated in a self sealing coupling on the outer face of the fairing structure, forward of the hinge. A probe located correspondingly in the root of the outer wing engaged with the coupling in the appropriate condition.

6.1.2. Fuselage

The layout chosen for the aircraft was such that substantial difficulties were encountered in the structural design of the fuselage. In the centre fuselage it was necessary to provide substantial cutouts in the lower surface for the weapons and main undercarriage bays in the same region as the wing box and the occurrence of the highest shear and bending loads. The wing hinge positions were within the fuselage cross section above the undercarriage bay. It was convenient to support the centre wing box on a pair of longitudinal shear webs which coincided with the edges of the weapon bay. Unfortunately the width of the fuselage forward of the air intakes and weapon bay was significantly greater than the 4 ft distance between these webs due to the choice of side by side seating for the crew. Thus it was not possible to extend the webs forward to coincide with the more or less vertical sides of the forward fuselage. Because of this there is an awkward region where the webs are partially discontinuous and this is complicated by the passage of the air intake ducts through the webs. Aft of the weapons bay these central shear webs must terminate because of the presence of the powerplants. In this region a single central web was used to provide shear stiffness and act as a firewall between the two engine bays. Engine removal was through large doors located in the lower surface of the fuselage and of necessity were wider than the spacing of the centre fuselage webs.

a) Basic Structure

The centre fuselage longitudinal webs were full depth over the length of the weapon bay and cutout only for the passage of the wing centre box and actuators. Each web was built up on a 0.1 ins thick plate. It was stiffened by horizontal zed stringers of 1.5 ins depth and 0.05 ins thickness. Apart from those frames coincident with the webs of the wing box the frames were attached to the web by 2.0 ins deep tee section extrusions. Double extruded angle sections were employed at the wing spar frames. Longerons were located along both the upper and lower edges of the webs. These were 2 ins deep I section extrusions. Machined reinforcements of up to 0.32 ins thickness were attached around the cutouts for the wing actuators. Forward of the weapon bay the lower part of the webs was kinked inwards to join up to the nose undercarriage bay sidewalls and finally terminate on the rear face of the cockpit sloping bulkhead. The upper part of the webs was terminated at a suitable bulkhead and the shear load transferred outboard to the forward sidewalls.

Over the greater length of the fuselage the frame pitch was approximately 1.5 ft, and the outer shell was supported by zed section stringers. These were typically positioned at 4 ins pitch around the periphery and were 1.5 ins deep and 0.064 ins thick. In the cabin region the frame pitch was about 1.0 ft and the stringers 1.0 ins deep. The design of the intermediate frames varied according to their location along the fuselage. In the majority of instances they were of pressed channel or zed section. Full depth webs were used at the end of the integral fuel tanks and at the forward extremity of the pressure cabin.

The cockpit sloping pressure bulkhead was machined in one piece with a 1.25 ins deep reinforcing grid at about 8 ins pitch both horizontally and vertically. The nose undercarriage leg was attached to the lower edge of this bulkhead and the drag strut to the roof of the bay. Steel machinings were used for the attachment fittings. A major bulkhead was located at the aft end of the main undercarriage bays. This was also a single piece machined item with a web thickness varying in the range of 0.1 ins to 0.2 ins. The 1.5 ins deep supporting grid had a horizontal pitch of 5.5 ins and a vertical pitch of 4.5 ins. The main undercarriage skewed hinge pintles were attached to the forward face of the bulkhead. The outer attachment lug was machined integrally with the bulkhead but the inner one was substantially further forward and a separate structure was built up from the bulkhead to accommodate it. The pintles were mounted in split trunnions.

At the rear end of the fuselage the structure terminated in two substantial bulkheads. These were of mixed forged and built up construction. The more forward of the two was a double unit with the taileron spigot attached between the two parts. The fin front spar also connected to this bulkhead. The rearmost bulkhead supported the fin rear spar and the reheat nozzles.

Each engine was supported on a pair of horizontal trunnions and a single swinging link. The former were located in fittings attached to the centre web and the outer sidewalls and the latter was suspended from the roof of the bay which was also the upper fuselage section.

b) Air Intakes

The intake system and ducting was built into the fuselage structure. Variation of intake area and shock angle was achieved by moving the semi-conical bullets in each intake. The bullets were located on the vertical sidewalls of the fuselage and each one was made in two parts. The geometry of these two parts was such that a smooth contour was maintained as each part was moved independently about a vertical pivot located at their opposite extremities. Movement was obtained by hydraulic jacks operating both parts through a linkage system. The bullet and intake lips were made in titanium alloy.

The light alloy ducting was stiffened by zed section stringers and passed through the longitudinal shear webs and integral fuel tanks.

c) Windscreen and Canopy

The vee shaped windscreen framing was cast in L52. The front glazing panels were each approximately 34 ins by 23 ins in size. The panels were designed to cope with bird impacts in the low level maximum operating speed condition. Their cross section consisted of an outer 0.125 ins thick annealed splintershield; 0.75 ins thick toughened glass; 0.375 ins vinyl interlayer; 0.25 ins alumina silicate glass; 0.5 ins wide conditioned gap and a 0.4 ins alumina silicate internal facing. The side and eyebrow glazing panels used two panels of 0.375 ins thick alumina silicate with a 0.5 ins gap between them.

A double plexiglass construction was used for the canopy. It was built in two parts, each of which was hinged along the upper centreline member. Entrance to the cabin was through the canopy. Both canopy sections were designed to be blown off for crew ejection.

d) Weapon Bay Doors

The weapon bay doors were of conventional clamshell design. Each door was hinged at four points along the lower edge of the centre fuselage shear webs. The construction used consisted of 0.08 ins thick double skins separated by channel section ribs and fore and aft edge members. The hinge ribs were machined items and the remainder pressed.

e) Air Brakes

The air brake system consisted of three separate 'barn door' surfaces located round the rear fuselage. One was located on the lower centreline just forward of the engine bays. The other pair were located on the upper fuselage sides, somewhat further forward.

The construction of both the lower and upper doors was similar although the former was somewhat larger than the other two. Each brake was hinged at two points on the fuselage just forward of the door edge. The hinge members were machined from titanium alloy and formed the basis of the internal grid structure. All the grid members were of I section and apart from the heavily loaded hinges were machined from light alloy. The double skinning was in 0.08 ins thick light alloy.

Each door was moved by its own ball screw actuator. These were driven from a single gearbox powered by duplex hydraulic motors. The gearbox was located on the rear face of the aft weapon bay bulkhead.

6.1.3 Tail Unit

The tail unit consisted of the single fin with conventional rudder and independent halves of the tailplane, or tailerons, and their flaps.

a) Fin

The critical design load for the fin was 36,400 lbs proof and it occurred as the result of sinusoidal movement of the rudder at M = 1.1 and sea level.

Two main spars were used for the fin construction. These were built integrally with the two rear fuselage bulkheads and were curved in side elevation at the root to accommodate the basic sweep of the structure. In this curved region the ribs were placed along radial lines about a point corresponding to the intersection of the fin trailing edge of fuselage top line. Torsional stiffness requirements were found to be dominant in determining the skin thickness. The covers were integrally machinedwith a skin thickness varying from 0.2 ins at the root to 0.074 ins at the tip. The shallow stiffeners were placed at about 6 ins pitch across the chord. Extrusions of channel section were used for the spars which were stretch formed in the root region. Spar web thickness was 0.15 ins at the root and 0.04 ins at the tip. In the root region the ribs were machined but above the curved spar section pressed channels were used. Pressed ribs placed perpendicularly to the front spar were employed to support the leading edge skins.

b) Rudder

The maximum load on the rudder was approximately 10% of that on the fin in the same critical case. The component was hinged from the fin rear spar at two points but was operated through a torque tube attached to the root rib. The actuator was located in the top of the fuselage.

The construction used a single pressed channel spar with the 0.05 ins thick skins supported by full depth honeycomb. Pressed ribs were used at the two ends and hinge points.

c) Tailerons

The tailerons were designed by the combined rolling/pitch out case at M = 2.5 and 36,000 ft when the proof load on one half was 24,400 lbs. Each taileron was pivoted on a spigot extended out from the fuselage side, and operated by an actuator located within the fuselage which connected to a boss extending inwards from the taileron front spar.

A swept and tapered single cell box was used as the basis for the structure. Over the inner part of the span integrally machined covers were employed, but outboard of the spigot bearing rib the covers were built up with top hat section stiffeners. The skin thickness varied from 0.21 ins at the root to 0.11 ins at the transition of construction and 0.056 ins at the tip. The integrally machined stiffener pitch was 3.6 ins and the depth 1.1 ins. Channel section pressings were used for the spars the basic thickness being 0.1 ins at the root and 0.048 ins at the tip. The taileron planform ahead and behind the basic structural box was constructed using 0.064 ins thick skins supported by pressed channel section ribs.

A section through the taileron spigot and bearings is shown in Figure 15. The inner of the two bearings was a dry PTFE reinforced item of the Fibreslip type. It was mounted on the machined root rib just forward of the rear spar. The outer bearing was also located on a rib but in this case the chordwise position was just aft of the front spar. A double row tapered roller bearing in a self-aligning housing was used. The outer bearing reacted all the tangential loads and provision was made for the adjustment of end float. The spigot itself was a hollow steel forging in S99 which was bolted between the two parts of the fuselage mounting bulkhead.

d) Taileron Flaps

The subsidiary surface at the taileron trailing edge was used simply as a flap to augment the lift coefficient available to trim the aircraft longitudinally with the wing flaps deployed and blown. A single spar construction was used with two hinges connecting it to the primary surface. The 0.13 ins thick skins were supported by full depth honeycomb and pressed ribs at the ends and hinge positions. The flap was hydraulically operated by an actuator housed in a blister below the surface at the inner hinge position.

6.1.4 Undercarriage

A conventional tricycle undercarriage was employed with two wheels on each of the three units.

a) Main Undercarriage

The layout of the main undercarriage presented some difficulty. The major problem was that of obtaining adequate track and at the same time satisfactory stowage in the volume available in the bays. The latter was restricted by the cross section available outside the intake ducting but within the cross section. Initially a lever suspension unit with a canted out leg was proposed. Retraction was found to be difficult. The skewed hinge angle needed to bring the leg into the correct retracted position resulted in the twin wheels lying across the bay at an angle which required greater width than was available. This could have been overcome by introducing a second skew hinge part of the way along the main leg and the geometry required for this was established. However this solution was considered to be mechanically unacceptable. The solution finally adopted was unusual in that the twin wheels were both positioned on the same side of the leg to give the required track. The bending loads at the base of the leg were very high but it was possible to maintain the leg more or less vertical in the front elevation. The general arrangement of the final layout is shown in Figure 16. The proof design loads were 57,400 lbs vertically, 23,000 lbs drag and 19,400 lbs side. The first of these two resulted from a two point landing and the latter during turning and swinging.

A liquid spring shock absorber was used, the proof reaction factor being 3.4. Steel was used throughout the main leg. The piston rod was S96 but the other parts were forged in S98. The main leg and side stays supporting the pintles were in one piece. The long drag strut was basically a T59 tube hinged part of the way along its length for retraction. The down lock was incorporated in the knuckle joint.

b) Nose Undercarriage

Although various arrangements for the nose undercarriage were considered a straightforward telescopic unit was chosen. The proof design loads were 31,000 lbs vertically, 12400 lbs drag and 7750 lbs side, which occurred in the three point landing cases.

The steering mechanism was mounted across the top of the main support trunnions. The triple tube oleo-pneumatic shock absorber assembly consisted of an S99 sliding tube surrounded by an S96 steering tube and an L77 light alloy outer casing. Both the steel components were hard chrome plated. The lower end of the sliding tube terminated in an L77 axle fitting bolted on to it. The axle itself was a separate S96 member and the toggle links were machined in S99.

The undercarriage bay doors were in three parts. The forward unit was attached to the leg itself and there was a rear pair of clamshell doors operated mechanically from the leg. The retraction jack was located across the drag strut knuckle joint and the bay roof. It incorporated the mechanical down lock.

6.2 Powerplant System

6.2.1 Engine installation

The two Rolls Royce BS 146-01R engines were located side by side in the rear fuselage. The basic engine, complete with all accessories, was installed from below and two large access doors were provided in the rear fuselage. Smaller panels in the large doors were located to enable normal servicing to be undertaken without opening the main access. The reheats and short jet pipes were removed by aft withdrawal along rails built into the structure for this purpose. It was possible to remove both the engine and reheat assemblies independently of one another. The arrangement of the engine accessories and systems is shown in Figure 17.

