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AIRCRAFT DESIGN STUDIES - COUNTER INSURGENCY AIRCRAFT
WITH SUCTION BOUNDARY LAYER CONTROL

by

D. Howe

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D. Howe, S. M., D. C. Ae., C. Eng., F. I. Mech. E., F. R. Ae. S.

Summary

A design study of a counter insurgency aircraft with a suction boundary layer controlled wing to give high lift has been undertaken. The work was carried out by the students in the Department of Aircraft Design during the 1966 academic year and was intended to provide evidence on the feasibility of the configuration employed.

The aircraft has a gross weight of 9800 lb. and is designed to carry a variety of payloads of up to 2000 lb. at a maximum speed of 380 m.p.h. The flight usable lift coefficient of five is achieved at an incidence of approximately 30° which introduces particular layout and undercarriage problems. A twin boom configuration with a variable geometry undercarriage was adopted.

It is concluded that the use of a suction boundary layer control system can confer significant performance benefits but the aircraft might well be handicapped by climatic operational limitations. The variable geometry undercarriage is complex and an alternative layout using a tilt wing might be preferable.

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1.0 INTRODUCTION

During the past two decades considerable emphasis has been placed on the use of boundary layer control to improve aircraft performance. In particular extensive use has been made of the addition of energy to the boundary layer by blown air as a means of obtaining high lift coefficients at low speeds. The use of suction has also been the subject of a substantial research effort but it has not been applied to any great extent in production aircraft. Whilst one of the possible uses of suction is to maintain laminar flow in the cruise condition and hence improve the cruise lift to drag ratio, an alternative application is at low speed when the separation of the turbulent boundary layer can be delayed and high values of lift coefficient realised. The effectiveness of this latter technique has been demonstrated by a number of converted or specially designed light aircraft. The investigations at Mississippi State University are worthy of special mention in this respect. (Refs. 1 and 2). In the United Kingdom an Auster Mk. 7 aircraft was extensively modified by Marshalls to the requirements of Cambridge University for the purpose of carrying research on aerofoils having distributed suction at high lift coefficients, (Ref. 3). After an initial period of development flying this aircraft, which was known as the MA4, was used to obtain a substantial amount of data on suction aerofoil shapes, pressure distributions and stalling characteristics both with and without the use of a trailing edge flap. Lift coefficients in excess of six were achieved well away from the ground. The lift curve slope is not significantly changed by the use of suction and therefore even when a flap is used a wing angle of attack of the order of 30° is required to obtain these high values of lift coefficient. The MA4 was not designed to be able to operate near to the ground at such high angles and the excellent STOL performance conferred by suction could not be demonstrated. Because of this a decision was taken to consider a possible application for high lift suction boundary layer control and to assess its merits relative to a more conventional design. In addition the peculiar layout problems associated with this type of aircraft could be investigated with the eventual possibility of building a demonstrator aircraft should interest justify it. A demonstrator aircraft would be valuable in establishing the lift characteristics in the proximity of the ground, but for the purpose of the present design study it was necessary to assume that the results obtained from the MA4 work applied to this condition

It is apparent that the most likely application of suction to obtain high lift is in those aircraft where there is little or no air readily available for blowing since one of the particular advantages of suction is the relatively low powers required. (Ref. 4). Thus suction could conceivably find application in a relatively small STOL transport aircraft powered by propeller engines, possibly in conjunction with a tilting wing, (Ref. 4), to obtain the required high angles of attack. This class of aircraft was not selected, however, since it was by no means obvious that the desired performance could not be achieved by more conventional means. A more promising application was considered to be to a counter-insurgency design where the use of any simple and cheap device can confer highly desirable versatility. Several aircraft have been specifically designed for this role in recent years, and of these the North American OV-10A Bronco has seen extensive battlefield service in Vietnam. The CI66 project study was conceived as a very similar design but with the additional use of the suction boundary layer control system. A total of thirteen students were engaged in the study and their individual responsibilities are listed in Appendix A.

2.0 SPECIFICATION

A somewhat arbitrary specification was proposed as a basis for the project investigation. The following requirements were tentatively stipulated:-

- 1) The aircraft should be capable of operating in three distinct roles:-
 - (a) Observation and liaison with two crew members.

- (b) Strike operations. The payload to consist of externally mounted bombs or a standard Hawker Hunter 4 x 30 mm Aden gun pack.
 - (c) Light transport for personnel and freight. Apart from general purpose loads the carriage of four stretcher cases with an attendant or six equipped paratroops must be possible.
- 2) Maximum speed to be as near to 400 m. p. h. as possible.
 - 3) A normal payload of 2000 lb. to be carried over an operational radius of not less than 60 n. miles. The range with the normal fuel tanks full to be not less than 500 n. miles.
 - 4) The aircraft should be capable of taking off from and landing onto a 500 ft. long semiprepared airstrip with 50 ft. height clearance in either case.
 - 5) The design diving speed should be 400 knots E. A. S. and the normal manoeuvre factor 6.

3.0 OVERALL CONFIGURATION OF THE DESIGN

Figure 1 is a general arrangement drawing of the aircraft and Figure 2 is a photograph of a scale model. The predicted weight breakdown is shown in Table 1 and Appendix B lists the design geometry and characteristics. With a design gross weight of 9800 lb. the aircraft has a wing span of 32 ft. and an overall length of 35.6 ft. Power is provided by two Bristol Siddeley Gnome P1200 powerplants, although Turmo IID series engines could also be used.

The twin boom layout used is very similar to that of the OV-10A and is virtually dictated by the freight and personnel carrying requirement. The fuselage arrangement with typical payload is shown in more detail in Figure 3. Two crew members are arranged in tandem and are provided with ejector seats. The volume below the crew seats is used for the nosewheel bay and fuselage fuel tanks and there is space in the nose for a radar unit should this be required. The cockpit is unpressurised. Armour protection is provided for the crew and fuel tanks and this includes a bullet proof windscreen, heavy gauge crew floor and local armour plate on the cockpit bulkheads. The payload bay extends aft of the rear cockpit, below the wing and terminates in a large rear door which is hinged at one side and can be removed completely if paratroop dropping is to be undertaken. The payload bay has a floor length and width of 10.7 ft. and 4.0 ft. respectively and the minimum height is 3.65 ft. A portion of the floor is removed with the lower surface when the gun pack is carried.

The powerplants are located in the nose of each boom as shown in Figure 4. Each one drives a single 11 ft. diameter three blade constant speed propeller. The engines are interconnected by cross shafting which connects to the gas generator output gears, and the propellers are arranged to rotate in opposite directions. The cross coupling is necessary to equalise propeller thrust in the event of a single failure and thereby maintain the aerodynamic symmetry of the aircraft. The fin size required to cater for single propeller flight is prohibitive. The booms are used to house the retracted main undercarriage units and also the rear fuel tanks. Swept back twin fins support a high mounted, variable incidence tailplane. Both the fins and tailplane are provided with conventional surfaces for primary yaw and pitch control.

The wing planform is only slightly tapered and has an aspect ratio of six. This is relatively low for a high lift aircraft and the wing loading of 58 lb/sq. ft. is relatively high. The wing span was dictated partly by high speed drag considerations and partly by the desirability of keeping the greater part of the wing within the slipstream from the two propellers. Both the leading and trailing edges are fitted with

plain flaps, as can be seen from the wing section shown in Figures 5 and 8. The leading edge flaps are necessary to obtain a large upper nose radius of curvature for use with the suction system, thereby reducing the peak suction differential required. The trailing edge flaps are connected with the ailerons which droop 12° for the landing condition. Although the use of suction enables a high lift coefficient to be obtained without the use of trailing edge flaps they are advantageous in that their effect is to reduce the angle of incidence at which it is achieved. They have the further merit of improving the landing performance when suction is not in use. Suction is applied over the whole of the upper wing surface, including the leading edge and trailing edge flaps and ailerons. Suction intensity is graded according to the local chordwise pressure levels.

The overall normal maximum flight lift coefficient of five is achieved at approximately 30° wing incidence, and the associated ground clearance requires the upswept booms, swept fins and high mounted tailplane. The wing lift curve slopes for various combinations of flaps, slipstream and suction are given in Figure 6. A further complication which arises due to the high ground attitude angles is the effective fore and aft movement of the centre of gravity as the aircraft rotates. This dictates the use of a variable geometry main undercarriage which is arranged so that the wheels move longitudinally to compensate for the motions of the centre of gravity. Each of the twin wheel main undercarriage units consists of an inclined shock strut and a drag strut which is attached to the axle thereby functioning as a radius rod.

4.0 ESTIMATED PERFORMANCE

The estimated variation of range with payload carried is shown in Figure 7. Normal cruising speed at sea level is 300 knots, 346 m.p.h., whilst at 15000 ft. altitude the true cruising speed is 308 knots, 354 m.p.h. Provision of an internal tank enables the normal maximum fuel load of 2000 lb. to be increased by 700 lb. and in this case a ferry range of 860 n.miles is achieved, the flight being at 20,000 ft. altitude.

The maximum level speed at sea level is 310 knots, 357 m.p.h. and at 15,000 ft. altitude the corresponding true airspeed is 330 knots, 380 m.p.h.

When the aircraft is climbing at 9800 lb. and at 150 knots the rate of climb varies from 4600 ft/min at sea level to 2150 ft/min at 20,000 ft. altitude.

The length of the take off run is sensitive to the technique used and the ground surface conditions. As designed the aircraft uses the main powerplants to provide the suction power directly and this results in a take off power loss of some 12%. Making allowances for this and assuming a dry, smooth runway surface the aircraft can take off to 50 ft. height in a ground distance of 490 ft. Of this the ground run is some 360 ft. and the aircraft climbs away at 70 knots forward speed.

Landing performance is also difficult to define precisely. With a normal braking deceleration of 11 ft/sec. after touchdown from an approach of 60 knots the landing distance from 50 ft. is some 750 ft., of which 450 ft. is required to bring the aircraft to rest after touchdown. The use of reverse thrust enables this distance to be reduced by some 200 ft.

5.0 DESCRIPTION OF DESIGN DETAILS

5.1 Aircraft Systems and Installations

5.1.1 Powerplant Installation

A Gnome P1200 powerplant is mounted in the nose of each boom. This particular version of the Gnome was not developed but is essentially a standard H1200 gas generator and gearbox with an additional propeller drive and reduction gearbox located above the basic engine. The form of the layout can be seen in Figure 8. The

Turbomeca Turmo IID engine is very similar and can be considered as an alternative power unit.

Each engine is mounted directly from the primary boom structure at four points. There are a pair of horizontal trunnion fittings on the main, rear gearbox, and a pair of suspension points on the forward, propeller reduction box. These front points incorporate swinging links to cater for fore and aft engine expansion and do not react loads in this direction. One of them is Y shaped and together with the corresponding rear trunnion fitting reacts side loads. The links are suspended from a stiffened boom frame and the trunnions are mounted in shaft housings which are located on a pair of wing ribs coincident with the sides of the boom. All the aircraft fittings are light alloy forgings in L65.

All the engine auxiliaries are located beneath the gas generator aft of the intake region. Access to these is obtained by opening the two large cowling doors which effectively form the lower surface of the boom between the air intake and wing. The doors are hinged longitudinally along a sloping coaming member and meet on the lower centreline. A removable panel is located in the top of the boom above the rear gearbox and mounting trunnions. The air intake is attached to the front face of the engine. Engine removal is basically forward and down, complete with the intake, but a small initial upward motion is necessary to enable the exhaust to clear the structure unless this has been previously disconnected.

The rear engine gearbox incorporates an output shaft and use is made of this for the engine interconnection. The starboard engine gearbox has an extra gear in the propeller drive train to reverse the rotation. Engine interconnection is made by three secondary bevel gearboxes and two split shafts as shown in Figure 10, the cross shaft speed being reduced to two thirds of that of the output shaft by the bevel gears. Each of the three gearboxes is basically similar but varies in detail assembly. The centrebox has a straight through shaft for the main cross connection and uses the bevel drive for the auxiliary power take off to the pneumatic system pump and generator. The shafting is designed to transmit 60% of the maximum power of one engine with a short life rating. Each shaft is made in two sections, partly for purposes of assembly and partly to enable a centre bearing to be used to prevent whirling. Universal and sliding joints are provided. The shafts are 2.5 inches diameter by 0.095 inches thick T60 steel tube.

5.1.2 Suction System and Air Intake

It is essential that the suction be obtained with both a minimum power and weight penalty. Various suction pumps were considered for use in the C166:

a) Separate suction gas turbine.

This would be a bulky unit and hence create installation problems within the closely constrained aircraft layout. It would also be relatively heavy and have operational problems such as in-flight starting.

b) Use of propulsion engines.

The use of the main propulsion engines to provide suction is particularly attractive since there should be little direct weight or volume penalty. On the other hand the depression caused in the engine intake can seriously impair the engine performance and this could result in an indirect weight penalty.

c) Mechanically driven suction pump.

The existence of the engine cross shafting and centre gearbox makes the installation of a mechanically driven pump relatively straightforward. Space for the pump is still a problem although this is less severe than in the case of the gas turbine, and the same trend is true of the weight penalty.

It was envisaged that the suction differential would be of the order of 2 p.s.i., but that the volume flows would be quite large. In view of this the use of the main propulsion engines seemed to be particularly attractive and accordingly the design was based on this premise. The basic concept was that the engine air intake should be designed to act either wholly or partly as a plenum chamber at low speed, but be able to make use of the beneficial ram effects at high speed.

