

R47049/A

CRANFIELD INSTITUTE OF TECHNOLOGY

AIRCRAFT DESIGN STUDIES - E67
TILT - WING EXECUTIVE AIRCRAFT

by

D. HOWE



R47049/A



Cranfield Report Aero. No.23

August 1973

CRANFIELD INSTITUTE OF TECHNOLOGY

AIRCRAFT DESIGN STUDIES - E67

TILT-WING EXECUTIVE AIRCRAFT

by D. HOWE

SUMMARY

The E67 design was for a vertical take off and landing aircraft for executive use. Vertical flight capability was achieved by using the tilt wing concept. In the initial version of the design power was provided by two Rolls Royce H1400 Gnome turbo shaft engines driving two 16 ft. diameter propellers. Cross shaft interconnection between the propellers was included in the layout. Subsequently the need for four engines to cater for an engine failure condition in vertical flight became apparent. The pressurised cabin was designed to accommodate up to 18 passengers in a high density feeder role. Conventional design techniques were used throughout.

A market survey showed that the design had significant advantages relative either to a helicopter or a twin jet executive type (Reference 3).

CONTENTS

	<u>Page</u>
1. INTRODUCTION	1
2. CONFIGURATION OF E67 DESIGN	2
3. CONTROL AND TRANSITION	3
4. PERFORMANCE	3
5. DESIGN CONDITIONS	4
6. STRUCTURAL DESIGN	4
6.1 Fuselage	4
6.2 Wing	6
6.3 Tailplane	9
6.4 Fin	9
6.5 Main undercarriage	10
6.6 Nose undercarriage	11
7. POWERPLANT AND TRANSMISSION	12
7.1 Powerplant (Gnome H1400)	12
7.2 Transmission	12
7.3 Auxiliary Power Unit and Power Supplies	13
7.4 Powerplant Mounting and Nacelle	14
7.5 Engine Installation	14
7.6 Four Powerplant Design (Garrett TPE 331)	15
7.7 Propellers and Tail Rotor	15
8. SYSTEMS	15
8.1 Fuel system	15
8.2 Cabin environmental system	16
8.3 Flying control system	16
9. EQUIPMENT AND INSTALLATIONS	18
10. MARKET SURVEY AND OPERATION ANALYSIS	18
11. DISCUSSION AND CONCLUSIONS	19
11.1 Discussion	19
11.2 Conclusions	21
REFERENCES	22
APPENDIX A	A1
APPENDIX B	B1
TABLE 1	
FIGURES	

LIST OF FIGURES

1. General arrangement of E67 design
2. Fuselage layout
3. Variation of wing tilt angle with transition speed
4. Variation of tailplane and flap angles during wing tilt
5. Payload range diagram
6. Photograph of cut away model of E67
7. Structure and mechanism of wing tilt
8. Layout of wing
9. Inner wing structure
10. Wing tilt actuator
11. Tailplane structure
12. Main undercarriage unit
13. Nose undercarriage unit
14. Powerplant and nacelle
15. Centre gearbox
16. Tail rotor gearbox
17. Fuel system
18. Aileron roll-yaw interconnection mechanism
19. Tailplane actuation mechanism
20. Pilots instrument panel layout.
21. Slipstream effects.

1. INTRODUCTION

One of the features of recent aviation history has been the rapid increase in the number of major organisations who provide aircraft for the use of their executive staff. The aircraft used range from basic piston engine types to sophisticated jet machines. In some cases the organisation is fortunate enough to be located close to a suitable airfield, but in many instances lack of suitable facilities places a limitation on the benefits which can be derived from the use of the executive aircraft. This is especially likely to be the case at destination points. Whilst it is relatively easy to arrange for a car to meet the aircraft when it lands, the inconvenience of the transfer and the time lost in having to complete a journey over frequently congested roads are difficulties which are severe enough to justify study to see if they can be eliminated. One possible solution is the use of a vertical take off and landing aircraft capable of operating from relatively easily prepared sites adjacent to departure and destination points. Even in this case there are possible restrictions with regard to flying over built-up areas and integration into air traffic control. However the fact that some organisations make an appreciable and successful use of helicopters shows that these difficulties can be overcome. Unfortunately the helicopter is a relatively expensive type of aircraft with limited speed performance and many business concerns prefer to buy faster and more prestigious fixed wing aircraft.

A possible way of overcoming the conflict of requirements emphasised by the fundamental differences in operating performance characteristics of fixed and rotating wing craft is the use of the tilt-wing concept. In this particular form of vertical take off and landing aircraft the cruise performance can be as high as any fixed wing type with the exception of those employing jet propulsion. Of course the tilt-wing aircraft is more complex and expensive than a comparable fixed wing machine in that it has some of the elements of the helicopter. Nevertheless it seems possible that the operating cost would be comparable with a jet executive aircraft and it could have similar departure to destination block time over typical stage lengths with much greater convenience.

During the 1967/8 academic year the students in the Aircraft Design course at Cranfield undertook a design study of a tilt-wing executive aircraft. The individual responsibilities in the investigation are listed in Appendix A. The design was known as the E-67 and it was intended to have cruise performance characteristics as near as possible to those predicted for the Handley-Page HP137 Jetstream turboprop executive aircraft.

2. CONFIGURATION OF THE E-67 DESIGN

A general arrangement drawing of the aircraft is shown in Figure 1. Although intended primarily to fulfil the executive role the 19 ft long cabin was designed to be capable of accommodating up to 18 passengers for third level airline operations. In this high density role there are six rows of three seats with a dividing aisle having 6 ft clear headroom. This arrangement is shown in Figure 2.

The nature of the tilt-wing concept dictated a high wing for reasons of ground clearance and this resulted in a 7ft external diameter for the pressurised fuselage. This is a penalty of approximately 0.5 ft relative to a comparable low wing layout, but does confer an advantage in terms of cabin width.

Provision in the cockpit was made for a flight crew of two. This was regarded as a standard arrangement for third level type operations but an alternative single pilot layout was prepared for executive use.

As originally designed the aircraft was powered by two 1400 HP Rolls Royce Gnome H 1400 shaft turbines. Each of these drove a 16 ft diameter propeller through reduction gearing mounted at the rear of the engine. A power offtake from the reduction gearbox was used to interconnect the two propellers, the whole arrangement being very similar to that employed for propeller interconnection on the Breguet 941 aircraft. A central gearbox on the cross shafting was provided for accessories and a drive for a small horizontal pair of contrarotating tail rotors. As the design study progressed it became apparent that whilst the propeller interconnection was valuable in alleviating control problems after failure of an engine in vertical or transition flight, the remaining engine was of insufficient power to maintain altitude when the forward speed was zero. It was therefore considered to be essential to revise the powerplant arrangement and replace each of the Gnome engines by a pair of interconnected smaller units, such as the Garrett TPE 331 shaft engine. This would confer four engine reliability without changing the configuration, as there would still be only two propellers, and also enable the aircraft to operate safely in the event of single engine failure.

The design vertical take off weight was 13,000 lbs of which some 4135 lbs was predicted to be available as disposable load. A maximum design weight of 15,000 lbs was used for short take off and landing operations. The disc loading in the vertical take off case was 32 lb/sq ft whilst with the wing area of 260 sq ft the wing loading was 50 lb/sq ft. The detailed predicted weight breakdown is given in Table 1.

The wing was hinged at the mid depth of the section 9.7 ins forward of the unswept 0.6 chord line. The wing incidence could be varied from +2 degrees to +102 degrees relative to the body datum. Kruger flaps were placed along the whole of the leading edge and 35% chord double slotted flaps positioned along the trailing edge inboard of the powerplant nacelles. The low mounted tailplane was of the all moving type capable of being operated over the range of 15 degrees down to 45 degrees up.

A conventional twin wheel main undercarriage was mounted from fuselage side beams and it was retracted inwards into a bay located below the cabin floor. The nosewheel retracted forwards into the fuselage nose cone.