Each engine unit was attached to the airframe at three points. There were two side trunnions mounted in split housing fittings positioned on the central shear web and outer walls. A forward suspension point was attached to the top fuselage structure. Small panels in the upper fuselage skinning gave access to this forward suspension. Installation of the engine in the bay could be achieved either by using small winches attached to the roof or with the aid of a special elevating trolley. This trolley was designed to be suitable for reheat removal also.

The main bay doors were hinged at two points on the lower edge of the centre shear web. At their front ends the doors were cutout for the lower airbrake. Each door was some 7.5 ft long and 3.2 ft wide. The construction consisted of a double skinning supported by an 'egg-box' arrangement of zed section stringers. Titanium alloys were used both to impart stiffness and to cope with the relatively high temperatures in the bay region. The doors were not designed to carry the primary structural loads and Dzus type fasteners were used around their whole peripheries.

6.2.2 Fuel system

Each of the engines was fed with fuel from an independent system although there was provision for cross feeding so that either all remaining fuel could be used for one engine or both engines fed from one half of the system. Schematic diagrams of the main features of the complete system are shown in Figure 18. There were five tanks in each half. Tanks 1 to 4 fed into tank 5 port which acted as a collector for the port engine system. Similarly tanks 6 to 9 fed into collector tank 5 starboard. The total capacity of the system was some 2490 Imperial gallons. This was contained in integral tanks located in the fuselage. The system was primarily designed for the use of AVTUR although provision was made for the use of AVTAG in certain circumstances.

All the tanks were pressurised to a differential of 6 lb/sq in. Transfer to the collector tanks was by the air pressurisation through four cell hydraulically driven proportioners. Each collector tank was fitted with twin A.C. booster pumps for the engine feed. The refuel/defuel system consisted of a main gallery running along the length of the tankage from the refuelling point located on the forward starboard side of the fuselage. Using a 300 gallon/minute bowser it was calculated that refuelling could be completed from empty in 8.5 minutes and defuelling in 18 minutes. Fuel could be jettisoned using, in the main, the feed lines. Gross imbalance in flight could be corrected manually by using the refuel system in conjunction with part of the jettison system.

7. DISCUSSION

7.1 Wing Hinge Location and Fairings

The initial concept for the variable geometry was based on the use of only the two extreme sweep positions. The choice of hinge location and fairing geometry was based on the principle of obtaining the best possible match between the fixed and moving wing panels in these two conditions. The fully swept, high speed position was given first priority and a discontinuity in the aerofoil contour during transition was accepted. Subsequent detailed performance analysis showed that the advantages of using intermediate sweep positions were greater than was originally anticipated and that the requirements in the high subsonic speed range could only be achieved in this way. The modifications required to give an acceptable compromise on the fairing geometry were not very great. The major change involved the elimination of the step in contour which lined up with the fence in the low speed position and matched the edge of the inner wing fairing in the high speed case. This change resulted in a 0.6 ins step down in contour around the edges of the inner wing fixed fairing behind the hinge point.

A much more severe penalty which had been accepted to enable a good fairing geometry to be achieved was the trim change with wing sweep consequent upon the use of an aft location for the hinge. It is considered that a more forward hinge location would have been beneficial in reducing trim changes and enabling a greater reduction of wing area in the high speed position to be achieved. It should have been possible to accommodate this more forward hinge location without undue penalties in the fairing geometry especially once the strict requirements for the high speed position had been relaxed.

7.2 Fuselage Layout and Main Undercarriage

Comments have previously been made of the structural layout difficulties encountered in the fuselage which were a consequence of the side by side seating for the crew and proximity of cutouts for the main undercarriage and weapon bays. The use of tandem seating would have enabled the overall width of the forward fuselage to be reduced and coincide with spacing between the two centre fuselage longitudinal webs. The air intakes would then have been mounted outside the main structure, but it would still have been necessary to pass the intake ducts through the webs. In spite of this the structural arrangement would have been simpler and the cross sectional area at the intakes somewhat less.

On the other hand there would have been a narrow space for the stowage of the main undercarriage units, and an alternative layout would have become necessary. It is not obvious just what form this would have taken but it might well have necessitated the use of a bogie to keep the overall width down to be compatible with the bay. The main difficulty would have been to maintain adequate track with satisfactory shock absorption characteristics. The reduced fuselage cross section area would have implied a longer fuel tankage arrangement and with the tandem seats a longer fuselage. The total wetted area would almost certainly have been significantly greater but the volume wave drag less. It is not possinle to be dogmatic as to which of the two layouts would have resulted in the best design since both introduce significant paneltles. However the far reaching effects of the choice of seating layout are of note as are the problems of undercarriage arrangement for this class of aircraft, especially when internal weapon stowage is required.

8. CONCLUSIONS

1) On balance it would appear that the selected hinge position for the wing was too far aft and a better design would have resulted had its location been somewhat further forward on the chord.

2) The choice between side by side or tandem seating for the crew has important implications to the layout of fuselage structure in an aircraft of the size considered.

3) The design of a satisfactory undercarriage for this class of aircraft introduces considerable difficulties especially when internal weapon stowage is a requirement.

REFERENCES

1. HOWE, D. Aircraft Design Studies - Variable Sweepback Naval Aircraft
College of Aeronautics Rep. 206. 1968

APPENDIX A

Allocation of Components for S68 Study

Brain, C.J.	Main undercarriage
Butterworth, D.L.	Fin and rudder
De Lessenne Coulander, C.M.	Wing trailing edge flap system
Draper, J.M.	Outer wing and leading edge flaps.
El-Bahaie, B.E.A.	Inner wing and fairings
Hall, S.J.	Tailerons
Hutchinson, B.	Hinge structure and wing sweep system.
Jackson, R.K.	Airbrakes
McNaughton, J.B.	Powerplant installation.
Patel, N.G.	Rear fuselage
Perry, P.J.	Nose undercarriage
Potter, J.G.	Forward fuselage including air intakes.
Smith, D.	Nose fuselage and cockpit
Spragg, P.J.	Fuel system
Ward, R.E.	Centre fuselage structure.

APPENDIX B

Specification of Aircraft

1. GEOMETRY

1.1 Wing

Aerodynamic Reference Area		520 sq ft.
Lowspeed:	Gross area	530 sq ft.
	Span, actual	52 ft
	Aspect ratio	5.1
	Leading edge sweepback, inner panel	66^{o}
	Leading edge sweepback, outer panel	20^{o}
	Trailing edge sweepback, outer panel	11^{o} approx.
	Area of outer panel	234 sq ft.
	Standard mean chord	10.2 ft
Intermediate:	Span, actual	45.4 ft
	Aspect ratio	3.9
	Leading edge sweepback, outer panel	45^{o}
Highspeed:	Gross area	522 sq ft
	Span, actual	34.2 ft
	Aspect ratio	2.24
	Leading edge sweepback	66^{o}
	Trailing edge sweepback	58^{o} approx.
	Standard mean chord	16.2 ft
	Wing angle to body datum	2^{o}
	Location of wing datum above fuselage datum at hinge line	3.05 ft
	Location of hinge line aft of fuselage nose datum	40.7 ft
	Location of hingeline aft of leading edge intersection with aircraft centreline	22.47 ft
	Location of hinge point outboard of centreline	3.75 ft
	Location of fence outboard of centreline	8.0 ft
	Wing chord at fence	8.0 ft
	Aerofoil sections:	

The wing aerofoil sections are based on a 10 per cent thickness to chord ratio RAE 104 shape, but have been substantially modified to suit the root fairing geometry and to move the thickness aft to the semi-chord point. See Figure 6.

1.2 Trailing edge flaps

The trailing edge flaps occupy the full span of the lowspeed wing, outboard of the wing fence. They are of the plain type with blowing over the leading edge.

Flap chord/wing chord	0.3
Take off flap angle	30^{o}
Landing flap angle	60^{o}

1.3 Leading edge flaps

The wing leading edges are drooped in conjunction with the trailing edge flap deflection. Drooping occurs over that part of the leading edge of the lowspeed wing outboard from the fence to a station 24.0 ft from the centreline. The flaps are blown at the knee.

Flap chord/wing chord	0.08
Flap angle	40^{o}

1.4 Tailplane

Gross area	240 sq ft
Net, reference area	125 sq ft
Span, gross	24.8 ft
Net aspect ratio	2.25 ft
Leading edge sweepback	55^{o}
Trailing edge sweepback	11.5^{o} approx.
Centreline datum chord	17.3 ft
Nominal tip chord	2.35 ft
Net, standard mean chord	7.45 ft
Location of tailplane datum above fuselage datum	1.2 ft
Location of leading edge intersection with centreline, aft of wing hinge line	7.25 ft
Angular movement	$\pm 10^{o}$

Aerofoil section 6% thick RAE 104.

1.5 Tailplane flap

The tailplane flap extends over the whole of the net tailplane span. It is a plain flap with blowing at the knee of the undersurface, and is deflected in conjunction with the wing flaps.

Flap chord/tailplane chord, at inner end	0.24
Flap chord/tailplane nominal tip chord	0.48
Movement (up only)	-30^{o}

1.6 Fin

Reference area above fuselage	66 sq ft
Height above fuselage datum	13.2 ft
Net aspect ratio	1.43
Leading edge sweepback	52^{o}
Trailing edge sweepback	15^{o} approx.
Chord at fuselage datum	15.4 ft
Nominal tip chord	2.1 ft
Net, standard mean chord	6.8 ft
Location of leading edge intersection with fuselage datum, aft of wing hinge line	10.4 ft

Aerofoil section 5% RAE 104.

1.7 Rudder

The rudder is a plain, round nose, control and effectively extends over the whole of the fin height above the fuselage.

Rudder chord/fin chord at fuselage top, in line of flight	0.29
Rudder chord/fin nominal tip chord	0.72
Sweepback of hingeline	23° approx.
Angle of rudder root chord to hingeline	90°
Distance of rudder root at hingeline, above fuselage datum	4.65 ft
Angular movement, lowspeed	$\pm 20^{\circ}$

1.8 Fuselage

Overall length, including pitot	71.7 ft
Length of pitot forward of nose datum	4.0 ft
Maximum width	9.6 ft
Maximum depth	5.7 ft
Surface area, including centre wing across fuselage	1420 sq ft

1.9 Speed brakes

The speed brakes consist of three interconnected units and are located on the rear fuselage. One unit is in the lower surface, aft of the bomb bay, and the others are in the upper, side surfaces, aft of the wing. They open to an angle of 60°.

Lower brake: Reference area	16 sq ft
Distance of leading edge aft of wing hinge datum	10.5 ft
Length	4.0 ft
Width	4.0 ft
Upper brakes: Total reference area	17.5 ft
Distance of leading edge aft of wing hinge datum	8.0 ft
Length	3.5 ft
Width around fuselage contour	2.5 ft

1.10 Undercarriage

Type:- Nosewheel	
Wheelbase (50,000 lb static)	27.9 ft
Track, between centre of wheel units	10.0 ft
Distance of static ground line below fuselage datum	5.7 ft
Design proof vertical velocity	14 fps

Main units:- Cantilever, offset twin tyre
Tyres 33 inches dia. by 9 inches width
Tyre pressure at 50,000 lb 100 psi

Nose unit:- Cantilever, twin tyre
Tyres 22 inches dia. by 6 inches width
Tyre pressure at 50,000 lb 100 psi

Location of mainwheel axle aft of wing hingeline, 50,000 lb static	2.9 ft
Location of nosewheel axle forward of wing hingeline, 50,000 lb static	25.0 ft

2. POWERPLANTS

Number	2
Type:- Rolls Royce, Bristol, BS146-01R,	
Bypass ratio	1.3
Maximum reheat temperature	2000°K
Rated sea level static thrust, dry	8120 lb
Rated sea level static thrust, max. reheat	14000 lb
Overall length (including reheat and nozzle)	14.2 ft
Maximum diameter	2.79 ft
Intake diameter	2.47 ft
Location: Side by side in rear fuselage	
Distance of engine front face aft of wing hingeline	12.15 ft
Distance of engine centreline above fuselage datum	1.75 ft
Distance of engine centreline outboard of centreline	1.75 ft

Air intake:- The engine air intakes are located on the sides of the fuselage, forward of the wing. They are of the semi-annular type, the area being varied primarily by lateral translation of the central bullet. Secondary ground intake doors are incorporated in the sides of the outer cowl.