The suction system was designed to enable a lift coefficient of six to be obtained in both the take off and landing conditions and the suction differentials and mass flows estimated to be necessary are shown in Table 2. As can be seen the most severe case arises during take off at the highest lift coefficient, when the required suction differential is 2.7 p.s.i. This is equivalent to the ambient air pressure at 5500 ft. altitude and implies an engine power loss of some 12% relative to sea level standard conditions. The corresponding mass flow of 0.18 slugs/sec is equivalent to a mass flow coefficient of approximately 0.005. It should be noted that all the figures of Table 2 are based on the assumption of zero slipstream effect. There is little evidence of the real effect of the slipstream but calculations indicated that it reduces the differential somewhat and increases the mass flow by some 25% due to the higher surface air velocities.

The aircraft has a high installed power to weight ratio which is determined by high speed flight requirements and hence the loss of take off power due to the intake depression is less significant than would otherwise be the case. However it does amount to an approximately 10% increase in the take off run and is obviously undesirable. In an attempt to partially overcome this defect it was decided to investigate the use of a venturi in the intake. This would enable the air pressure at the compressor to be restored to very nearly the true ambient value, with total intake losses of no more than 10% provided the suction mass flow is small in comparison with the gross engine throughput. At maximum power each engine has a mass flow of approximately 14 lb/sec, 0.435 slugs/sec. Thus the total suction flow amounts to some 25% of the total engine flow at take off. It is very doubtful if the venturi system could be designed to enable sufficiently high recovery pressures to be achieved to make it worthwhile in these circumstances. There is, however, a further possibility. The suction requirements can be divided into low and high pressure components. The latter occur over the leading edges of the wing and flaps and account for some 65% of the total mass flow. If a separate suction pump, say mechanically driven, were to be provided for this then the main engines could reasonably cope with the remaining 35%, low pressure flow. In this case it would represent only 8% of the engine mass flow and the venturi concept could be used effectively with very little weight or power penalty. Such a venturi requires a throat area which is about half that of the intake and is shown in outline in Figure 8. The mechanical suction pump installation has not been designed since it was decided to accept the power loss and use the intake to provide all the suction flow. Whether the venturi is used or not it is necessary to provide a means of compensating the intake for the different suction demands at take off and landing. This is done by inserting a wire screen in the intake. It is arranged to act as a door and rotates about a vertical axis to come into effect during landing. The 0.25 inches diameter holes in the screen are located at 0.4 inches pitch and pass 9 lb/sec of air with a pressure drop of 100 lb/sq.ft at sea level.

The suction distribution from each engine is by two primary 5 inches diameter ducts which connect the intake to the main wing box, as is shown in Figure 8. The box acts as a balancing duct of substantial volume and extends over the whole span. Shut off valves are placed in the primary ducts. Secondary flexible ducts of approximately 2 inches diameter are located both forward and aft of the wing box and connect to the leading and trailing edge flaps and ailerons, as can be seen in Figure 9. The local suction distribution to the porous skins consists of a series of spanwise top hat section cells which are divided into 4 inch lengths. High density suction regions are located over the top surfaces of the leading edge flap and the noses of the trailing edge flaps and ailerons. Elsewhere across the chord the suction is of lower density and is confined to local spanwise strips. A typical chordwise arrangement is shown in the wing

cross section in Figure 8. Each of the short spanwise cells incorporates a non return valve to prevent outflow in high speed flight. A typical valve and cell arrangement is illustrated in Figure 9. The cells are made from injection mouldings in polypropylene and the valves are of neoprene rubber reinforced with stainless steel where necessary. The cells are bonded to the predrilled porous skins which consist of staggered rows of 0.05 inches diameter holes placed at 0.10 inches pitch.

In the event of an engine failure the remaining engine is able to remove air from the whole wing because of the large volume wing box. However the suction mass flow is less than desirable and the achievable lift coefficient is restricted. Had a separate mechanical pump been provided for the high differential suction it would have been possible to maintain a substantially complete suction distribution. This additional element of safety is a further point in favour of the alternative system.

5.1.3. Fuel System

The fuel system has a capacity for 2000 lb of kerosene. There are five main and two collector tanks. The fuselage fuel bay contains three tanks which are located side by side and are inserted through a central cutout in the fuselage lower surface. Their total capacity is 1000 lbs and the side tanks gravity feed into the centre one through non-return valves. Each of the main boom tanks has a capacity of 400 lb and gravity feeds into the small 100 lb collector tank located immediately behind the engine. It is positioned above the main undercarriage and is installed through a removable panel in the roof of the bay. All the tanks are of welded construction in 22G magnesium alloy. They are covered by a rubberised fabric, FPT/446/LS, to minimise the effect of small arms strikes. In addition there is a degree of armour plating above and round the sides of the fuselage tanks. The tanks are located on their bearers by webbing straps. Provision is made for overwing refuelling. Duplicated electrically driven transfer pumps are positioned in both the fuselage centre and collector tanks with arrangements for engine cross feeding. The collector tanks contain sufficient fuel for eight minutes of flight at full power and internal recuperator chambers are included for inverted flight. Each tank is separately vented and fuel content is measured by a capacitor system.

The fuselage tanks are only used when long range is required or when the payload centre of gravity is aft. In the former case they are used first. It is not essential to use the fuselage tanks in the strike or liaison roles.

5.1.4 Power Operation

The aircraft is provided with a generator driven off the central gearbox for supplying the general electrical and radio services. The tailplane incidence is adjusted for landing by means of an electrically driven screw jack.

Power operation is restricted to the flaps, undercarriage and canopy. Air motors are employed for driving the flaps and also for prespinning the wheels. Pneumatic rams are used for main and nose undercarriage retraction and downlocks and for the canopy opening. The air pump is also driven from the centre gearbox.

The mainwheel brakes are operated through a self contained hydraulic circuit.

5.2 Structural and Mechanical Components

Figure 11 is a key diagram of the structural members. The greater part of the construction is of light alloy using L72 sheet and L65 extrusions and forgings.

5.2.1 Wing Structure

The critical wing loading case is the 6g pullout when the factored maximum shear

forces and bending moments amount to 30,750 lb and 132,000 lb ft respectively. The wing is of three spar construction, the vertical webs being located at 15%, 40% and 61% of the chord. Whilst both the centre and rear spars are continuous from one tip to the other, the engine installation requirements dictate that the front spar be discontinuous in the way of the boom. A degree of continuity is provided by arched boom frames located across the ends of the spar. The top boom structure between the spars is built with the wing, and the wing is attached to the remainder of the boom by bolting around the side ribs and top frames. A non-buckling design is used for the inboard portions of the front and rear spar web and the thickness between the booms is 12G whilst outboard it is 17G. The centre web is of the tension field type and is of 16G material. The whole of the main structural box is used as the primary suction duct and equalising chamber.

The wing skins are made as spanwise planks joined on the spar edge members. Inboard of the booms the skin is of 12G and L71 material and is chemically etched locally. This skin thickness was determined partly by the stress concentration effect of the porous skin design. Outboard of the booms the loads are much less and the skin is 22G in L72. The spanwise stringer arrangement is a combination of zed and top hat sections, the latter being used as local suction ducts. The mean pitch on the top surface is 3.3 inches and on the lower it is 4.4 inches. The zed section stringers are 1.4 inches deep inboard which decreases to 1.0 inches outboard and the corresponding thicknesses are 16G and 22G. The maximum design stress of the inboard skin-stringer panels is 34,000 p.s.i. and the steady level flight design stress is 3750 p.s.i. Pressed channel section ribs placed at approximately 16 inches pitch inboard and 13 inches pitch outboard support the skin panels.

The wing to fuselage joint is made at the front and rear spars only. The two front spar pickups have single pins located in the lateral direction to transmit both vertical and drag loads whilst the pair of rear spar attachments uses fore and aft pins to transmit the vertical and side loads. Forged L65 brackets attached to the forward face of the front spar mate with the corresponding fuselage fittings and forged bosses are employed for the rear joints.

Leading Edge Flaps and Shrouds

The leading edge flap is designed by the loads which occur during high speed flight. For example the inboard flap sections each have to withstand a factored air load of 6850 lb at the design diving speed compared with 5130 lb when they are extended at low speed. The corresponding factored hinge moment at high speed is 1300 lb ft. A typical cross section is shown in Figure 9. As can be seen the flaps are of corrugated construction, the upper surface corrugations being used as the high density suction ducts. Each of the four flap sections is hinged at two points. They are operated by a single air motor which drives through leavers and cables to a screw jack at each hinge. The skin thickness is 24G and the pressed spar is 20G thick. Light supporting riblets are located at 9 inches pitch inboard of the booms and 12 inches pitch outboard.

The leading edge top shroud is also corrugated for suction purposes but the lower shroud is a simply stiffened 24G sheet. A spring steel rubbing strip is provided to seal the top shroud to flap gap.

Trailing Edge Flaps and Shrouds

The two trailing edge flap sections are located between the fuselage and booms. Each has two hinges and they are driven by an air motor. The drive uses gearboxes and rods to the screw jacks positioned at each hinge. The design loads occur when the flaps are deflected to the take off position of 30° and the aircraft encounters a 25 ft/sec gust, with the suction off. The factored load in each section is then 5120 lb and the hinge moment is 2550 lb ft. For landing the flaps are deflected to 60°. A two cell construction is used with light pressed channel section ribs placed at 12 inches pitch. A

typical cross section is shown in Figure 9. The spar and nose skin are of 24G whilst the rear skins are 22G and 20G thick on the upper and lower surfaces respectively. Light intercostal angles are used to stabilise the skins. Bonded construction is used. The hinge brackets are machined from L65 extrusions.

Other details including the shroud and seal design, are similar to the leading edge flaps.

Ailerons

The ailerons have an internally sealed balance together with spring and trim tabs. The inboard of the two hinges on each aileron is located at the side of the boom and consists of a hollow shaft. It is used both for operating the surface and as a suction duct. A basically single cell construction is employed with 24G skins and a 20G spar. The spanwise top hat section suction ducts are used to stabilise the skins in conjunction with pressed ribs placed at the same pitch as on the main wing. Mass balance is located along the nose.

The ailerons are drooped through an angle of 12° when the trailing edge flaps are fully deflected to 60° . This is achieved through a double bell crank lever mechanism which is illustrated in Figure 12.

5.2.2 Fuselage

The fuselage is relatively lightly loaded since the main purpose of the structure is to transmit nose undercarriage and local inertia loads to the wing spars. The maximum factored vertical shear force and bending moment of 18300 lb and 142,000 lb ft. respectively occur in a 6g pullout manoeuvre.

There are a large number of cutouts in both the upper and lower surfaces of the fuselage. At the top these are for the cockpits and wing whilst at the bottom they are for the nose undercarriage, fuel tanks and gun pack. In view of this the construction is based on a simple four boom arrangement with the skins supported by closely spaced frames. The skins are 18g thick and are designed to buckle elastically at 65 per cent of the proof loading. The top longerons are of 16g drawn lipped angle section and are coincident with the sills in the region of the cockpits. The bottom longerons are built up from a drawn zed section which is connected back to the skins by a wide drawn angle to form a box. A pitch of 10 inches is used for the frames which are 20g pressed channels varying in depth from 2 to 3 inches. Complete bulkheads are provided at the ends of the nosewheel bay and at the rear of each of the cockpits. The nosewheel bay is completed by fore and aft vertical webs which react the drag loads. Two fore and aft channel sections support the crew floor which is 10g thick to give a measure of armour protection. The payload floor is supported by 18g channels and consists of a 0.5 inches deep honeycomb core sandwich with 20g faceplates.

The front spar frame is open at the bottom because of the gunpack installation. It consists of separate top and side light alloy forgings which are bolted together. The two 1.125 inches diameter front wing pick up pins are located in forged fittings which extend aft along the payload bay roof to the next frame. The rear spar frame has a top forging in L65 but the sides and bottom are fabricated 15g channel members in L73. Split conical housings are used for the two rear wing pick ups. The complete wing to fuselage attachment is designed so that 80 per cent of the ultimate load can be carried with any one of the four points failed. The gunpack is mounted off the lower longerons by four forged fixing brackets. When it is not in use the fuselage surface and payload floor is completed by means of a panel which is attached to the gunpack mounting points. The rear freight door is hinged off the longerons on the port side of the fuselage. It is of fibreglass construction and can be removed completely for paratroop operations.

The pilot's windscreen is of substantial thickness and is designed to be both bullet and bird impact proof. It consists of two acrylic panels of 0.75 inches thickness separated by a 0.15 inches air gap. The canopy is of similar construction but uses 0.25 inches thick material. It is opened by an upwards movement, the rear edge being constrained to run in a near vertical direction by rollers and tracks. This is shown in Figure 12. Pneumatic rams are provided to assist the crew in opening the canopy. For emergency ejection conditions the complete canopy is jettisoned. This is achieved by using explosive bolts to release the rollers from the ends of the tracks.

5.2.3 Booms

The booms are designed by a combination of fin, tailplane and main undercarriage loads. A high drag landing gives rise to the critical vertical shear force of 17400 lb whilst the factored vertical and lateral bending moments of 107,000 lb ft and 44,000 lb ft arise in pitching and yawing manoeuvres respectively. Powerplant loads have a local effect on the forward boom structure.

A semi-monocoque construction is employed with a basic skin thickness of 20g supported by 18g zed section stringers placed at 3.5 inches pitch. The skin thickness is increased to 18g locally around the main undercarriage bay and fin attachment. The skin panels are designed to remain unbuckled during a trimmed landing with a normal acceleration factor of two. The stringers at the edges of the undercarriage bay cutout are of top hat section and form a closed box with the skin. Intermediate frames are 2.5 inches deep channel pressings in 16g and are located at 10 to 11 inches pitch. They are cut to allow the stringers to pass them. The structure is open at the bottom in the region forward of the wing contrespar to give access to the engine. Over the length of the cutout there are four 20g frames placed 14 inches apart and they terminate at their lower edges on the 16g sloping coaming member which is of top had section. The stringers, which are 20g angles in this area, run out at the coaming member.