3. CONTROL AND TRANSITION

During cruising flight the aircraft was controlled in a conventional way using the tailplane, rudder and ailerons. Whilst the aircraft is in vertical and transition flight modes the control forces were produced by a combination of rotor and aerodynamic surface effects. Roll control was achieved simply by differential use of the collective pitch change on the two propellers. Simultaneous collective pitch change used in conjunction with throttles was employed to control vertical motion. Longitudinally use was made of the contrarotating tail rotor in conjunction with overall collective pitch variation. When the propellers are in the horizontal plane the ailerons were aligned vertically beneath them so that they could be used for yaw control. As an alternative to this system consideration was given to rotation of the tail rotor about a fore and aft axis to give a lateral thrust component. This concept was not found to be essential. Tilting of the wing and hence the overall thrust vector gave fore and aft control.

Transition from hovering to forward flight was accomplished by a gradual tilt-forward of the wing to increase the forward speed. As the wing started to rotate towards the horizontal position the slipstream and forward speed effects enable a vertical component of aerodynamic force to be developed to offset the loss of vertical powered lift. This process was assisted by the programmed deployment of the leading and trailing edge flaps. The anticipated variation of wing-tilt angle as a function of transition speed is shown in Figure 3, whilst Figure 4 shows the flap and tailplane angle variation during the tilt process. The transition was effectively complete at 60 knots forward speed with the wing angle at approximately 12 degrees. At this speed there was still a very large beneficial slipstream effect on the wing which was some 16 knots below the estimated zero power stalling speed. During the wing tilt process the role of the ailerons was automatically transferred progressively from yaw to conventional roll control.

4. PERFORMANCE

The estimated vertical thrust during vertical take off was 14000 lbs, the margin over the take off weight being allowed for acceleration and control. The power installed to provide this vertical thrust was some 40% greater than would normally be expected for a conventional aircraft of comparable weight and role. Hence the forward speed performance was better although the effect of the extra power available was somewhat offset by a lower cruise propeller efficiency due to the compromise with take off performance. The usual cruise condition was predicted to be some 285 knots true airspeed at 20,000 ft altitude.

This is some 30 knots faster than the Handley Page Jetstream in similar conditions. The maximum normal cruising speed at low altitude and 9000 lbs was estimated to be 290 knots.

The payload-range characteristics for the 20,000 ft altitude cruise case is shown in Figure 5. The only reserve contingency allowed for is sufficient fuel for a second approach after a baulked vertical landing. Two curves are shown in the figure. One is for the basic case of a 13,000 lb take off weight condition with both vertical take off and landing. The other is for 15,000 lbs short take off case. In the basic VTOL condition a 2000 lb payload could be lifted over about 500 nautical miles still air range, no reserves. With short take off operations this could be increased to about 1100 nautical miles. The Handley Page Jetstream was designed to have a range performance between these two values. For example at 12500 lbs take off weight the still air range at 20,000 ft was predicted to be 700 nautical miles with 2000 lbs payload and a reserve allowance.

5. DESIGN CONDITIONS

The aircraft was designed to meet the requirements specified in British Civil Airworthiness Requirements. In the main the stipulations of Section D for fixed wing aircraft were applied, but where appropriate reference was made to Section G, Rotorcraft.

The maximum design weight of 15000 lbs in the STOL role was associated with a design diving speed, V_D , of 355 knots and a normal acceleration factor of 3.

The design cabin differential pressure was 6.5 lb/sq.in. Cabin floor loading of up to 200 lb/sq ft was stipulated to cater for use of the aircraft in the freight role.

For airframe and mechanical systems life evaluation it was assumed that the utilisation of the aircraft would total 20,000 hours over a period of 15 years. This is equivalent to 30,000 flights of 40 minutes average duration. Of these flights 10,000 were assumed to be in the full VTOL role and the remainder STOL or conventional operations.

6. STRUCTURAL DESIGN

The structural design was basically conventional, using light alloy materials except in certain localities. An indication of the general layout of the main structural members may be obtained by reference to Figure 6 which is a photograph of a one tenth scale cut away half model of the design.

6.1 Fuselage

It was found that several design cases contributed to the critical loading envelopes for the fuselage. As far as vertical bending is concerned the two point landing case was found to be critical over much of the length with a peak value of about 170,000 lb ft factored. As would be expected

the critical fin load case gave rise to a maximum factored bending moment of 180,000 lb ft associated with a shear force of 10,000 lbs. The maximum factored vertical shear force of 45,000 lbs arose in a pitch down case from a 3g manoeuvre at 15000 lbs weight and the cruise speed.

The fuselage frames had a nominal pitch of 20 ins although locally this was reduced to as low as 10 ins to suit the layout. The standard frames were 3.0 ins deep pressed channels in L72. Over much of the shell the skin thickness was 0.024 ins but this was increased locally to 0.032 ins in the region of the wing and main undercarriage attachments. The skins were supported by zed section stringers located at a basic pitch of 3.5 ins round the cross section. These were passed through and cleated to the frames. The cockpit floors were supported by a grid of channel section stiffeners. The cabin flooring used an end grain balsa sandwich of 0.4 ins depth with 0.028 ins thick face plates and was supported transversely by the frames and longitudinally by the extruded seat rails. The front pressure bulkhead was a flat panel reinforced by vertical back to back channel members. At the rear of the cabin there was a domed pressure bulkhead having a membrane thickness of 0.028 ins. Circumferential and radial crack stopping strips were incorporated into the design.

The nose undercarriage was attached to a structure located off the forward face of the frame at stn.224 ins forward of the datum which is at the wing 0.6 chord line. This structure consisted of vertical reinforcing members on the frame and fore and aft channel members along the inside of the lower skin at the edges of the bay cutout. The main pintle bearings were located at the intersection of the members whilst the drag strut was attached to the forward extremity of the longitudinals. The main undercarriage units were also attached to the fuselage, the pintles being located between forged and machined extensions from the bottom of the frames at stns.9.5 ins forward and 18.5 ins aft. The units retracted sideways into a bay located below the cabin floor.

The fin was attached at four points along the top of the rear fuselage. These were arranged in pairs, two on the frame at stn.201 ins aft and two at stn.233 ins aft. The tailplane pivot bearings were mounted between the second of these frames and another one some 4 ins aft of it. Cover plates across the insides of the tailplane frames completed a local box structure.

There were five specific attachment points between the wing and the fuselage. Two of these were the main hinge points which were located at the upper extremities of triangular shaped L65 forgings positioned across the undercarriage frames. The arrangement used is illustrated in Figure 7 which shows that the triangular forgings were mounted at the sides of the fuselage on channel section L65 longitudinal extrusions. Two more of the wing connection

points were the actuator attachments positioned towards the forward end of the longitudinals. The fifth point was a latch on the front spar which reacted lateral loads in cruising flight. The high wing was faired into the top of the fuselage in the conventional position and hence it was necessary to provide an inner, flat, pressure skin below it. This was constructed in 0.032 ins thick L70 and apart from the frames was supported by 1.63 ins deep longitudinal channels placed at 4 ins pitch across the width of the fuselage.

A design was undertaken to provide for the use of the aircraft in the freight role by incorporating a large door between stns. 80 ins and 145 ins aft. This had a maximum true depth of 65 ins at the front edge. It was intended to replace the 30 ins wide by 65 ins deep standard passenger door. The windows were slightly elliptical in shape and were located between every other frame. They were of double construction with 0.375 ins and 0.312 ins thick outside and inside Plexiglass panes respectively. The design of the pilots' windscreens varied according to their location. The front panels had a total thickness of about one inch, the main laminates consisting of a 0.4 ins thick chemchor glass panel backed by a 0.418 ins visual layer and having inner and outer faces of semi-toughened glass.

6.2 Wing

The bending moment and shear force case which was critical for much of the span was the 3g manoeuvre case at 15000 lbs weight and the cruising speed. The factored root values were approximately 200,000 lbs ft and 22,000 lbs respectively. The flap design speed was established to be 149 knots equivalent airspeed. This gave rise to factored loads of 2300 lbs, 3100 lbs and 1700 lbs on each segment of the trailing edge, inboard and outboard leading edge flaps respectively. Each aileron was designed for a load of 2500 lbs which occurred with maximum control deflection at 187 knots equivalent airspeed.