Nominal diameter at intake lip	3.0 ft
Area of secondary side door, per side	1.5 sq ft
Location of intake lip, at intake centreline, forward of wing hingeline	15.65 ft
Location of intake lip, at intake centreline, above fuselage datum	1.7 ft
Angle of intake lip to fuselage datum	85°

3. AERODYNAMIC INFORMATION

3.1 General

Maximum lift coefficients at low speed, (untrimmed):-

Basic wing	1.0
Increment due to flaps at 30°, no blowing	0.45
Increment due to flaps at 30°, $C\mu = 0.015$	0.82
Increment due to flaps at 60°, no blowing	0.63
Increment due to flaps at 60°, $C\mu = 0.055$	1.47

(N.B. The leading edge flap is assumed to be drooped and blown, $C\mu = 0.015$, when the trailing edge flaps are blown. $C\mu$ is the blowing momentum rate coefficient, defined as the product of the blowing mass flow and slot velocity, non dimensionalised in terms of the flight dynamic pressure and the surface area equivalent to the span over which the blowing occurs).

Drag coefficients:-

Zero lift drag coefficient, clean aircraft	see Fig.10
Induced drag coefficient, clean aircraft	see Fig.11

Drag polars, lowspeed, sea level

Clean aircraft	$C_D = 0.0155 + 0.06C_L^2$
Undercarriage down, flaps retracted	$C_D = 0.0448 + 0.06C_L^2$
Undercarriage down, flaps 30° blown	$C_D = 0.204 + 0.06C_L^2$
Undercarriage down, flaps 60° blown	$C_D = 0.555 + 0.06C_L^2$

Pitching moment coefficients, lowspeed

Zero lift, wing-body, C_{MO}	-0.035
Increment due to flaps at 30°, blown	-0.24
Increment due to flaps at 60°, blown	-0.43

Stability derivatives (per radian)

Rolling moment coefficient due to rate of roll, ℓ_p:-

lowspeed	-0.35
high speed, M = 2.0	-0.21

Rolling moment coefficient due to sideslip, ℓ_v, lowspeed:-

$$-\{0.11C_L + 0.106 + 0.034(0.73-C_L)\}$$

Rolling moment coefficient due to rate of yaw, ℓ_r, lowspeed:-

$$\{0.22C_L + 0.034(0.73-C_L)\}$$

Yawing moment coefficient due to sideslip, n_v, lowspeed

0.126

Yawing moment coefficient due to rate of yaw, n_r, lowspeed

$$-\{0.15 + 0.006 + 0.022C_L^2\}$$

Side force coefficient, y_v, lowspeed 0.204

(C_L is the overall aircraft lift coefficient)

3.2 Aerofoil characteristics

Wing-body lift curve slope, a_1:-

lowspeed	4.1/rad
highspeed wing, M=1.0 to M=1.8	2.1/rad

Tailplane lift curve slope, a_{1T}:-

lowspeed	2.0/rad
M = 1.1	3.0/rad
M = 2.0	2.0/rad

Fin lift curve slope, a_{1F}:-

lowspeed	3.3/rad
M = 1.0	5.0/rad
M = 2.0	2.8/rad

Ratio of fin/rudder lift curve slopes, a_{2F}/a_{1F},

lowspeed	0.51
M = 2.0	0.28

4. INERTIA CHARACTERISTICS

Basic equipped weight	27720 lb
Operating weight, zero fuel	30080 lb
Maximum internal fuel load	21920 lb
Maximum internal payload	4000 lb
Weight breakdown	See Table 1
Centres of gravity	See Table 2
Moments of inertia	See Table 2

TABLE 1

WEIGHT BREAKDOWN

Component	Weight, lb		% AUW
Fuselage; basic 5360 lb intake structure 970 lb fixed controls 100 lb	6430		12.8
Wing; inner 3170 lb outer and flaps 3560 lb	6730		13.5
Tailplane, and flap	670		1.3
Fin and rudder	350		0.7
Undercarriage; main 2140 nose 330	2470		5.0
Total Structure		16650	33.3
Engines, excluding reheat	4320		8.6
Engine installation	200		0.4
Total Powerplant		4520	9.0
Fuel system, including tanks	700		1.4
Auxiliary air system	800		1.6
Electrical system	1100		2.2
Hydraulic system, including power controls	1200		2.4
Accessory drives	300		0.6
Total Systems		4100	8.2
Instruments	150		0.3
Communication, navaition, suto flight	1600		3.2
Furnishings, including seats	400		0.8
Fixed armament	100		0.2
Miscellaneous cockpit equipment	200		0.4
Total Equipment		2450	4.9
Basic equipped weight		27720	55.4
Crew	360		0.7
Normal payload	2000		4.0
Operating Zero Fuel Weight		30080	60.1
Fuel	19920		39.9
Take Off Gross Weight		50000	100.0

TABLE 2

CENTRES OF GRAVITY AND MOMENTS OF INERTIA

	Case	Weight lb	$\bar{x}$ ft positive below fuselage	$\bar{z}$ ft positive below fuselage datum	Moments of Inertia about hinge and fuselage datums lb ft^2		
					Roll	Pitch	Yaw
Undercarriage retracted	Basic equipped: low speed	27720	0.28	-0.99	1.25×10^6	6.35×10^6	7.30×10^6
	Basic equipped: high speed	27720	0.77	-0.98	0.85×10^6	7.05×10^6	7.60×10^6
	Zero payload, full fuel: low speed	48000	Fuel disposed and used to keep c.g. within limits		1.47×10^6	8.04×10^6	9.17×10^6
	Zero payload, full fuel: high speed	48000			1.07×10^6	8.74×10^6	9.47×10^6
	A.U.W. low speed*	50000			1.50×10^6	8.10×10^6	9.20×10^6
	A.U.W. high speed*	50000			1.10×10^6	8.80×10^6	9.50×10^6

* Payload assumed is 2 x 1000 lb bombs

When the undercarriage is extended at the basic equipped weight

$\bar{x} = 0.4$ ft and $\bar{z} = -0.48$ ft.