The front engine mounting frame is an 18g pressed channel in L73 which is reinforced by web angle stiffeners at the engine suspension link attachment points. It is also cleated back to the stringers by triangular gusset plates. The front spar frame is built up from extruded angle booms with an 18g web reinforced 14g doubler over the attachment region. It varies in depth from 3.5 inches to 7.0 inches. The rear spar frame also uses extruded angle booms but the portion of it above the wing is a 20g plate bulkhead with 8g doublers at the attachments. Both spars are attached to the frames by two groups of four bolts each. Angle members running between the front and rear spar frames are bolted to the wing ribs at the boom side stations.

The tank floors are connected to the frames by 18g rolled angles. They employ a 0.25 inches deep honeycomb construction with 26g faceplates. The main undercarriage leg is attached at two split bearing housings which are located on a local 16g box beam built up between two adjacent frames. The drag strut is attached on machined bosses which are part of the appropriate frame. The extreme tip of the boom is a fibreglass fairing.

5.2.4 Fin and Rudder

The fin is designed by combined yawing and pitching cases which arise in both symmetric and asymmetric flight. In the former case the critical combination is a tail load of 5200 lb and a fin load of 500 lb with a tailplane torque of 450 lb ft per side. The fin load in the latter case is 2580 lb with 1610 lb on the tail, per side.

The two fin spars are positioned at 15 per cent and 65 per cent of the chord. Each one consists of back to back rolled angles attached to an 18g plate web. All the skinning is in 20g with 22g zed section stringers placed at 2.0 to 2.5 inches pitch. The majority of the ribs are pressed from 24g and the average pitch is 12 inches. However

at the tip the tailplane attachment rib is built up with a 16g web and 16g angle booms, and the root rib is pressed in 18g. The leading edge is made in three separate sections, the centre one of which is removable for access to the elevator control rods which run inside. The tailplane is attached at two points on each fin. The forward point consists of an electrically driven screw actuator which is connected to the fin front spar. The aft point is a bearing housed in a 12g pressed bracket. At the root of the fin the rear spar continues to form the rear boom end bulkhead. A forged S96 bracket is used to connect the front spar web and booms to the appropriate frame and root rib.

Rudder

Each rudder is attached to the appropriate fin by two hinges. The lower one of these is employed for actuation. A trim tab is incorporated in the design, which is very similar in detail to that of the elevator.

5.2.5 Tailplane and Elevator

The factored design tailplane loads of 15500 lb up and 6650 lb down occur when the aircraft is pitching in and out of the normal acceleration manoeuvre.

The tailplane is of constant cross section and structurally has two spars located at 15 per cent and 58 per cent of the chord. The rear spar carries the four elevator hinges of which two are located 37.5 per cent of the semispan out from the centreline and the other two at the tips. The front spar is built up from 14g back to back angles and a plate web of 16g to 20g thickness. The rear spar is of similar design with 18g booms and a 22g web. The main structural box skins are 16g supported by 18g zed section stringers at 2.3 inches pitch. Over the centre region between the inner elevator hinges the stringer area is supplemented by 20g capping strips. The leading edge is assembled in five separate lengths and is stiffened by light riblets. Shear pegs locate the riblets onto the front spar and the skin joint is bolted. Rib location is dictated to some extent by the elevator hinges and the pitch varies from 12 inches at the centre to 17 inches outboard. The ribs are channel pressings the thickness being 16g for those which coincide with the elevator hinges and 22g elsewhere. The tailplane is hinged to the fin about the 45 per cent chord position, a self aligning roller bearing being housed in a fitting located on the outboard rib at each end. The electric actuators are also attached to the outboard ribs on brackets positioned just forward of the front spar.

Elevator

The elevator has a full span geared tab. The tab chord is 8 per cent of that of the tailplane and the operation is from the port end only except for trimming purposes when the starboard end is used. The control runs pass from the booms where cables are used, up the fin leading edges and then across the fin tips below the tailplane. Critical design values of the elevator load and hinge moment are 3660 lb and 360 lb ft respectively.

A two cell box construction is employed. The 18g mainspar is positioned 0.6 inches behind the hingeline. The leading edge and main box skin thicknesses are 20g and 22g respectively and a 22g subsidiary spar is used to mount the tab piano hinge. The 22g pressed ribs vary in pitch from 5 inches to 6 inches. The hinge brackets are machined in L65 and there is a distributed mass balance. The geared tab is a simple 22g box built up from a rear skin and a zed section nose.

5.2.6 Undercarriage

The undercarriage is of conventional nosewheel layout but is unusual in the variable geometry configuration adopted because of the large pitching attitude variation during landing and take off, and the implied fore and aft centre of gravity movement relative to a ground

datum. The geometry chosen is such that the mainwheels move in a fore and aft sense during the change in pitching attitude and thereby compensate for the centre of gravity movement.

Main Undercarriage

The layout of the main undercarriage is shown in Figure 13. It is designed to operate satisfactorily with touchdown attitude angles of up to 35° relative to the horizontal although the normal value corresponding to a lift coefficient of five is 30° . Basically each unit consists of an inclined shock strut with a drag strut which is attached near to the axle of the twin wheels and acts as a radius rod. As the shock strut closes the wheels move forward relative to the airframe so that as the lift is shed by the wing on landing the wheels follow the forward motion of the centre of gravity which results from the reduction of incidence. Touchdown speeds in the range of 100 ft/sec to 164 ft/sec were considered in the design, and the critical loads were found to be 13500 lb vertical 4500 lb side and 9100 lb drag. These do not all occur simultaneously, the side load arises during ground manoeuvres, but the vertical and drag loads are due to a high drag landing. This case proved to be extremely important in determining the geometry and design of the unit. Considerable variation is possible in the characteristics of the layout chosen. The geometry should be such that the resultant of the vertical and drag loads never acts behind the top pivot of the radius rod, since if it does the shock strut will tend to open rather than close. Unfortunately the restrictions of the overall layout of the CI66 were such that it was not found to be possible to ensure this in all cases and an extensive analogue computer study of the problem was undertaken by Hayden (Ref. 5) as part of the design investigation. The high drag force which causes the difficulty is due to the wheel spin up and only persists for a relatively short time during touchdown. Amongst the possible solutions investigated were the possibility of the tyres being able to absorb the initial impact energy, the use of a tension spring in the shock strut and prespinning the wheels. The tyres were not found to have adequate capacity to enable them to absorb all the vertical energy up to the time the drag force dropped to a level which enabled the strut to start closing. The use of a tension spring appeared to be a possibility but since it further complicated an already difficult problem it was not proceeded with. Wheel prespin was therefore adopted. Prespin has been used in the past primarily in an attempt to reduce tyre wear, but it has not found any general application as it is known that most tyre wear occurs during taxiing rather than on landing. In this application the purpose of the prespin is quite different as it is used solely to reduce the high drag forces to a tolerable level so that the resultant force in the plane of symmetry acts forward of the drag strut top pivot. Various methods of spinning up the wheels were considered. Vanes were not found to give sufficient rotation to achieve the desired results, only about 60 per cent to 70 per cent of full speed being possible for a 60 lb weight penalty. On the other hand the use of duplicated air motors was found to give 100 per cent prespin with an acceptable measure of safety for a penalty of 30 lb. Should both motors on one undercarriage unit fail, the aircraft landing attitude and speed must be chosen to ensure satisfactory shock absorber performance, but a safe landing is possible if not straightforward. The shock absorber has a stroke of 28 inches and the estimated maximum effective vertical reaction factor is 2.5.

Structurally the main leg consists of a Y shaped L65 forging which is interchangeable port to starboard. The lower part of the forging contains the oleo pneumatic shock strut. The S96 sliding tube member carries the axle fitting which, like the torque links, is an L65 forging. The twin wheels are magnesium castings and are mounted on a semi-floating live axle. One prespin motor is mounted adjacent to each wheel and drives it through exposed PTFE gears. The gears are mounted away from the single disc brakes and the air motors are automatically switched off when the shock strut closure exceeds 50 per cent. Split bearing caps are used to attach the leg pintle fittings to the boom structure.

The drag strut assembly is also interchangeable on both sides of the aircraft. The

top section consists of an A frame built up of L65 forged components. A simple L63 tube with S96 end fittings is used for the lower part. The drag strut folds for retraction and the downlock is incorporated in the joint. The lock is spring loaded and engages automatically. A pneumatic ram is used to unlock the joint. The retraction jack is also a pneumatic unit and is located across the tops of the main leg and drag strut. The lever ratios are chosen so that there is no change in jack length as the whole unit moves with shock strut closure.

Nose undercarriage

The nose undercarriage is a conventional telescopic oleo pneumatic unit. The stroke is 16 inches and the maximum reaction factor 2.1. Critical design loads were found to be 8000 lb vertically, during dynamic braking, 4250 lb drag in a high drag landing and 1910 lb side load.

An L65 forging is used for the outer case of the leg and both the drag and downlock fittings are incorporated into it. The sliding tube is S96 and the twin wheels are mounted on a live axle carried in the L65 bottom fitting. The wheels are designed to castor up to 60° in either direction and they can be steered through 30° either way. Steering power is taken directly from the rudder pedals and the linkage is attached to the top of the leg. The axle has a positive trail of 2.6 inches and friction shimmy dampers are incorporated in the leg unit. Retraction is rearwards with the aid of a pneumatic ram and during retraction the steering automatically disengages and a self centering spring comes into action. The downlock is spring loaded with a pneumatic release and the three undercarriage doors have a mechanical linkage connection to the leg.

6.0 DISCUSSION

6.1 Performance

Without undertaking the design of an exactly comparable conventional aircraft it is not possible to draw precise conclusions with regard to the benefit obtained by using the suction boundary layer control. However some indication of the improvements in performance to be gained can be obtained by comparison of the CI66 with existing types of aircraft. If this is done it would appear that the boundary layer control system used confers an additional flexibility in the design which is equivalent to approximately 10 per cent of the gross weight. The designer can use this in various ways, for example to have a larger wing to improve the STOL performance still further, or to carry more payload. In the case of the CI66 the benefit was used to install relatively powerful engines and this enabled the top speed to be increased by about 100 m.p.h.

6.2 Operational Considerations

The potential performance improvements resulting from the use of suction boundary layer control have been adequately demonstrated by the various research aircraft which have been tested. However one consideration which must be mentioned is the operational availability and limitations of this type of system. The basic difficulty in this respect is associated with the porous skin and relatively low suction differential. Operational difficulties can be anticipated in certain climatic conditions.

1) Very heavy rainstorms. The mass of water might effectively choke the hole in the skin. The suction pressure and skin hole size are sufficiently great to suggest that this is likely to be a fairly remote possibility, but it does require investigation. The action of the suction would automatically dry the system out so corrosion from this source is not likely to be of consequence.

2) Insects or dust. Again clogging is the problem. This difficulty is much less critical than in the case of a suction laminar flow wing and would not normally need any special precautions. Small size dust particles would be passed by the system and could be filtered at a convenient point, and in fact they would probably tend to collect at certain points in the main wing box. How serious this problem would be in practice would have to be established.

3) Icing. Operation of the aircraft in icing conditions obviously introduces special difficulties and since the frequency of such conditions could be relatively high, icing is likely to be the most significant operational problem. It would be necessary to anti-ice a large part of the wing surface to enable it to function satisfactorily. Electrical anti-icing would appear to be most suitable but the weight and power penalty would be considerable.

It may be concluded therefore that there are certain conditions where the operations of the aircraft could be severely restricted. Whilst ways of alleviating this problem exist, more investigation is required.

6.3 Suction Source

As designed the aircraft uses the main propulsion engines as the sole source of suction. Some measure of duplication in the event of a single engine failure is implied by the large volume of the wing box reservoir. There is a significant reduction in take off performance due to the reduction of intake pressure and an alternative means of suction might well prove to be better overall. In particular the use of the main propulsion engines with a venturi in the intake to give only sufficient suction mass flow for the low differential component can considerably reduce the power loss. An alternative method of suction for the high differential component is then necessary and the form of this requires investigation. The use of a mechanical pump driven either from the cross shafting or independently has much to recommend it. A greater degree of safety would be conferred although there would probably be a weight penalty of the order of 100 lbs. This must be viewed in the context of the effective weight penalty due to loss of take off power, which is equivalent to about 200 lbs, if the take off rather than high speed flight is critical.

6.4 Undercarriage Design

The configuration chosen for the aircraft is such that if conventional landing techniques are retained it is necessary to employ a variable geometry undercarriage. This proved to be a major problem in the design. Although it is feasible provided wheel prespin is used, the adopted solution is more complex than is desirable for this class of aircraft. It may be possible to develop alternative landing techniques where the pilot would retain control of his attitude to a very late stage in the landing run, in which case the need for a variable geometry undercarriage might be obviated. The aircraft would then inevitably be statically unstable on the ground during the initial phase of the landing and this can hardly be regarded as desirable. A possible alternative would be the use of completely different aircraft layouts.

6.5 Tilt Wing

The use of a tilt wing would seem to be the most obvious and viable way of removing the undercarriage geometry problem. It has the additional merit of giving the pilot considerably improved vision for landing. Some weight penalty must result, but it need not be large and the tilting of the wing through say 30° can be simple mechanically. It is not possible to use a tilt wing in the context of the CI66 layout with twin booms and for an aircraft of this type there could be problems associated with the freight bay in any alternative arrangement. A tilt wing would seem to offer considerable advantages however on a STOL transport aircraft with a conventional fuselage, and a

suction wing design for this type of aircraft is worthy of very careful consideration.