6.2.1 Structure

The 0.6 chord line of the wing was perpendicular to the fore and aft datum of the aircraft as can be seen by reference to Figure 8. The two hinge pivots were positioned 35 ins outboard of the centreline and 9.7 ins forward of the 0.6 chord datum. Vertically the hinges were located on the mid depth of the section. The structural layout of the inner portion of the wing is shown in Figure 9. The rib pitch was approximately 18 ins inboard increasing to 20 ins in the tip region. Although the wing was basically of 3 spar construction the rear spar was discontinued over the centre section. This was to give a rectangular cutout some 85 ins wide in the wing for clearance during tilting. The unusual stress distribution problems which resulted from this layout were the subject of a separate investigation by El-Bahaie, Reference 1. All the spars used plate webs with back to back extruded L65 angle booms. The webs were stiffened by vertical angles. Skin thickness varied from 0.08 ins inboard to 0.024 ins outboard. In the inner portion it was reinforced by bonded finger plates, and outboard extensive use was made of chemical etching. The skins were supported by zed section stringers placed at approximately 4 ins pitch across the chord.

The light ribs were pressed channel items which were located within the stringers. They were attached to both the skin and stringers by continuous cleat members. A pair of closely spaced ribs were used to react the hinge loads and these utilised the full depth of the section. At these sections the stringers were stopped off and the end load continuity was retained by means of forged dagger fittings which passed through the rib webs. This design was used partly because of the high torsional shears at these points and partly because the ribs were also tank ends. The other ribs in the centre section were built up of tubular construction. This was chosen to enable the inside of the single cell box portion of the wing to be easily inspected. The tilt actuators located between the pairs of hinged ribs and the actuator bodies were attached to the inside booms.

The wing hinge fittings were L65 forgings which were attached to the inside faces of the hinge ribs. The distribution of load into the ribs was assisted by providing angle cleats on the outside of the rearward extension of the ribs. Brackets for supporting the trailing edge flaps were attached to the rear face of the rear spar and the lower skin surface just forward of the rear spar. The flap shroud ribs were light pressings. The leading edge flap supports were mounted off the front face of the forward spar.

The whole of the front wing box, except the volume between the pairs of hinge ribs, was used as integral fuel tanks. Outboard of the nacelle the rear box was also used for tankage. Large stressed access panels were provided between alternate ribs in the lower surface of the wing. Access to the centre wing tank was through a single stressed panel in the centre of the front spar web.

The engine mounting was attached to the wing at four points, two each located off the front faces of the front and centre spars at Stns. 137 ins and 155 ins outboard from the centreline.

6.2.2 Wing actuators

The wing was tilted by a pair of ball screw actuators. These actuators were mechanically interconnected and driven by tandem hydraulic motors. Details of the actuator design are shown in Figure 10. Interesting features were the very long stroke which meant that the screw had to pass right through the body, and the tie rod intended to give tensile integrity to the screw should a failure occur. As can be seen in Figure 7 the screw was attached to the fuselage and the body to the wing, this being necessary for clearance reasons.

6.2.3 Leading edge flaps

Kruger type flaps were located along the greater part of the leading edge. They were hinged at each rib station. Structurally each segment consisted of a 0.028 ins thick

L72 closed box with two internal spanwise channel section members. These spanwise members were interrupted by the pressed channel section riblets which carried the simple hinge pins. The flaps were actuated by means of spanwise push-pull rods which were connected to each of the flap hinge ribs by oblique rods.

6.2.4 Trailing edge flaps.

Double slotted flaps of 35% chord were fitted along the trailing edge inboard of the powerplant nacelles. The nose slat was fixed relative to the main flap and the whole unit was simply hinged about a point approximately 14 ins below the trailing edge,

The main flaps were of three cell construction with ribs placed at about 6 ins pitch. The spars and ribs were pressed channel sections. A thickness of 0.036 ins in L72 was used for the front spar and forward skin whilst that of the rear spar and skins was 0.022 ins. The slat was similarly constructed in 0.022 ins L72 with an extruded L65 trailing edge piece.

Each segment of the flap was hinged at two points on the spar. A pair of back to back ribs were used to attach the 0.064 ins thick L72 box section hinge arms. The hinge bearings and actuation attachments employed local fittings which were machined in L65. The hinges were supported off the main wing structure by 0.08 ins box section L72 members. Actuation was by synchronised hydraulic jacks placed across the hinge support brackets and hinge arms.

6.2.5 Ailerons

Conventional construction was used for the ailerons with a single pressed channel section spar located just aft of the hingeline. The normal pitch of the pressed ribs was 5.0 ins and much of the component was fabricated in 0.022 ins thick L72. The leading edge of the control surface was sealed to the wing rear spar by a flexible membrane and provision was made for a distributed mass balance. The hinge loads were found to be relatively small and the hinge fittings and support brackets were largely constructed of pressed angles and channels in 0.028 ins thick L72. Each aileron was hinged at two positions and operated from the inboard one. A spring tab system was incorporated. The tab was a simple box construction in 0.22 ins thick L72 hinged along its whole length. The spring tab occupied approximately the inner half of the aileron span, the outer half being allocated to a trim tab.

6.3 Tailplane

The net factored tailplane design load of 10,000 lbs occurred during a rolling pitch up from zero g at the design weight and diving speed. An all moving surface was used.

The structure consisted of a three cell box although the major load carrying capacity was concentrated between the two spars. The general layout of the structure is shown in Figure 11. The centre cell employed corrugated reinforcing for the 0.024 ins thick skins. An identical thickness was used for the 0.7 ins deep corrugations which had 0.4 ins wide flats and were bonded to the skins. Additional reinforcing plates were incorporated in the root region adjacent to the attachment of the centre pivot tube. The spars used a built-up construction with 0.048 ins L73 plate webs and back to back angle booms. Angle section vertical stiffeners were placed on the web. Rib pitch in the centre box varied from about 14 ins inboard to 23 ins outboard. The nose and rear cells were of lighter construction using 0.028 ins and 0.022 ins thick L72 skins respectively. These were supplemented by light pressed ribs at about 6 ins to 12 ins pitch with intermediate angles near to the root.

The pivot tube was continuous across the fuselage. It had a nominal maximum diameter of 5.0 ins and wall thickness of 0.1 ins and was machined from an S96 forging. Local thickening was arranged at the actuator arm attachment, the bearing locations and connections points. The latter were arranged to coincide with two rib positions on each half of the tailplane. The attachments to these ribs were splined nut plates which like the operating arm were machined from S96 forgings. The bearing housings were L65 machinings fitted across the pair of tailplane mounting frames. Reinforced p.t.f.e. was employed for the bearing surface.

Both the root attachment ribs were of built-up construction. The inboard one of the pair was Y shaped, one arm being perpendicular to the pivot tube and the other running along the side of the fuselage. The full depth plate web was 0.048 ins thick L72 with angle section booms on the inside face. The skin corrugations terminated at the outer face and were connected to the rib by angles formed in 0.064 ins thick L72. Additional 0.064 ins thick reinforcing plates were located around the cut outs for the pivot tube to spread the load from the splined nut plates. A similar design was used for the outer attachment rib except that the skin corrugations passed over the angle booms.

6.4 Fin

The critical design fin and rudder load of 9500 lbs, factored, occurred when the aircraft reached the overswing angle of 18.8° following instantaneous rudder application at 385 knots at 15000 ft altitude. The corresponding maximum

rudder load was 1250 lbs. The maximum lateral acceleration at the tail was 0.6g.