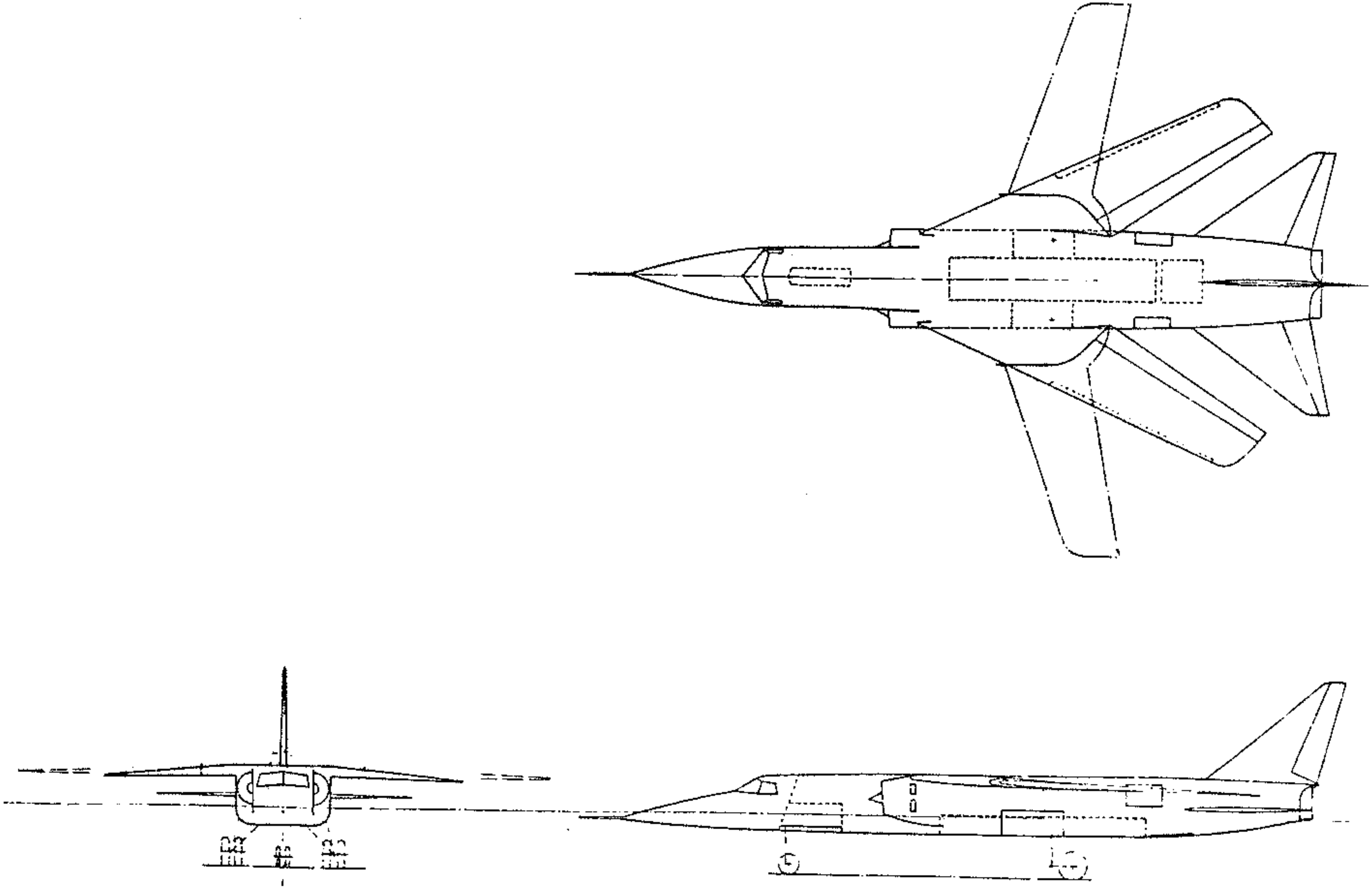


FIG. 1 GENERAL ARRANGEMENT OF AIRCRAFT

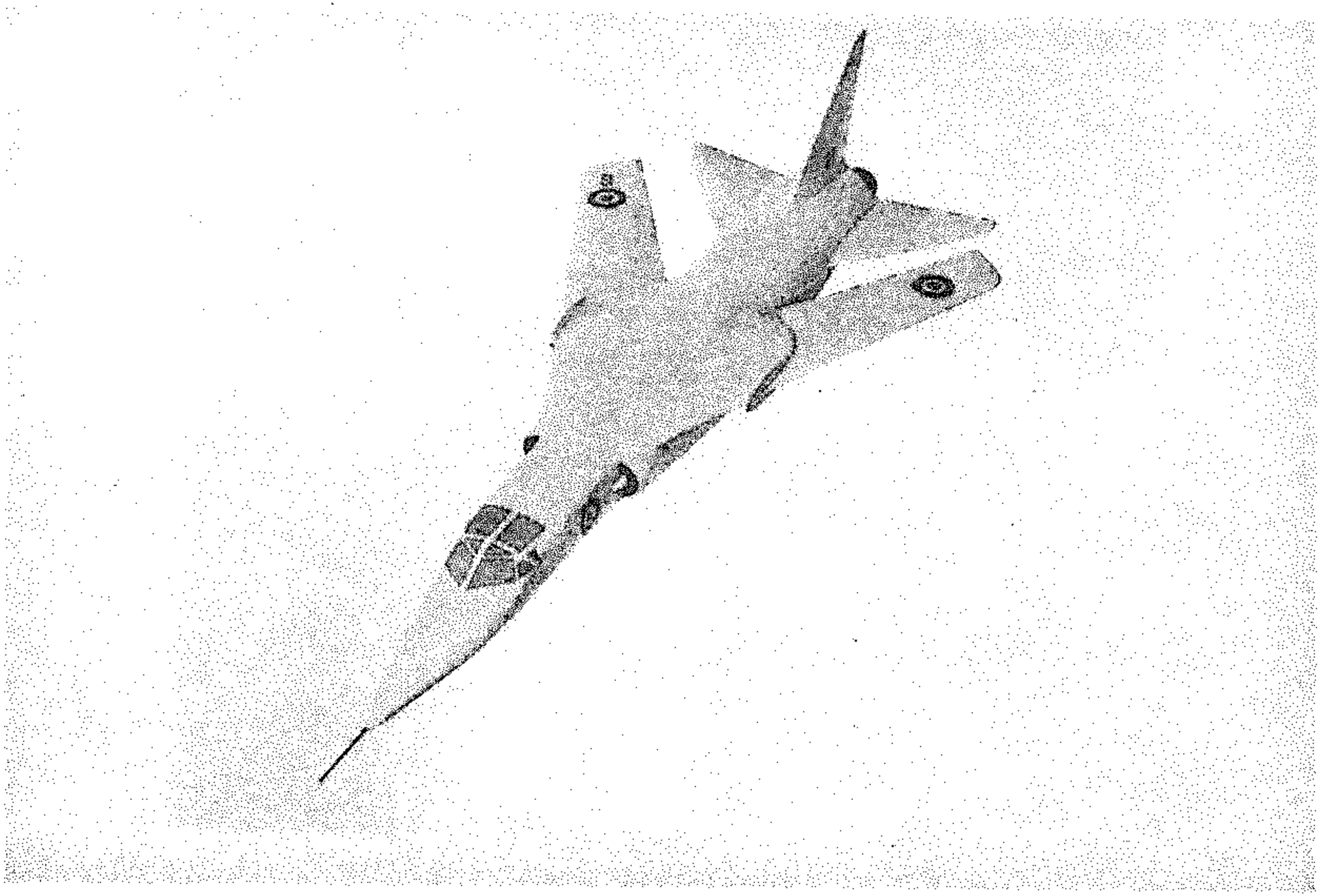


FIG. 2 PHOTOGRAPH OF MODEL WITH WINGS FULLY SWEPT

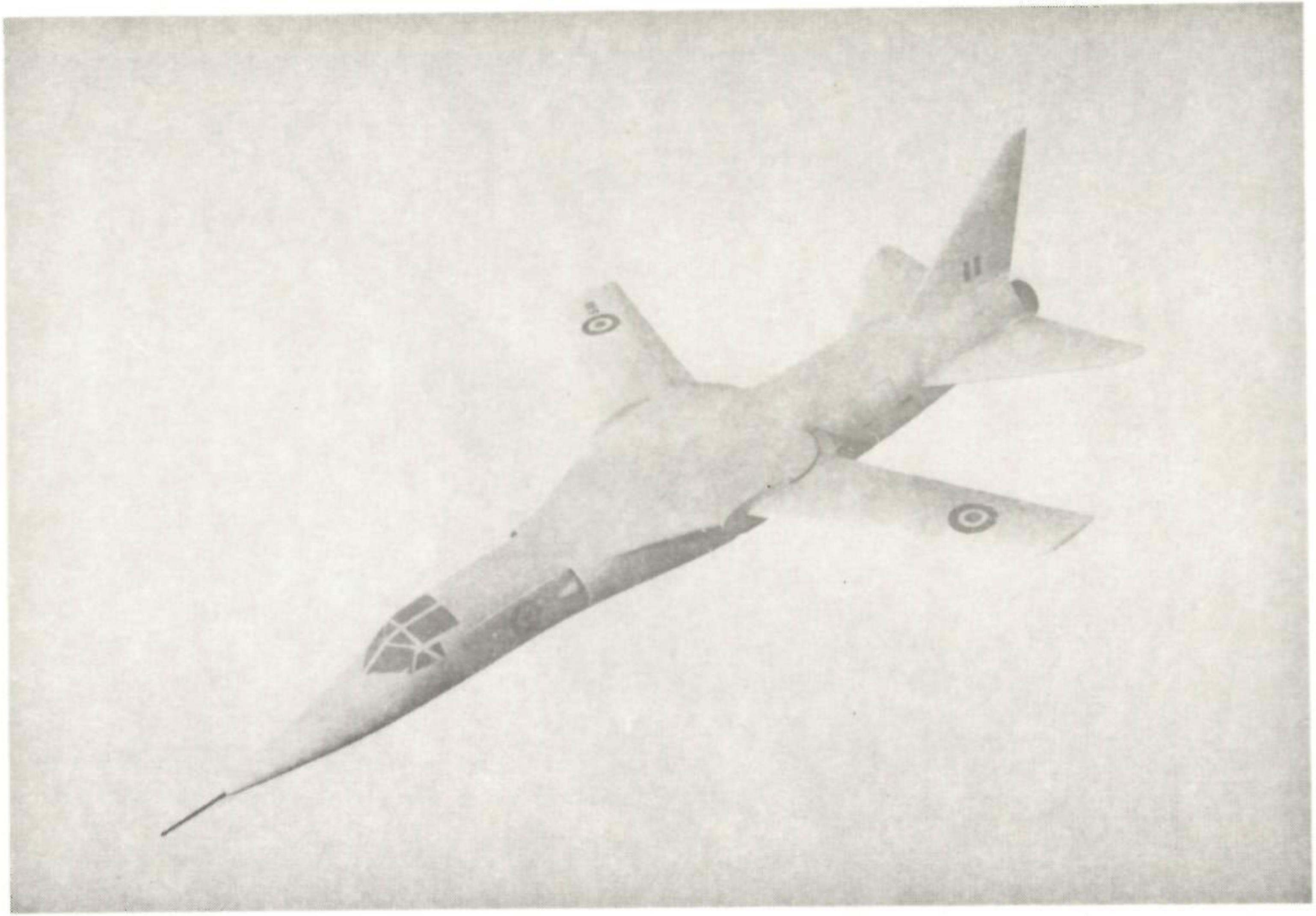


FIG. 3 PHOTOGRAPH OF MODEL WITH WINGS IN LOW SPEED POSITION

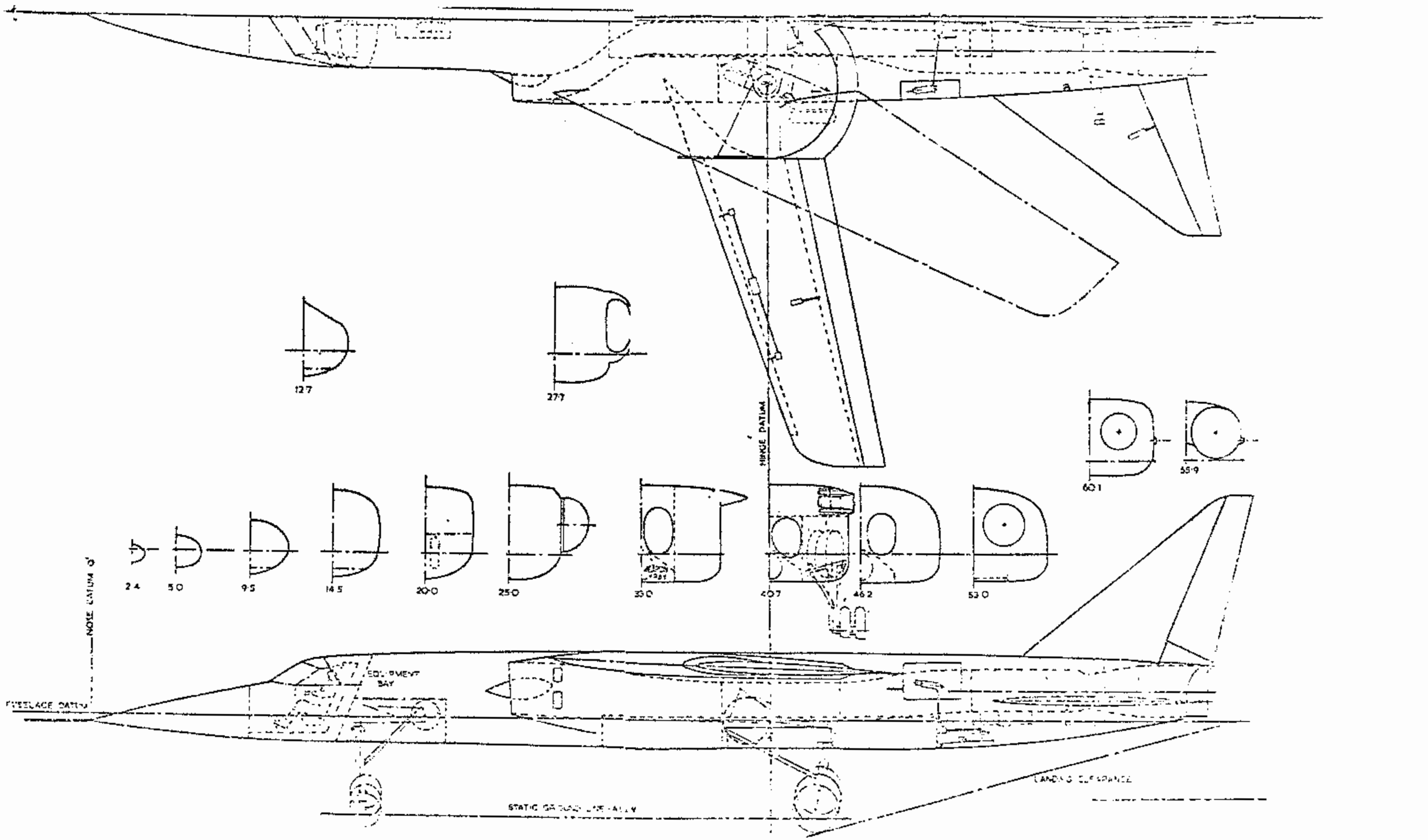
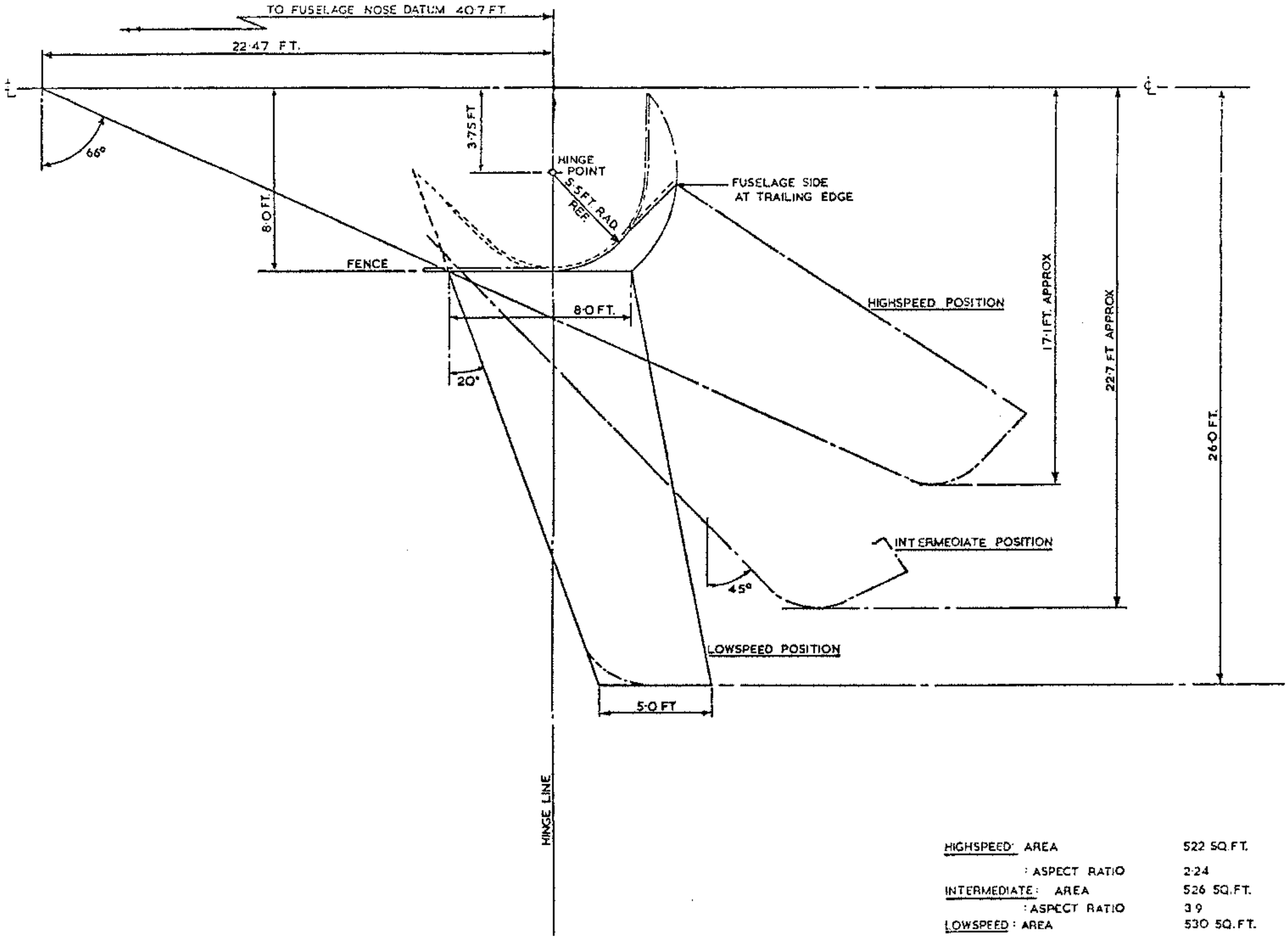