7.0 CONCLUSIONS

- 1) The use of a suction boundary layer control system for a counter insurgency aircraft can enable significant performance gains to be made. In the case of the CI66 a maximum speed increase of some 30 per cent relative to a conventional design has been predicted.
- 2) The nature of the porous wing surface introduces the possibility of operational restrictions due to water, dust, insect or ice accretion. The true extent of these restrictions requires to be established by further investigation although in some cases methods of alleviation can be suggested.
- 3) There is scope for further work into the best way of providing suction power. It is not obvious that the proposed sole use of the main propulsion engines for this purpose is the best overall method.
- 4) The use of a variable geometry undercarriage to cater for the high landing and take off attitudes introduces a significant complexity in the design. Although it is not possible for the layout of the CI66 aircraft, a tilt wing configuration can obviate the need for the variable geometry undercarriage and has much to commend it.

REFERENCES

1. RASPET, A. ,
CORNISH, J. J. and
BRYANT, G. D. Delay of the stall by suction through distributed perforations.
I. A. S. Preprint No. 587, 1956.
2. BRYANT, G. D. The Marvel Project - The Marvelette aircraft background and description.
Mississippi State University Rep. 45, May 1963.
3. HEAD, M. R. , and
CLARK, D. G. Flight experiments on suction high lift
AIAA Paper 65-750, Nov. 1965
4. MAIR, W. A. STOL - some possibilities and limitations.
R. Ae. S. 70, 669, September 1966.
5. HAYDEN, J. G. An investigation into the behaviour of the undercarriage of the C166.
C. of A. Thesis, 1968.

APPENDIX A

Allocation of Components for CI66 Study

Arnett, R. R.	Forward fuselage.
Booth, D.	Flaps
Burgess, D. M.	Fins
Clarke, W. J.	Engine installation and fuel system
Clifford, P.	Booms
Collins, M. J.	Suction system
Cowan, D. M.	Elevator and rudder
Edwards, R. H.	Nose undercarriage
Hayden, J. G.	Main undercarriage
Kandil, Z. S.	Centre wing
Machin, W. R.	Rear fuselage
Neal, M. E.	Outer wing and aileron
Rahman, K.	Tailplane

APPENDIX B

1.0 GEOMETRY

1.1 Wing

Gross area	170 sq ft
Span	32 ft
Aspect ratio	6.0
Sweepback of quarter chord line	0°
Root chord (constant to 0.595 semi span)	5.5 ft
Tip chord (nominal)	4.6 ft
Standard mean chord	5.33 ft
Aerofoil sections NACA 63 ^A 215	
Wing-body angle (centreline chord to body datum)	+3°
Dihedral	0°
Location of 0.25 S.M.C. aft of fuselage nose	14.1 ft

1.2 Trailing Edge Flaps

Type: - Plain, hinged on lower surface of aerofoil

Flap chord (aft of hinge)/Wing chord	0.3
Take off flap angle	30°
Landing flap angle	60°
Inboard end of flap from aircraft centreline	2.50 ft
Outboard end of flap from aircraft centreline	7.40 ft

1.3 Leading Edge Flap

Type:- Plain droop nose, hinged on lower surface of aerofoil

Flap chord (forward of hinge)/wing chord	0.1
Flap angle of droop	30°
Inboard end of flap from aircraft centreline	2.50 ft
Outboard end of inner flap section	7.40 ft
Inner end of outer flap section	9.60 ft
Outboard end of flap from aircraft centreline	16.0 ft

1.4 Ailerons

Type:- Internally sealed 27 per cent aerodynamic balance.

Aileron chord (aft of hinge line)/Wing chord	0.3
Balance chord (forward of hinge line)/Aileron chord (aft of hinge line)	0.33
Aileron droop for landing condition	12°
Aileron movement (in addition to droop)	20° up 18° down
Inboard end of aileron from aircraft centreline	9.60 ft
Outboard end of aileron from aircraft centreline	16.0 ft

1.5 Tailplane

Gross area (actual)	58 sq ft
Span (between fin centrelines)	17.0 ft
Span (actual)	16.5 ft
Aspect Ratio (actual)	4.7
Sweepback	0°
Chord (constant across span)	3.5 ft

Aerofoil section NACA 23 _A 012	
Tail setting angle relative to body datum:-	
Lowspeed (approach and take off)	-12.0°
Normal flight	- 3.0°
Vertical location of tailplane 0.25 S.M.C. above body datum	8.4 ft
Distance of tailplane 0.25 S.M.C. aft of wing 0.25 S.M.C.	19.0 ft
Tail volume coefficient	1.22
1.6 <u>Elevator</u>	
Type: Internally sealed 27 per cent aerodynamic balance.	
Elevator chord (aft of hinge line)/Tailplane chord	0.30
Movement of elevator	27° up
	12° down
	20°
Angle of cut off of elevator tip	
1.7 <u>Fins</u>	
Nominal area, per fin	18.0 sq ft
Nominal height (mean)	4.15 ft
Aspect Ratio (based on the above dimensions)	1.9
Root chord (at bottom of rudder), nominal	5.1 ft
Tip chord (at top of rudder), nominal	3.4 ft
Height of bottom of rudder above body datum	4.15 ft
Sweepback of leading edge	50°
Aerofoil section: NACA 23 _A 015	
Distance of nominal root leading edge aft of wing 0.25 S.M.C.	12.5 ft
Fin volume coefficient, both fins	0.115
1.8 <u>Rudder</u>	
Type: Internally sealed 27 per cent aerodynamic balance.	
Rudder chord (aft of hinge line)/fin chord	0.25
Balance chord (forward of hinge line)/Rudder chord (aft of hinge line)	0.33
Movement	+20°
1.9 <u>Body</u>	
Overall length	25.3 ft
Maximum width	4.8 ft
Maximum depth (over canopy, no gun pack)	6.25 ft
Length of floor of freight bay	10.7 ft
Width of floor of freight bay	4.0 ft
Height of freight bay	3.65 ft
1.10 <u>Booms</u>	
Overall length, inclusive of power plant	24.5 ft
Location of nose forward of wing 0.25 S.M.C.	6.95 ft
Maximum depth	3.75 ft
Maximum width	1.83 ft
1.11 <u>Undercarriage</u>	
Type: Nosewheel	
Wheelbase: parallel to body datum	
Shock absorbers fully extended	17.7 ft
Shock absorbers fully closed	14.3 ft

Track (to centres of main legs)	17.0 ft
Design vertical velocity (proof)	12 ft/sec

Main undercarriage units

Type: Twin wheel, radius rod layout	
Tyres: 20 ins diameter x 5.25 ins width	
Tyre pressure	70 lb/sq in
Wheel track	0.92 ft
Shock absorber closure	2.4 ft
Tyre closure (max.)	0.25 ft
Vertical axle travel (parallel to datum)	1.41 ft
Location of mainwheel axle aft of 0.25 S. M. C:-	
Fully extended	5.4 ft
Fully closed	2.0 ft

Nosewheel unit

Type: Twin wheel, cantilever.	
Tyres: 15 ins diameter x 4.75 ins width.	
Tyre pressure	80 lb/sq in
Wheel track	0.83 ft
Tyre closure (max)	0.19 ft

2.0 Power Plants

Type: 2 Bristol Siddeley Gnome P1200 Turboprops (1150 H. P. sea level static)	
Propeller: 3 blade constant speed	
Diameter	11.0 ft
Polar moment of inertia	1100 lbf ft ²
Location of propeller axis above body datum	2.88 ft
Accessory Drive. The engines are coupled by a spanwise shaft located immediately aft of the main wing spar. The accessory drive gearbox is located in the fuselage and is driven by the coupling shaft.	

2.1 Suction system

The main propulsion engines are used as the source of suction power for the wing boundary layer control.

3.0 Weights, Centres of Gravity and Moments of Inertia

Design all up weight	9800 lbf
Maximum landing weight	9600 lbf
Minimum landing weight	7200 lbf
Basic operating weight (no crew)	6695 lbf
Disposal load (2 crew)	2705 lbf
Maximum normal payload	2000 lbf
Maximum normal fuel load	2000 lbf
Body tank (divided into three compartments)	20 cu ft
Upper boom tanks (2)	8 cu ft each
Collector tanks (2)	2 cu ft each
Weight breakdown - see Table 1.	

Centre of gravity position at 7095 lbf (basic operating weight with two crew):-

- (a) Undercarriage extended
 - 0.15 ft forward of wing 0.25 S.M.C. and
 - 1.97 ft above body datum.
- (b) Undercarriage retracted
 - 0.07 ft forward of wing 0.25 S.M.C. and
 - 2.17 ft above body datum.

Allowable centre of gravity range
 0.267 ft forward to 0.693 ft aft of wing
 0.25 S.M.C. (0.2c to 0.38c)

Moments of Inertia, at 9800 lb including 2000 lb of payload

Pitch	412,000 lb ft ²
Roll	425,000 lb ft ²
Yaw	714,000 lb ft ²

4.0 Aerodynamic Information

Maximum lift coefficient (untrimmed)	
Basic wing (no flaps, suction or slipstream)	1.4
Increment due to leading edge flap	0.3
Increment due to trailing edge flap at 30°	0.38
Increment due to trailing edge flap at 60°	0.55
Increment due to ailerons drooped at 12°	0.13
Design low speed lift coefficient (absolute value)	5.0

Drag polars:

Cruise configuration at 15000 ft altitude, 244 knots E.A.S.

$$C_D = 0.033 + 0.058C_L^2$$

Take off configuration (30° flap and full power)

$$C_D = 0.175 + 0.065C_L^2$$

Landing configuration (60° flap and half power)

$$C_D = 0.29 + 0.062C_L^2$$

Pitching moment coefficient at zero lift, (clean)	-0.08
Increment due to leading edge flap	Negligible
Increment due to trailing edge flap at 30°	-0.18
Increment due to trailing edge flap at 60°	-0.25
Increment due to aileron drooped to 12°	-0.07
Total in take off configuration (30° flap and full power)	-0.44
Total in landing configuration (60° flap, 12° aileron and half power)	-0.54
Location of wing-body aerodynamic centre on wing S.M.C.	0.24
Location of tailplane aerodynamic centre on tailplane S.M.C.	0.24
Location of fin aerodynamic centre of fin S.M.C.	0.24
Wing no lift angle, relative to body datum, basic	-2°
Slope of wing lift curve, a ₁ :-	
Basic wing	4.0
With leading edge flap	4.4
With leading edge flap and take off slipstream	7.4
With leading edge flap and landing slipstream	5.8
Two dimensional ratio of aileron lift curve slopes, a ₂ /a ₁	0.55

Slope of aileron hinge moment due to wing incidence, b_1	-0.8
Slope of aileron hinge moment due to aileron angle, b_2	-0.38
Rolling moment coefficient due to aileron angle, ℓ_{ξ}	-0.21
Aileron derivative slipstream factors:-	
Take off	1.96
Landing	1.44
Slope of tailplane lift curve, a_{1T}	3.7
Ratio of elevator lift curve slopes, a_{2T}/a_{1T}	0.58
Slope of elevator hinge moment due to tailplane incidence, b_{1T}	-0.08
Slope of elevator hinge moment due to elevator angle, b_{2T}	-0.42
Slope of fin and rudder lift curve, a_{1F}	2.5
Ratio of rudder lift curve slopes a_{2F}/a_{1F}	0.54
Slope of rudder hinge moment due to fin incidence, b_{1F}	-0.02
Slope of rudder hinge moment due to rudder angle, b_{2F}	-0.28/rad
Fin and rudder slipstream factors:-	
Take off	2.37
Landing	1.64
Downwash at tailplane 0.25 chord, ϵ	$3.8C_L$ degrees
Rolling moment coefficient due to rolling, ℓ_p	
Cruise	-0.45
Take off power	-0.58
Landing power	-0.52
Rolling moment coefficient due to yawing, ℓ_r	
Cruise	$0.24C_L + 0.106$
Take off power	$0.24C_L - 0.07$
Landing power	$0.24C_L - 0.059$
Rolling moment coefficient due to sideslip, ℓ_v	
Cruise	-0.16
Take off power	+0.006
Landing power	-0.005
Side force coefficient due to sideslip, y_v	
Cruise	-0.5
Take off power	-0.77
Landing power	-0.64
Yawing moment coefficient due to sideslip, n_v	
Cruise	0.082
Take off power	0.185
Landing power	0.133
Yawing moment coefficient due to yawing, n_r	
Cruise	$-0.33 - 0.023C_L^2$
Take off power	$-0.47 - 0.023C_L^2$
Landing power	$-0.39 - 0.023C_L^2$

Tailplane rolling moment coefficient due to sideslip, K

0.15

(All derivatives are based on the reference dimensions. Hinge moments only are based on control surface area and chord. The quoted slipstream effects are at a datum low speed flight condition of 120 ft/sec. All derivatives are per radian unless otherwise stated. Control derivatives do not include the effect of tabs).