6.4.1 Fin structure

The fin structure consisted of a primary single cell box located between the two spars and a subsidiary nose cell. The primary cell ribs were placed at approximately 20 ins pitch with the nose and rear shroud riblets at half this value. Connection to the fuselage was by four pins at the lower extremities of the spar booms. Two of these were aligned fore and aft and two laterally. The main cell skins were 0.028 ins thick L72 and they were supported by 1.05 ins deep zed section stringers placed at 4 ins pitch. Both the spars used a built-up construction with plate webs and back to back booms consisted of nested angle sections. The rear spar web was stiffened by a spanwise angle located on the centre of the depth which was also used to attach the rudder nose sealing fabric. Pairs of S96 machined fork fittings directly bolted onto the booms were used for the fin attachments. A channel section was employed for the pressed ribs, the stringers being passed through but cleated to them. Additional angles located below the stringers were incorporated into the two rudder hinge ribs. The leading edge was of similar, but somewhat lighter construction.

6.4.2 Rudder

The rudder was hinged at three points, two on the fin and the lower one on the top of the fuselage. A single spar construction was used with nose riblets at 4.75 ins pitch and rear cell ribs at double this. The thickness of these ribs varied from 0.048 ins at the root to 0.022 ins at the tip. The nose cell skin was 0.022 ins thick and that of the rear cell 0.018 ins, L72. Stiffening channels were placed between the rear cell ribs. The lower hinge consisted of a torque tube attached to the root rib by a spool fitting and mounted in a pair of fuselage supported bearings. A quadrant attached to its lower end was used for the control connection. The other two hinges employed machined fittings attached to appropriate ribs. A trim tab was provided over approximately the lower third of the span. It was operated through the centre of the lower hinge tube. The nose of the rudder was sealed to the fin rear spar and a distributed mass balance was incorporated.

6.5 Main undercarriage

A two point landing gave rise to the main undercarriage design vertical and side loads of 20,000 lbs and 5100 lbs, factored, respectively. The maximum drag load was 16500 lbs and this occurred in a high drag loading.

Each of the two main undercarriage units carried twin wheels on a single axle and retracted sideways and inboard into a bay in the underside of the fuselage. The units were mounted between side extensions of two fuselage frames placed 28 ins apart. A general arrangement of one of the units is

shown in Figure 12. The upper section of the leg was a double triangular forging in L65, the top horizontal member being a pintle. Both the side members and the top members had a rectangular section with their junction region adapted to carry the 1.5 ins diameter steel pivot pins. The tubular central vertical member formed the outer tube for the oleo pneumatic shock absorber. The inner sliding tube was fabricated in FV520 steel and incorporated the axle mounting at the lower end. This latter item was flash butt welded to the tube, and was designed to include the brake attachment lugs. The torque links were L65 forgings. The wheels were split aluminium alloy castings and provided housings for the single plate disc brakes.

The side strut was built in two sections. The upper part was an L62 tubular structure of triangular shape, its inner extremities incorporating the hinge pins. The pins located in bearings mounted in the main support frames. The outer vertex of the triangle consisted of an S96 fitting which connected to the other rectangular part of the sidestrut which was shown in S96. The downlock was incorporated in the pivot between the two parts of the sidestrut. It was held in position by dual springs and the retraction jack. The latter was connected between the downlock and the fuselage structure in such a way that the initial movement released the lock. The uplock was suspended from a bracket off the fuselage structure and engaged with a boss on the main leg casing.

Hinged doors mechanically connected to the main leg partially closed the undercarriage bays when the units were retracted.

6.6 Nose undercarriage

Although the maximum factored drag load of 13000 lbs occurred in a high drag landing the nose undercarriage design vertical and side loads of 16000 lbs and 4000 lbs respectively arose during a three point landing.

Figure 13 is a general arrangement drawing of the nose undercarriage unit. Retraction was forwards into a bay in the nose fuselage, a special structural unit being provided for the mounting. The twin wheels were carried on a live axle.

A Y-shaped light alloy forging was used for the upper part of the main leg. The steel pivot pins were positioned in the extremities of the arms. The oleo-pneumatic shock absorber was a three tube design. The central tube was mounted in bearings and used to transmit the steering torque as well as act as the outer element of the shock absorber. Both this tube and the inner, sliding one were machined from S99 forgings. The axle bearing housing is an S96 forging fitted over the bottom end of the sliding tube and locked in place by a double nut. The torque links were L65 forgings. The axle itself was in hard chrome plated S98.

The down lock was incorporated in the drag strut. The upper part of the strut has a vee geometry and was forged in light alloy whilst the lower part was two separate L65 links. The lock on the pivot between them was held in position by a pair of springs located in single sided jacks. Hydraulic pressure applied to both jacks was used to unlock the unit against the spring force. The retraction jack was connected from the main leg casing to the structure on the forward face of the mounting bulkhead. A hydraulic actuator was also used for the nosewheel steering, the mechanism being located in the space between the arms of the main casing. The uplock was mounted in the roof of the bay. It was locked by spring force and unlocked either hydraulically or manually direct from the cockpit.

7. POWERPLANT AND TRANSMISSION

7.1 Powerplant (Gnome H1400)

The Gnome H1400 engine installation used in the initial concept of the design was the one on which most of the detailed investigation was undertaken. The powerplant was an adaptation of the basic unit employed in helicopter application. A similar arrangement to that employed for the E67 was proposed as the P1200 variant for STOL applications and used in the C166 counter insurgency project study, Reference 2. An outline of the arrangement can be seen by reference to Figure 14. The Gnome has a rear offtake for the power shaft from an aft turbine. In the case of the E67 and the P1200 proposal the drive was taken vertically upwards through spur reduction gearing before being taken vertically upwards through spur reduction gearing before being taken forwards by shafting to an epicyclic nose reduction box and the propeller output shaft. A bevel gear pair at the upper end of the first stage of reduction was used to provide a drive for a lateral cross shaft. Provision was also made in the gearbox for reversing the direction of rotation of the propeller, so that a contrarotating pair was used. It was considered that H1400 version of this system was a straightforward development of the P1200 variant and apart from initial layout of the rear gearbox no detail work was undertaken.

7.2 Transmission

Apart from the drives from the aft turbines of the powerplants to the associated propellers the transmission consisted of the spanwise interconnecting shaft, tail rotor drive and the associated gearbox.

7.2.1 Cross shafting

The spanwise cross shafting ran at 6000 R.P.M. It was designed to transmit 50% of the emergency power from one engine on a 10 hour life basis and 5% of the normal power on a 20,000 hour life basis. These figures were chosen to cover the emergency case of an engine failure and the normal power balancing condition respectively. The shafting was located behind the centre spar of the wing and ran parallel to the 0.6 chord line, along the line of the hinge. It passed through the centres of the hollow pins and was supported

in roller bearings at these points. Whilst the shaft was basically 2 ins outside diameter steel tubing it became a solid 1 ins diameter steel bar at the wing hinge bearing locations. Hooke's type universal joints were provided at the engine gearbox connections and inboard of the wing hinge bearings where there was also a sliding joint.

7.2.2 Centre gearbox

The cross shaft between the wing hinge bearings passed through the centre gearbox. The layout of this item is shown in Figure 15. It was mounted off the upper fuselage decking. As can be seen the cross shaft is not broken but has one element of a pair of bevel wheels mounted on it. The bevel pair was primarily used to drive the tail rotor shaft, but consideration was also given to the possibility of having accessory drives at this location. This would have had the merit of being independent of an engine failure but in the event the standard engine mounting of accessories was retained. The gearbox was a magnesium alloy casting and the gears were fabricated in a case hardening steel.

Just aft of the centre box there was a mechanical clutch and a small supplementary friction clutch. The latter was intended to spin the tail rotor drive shaft up to operating speed to enable the mechanical, power transmission clutch, to be engaged.

7.2.3 Tail rotor drive and gearbox

The tail rotor drive was at 4000 R.P.M. The shafting was located along the top of the cabin, outside the pressure shell in a special fairing. Behind the rear pressure bulkhead it passed into the fuselage and thence to the tail gearbox.