FIG. 4 LAYOUT OF FUSELAGE



HIGHSPEED	AREA	522 SQ.FT.
	: ASPECT RATIO	2·24
INTERMEDIATE	: AREA	526 SQ.FT.
	: ASPECT RATIO	3·9
LOWSPEED	: AREA	530 SQ.FT.
	: ASPECT RATIO	5·1

FIG. 5 WING GEOMETRY

AEROFOIL SECTION OUTBOARD OF [illegible]

x %	0	0.5	1	2	3	5	7.5	10	15	20	25	30	35	40	45	50	55	60	65	70	75	80	85	90	95	100
z %	0	0.75	1.05	1.4[illegible]	1.68	2.13	2.64	3.0	3.57	4.0	4.32	4.54	4.72	4.85	4.96	5.0	4.96	4.84	4.51	3.99	3.40	2.78	2.10	1.40	0.70	0

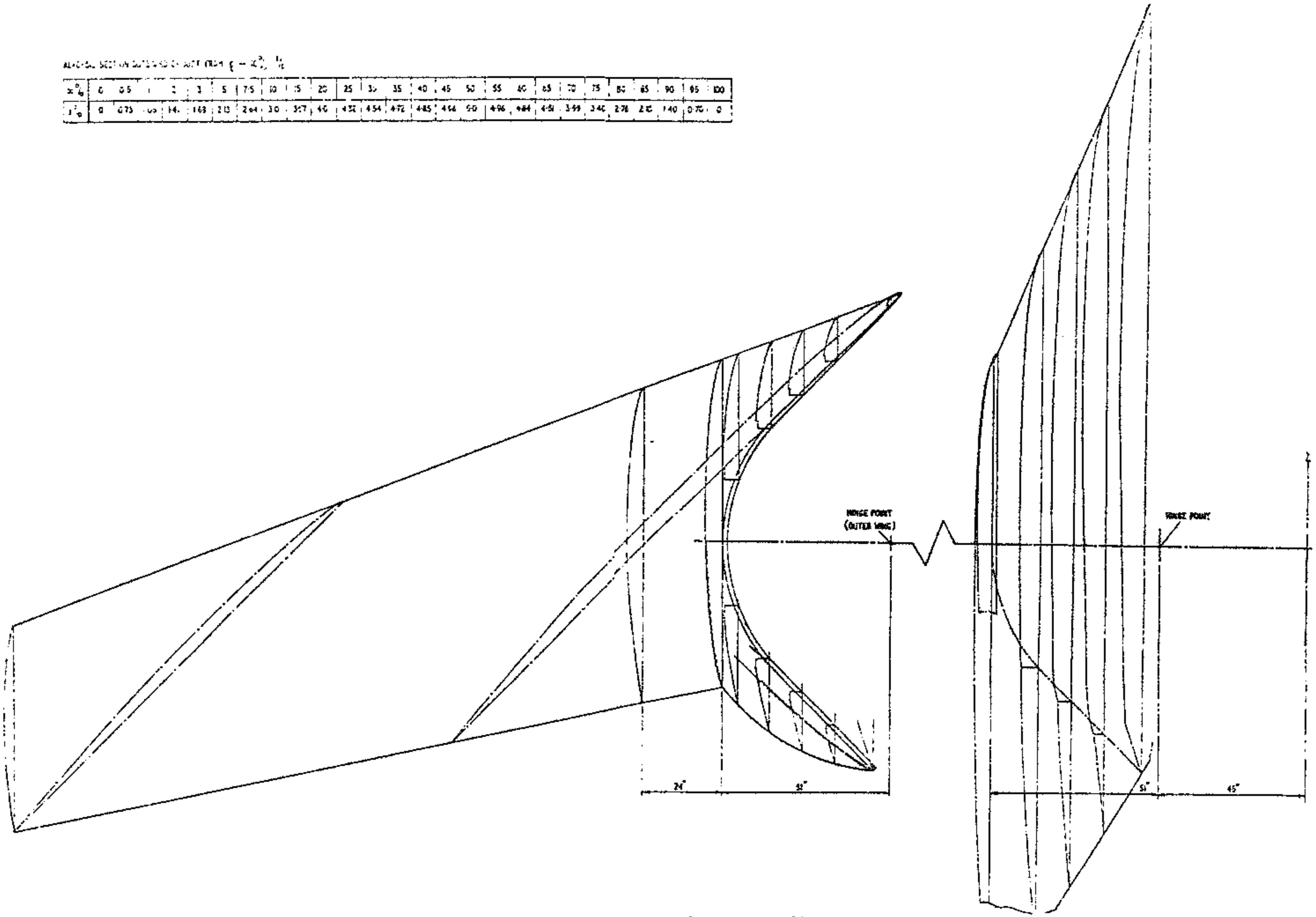


FIG. 6 WING AEROFOIL SECTIONS

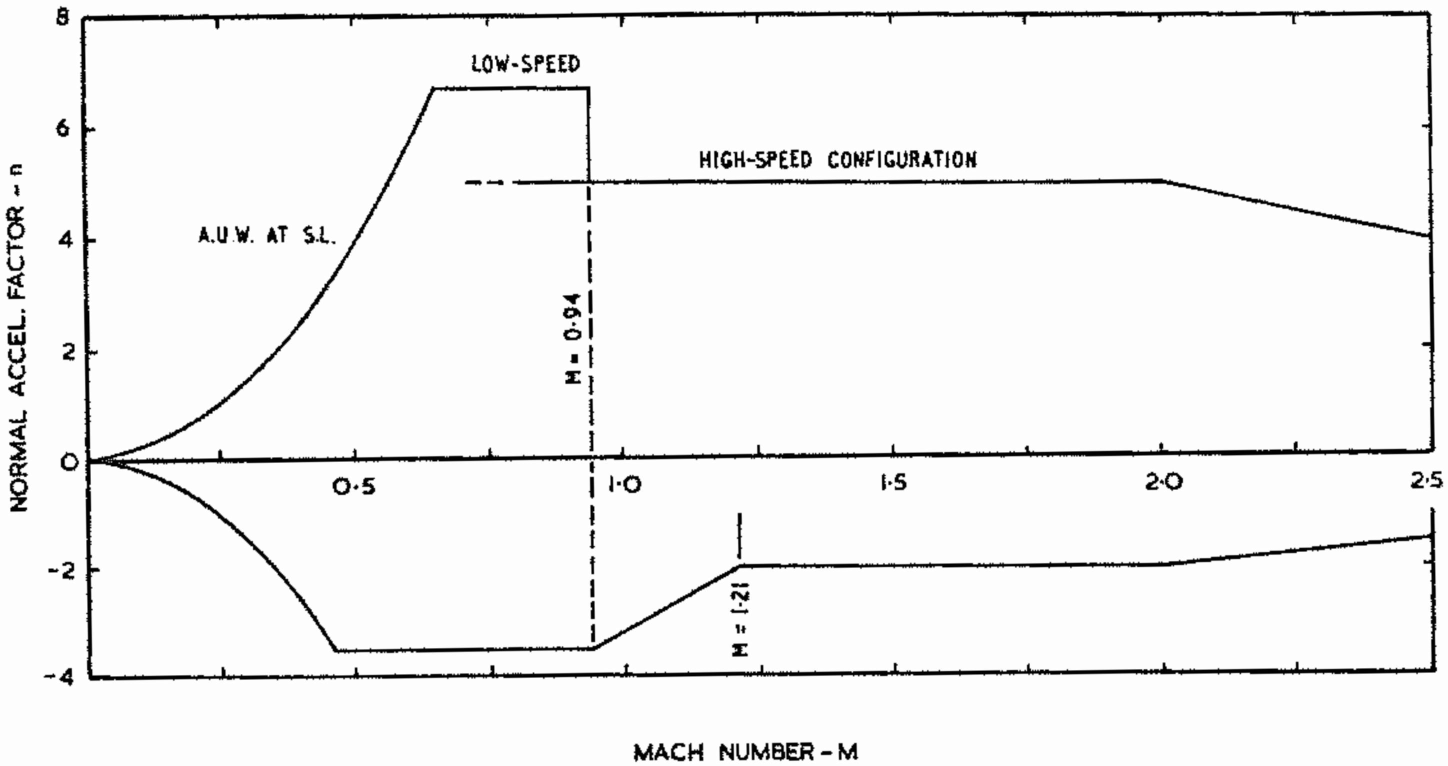


FIG.7 FLIGHT ENVELOPE

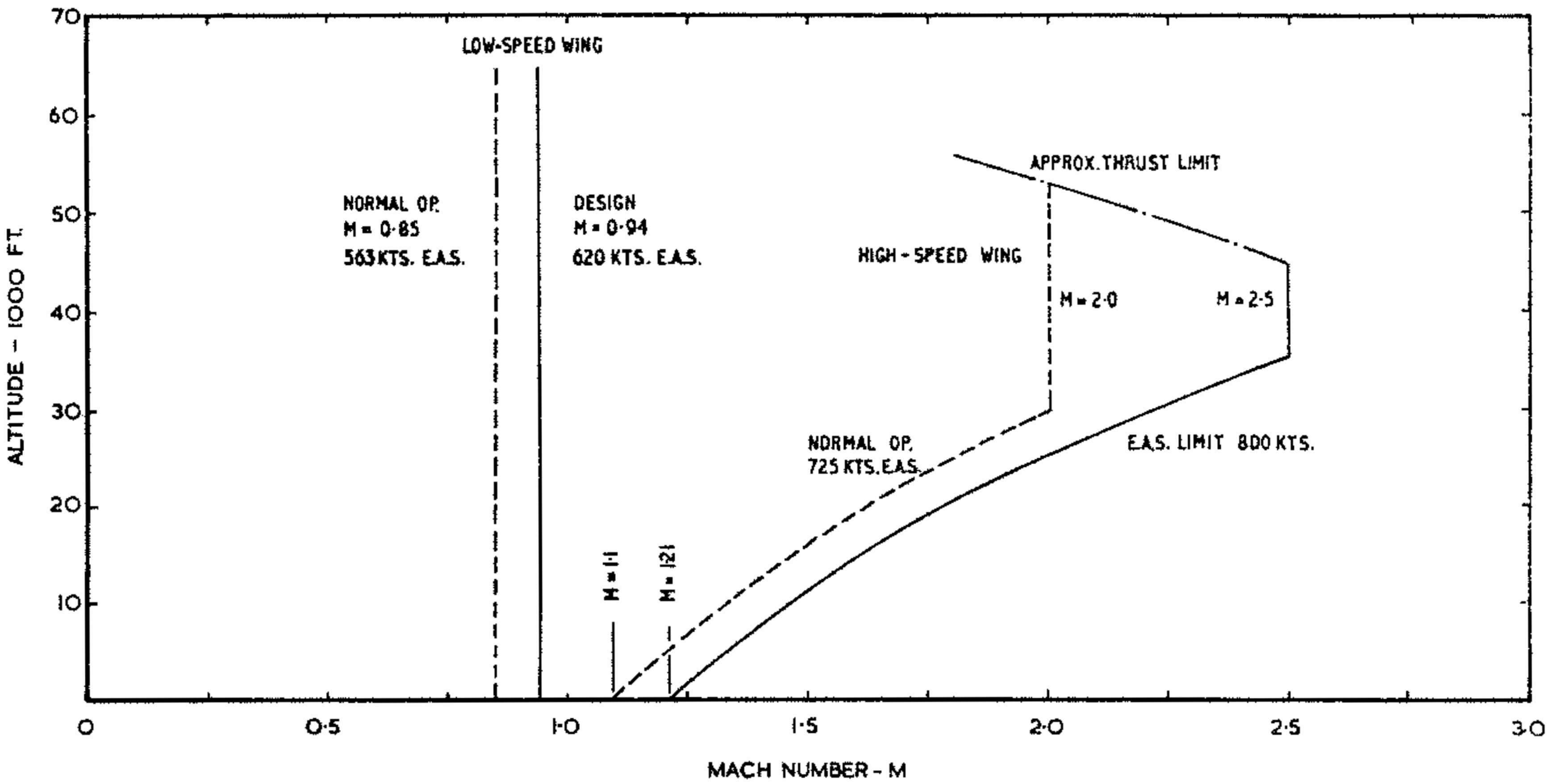


FIG.8 AIRFRAME FLIGHT LIMITATIONS

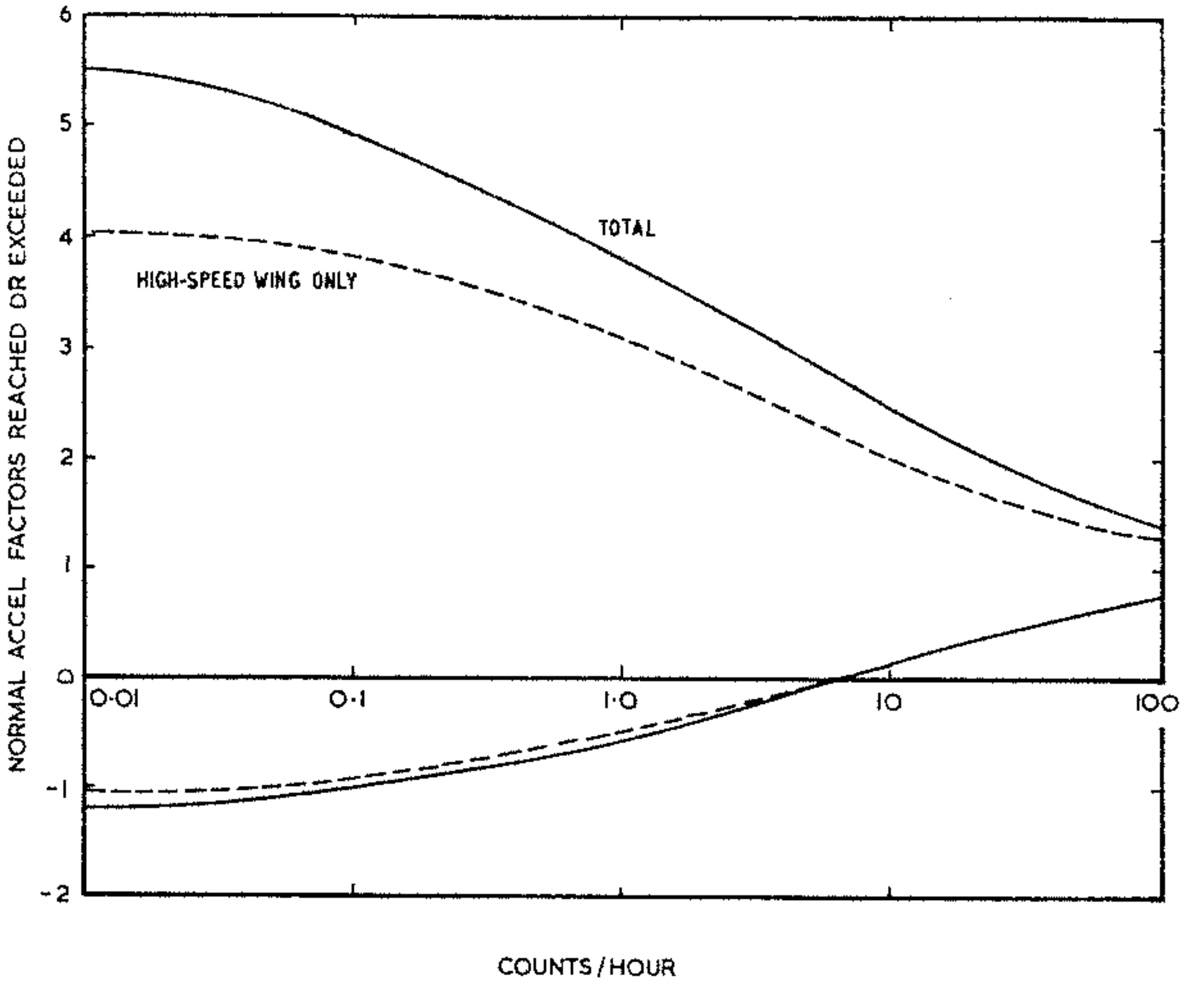


FIG.9 MANOEUVRE LOAD SPECTRUM

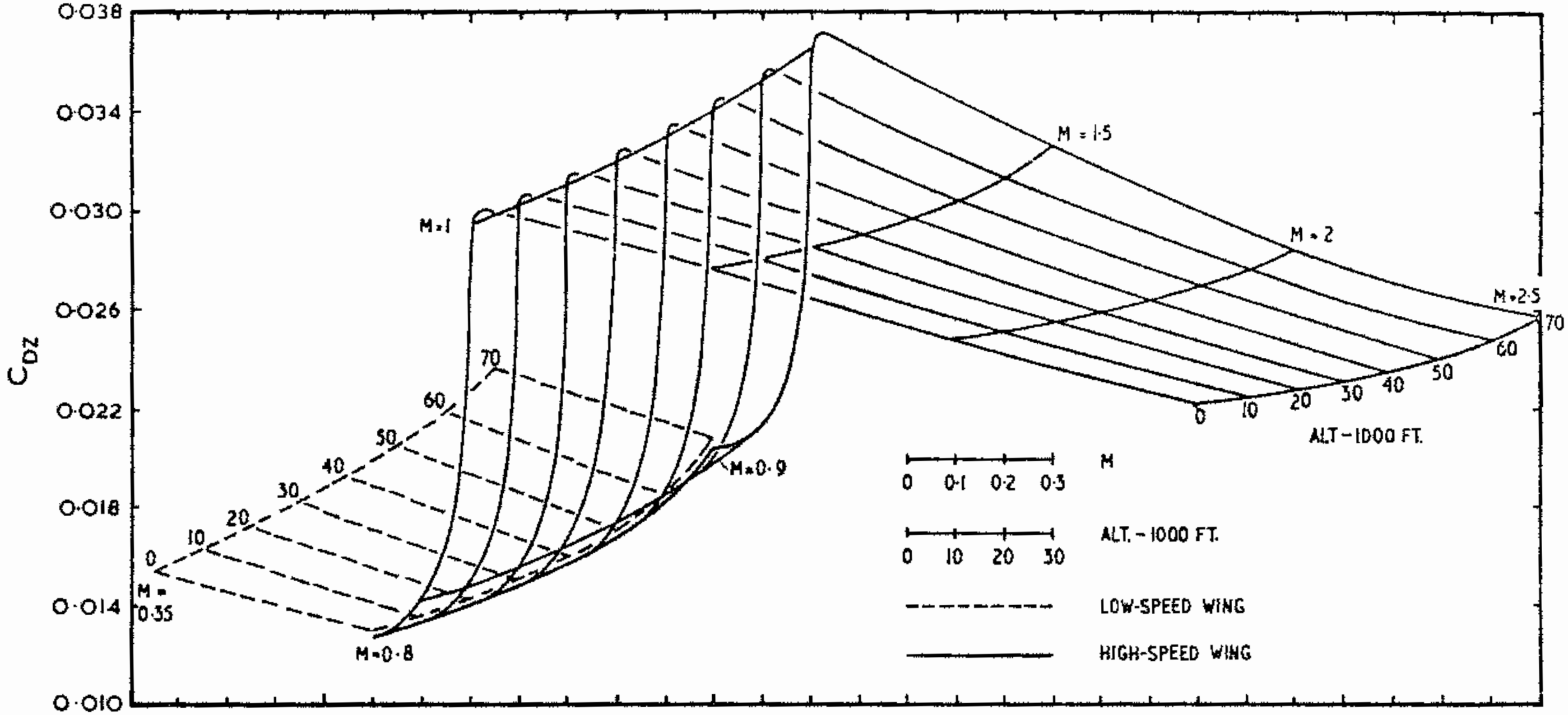


FIG. 10 ZERO LIFT DRAG COEFFICIENTS