TABLE 1

PREDICTED COMPONENT WEIGHTS

COMPONENT	WEIGHT lbf.	% A. U. W.
Fuselage	568	5.80
Booms, including engine mountings and intakes	840	8.60
Wings, flaps and ailerons	700	7.14
Tailplane and elevator	202	2.04
Fins and rudder	118	1.20
Main undercarriage	402	4.10
Nose undercarriage	71	0.72
Structure	2901	29.60
Engines, including exhaust	1472	15.00
Propellers	552	5.63
Gearboxes, cross shafts and accessory drives	140	1.43
Power Plant	2164	22.06
Fuel system	200	2.04
Power services (electrics and pneumatics)	400	4.08
Flying controls	100	1.02
Radio	180	1.83
Instruments	70	0.72
Furnishing and cabin conditioning	250	2.55
Armour protection	200	2.04
Fixed armament	100	1.02
Suction system	130	1.32
Systems and equipment	1630	16.62
Crew (2)	400	4.08
Empty weight	7095	72.36

TABLE 2
SUCTION REQUIREMENTS
 (No slipstream effects)

		Take off C _L			Landing C _L	
		4	5	6	5	6
		Suction Differential lb/sq ft	Surface Duct losses Total	119 102 221	199 102 301	288 102 390
Mass Flow	slugs/sec	0.21	0.19	0.18	-	-

FIGURES

1. General arrangements of the CI 66 counter insurgency aircraft.
2. Photograph of scale model of the design.
3. Layout of fuselage with typical payloads
4. Layout of boom
5. Aerofoil section details
6. Variation of lift coefficients with incidence
7. Variation of range with payload
8. Arrangement of powerplant and suction systems.
9. Leading and trailing edge flap sections and suction cell details.
10. Arrangement of gearboxes and cross shafts
11. Key diagram of structural members
12. Details of canopy and aileron droop mechanism
13. Arrangement of main undercarriage
14. Arrangement of nose undercarriage