The arrangement of the tail gearbox is shown in Figure 16. Double bevel gears were used to transmit the torque into the contrarotating shafts and a three bearing support design was needed to react the tail rotor bending loads. Provision was made in the design of this box for a rotation of the output about the fore and aft axis of the aircraft. This was intended to enable the tail rotor to be used to provide yaw as well as pitch control, but this feature was not adopted in the final design. Components of the box were similar to those in the centre gearbox.

7.3 Auxiliary Power Unit and Power Supplies

The auxiliary power unit was installed in the rear part of the starboard engine nacelle, as can be seen in Figure 14. The unit, a Saurer GT-15 turbine, was intended to be flight rated and provide electrical power either in a flight emergency or for ground running. A tubular substructure supported the unit from the rear of the wing structure.

The primary auxiliary supplies for the aircraft were derived from alternators and hydraulic pumps located on the main powerplants. This was the standard arrangement on the Gnome H1400 and was used in preference to drives on the centre gearbox.

7.4 Powerplant Mounting and Nacelle

The mounting structure for the powerplant was also used to support part of the nacelle fairing structure.

7.4.1 Mounting

The whole of the powerplant inclusive of the reduction gearbox and forward shafting was installed in the airframe as a single unit. As can be seen from Figure 14 the unit was suspended from two sets of three points. These were located in pairs on the front and rear gearboxes and the gas generator. A tubular framework was provided to connect these points to the wing box. This was constructed from T45 steel tube of approximately 1.5 ins diameter with welded end fittings. The connections between the tubes occurred at part frames and the wing centre spar. At these locations use was made of machined L65 fittings. The rear gearbox was suspended from the framework by forged L65 swinging links. A simple semi-circular channel pressing, 0.104 ins thick in L73 was used for the front part frame. The centre one was in the shape of an inverted U and was built-up with a plate web and boom angles in 0.08 ins thick L73. At the wing spar the connection was made on a pair of L65 vertical channel forgings.

7.4.2 Nacelle

The nacelle fairing structure was built up in a number of sections, some of which were removable. The forward upper portion, which extended as far back as the wing front spar, was attached to wing and engine mounting structure. It consisted of light pressed channel frames and zed section stringers supporting a 0.028 ins thick L72 skin.

The lower front portion included the engine air intake. A construction similar to the upper portion was used. This part was removable by disconnection of the bolts attaching it to the lower edges of the fixed top fairing. There was also a small removable fairing at the aft end of the powerplant in the region of the downward directed exhaust pipe. The whole of the rear section was also removable to give access to the auxiliary power unit and other equipment located in this volume. Construction was generally similar to that of the fixed front section.

The nacelle was completed by a pair of doors which gave access to the underside of the engine over its whole length. These were continuously hinged off the lower edges of the fixed fairing and joined together by quick release fasteners placed along the lower centreline of the nacelle. The doors were of double wall construction in 0.036 ins thick L72.

7.5 Engine Installation

The various accessories, including power supplies, were all located on the underside of the engine. Access to them was through the nacelle doors. A 0.015 ins thick stainless steel horizontal firewall was placed below the wing box in the fixed nacelle structure, and a further section of it isolated the rear nacelle volume from the exhaust region.

Powerplant removal necessitated the opening of the doors and removal of the lower forward and exhaust sections of the nacelle fairing. Access to the mounting points was then possible and after disconnecting these and the services the whole unit could be lowered away.

7.6 Front Powerplant Design (Garrett TPE 331)

Although a detail study of this proposal was not undertaken it is visualised that the basic layout would be unchanged. Each of the pairs of smaller engines would be located side by side in the nacelle and be coupled into a forward reduction box. In this arrangement the propeller output shaft would be taken from the top of this gearbox, as would a rear facing output for the cross shafting. The rear gearbox would be mounted off the rear face of the centre wing spar and would simply be a bevel unit. Geometrically the lower nacelle would be somewhat wider.

7.7 Propellers and Tail Rotor

It was envisaged that the 16 ft diameter, four blade main propellers would be conventional except for an unusual blade pitch layout to give acceptable static thrust performance.

The 7 ft diameter tail rotor consisted of a pair of two blade contrarotating elements. It was anticipated that each of these elements would be designed along conventional propeller practice.

8. SYSTEMS

8.1 Fuel system

A relatively simple fuel system was used. The schematic arrangement of which is shown in Figure 17. In all there were six integral wing tanks, two in the centre wing, two between the hinges and nacelles and two outboard. Basically each engine was fed from the appropriate collector tank, which was the outboard one in both cases. The two inboard tanks on either side fed the adjacent collector although provision was made for cross feed between the two sets. Engine supply was by means of twin electric booster pumps in each collector. These were located at the outboard end of the tanks. Fuel from the inner tanks was transferred by means of jet pumps driven by the main supply boosters and a further pair of boosters located at the inboard ends of the middle tanks. All feed points were located at the bottom aft ends of the tanks so that they were as low as possible in all wing tilt positions. Similarly the vent pipes were positioned along the front upper corners of the tanks. Gravity loaded vent valves were provided as were clack valves in certain of the ribs to cater for wing tilting and acceleration conditions. The vent pipes terminated in vent surge tanks in the wing tips.

Each tank could be refuelled independently or through a common pipe system. Fuel contents used a capacitance type system which was calibrated primarily for the cruise role of the aircraft. This was done as it was assumed that the time in the vertical mode would be relatively short and the crew would evaluate the situation either before take off or prior to transition for landing.

8.2 Cabin environmental system

The cabin environmental system was not designed in detail. It was envisaged that the system would use air tapped from the main powerplants for normal usage. During ground operating conditions air conditioning would be electrically driven from the auxiliary power unit supplies.

8.3 Flying control system

The different modes of operation of the aircraft from vertical to cruise flight through transition resulted in certain complications in the flying control system relative to a conventional aircraft. The tailplane was hydraulically operated but otherwise the controls were manual.

8.3.1 Cockpit controls

In the standard two seat cockpit layout the main flying controls were operated by interconnected pairs of floor mounted rudder pedals and push-pull type control columns located below the instrument panel. The latter had control stick extensions rather than wheels. Engine and other subsidiary controls were located on a console placed between the two pilots' seats.

8.3.2 Control runs

The main control runs in the fuselage and the runs to the aileron and rudder tabs used cables whilst the primary aileron control in the wing employed push-pull rods. In the case of the tailplane and rudder the cables were run under the floor in the bottom of the fuselage and then up the tailplane mounting frame to the tailplane booster valve and lower rudder hinge. The primary rudder cables were connected to a quadrant attached to the end of the hollow hinge pin. An automatic cable tensioner was incorporated in the quadrant. The rudder trim tab circuit terminated on a sprocket mounted on the nut of an irreversible screw actuator which in turn was supported on a pair of bearings within the hollow hinge pin. The screw operated light push-pull rods within the rudder and connected to the tab.

The aileron control cables in the fuselage passed from below the pilots' floor up the bulkhead behind the crew compartment and thence over the cabin ceiling to the wing hinge region. The transition from fuselage cables to wing push-pull rods was by means of a laterally sliding scissors linkage. The movement of the scissors was such as to cater for the wing-tilt movement, the control movement being lateral. The links were supported off the fuselage decking on the one hand, and the rear face of the wing centre spar on the other. In the wing the push rods were supported by a system of levers pivotted off the rear face of the spar. The rods were connected to a lever pivotted in the aileron and also connected to the aileron through a spring box. The other output from the lever was to the spring tab.

8.3.3 Aileron roll-yaw interconnection

A mechanical device was designed for changing the mode of aileron control from yaw to roll and vice-versa during transition. It was located below the pilots' floor between the outputs from the control columns and rudder pedals and the autopilot capstans. Details of the arrangement can be seen in Figure 18. Basically it consisted of three chain sprockets mounted on a single shaft. The centre sprocket, which connected to the aileron itself, was fixed to the shaft laterally, although free to rotate on it. The other two were free to move laterally within limits determined by sprung stops. One of these carried the input from the rudder pedals and the other that from the control column lateral movement and a propeller pitch control output. Hydraulic rams placed at either end of the shaft enabled it to be moved laterally so that the output sprockets to the aileron moved laterally relative to the two inputs. The faces of the output sprocket were provided with a cam surface and corresponding surfaces were incorporated on the inside faces of the other two. During vertical flight the rudder pedal input was engaged with the aileron sprocket whilst in horizontal flight the control column was engaged with it. Likewise in vertical flight the control column movement operated the differential propeller pitch but this was disconnected for horizontal flight at another point in the system. During transition the shaft was moved laterally by means of a hydraulic servo-system controlled by the wing tilt angle to give the appropriate sequencing of control.