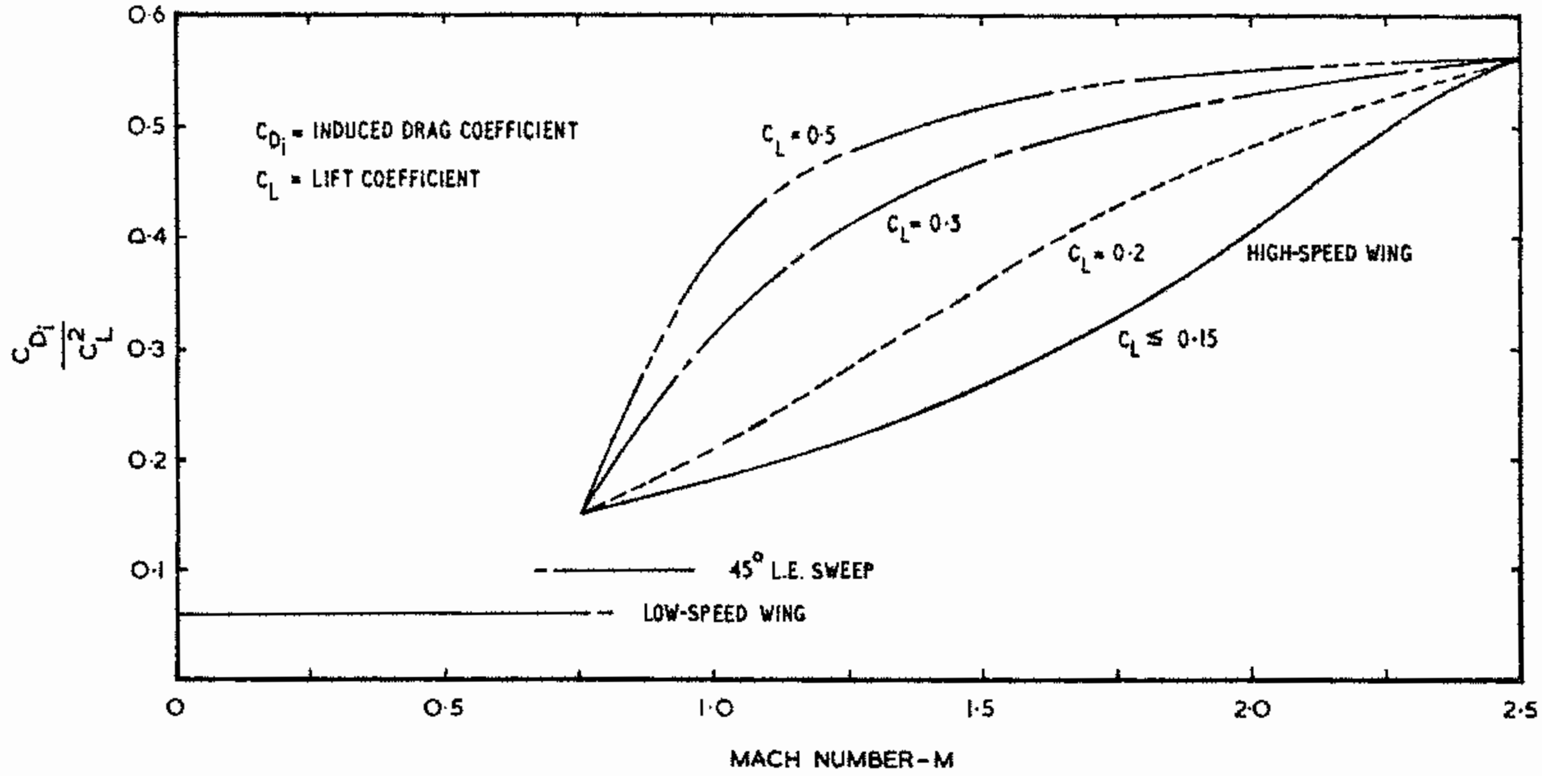


FIG. 11 INDUCED DRAG CHARACTERISTICS

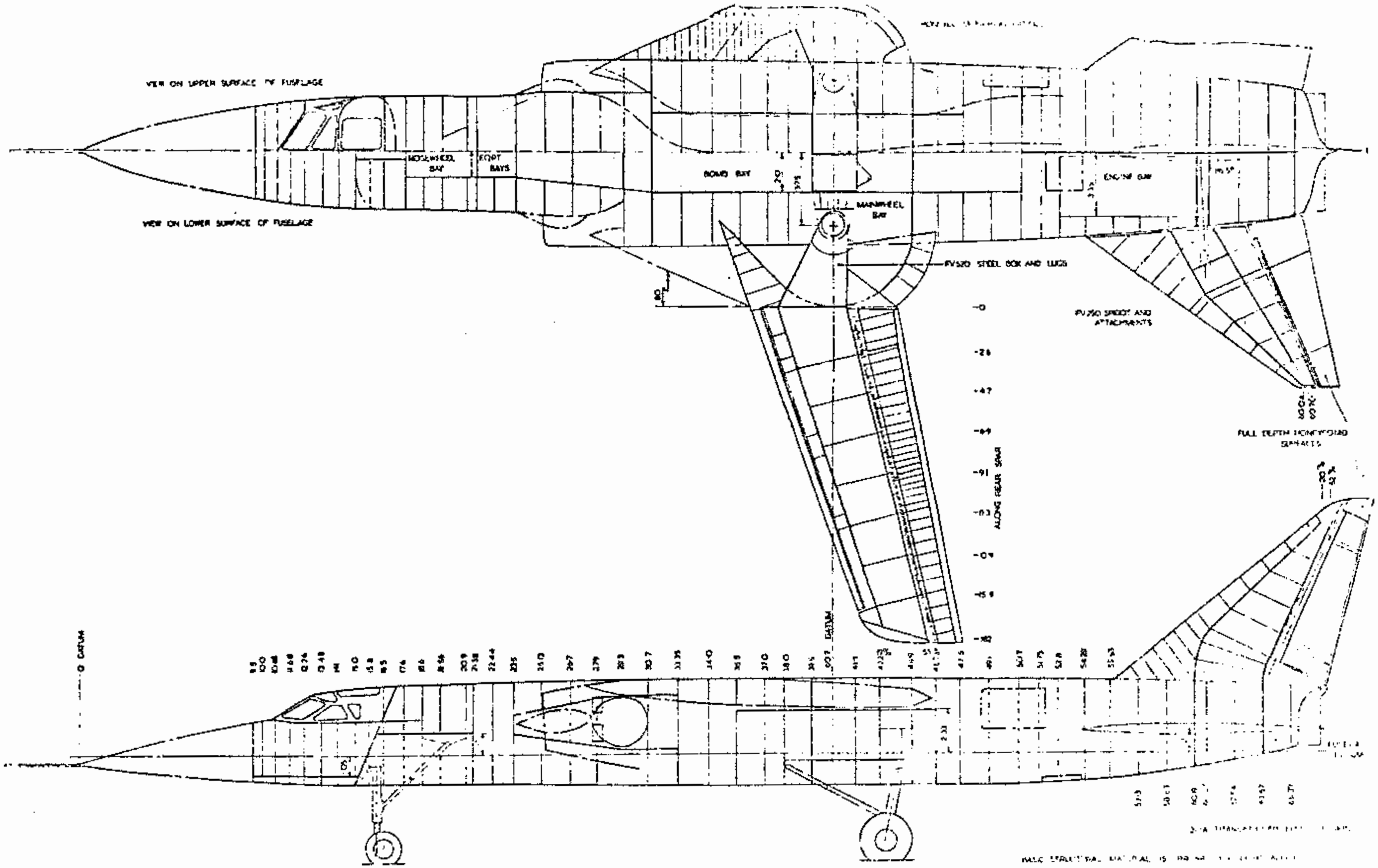


FIG.12 STRUCTURAL LAYOUT

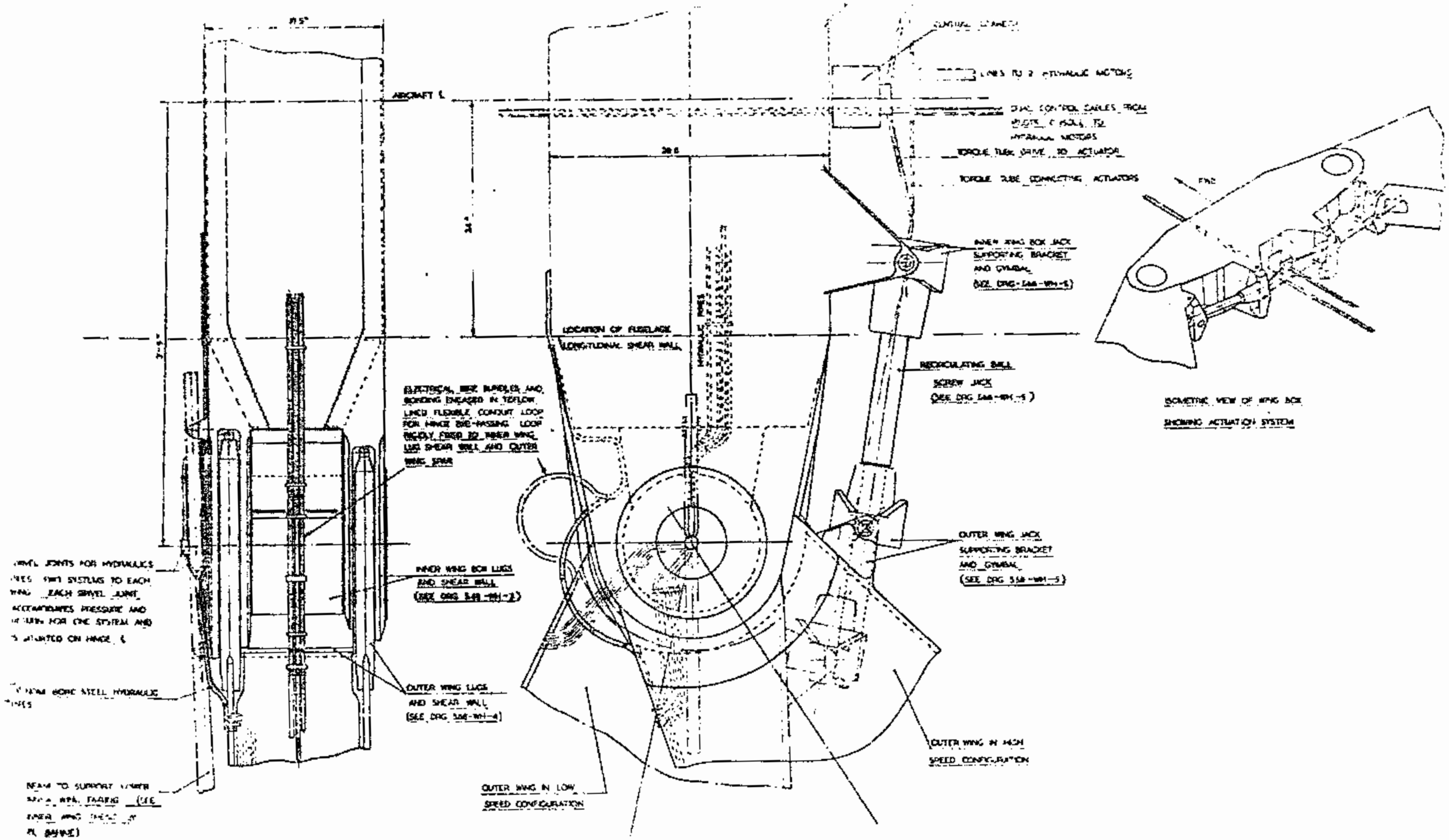
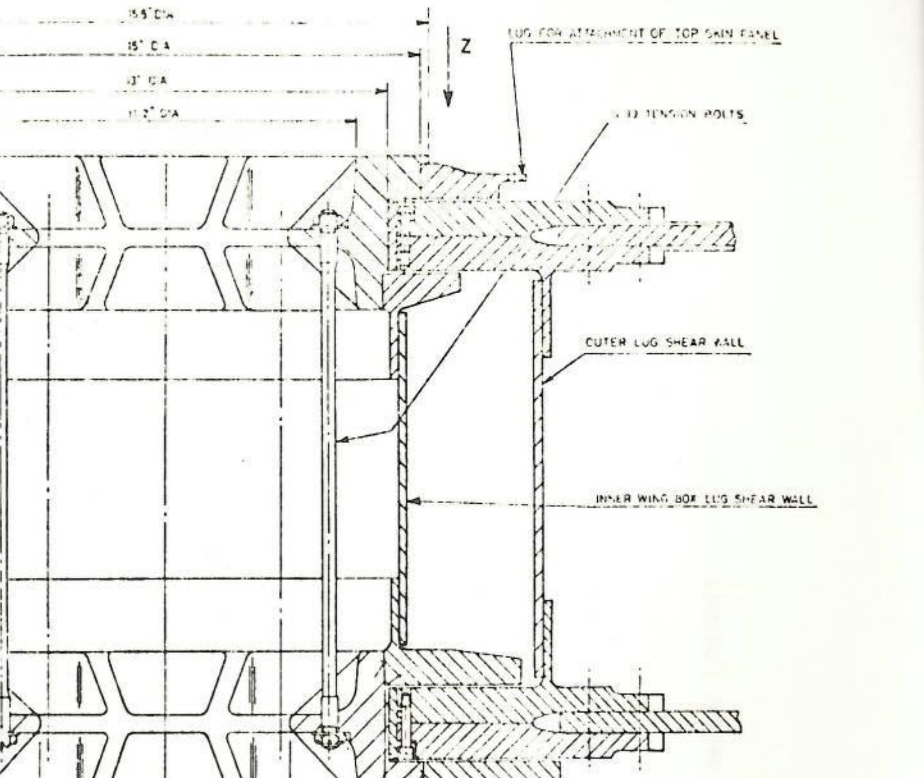


FIG.13 GENERAL ARRANGEMENT OF HINGE SYSTEM



15.5" DIA
15" DIA
13" DIA
11.2" DIA
Z
LUG FOR ATTACHMENT OF TOP SKIN PANEL
TENSION BOLTS
OUTER LUG SHEAR WALL
INNER WING BOX LUG SHEAR WALL