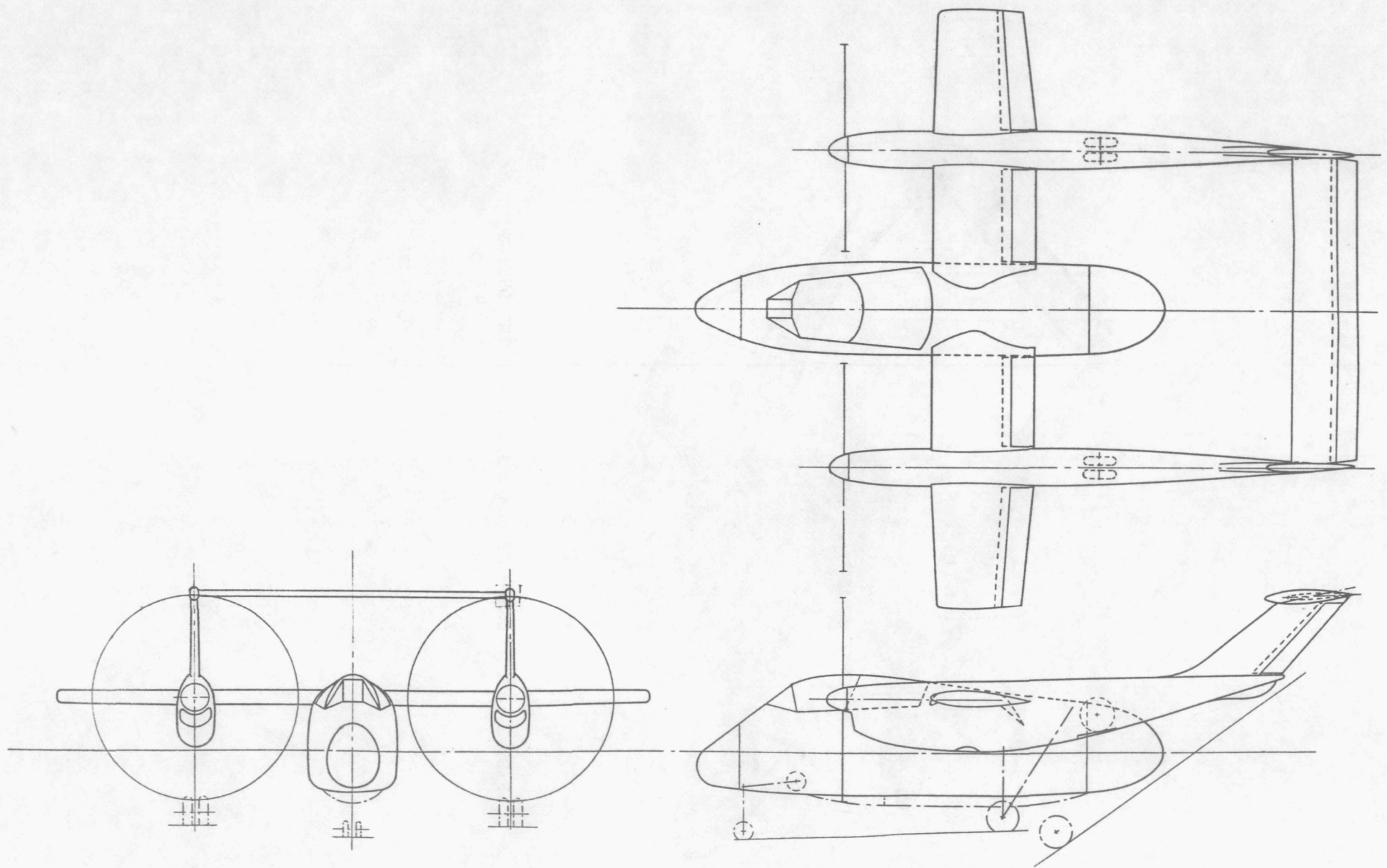


FIG.1. GENERAL ARRANGEMENT OF THE CI66 COUNTER INSURGENCY AIRCRAFT

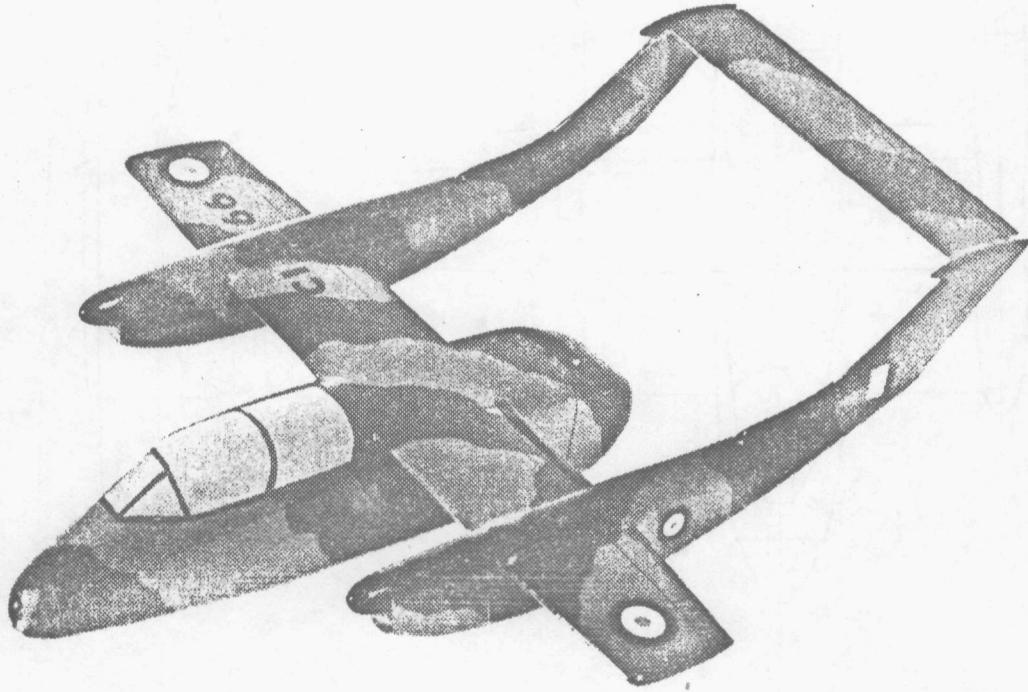


FIG. 2 PHOTOGRAPH OF SCALE MODEL AND THE DESIGN

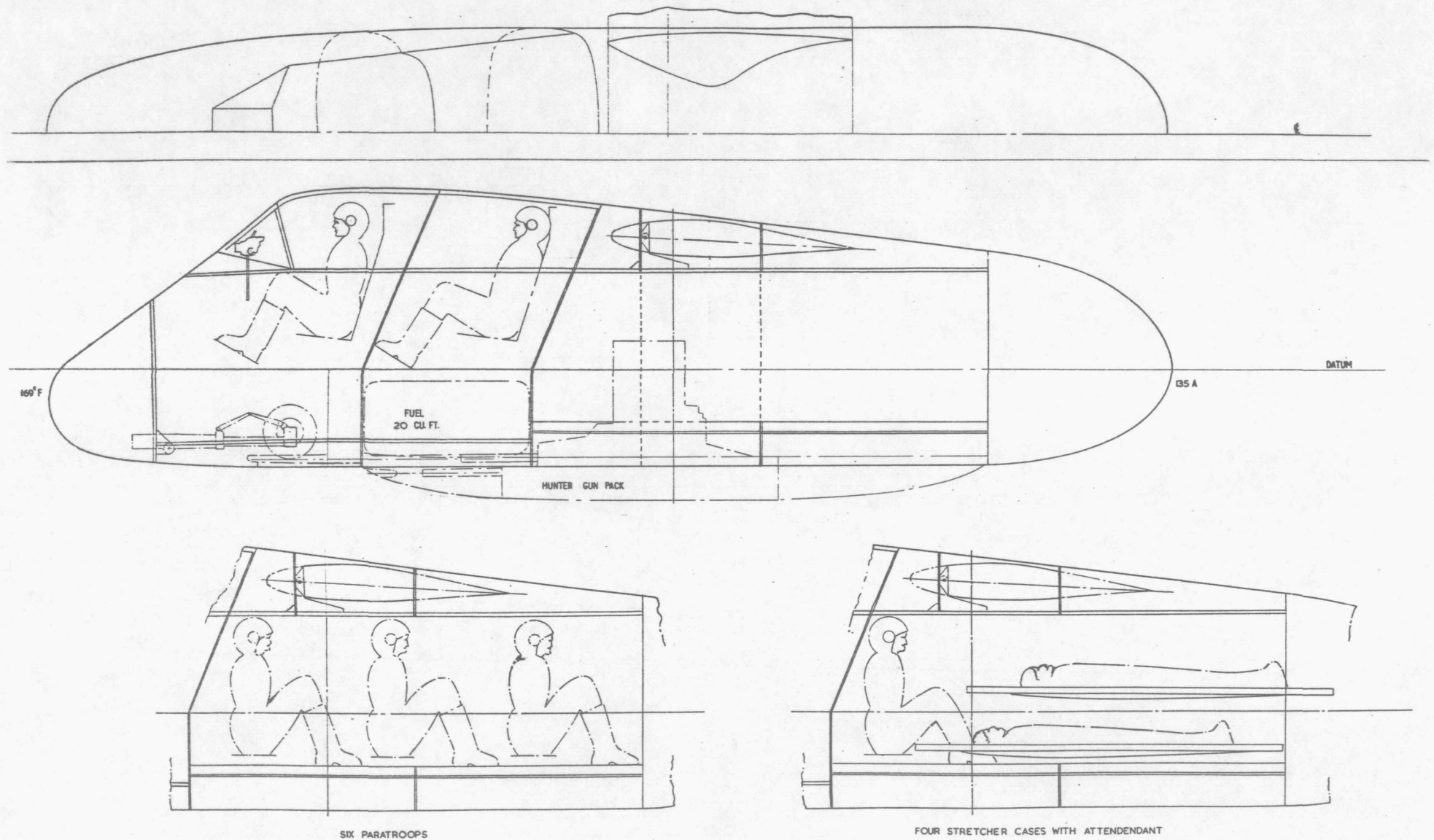


FIG. 3. LAYOUT OF FUSELAGE WITH TYPICAL PAYLOADS

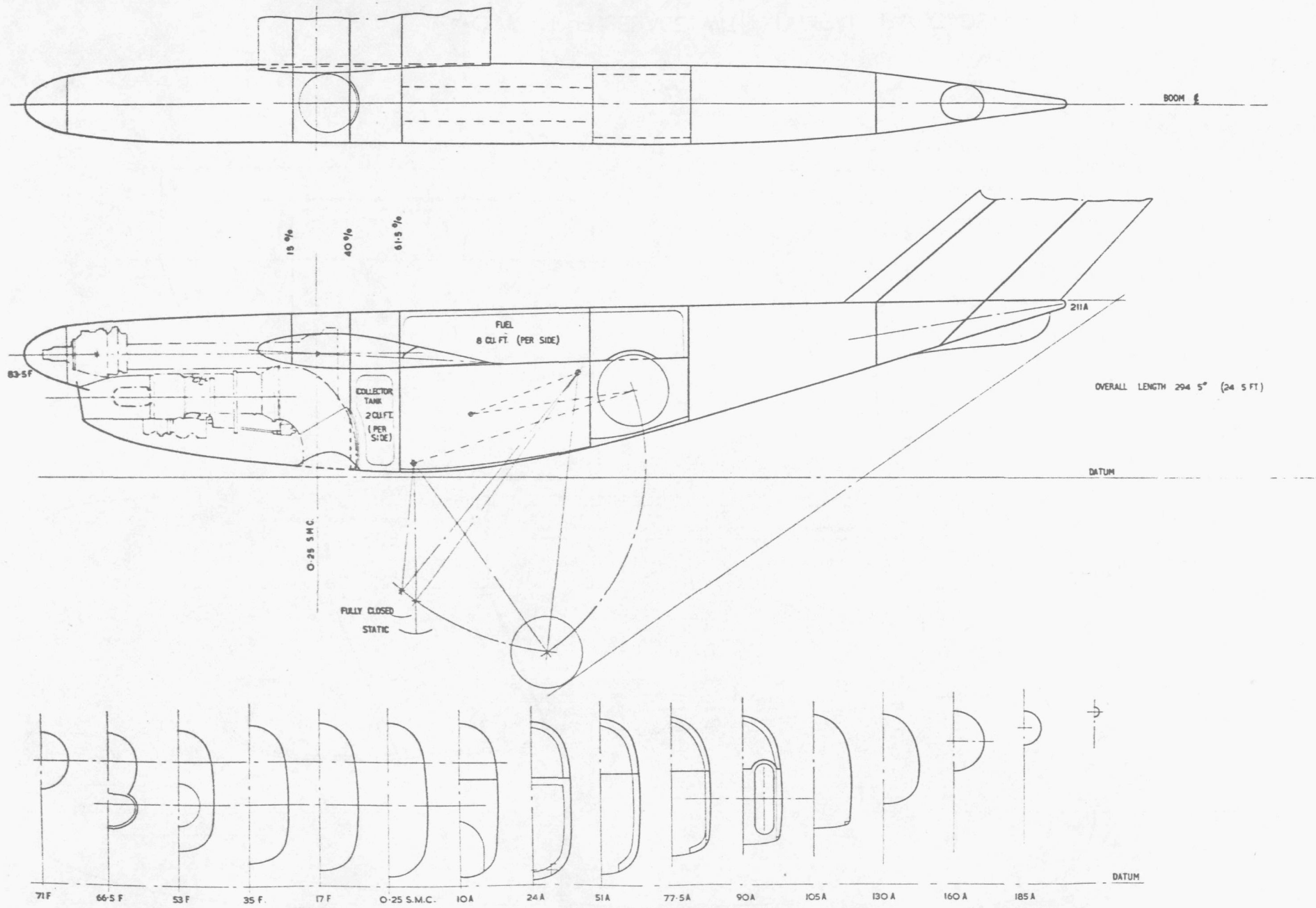


FIG.4. LAYOUT OF BOOM

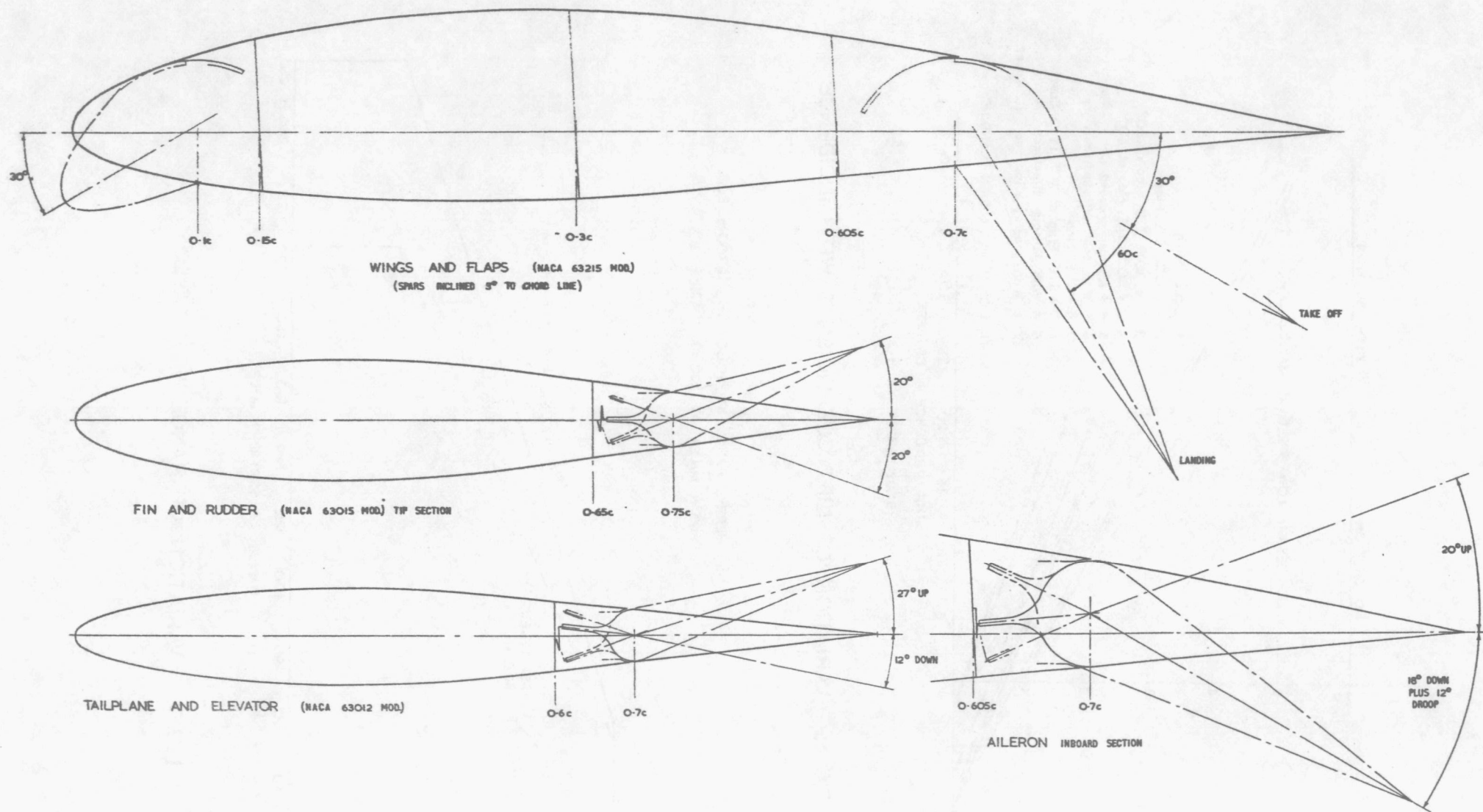


FIG. 5. AEROFOIL SECTION DETAILS

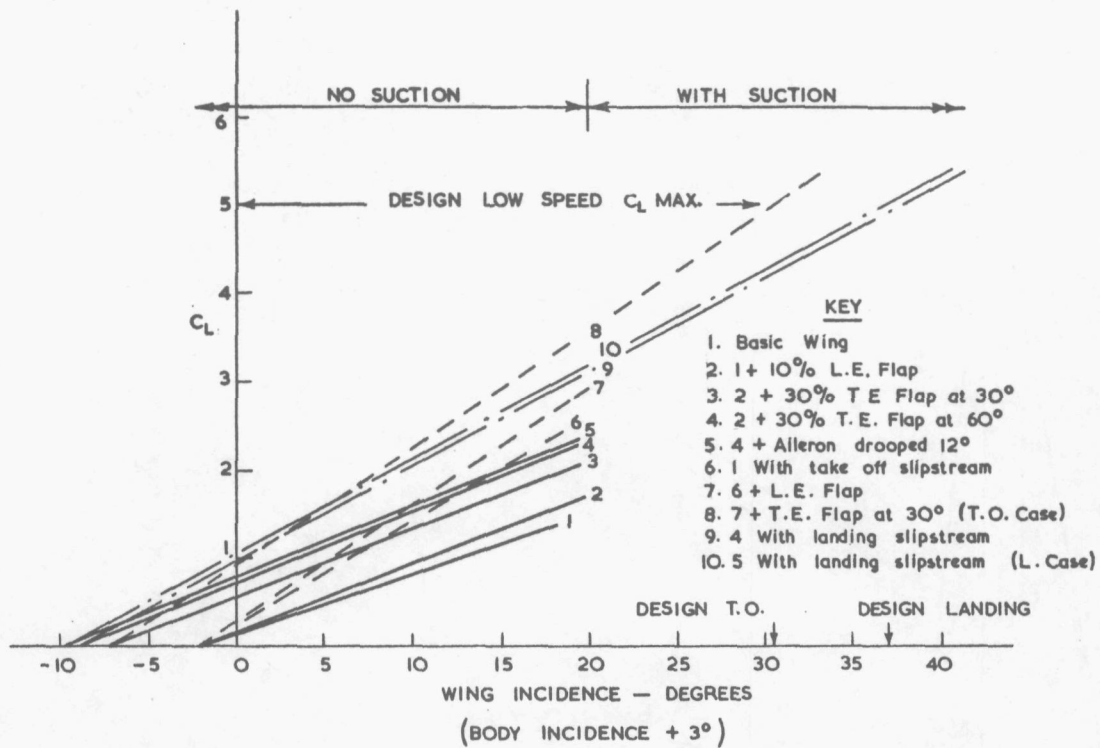