8.3.4 Tailplane actuation

A major difficulty associated with the operation of the tailplane was the large angular motion associated with the wing tilting during transition. The normal control function was superimposed upon this. This difficulty was overcome by using two actuators in series, as can be seen in Figure 19. The normal control function was achieved by means of a hydraulic booster unit, with manual reversion for emergency operation. The signals for this unit were provided from the control column through

the control cables. The output from the booster was not connected directly to the tailplane, but to the end of a lever pivotted off the rear face of the tailplane mounting frame. An irreversible screw actuator was connected between the lever and the tailplane operating arm to give a four bar chain mechanism moved by the hydraulic booster. The screw was moved by a hydraulic motor controlled by a position feedback from the wing tilt angle to cause it to adjust the geometry of the four bar chain and give the necessary tailplane movement.

8.3.4 Flaps and flap interconnection

The wing leading and trailing edge flaps were hydraulically operated through a push-pull rod signalling system to the selector valves. Manual selection was provided in the cockpit, both sets of flaps being operated by a single lever. There was also an automatic override which functioned during the wing tilt phase. This consisted of a cam mechanism driven from the wing which automatically locked the manual lever in the 30° position required at the end of transition and then took over the output to the valve.

8. EQUIPMENT AND INSTALLATIONS

Both the two pilot and single pilot cockpit layout were studied in detail. Whilst there was undoubtedly an advantage in sharing the work load between members of flight crew the automatic operation during transition did mean that it was possible to consider a single pilot and no great difficulty was found in laying out the cockpit for him. The proposed layout of the main instrument panel is shown in Figure 20. Standard format and instruments were used.

The avionics fit provided for was comprehensive. It included Decca navigator or doppler, DME, ADF, ILS, twin radio altimeters, duplicated navigation and communication VHF, a weather radar with nose mounted scanner and gyro compass. The navigation VHF, gyro compass and single channel autopilot provided signals to the flight director. Most of the avionic equipment was housed in racking located in the nose fuselage above the wheel bay.

10. MARKET SURVEY AND OPERATIONAL ANALYSIS

A study of the economic aspects of the concept was made by Brown, Reference 3, as a separate investigation. This was based largely on a series of case studies of executive aircraft operations interpreted in terms of the potential of the E67. The performance and operating cost of the design was evaluated relative to the use of the car, the twin turboprop aircraft, the twin jet aircraft and the helicopter. The general results of this investigation are of interest.

On a time basis the E67 had a better block time than the car for distances above about 30 miles and better than the twin jet below about 200 miles. The latter figure was conservative and assumed relatively favourable airfield locations for the jet. The E67 was always faster than the twin turboprop and helicopter.

The pure cost comparison showed a somewhat different trend since the car was always very much cheaper, although when a reasonable allowance was made for the value of time to the executive the E67 looked promising for journeys of more than about 50 miles. It was estimated that the operating cost of the E67 would be about £130 per hour (1969 values) which was some two thirds of that of a twin jet, three quarters that of a helicopter of comparable size but one and one third that of the twin turboprop aircraft. As a result on an overall cost benefit basis the E67 showed well against the jet aircraft for distances of up to about 400 miles, and was cheaper than the turboprop up to about 250 miles.

11. DISCUSSION AND CONCLUSIONS

11.1 Discussion

Although at first sight the E67 design appears to be unconventional a more detailed consideration shows that in most respects it employs accepted fixed wing or rotorcraft techniques. However there are some aspects of the concept which are worthy of further comment.

11.1.1 Transition aerodynamics

A theoretical evaluation of the aerodynamic characteristics of the design was undertaken as part of the project investigation. Whilst this study did not reveal any unusual difficulties the general interference problem associated with the interaction of the propeller slipstream and forward speed with the flow over the lifting surfaces is such that the results must be treated with caution. There is no doubt that this is the design feature which would require the most effort during development, especially as it must rely largely on experimental investigation.

11.1.2 Engine failure

There is no doubt that the original twin engine version of the aircraft is unsatisfactory in that the failure of an engine during vertical flight results in a loss of ability to maintain altitude. The proposed cross shaft coupling between the propellers eliminates many of the control problems which might otherwise arise. When each of the engines is replaced by a pair of smaller ones a single engine failure can be tolerated and the design requirements for the interconnecting shafts are eased. Recent provisional requirements for powered lift aircraft suggest that the case of two engine failure would need to be considered. Perhaps it would be best

to consider the E67 design relative to a twin engine helicopter which can cope with a single engine failure due to the lower disc loading employed. The four engine version of the E67 should have at least the same order of safety as this, especially bearing in mind that the conventional propeller presents a less severe problem than a low disc loading rotor system.

11.1.3 Tailplane movement

One of the unfortunate complications in the design was the need for the double hydraulic actuation of the tailplane. The primary reason for this was the large movement required in addition to the normal control function during transition. More recent work has suggested that if the tailplane is placed at the top of a tall fin it is possible to keep it out of the more severe flow region. In this case the large movements can be eliminated and the normal control function suffices. An immediate simplification results and indeed by reverting to a conventional fixed tailplane with a large elevator the need for a power control might be eliminated completely.

11.1.4 Wing tilt failure

Duplicated ball screw actuators were proposed for wing tilt operation. Nevertheless the possibility of a failure or jamming of the mechanism must be considered. Such an event need cause no more than the embarrassment of a diversion of the aircraft to an alternative airfield. The propeller ground clearance when the wing is in the cruise position is such that a conventional landing can be made. STOL landings with the wing in any position between the cruise and vertical position also present no insuperable difficulties.

11.1.5 Tail rotor and tail drive shaft

The need to provide the tail rotor for vertical and transition pitch control is a further complication. A possibly simpler solution would be to replace the tail rotor and drive shaft by a reaction control nozzle and ducting. A detailed study would be necessary to see if this arrangement would be preferable to the mechanical system but one problem associated with it would be the air offtake required from the powerplants. A further alternative would be to introduce cyclic pitch control to the main rotors, thereby eliminating the need for an additional pitch control and giving a yaw control capability to the rotor as well. The control system as such would be simpler at the expense of replacing the conventional propellers by either semi-articulated or non-articulated rotors. In as much as the tail rotor concept used accepted mechanical engineering practice it was preferred to the cyclic pitch arrangement with its associated rotor development problems. The alternative could, however, prove to be lighter.

11.1.6 General design concept

As envisaged the E67 used accepted design techniques. The only unusual feature requiring extensive development would be the behaviour during transition flight. The mechanical aspects of the design, such as the gearing and shafting, followed established rotorcraft practice and in many respects are similar to the arrangement used in the Breguet 241 STOL aircraft. The structural problems of the wing tilt are less than those associated with a retracting undercarriage. Whilst some improvement in performance might result from alternative means of control during transition this would only be achieved at the expense of a bigger development programme.

11.1.7 Operating characteristics and potential

The advantages of the concept of matching vertical take off capability with good cruise performance have been shown to be substantial when applied to business operations, (Reference 3). It would seem that the E67 has a number of characteristics which make it preferable to a helicopter with which it is perhaps best compared. Apart from the role as an executive aircraft it is possible to suggest a number of other applications, such as supply to offshore oil rigs and light military transport. The Canadair CL-84 and its developments are in the latter category and are very similar in design concept. Progress with the development of these aircraft could pave the way to a pressurised version for civil use.