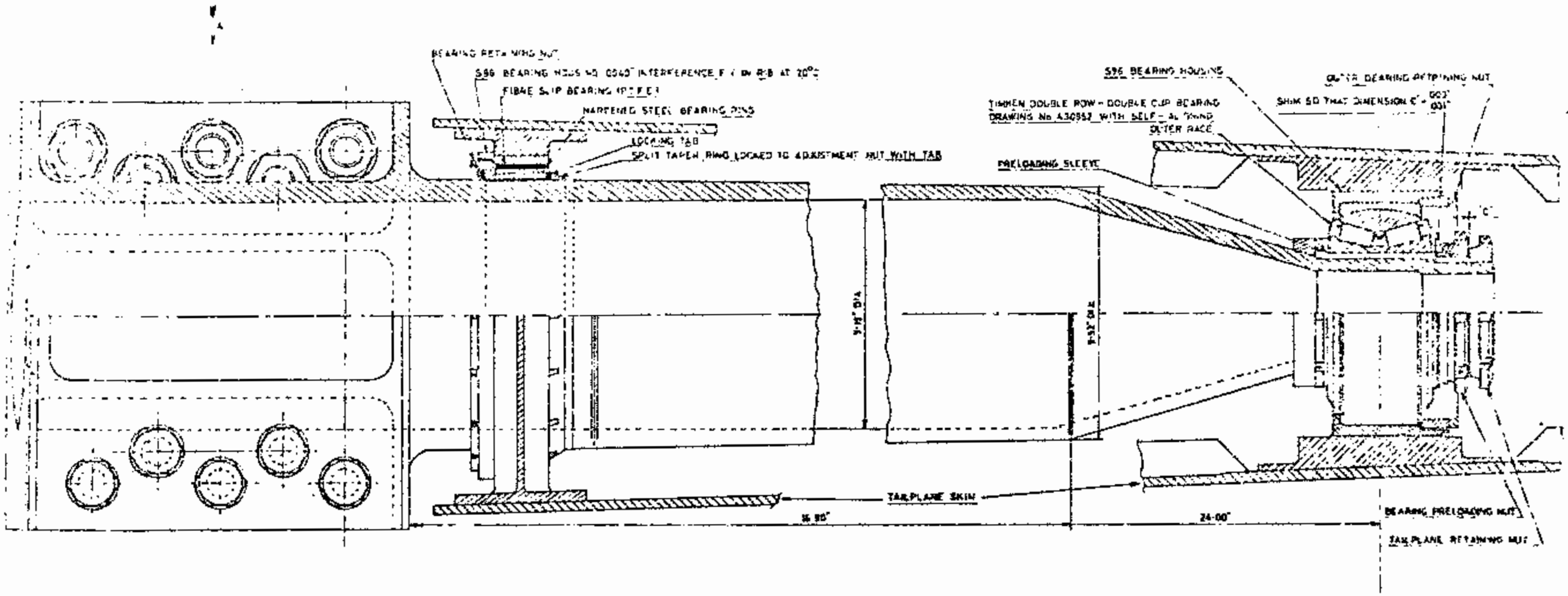


FIG. 15 CROSS SECTION OF TAILERON SPIGOT

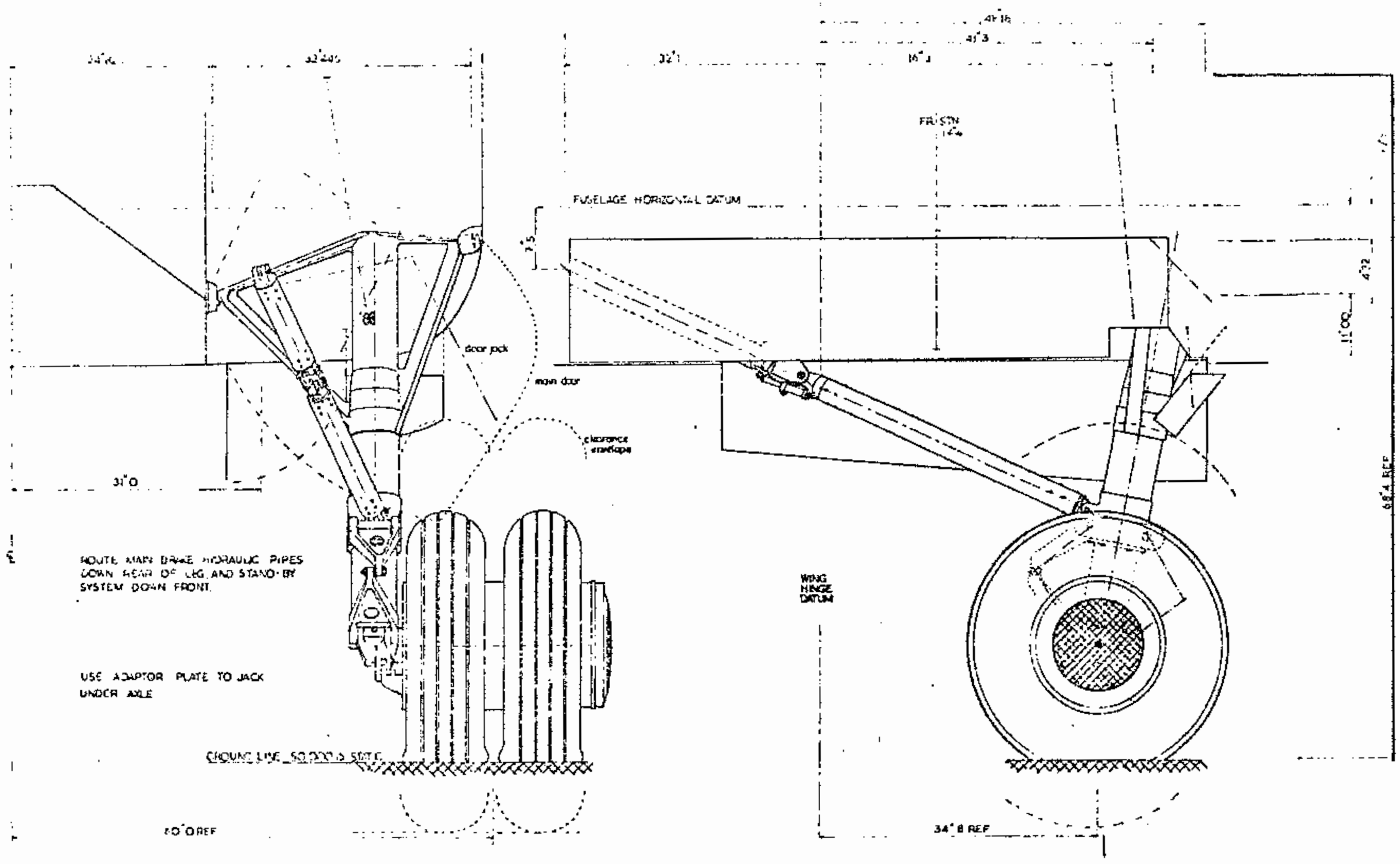
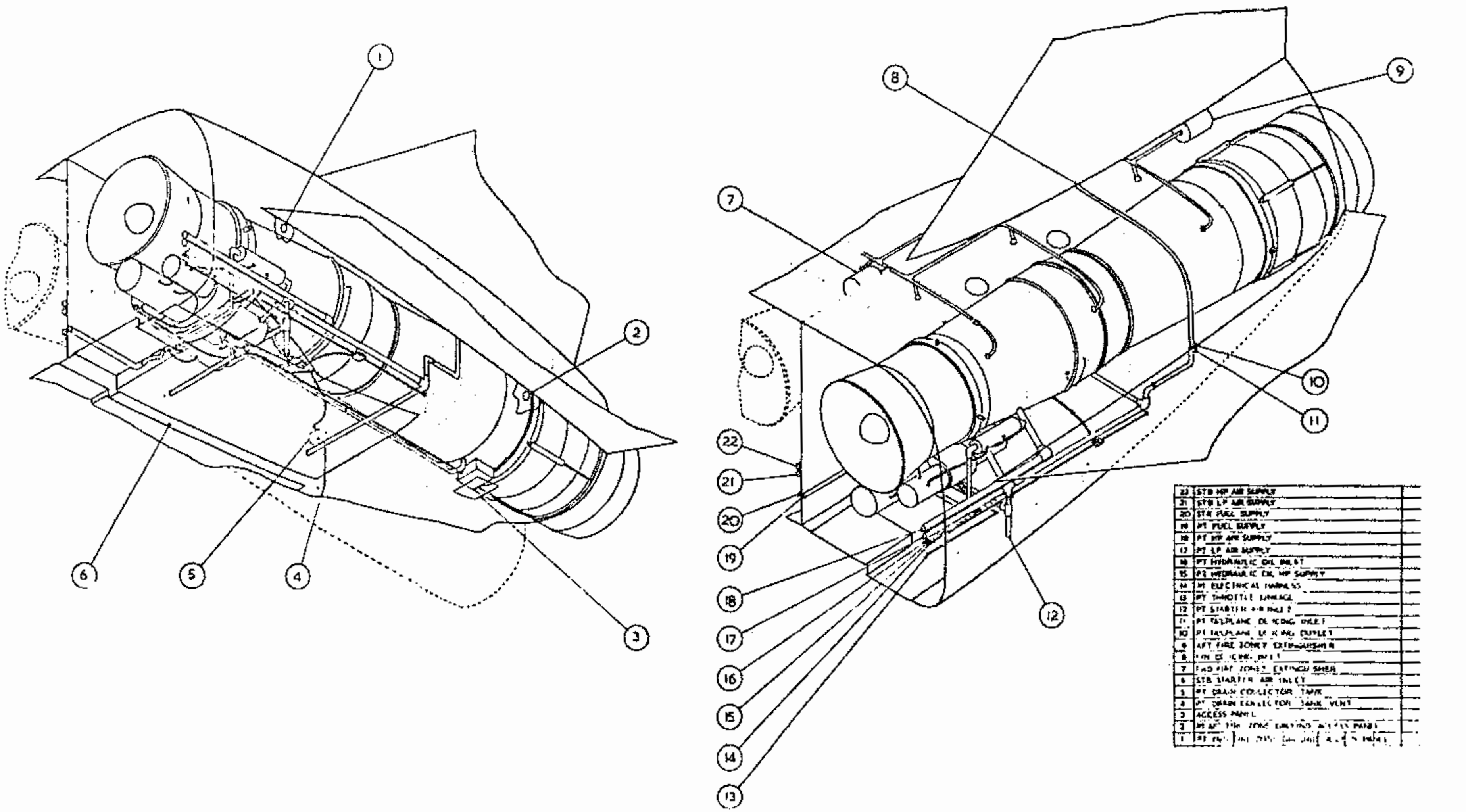
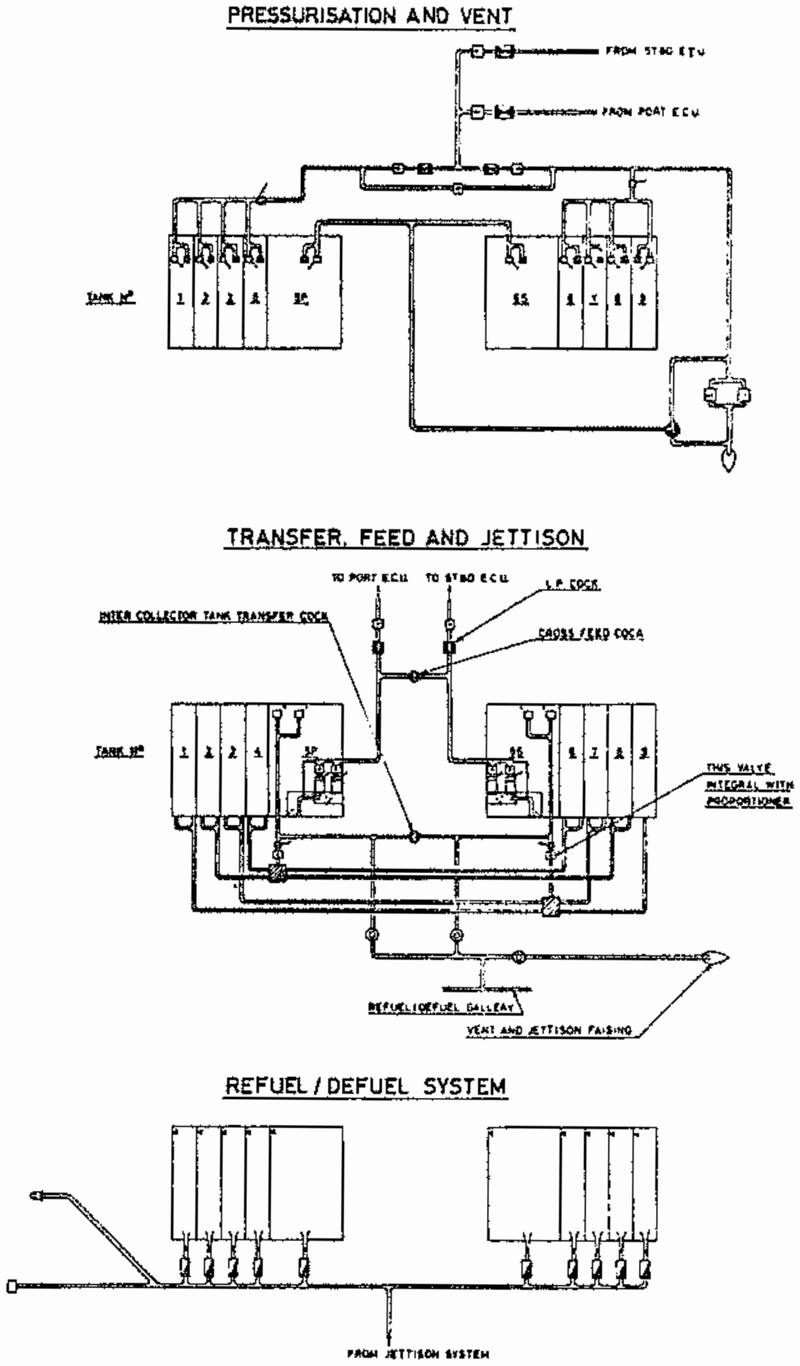


FIG.16 GENERAL ARRANGEMENT OF MAIN UNDERCARRIAGE



22	STB HP AIR SUPPLY
21	STB LP AIR SUPPLY
20	STB FUEL SUPPLY
19	PT FUEL SUPPLY
18	PT HP AIR SUPPLY
17	PT LP AIR SUPPLY
16	PT HYDRAULIC OIL INLET
15	PT HYDRAULIC OIL HP SUPPLY
14	PT ELECTRICAL HARNESS
13	PT THROTTLE LINKAGE
12	PT STARTER AIR INLET
11	PT TAILPLANE DE-ICING INLET
10	PT TAILPLANE DE-ICING OUTLET
9	AFT FIRE ZONES EXTINGUISHER
8	FIN DE-ICING INLET
7	FWD FIRE ZONES EXTINGUISHER
6	STB STARTER AIR INLET
5	PT DRAIN COLLECTOR TANK
4	PT DRAIN COLLECTOR TANK VENT
3	ACCESS PANEL
2	PT AFT FIRE ZONE [illegible] ACCESS PANEL
1	PT [illegible] ACCESS PANEL

FIG. 17 ENGINE INSTALLATION



KEY

- COMBINED REFUEL / DEFUEL VALVE
- RESTRICTOR AND DIFFUSER
- LIQUID LEVEL INDICATOR
- PRESSURE REFUELLING CONNECTION
- IN-FLIGHT REFUELLING CONNECTION
- HYDRAULICALLY DRIVEN PROPORTIONER
- A.C. ENGINE FEED PUMP
- JET PUMP
- ELECTRICALLY OPERATED COCK
- MECHANICALLY OPERATED COCK
- JETTISON VALVE
- FLOAT OPERATED TRANSFER VALVE
- NON RETURN VALVE
- CLACK VALVE
- PRESSURE SWITCH
- FLOWMETER
- FLOAT OPERATED FLAP-TYPE VENT VALVE
- 2-STAGE AIR OPERATED RELIEF VALVE
- OUTWARD VENT VALVE
- INWARD VENT VALVE
- PRESSURE REDUCING VALVE
- 3-WAY COCK

FIG.18 FUEL SYSTEM SCHEMATIC DIAGRAMS