FIG. 6. VARIATION OF LIFT COEFFICIENTS WITH INCIDENCE

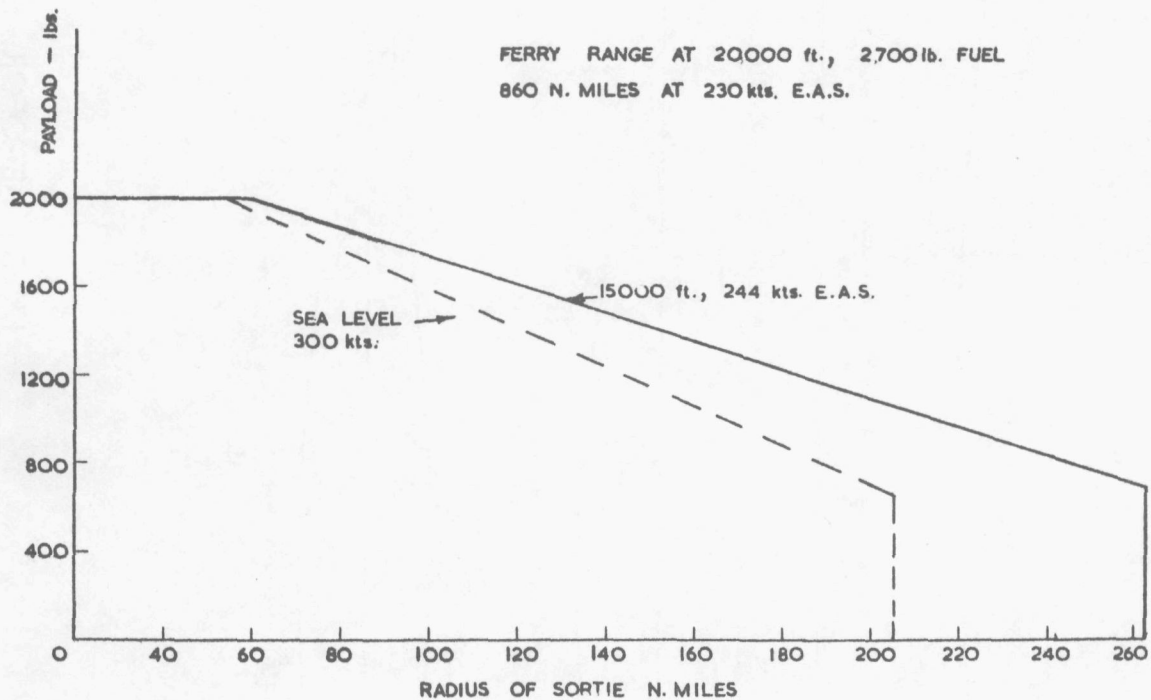


FIG. 7. VARIATION OF RANGE WITH PAYLOAD

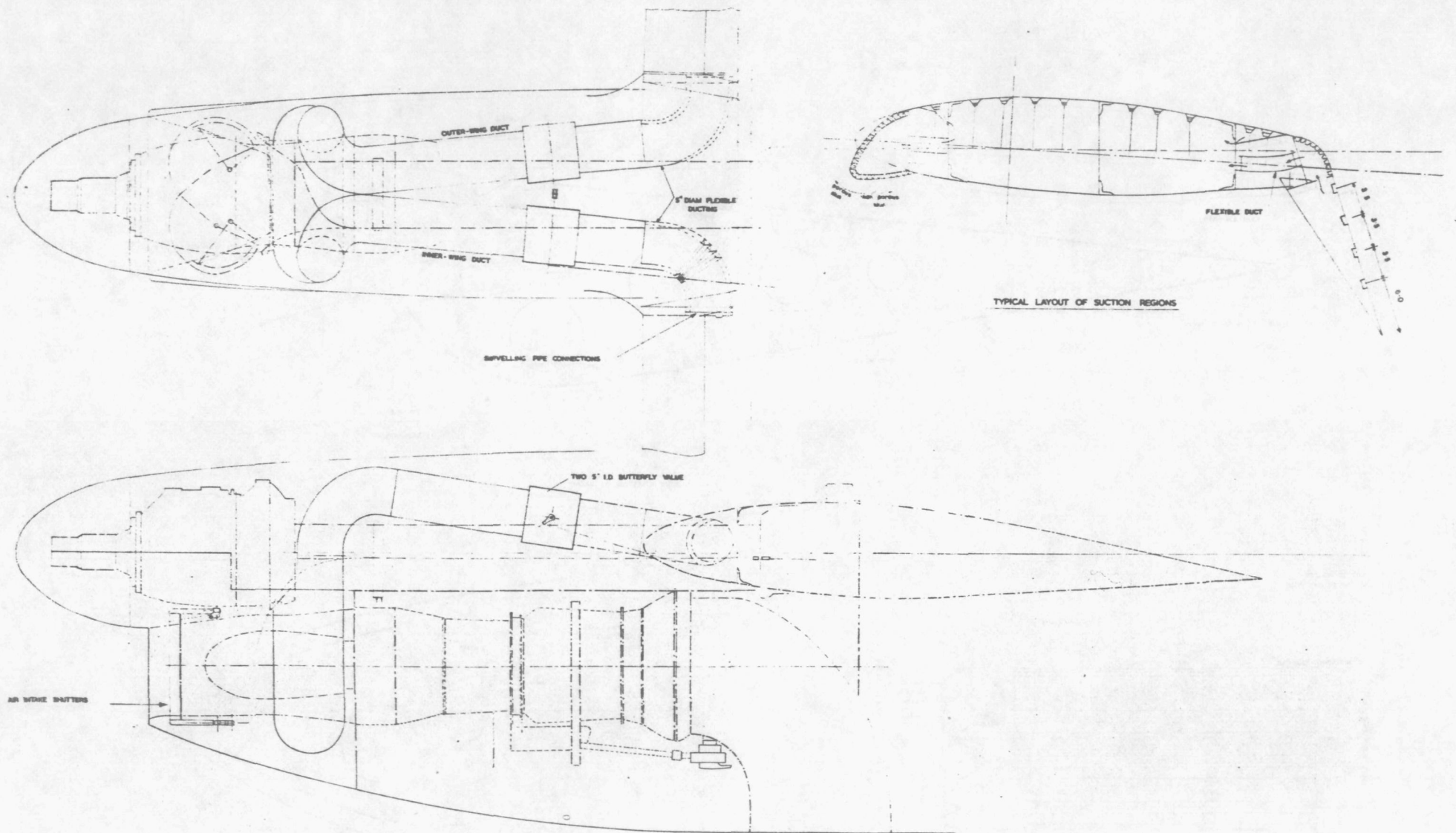


FIG. 8. ARRANGEMENT OF POWERPLANT AND SUCTION SYSTEMS

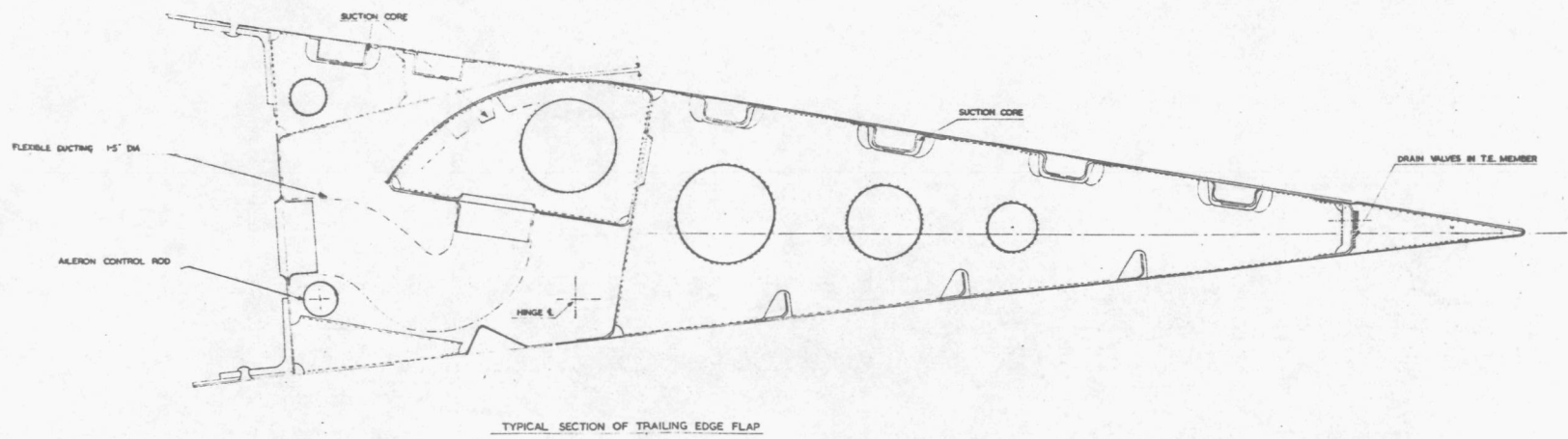
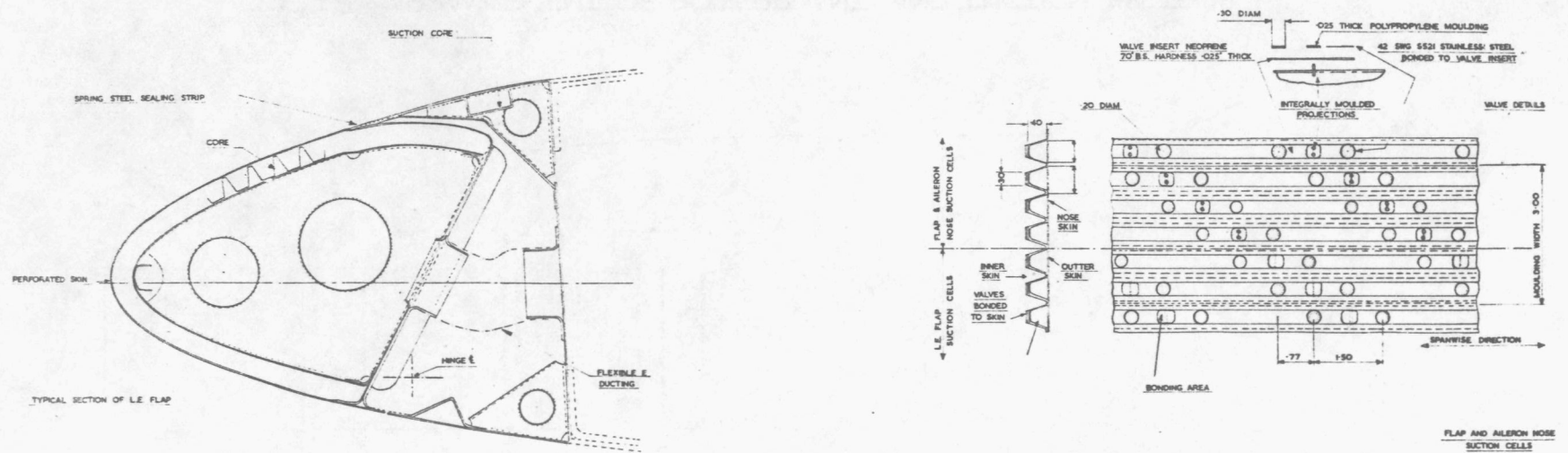


FIG.9. LEADING AND TRAILING EDGE FLAP SECTION AND SUCTION CELL DETAILS

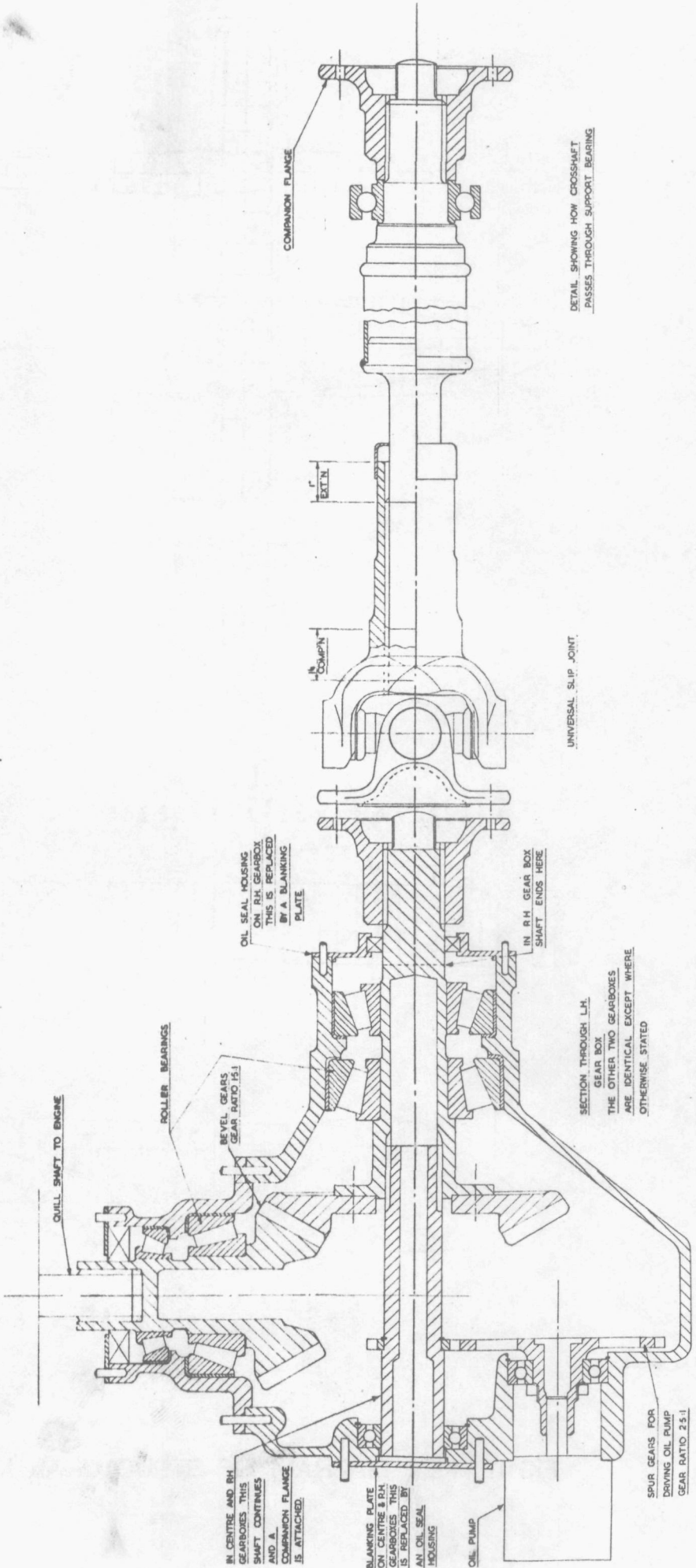
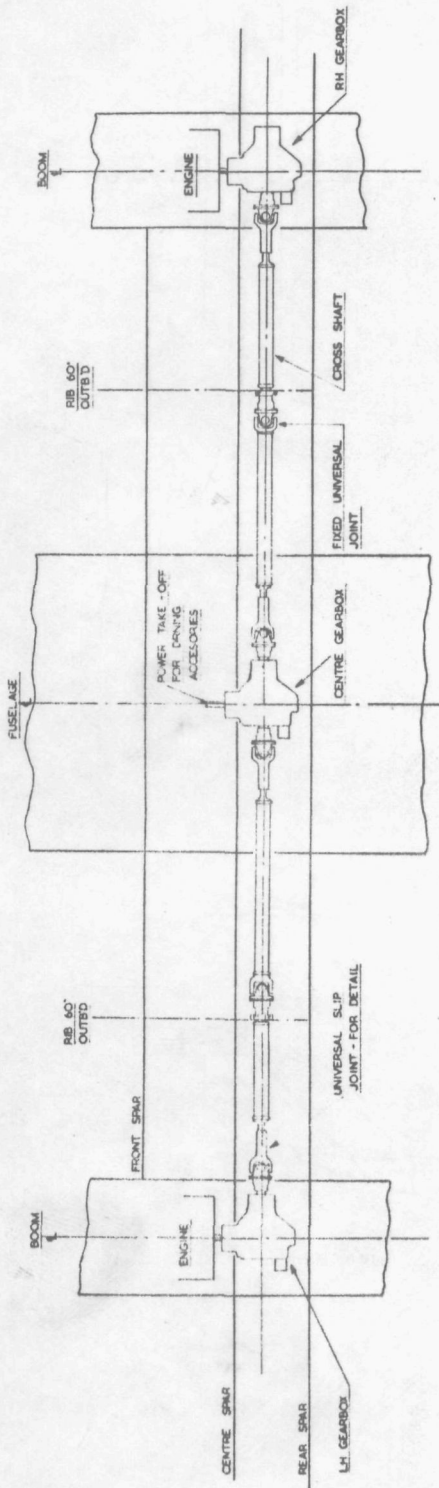