11.2 Conclusions

1. A four engine arrangement is essential for civil operation of this class of aircraft. In this configuration the safety of the aircraft should be at least as good as that of a twin engine helicopter.
2. The major development problem associated with the aircraft would be the transition aerodynamic performance.
3. A simplification of the tailplane actuation should result by locating it at the top of a tall fin.
4. Whilst alternatives to the mechanical tail rotor drive can be suggested it is likely that these would introduce more complexities in practice.
5. There appear to be significant advantages for a tilt wing aircraft of this size relative to both the helicopter and the jet executive type. Whilst executive applications are particularly promising, other applications are probable.

REFERENCES

1. EL-BAHAIE, E.A. Diffusion from a two cell to a single cell box.
Cranfield MSc Thesis, Sept. 1969.
2. HOWE, D. Aircraft Design studies - Counter Insurgency aircraft with suction boundary layer control.
Cranfield Rep. Aero. No.1, April 1970.
3. BROWN, D.M. The E67 VTOL aircraft considered as a transport system for business use.
Cranfield Thesis, June 1968.

ALLOCATION OF COMPONENTS FOR E67 STUDY

Bray, D.J.	Leading and trailing edge flaps
Brown, D.M.	Cockpit and cabin layout - single pilot operation.
Caesley, R.	Tail cone structure.
Clifford-Jones, J.B.	Rear fuselage - passenger version
Cox, P.J.	Main undercarriage
Farnsworth, J.	Ailerons and rudder
Harris, C.M.	Powerplant installation.
Kidd, J.	Transmission
Kyi, M.	Control system
Matthews, L.J.	Outer wing and aft nacelle structure
Nassar, E.E.	Nose undercarriage.
Pitt, A.J.	Centre wing
Robson, B.	A.P.U. installation and power supplies.
Rowlands, J.A.	Cockpit layout - two pilot operaton.
Rusbridge, P.J.	Fuel system
Stevens, M.B.	Nose fuselage structure
Swan, B.E.	Tailplane
Thirkettle, A.J.	Rear fuselage - freight version
Thorpe, J.	Fin
Webb, J.	Centre fuselage.

APPENDIX B

SPECIFICATION

1. WING

Gross area	260 sq ft
Span	40 ft
Aspect ratio	6.15
Sweepback of quarter chord line	2° approx.
Sweepback of 0.6 chord line	0°
Centreline chord (constant out to 3.5 ft)	7.75 ft
Tip chord (nominal)	5.0 ft
Standard mean chord	6.5 ft
Aerofoil section	NACA63 ₃ 218
Wing-body angle, basic setting (centreline chord to body datum)	2°
Wing movement, relative to basic setting position	0° to 100°
Dihedral	-0.75°
Location of 0.25 S.M.C. position forward of 0.6 chord line	2.3 ft
Location of 0.6 chord line relative to fuselage nose	24.67 ft

2. TRAILING EDGE FLAPS

Type:- Double slotted

Flap chord (aft of hinge line)/wing chord	0.35
Flap angle:-	65° max
Inboard end of flap from aircraft centreline	3.54 ft
Outboard end of flap from aircraft centreline	11.25 ft

3. LEADING EDGE FLAPS

Type:- Kruger

Flap chord/wing chord	0.1
Flap angle	140°
Inboard end of inner flap section from aircraft centreline	3.54 ft
Outboard end of inner flap section	11.25 ft
Inner end of outer flap section	13.21 ft
Outer end of outer flap section	19.6 ft

4. AILERONS

Type:- Set back hinge, sealed internal balance.

Aileron chord (aft of hinge line)/wing chord	0.25
Balance chord (forward of hinge line)/aileron chord (aft of hinge line)	0.33
Aileron movement	20° up 16° down

Ailerons ctd.

Inboard end of aileron from aircraft centreline	13.21 ft
Outboard end of aileron from aircraft centreline	19.6 ft

5. TAILPLANE

Gross area	90 sq ft
Net area	74.5 sq ft
Span	20 ft
Aspect ratio (gross)	4.44
Sweepback of trailing edge	0°
Centreline chord (nominal)	5.5 ft
Tip chord (nominal)	3.5 ft
Aerofoil section	NASA 63 ₂ 015
Tailplane movement	45° up 15° down
Vertical location of tailplane relative to fuselage datum (zero setting)	0.77 ft
Distance of trailing edge aft of 0.6 chord line	22.4 ft
Tail volume coefficient (gross)	1.22

6. FIN

Nominal area (above fuselage datum)	70 sq ft
Net area	52 sq ft
Height above fuselage datum (nominal)	11.0 ft
Aspect ratio (based on nominal dimensions)	1.73
Root chord (on fuselage datum)	10.0 ft
Tip chord (nominal)	2.92
Sweepback of leading edge	42°
Aerofoil section	NASA 63 ₂ 015
Distance of leading edge on datum aft of 0.6 chord line	12.35 ft
Fin volume coefficient (nominal)	0.153

7. RUDDER

Type:- Set back hinge, sealed internal balance.

Rudder chord (aft of hinge line)/fin chord	0.3
Balance chord (forward of hinge line)/fin chord (aft of hinge line)	0.33 (at root)
Height of rudder root hinge above fuselage datum	2.9 ft
Movement	±10°

8. FUSELAGE

Overall length	52.5 ft
Nominal maximum diameter (width)	7.0 ft
Maximum depth	8.5 ft
Length of passenger cabin (including entrance, etc.)	25.5 ft

Fuselage ctd.

Height of passenger cabin	6.0 ft
Maximum width of passenger cabin	6.5 ft
Fuselage geometry and sections - see Fig. 2	

9. NACELLES

Overall length (including spinner)	13.15 ft
Maximum width	2.0 ft
Maximum depth	3.65 ft
Location of extreme nose forward of wing 0.6 chord line	7.67 ft
Location of nacelle centreline outboard of aircraft centreline	12.17 ft
Location of nacelle datum above fuselage datum	2.50 ft

10. UNDERCARRIAGE

Type:- Nosewheel

Wheelbase	19.0 ft
Track	8.17 ft
Design vertical velocity (proof)	10 ft/sec

Main undercarriage units

Type:- Twin wheels (side retracting)

Tyres:- 24 inches diameter by 6 inches wide

Tyre pressure	70 p.s.i.
Track of twin wheels	0.82 ft
Shock absorber closure (approximate)	1.0 ft
Tyre closure (maximum)	0.3 ft
Location of axle aft of 0.6 chord line	0.37 ft

Nosewheel

Type:- Twin wheel (forward retracting)

Tyres:- 20 inches diameter by 5 inches wide.

Tyre pressure	70 p.s.i.
Wheel track	0.75 ft
Tyre closure	0.25 ft
Location of axle forward of 0.6 chord line	18.7 ft

11. POWERPLANTS AND TRANSMISSION

Type:- 2 Rolls Royce Gnome H1400 shaft turbines
(1400 H.P. sea level static)
Or 4 Garrett TPE 331-205 (715 H.P. sea level
static)

Considerable variation is possible with different combinations of payload and fuel.