FIG. 10. ARRANGEMENT OF GEARBOXES AND CROSS-SHAFTS

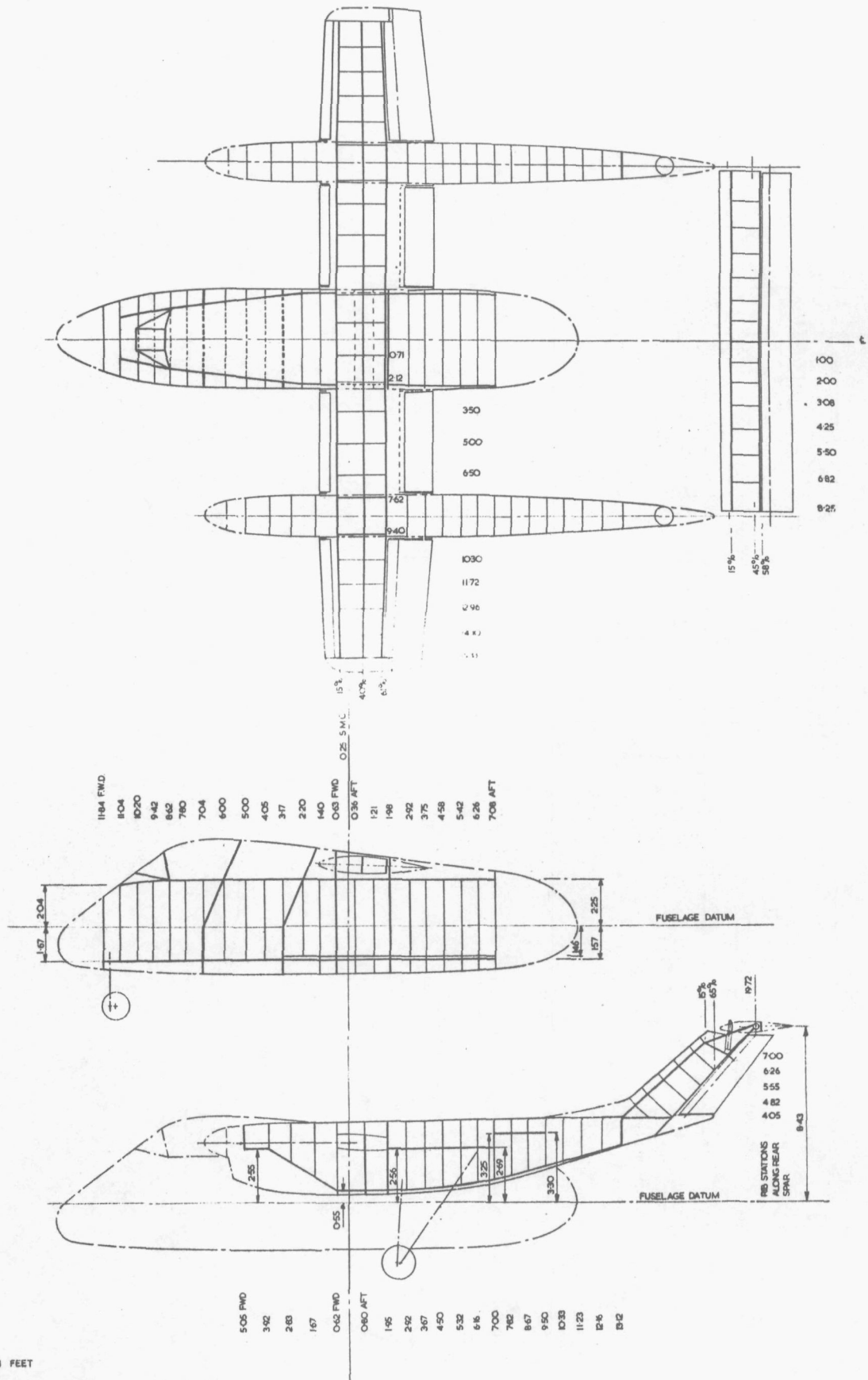


FIG. II. KEY DIAGRAM OF STRUCTURAL MEMBERS

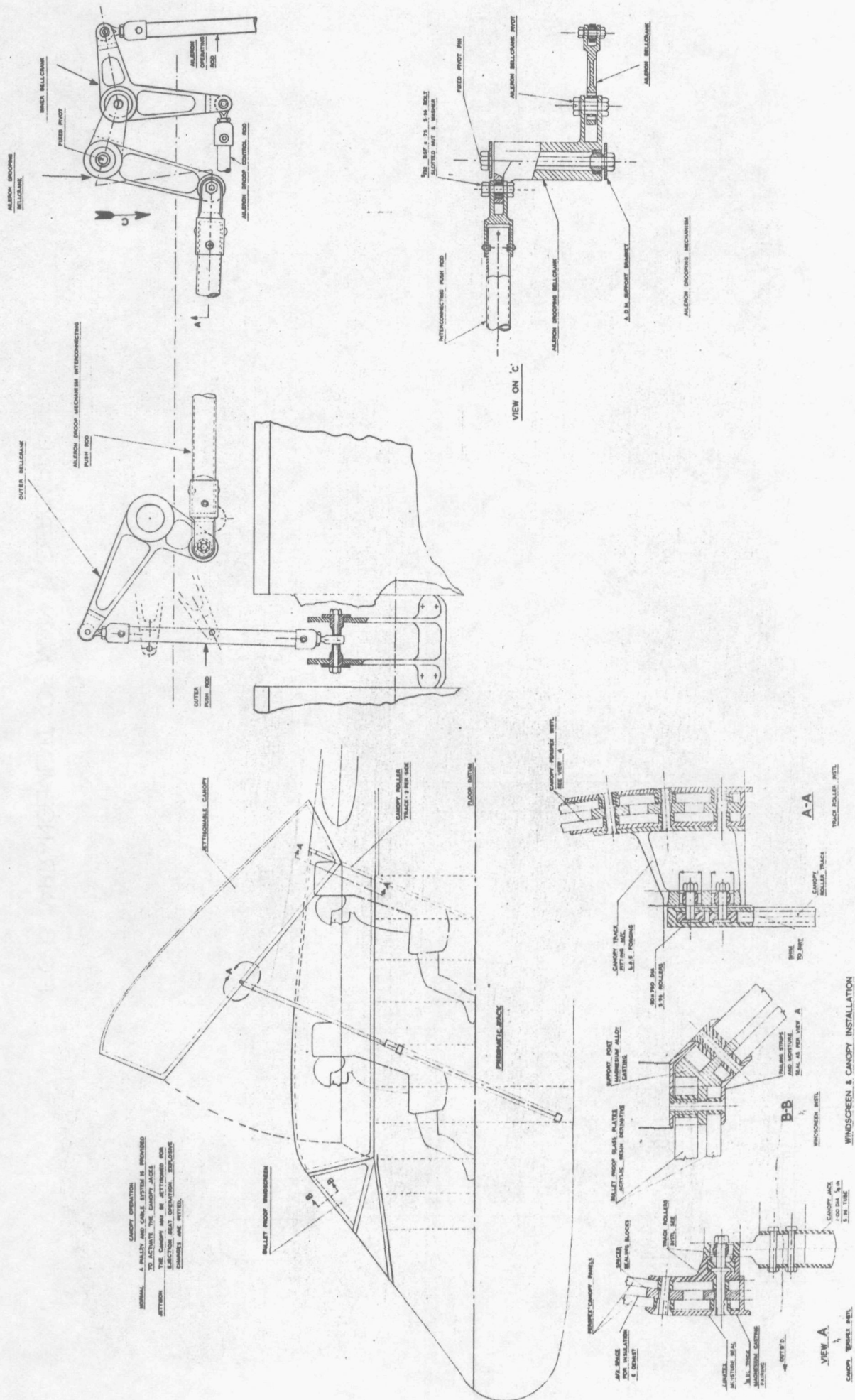


FIG. 12. DETAILS OF CANOPY AND AILERON DROOP MECHANISM

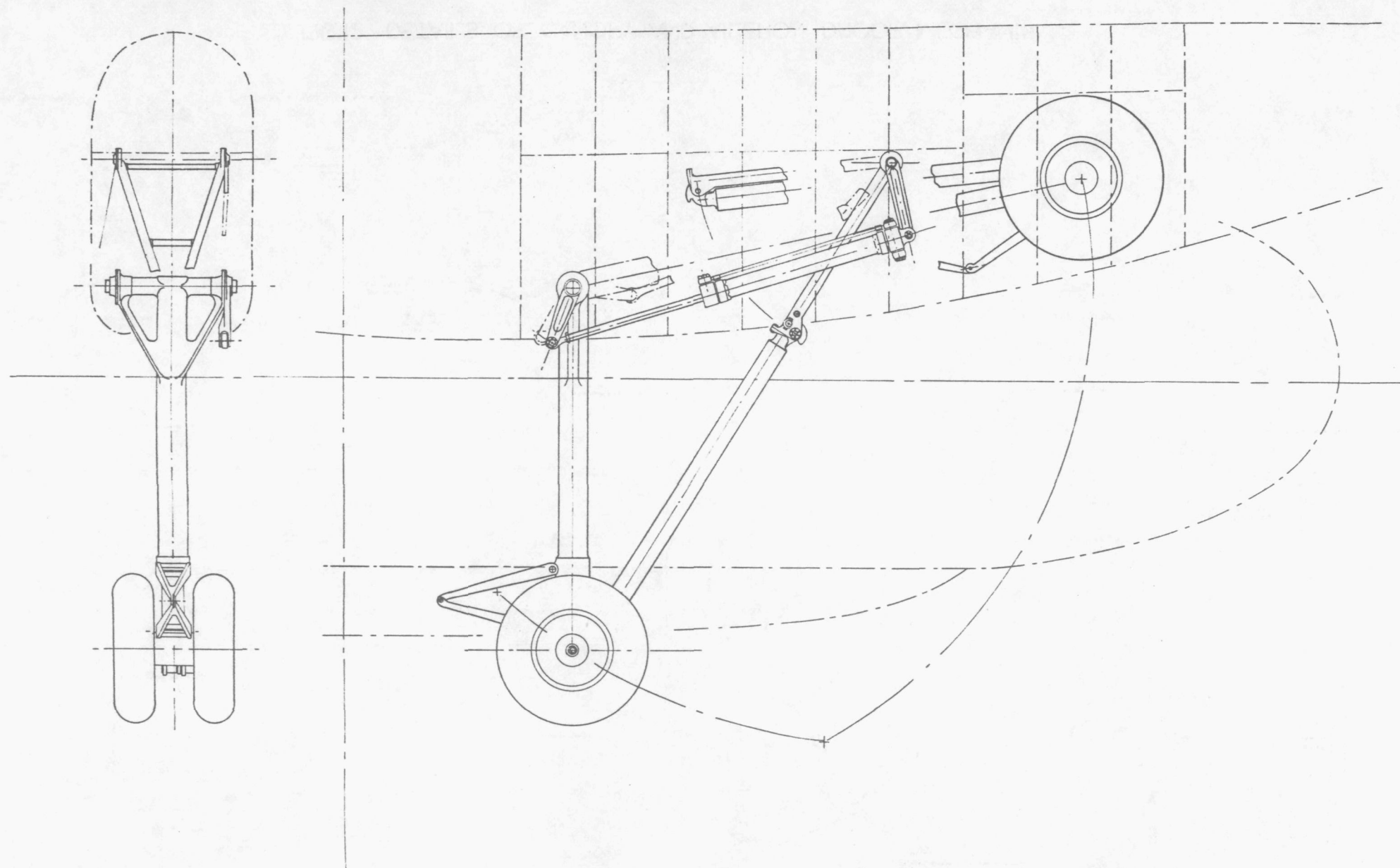


FIG. 13. ARRANGEMENT OF MAIN UNDERCARRIAGE

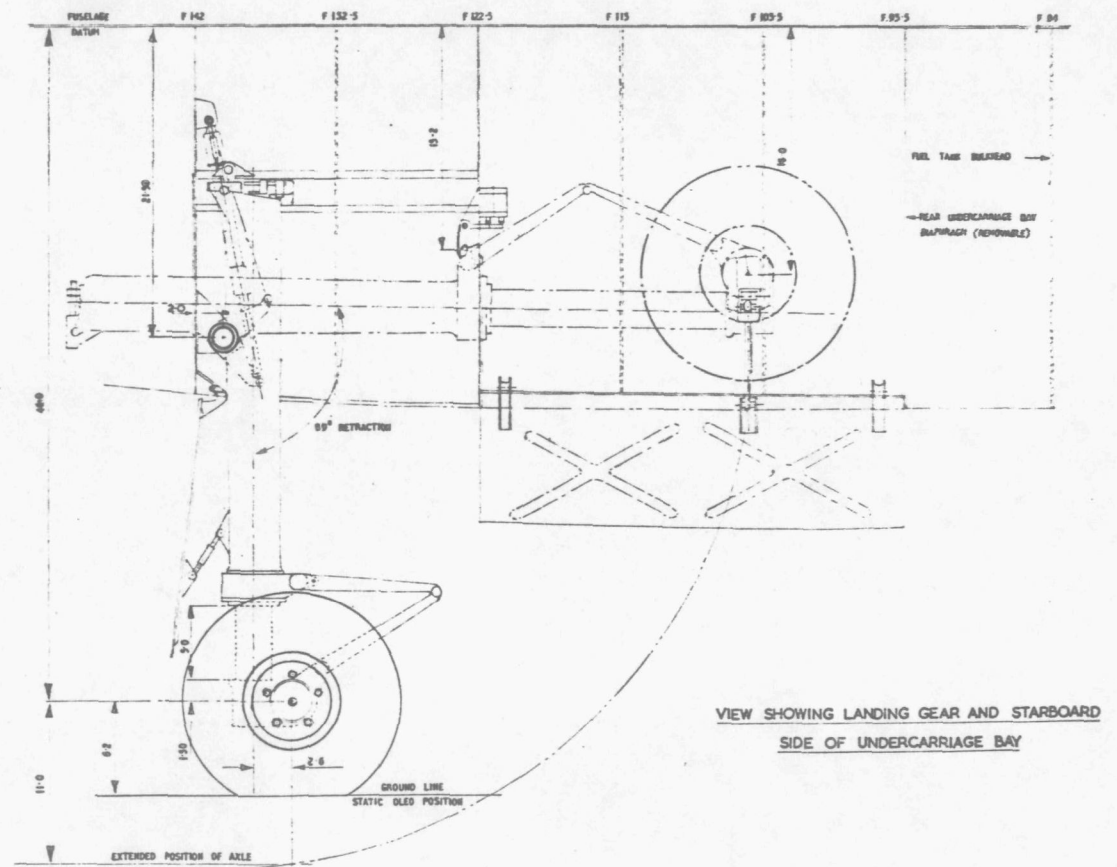
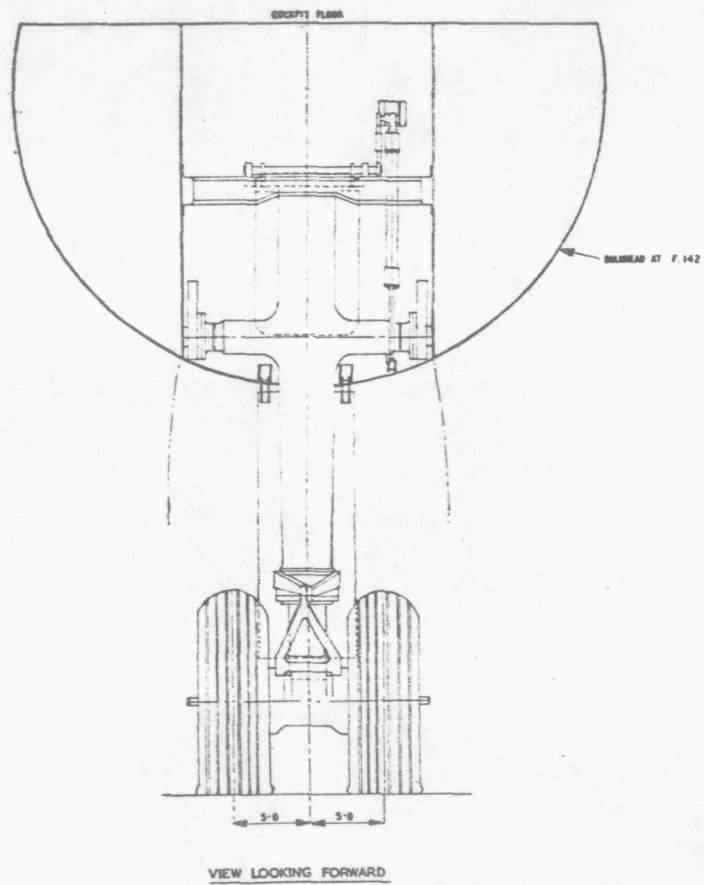


FIG. 14. ARRANGEMENT OF NOSE UNDERCARRIAGE