13. AERODYNAMIC INFORMATION

Surface and control characteristics

(a) Zero slipstream, conventional configuration

Maximum lift coefficient

Basic wing	1.45
Increment due to leading edge flap	0.27
Increment due to trailing edge flap at 30°	0.53
Increment due to trailing edge flap at 65°	0.81

Drag polars:-

Cruise configuration at 20,000 ft altitude 285 knots T.A.S.

$$C_D = 0.0315 + 0.062C_L^2$$

Low speed configuration, flaps down, undercarriage extended, zero slipstream:-

$$C_D = 0.162 + 0.062C_L^2 + 0.057\Delta C_L^2$$

(where ΔC_L is increment in C_L due to flaps)

Pitching moment coefficient at zero lift (basic wing + body)

-0.07

Increments in pitching moment about 0.25 S.M.C

Due to leading edge flap	+0.02
Due to trailing edge flaps at 30°	-0.067
Due to trailing edge flaps at 65°	-0.101

Location of wing-body aerodynamic centre forward of 0.6 chord line

2.3 ft

Wing no lift angle relative to chord line (basic wing)

-1.5°

Slope of wing lift curve, a_1 :-

Basic wing	4.3
With leading edge flap	4.55
With leading and trailing edge flaps	4.55

Two dimensional ratio of aileron lift curve slopes $(a_2/a_1)_0$

0.51

Slope of aileron hinge moment due to wing incidence:-

Two dimensional, b_{10}	-0.22
Actual, b_1	-0.20

Slope of aileron hinge moment due to aileron angle:-

Two dimensional, b_{20}	-0.55
Actual, b_2	-0.39

Rolling moment coefficient due to aileron angle, l_ξ

-0.14

Slope of tailplane lift curve, a_{1T}

3.8

Tailplane pitching moment coefficient at zero lift

0

Location of tailplane aerodynamic centre forward of tailplane trailing edge	3.3 ft
Slope of fin and rudder lift curve:-	
Net area, a_{1F}	2.03
Gross area, with body effects, a_{1FB}	2.86
Ratio of rudder lift curve slopes:-	
Two dimensional, $(a_{2F}/a_{1F})_0$	0.62
Actual, a_{2F}/a_{1F}	0.65
Slope of rudder hinge moment due to fin incidence:-	
Two dimensional, $(b_{1F})_0$	-0.24
Actual, b_{1F}	-0.02
Slope of rudder hinge moment due to rudder angle:-	
Two dimensional, $(b_{2F})_0$	-0.52
Actual, b_{2F}	-0.41
Mean rate of change of downwash angle at tail with wing incidence, $d\epsilon/d\alpha$	0.4

(b) Slipstream effects

The slipstream may be considered to be uniformly distributed over the whole span of the wing and tailplane, with negligible effect on the fin and rudder in non-yawed flight. When slipstream effects are present the surface and control data are therefore factored directly by the appropriate value of \bar{S} or \bar{S}_T as shown in Figure 21.

Stability Derivatives, Conventional Configuration

Rolling moment coefficient due to rate of roll, l_p :-	-0.42
Rolling moment coefficient due to yawing, l_r :-	$-0.14 + (0.14 + 0.23\bar{S})C_L$
Rolling moment coefficient due to sideslip, l_v :-	$-0.21 + (0.13 + 0.05\bar{S})C_L$
Side force coefficient due to sideslip, y_v :-	-0.15
Yawing moment coefficient due to sideslip, n_v :-	+0.1
Yawing moment coefficient due to yawing, n_r :-	$-0.47 - 0.03\bar{S}C_L^2$
Tailplane rolling moment coefficient due to sideslip, K	+0.15

Reference areas

All derivatives are based on the quoted areas and are per radian unless otherwise stated. The rudder control derivatives are based on net fin and rudder area, but the stability derivatives are based on gross fin area.

TABLE 1
WEIGHT BREAKDOWN

Component	Weight	% A.U.W.
Fuselage	1510	10.0
Wing	985	6.6
Tailplane	260	1.7
Fin	150	1.0
Nacelles and engine mounting	420	2.8
Main undercarriage	460	3.1
Nose undercarriage	100	0.7
Structure	3885	25.9
Engines	740	4.9
Engine gearboxes and propeller shafts	490	3.3
Engine accessories	140	0.9
Engine controls	40	0.3
Intakes and exhausts	90	0.6
Centre gearbox	35	0.2
Rear gearbox	30	0.2
Interconnecting shafts	70	0.5
Main propellers	660	4.4
Tail rotors	65	0.4
Powerplant and transmission	2360	15.7
Control system	260	1.7
Fuel system	180	1.2
Power supplies and APU	490	3.3
De-icing	60	0.4
Fixed cabin furnishing	260	1.7
Instruments and avionics minimum fit	230	1.5
Basic empty weight	7725	51.5
Crew and emergency equipment	400	2.7
Passenger seats (18) and removable trim	740	5.0
Operating empty weight (18 passengers)	8865	59.2
Passenger payload, typical	3100	20.6
Fuel	1035	6.9
VTOL operating weight	13000	86.7
Passenger payload, typical	3100	20.6
Fuel	3035	20.2
STOL, all up weight	15000	100

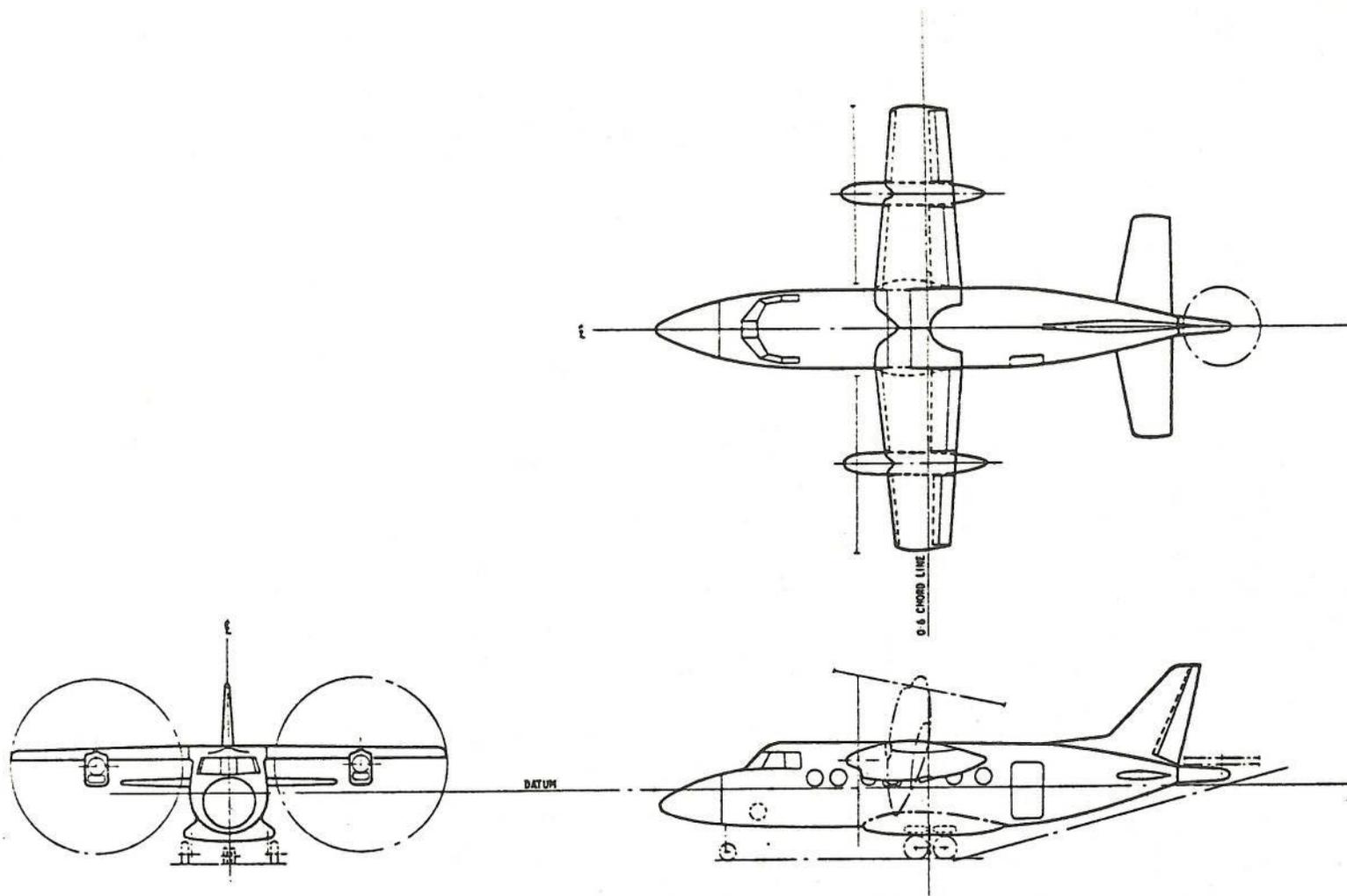


FIG. 1. GENERAL ARRANGEMENT OF E67 DESIGN.

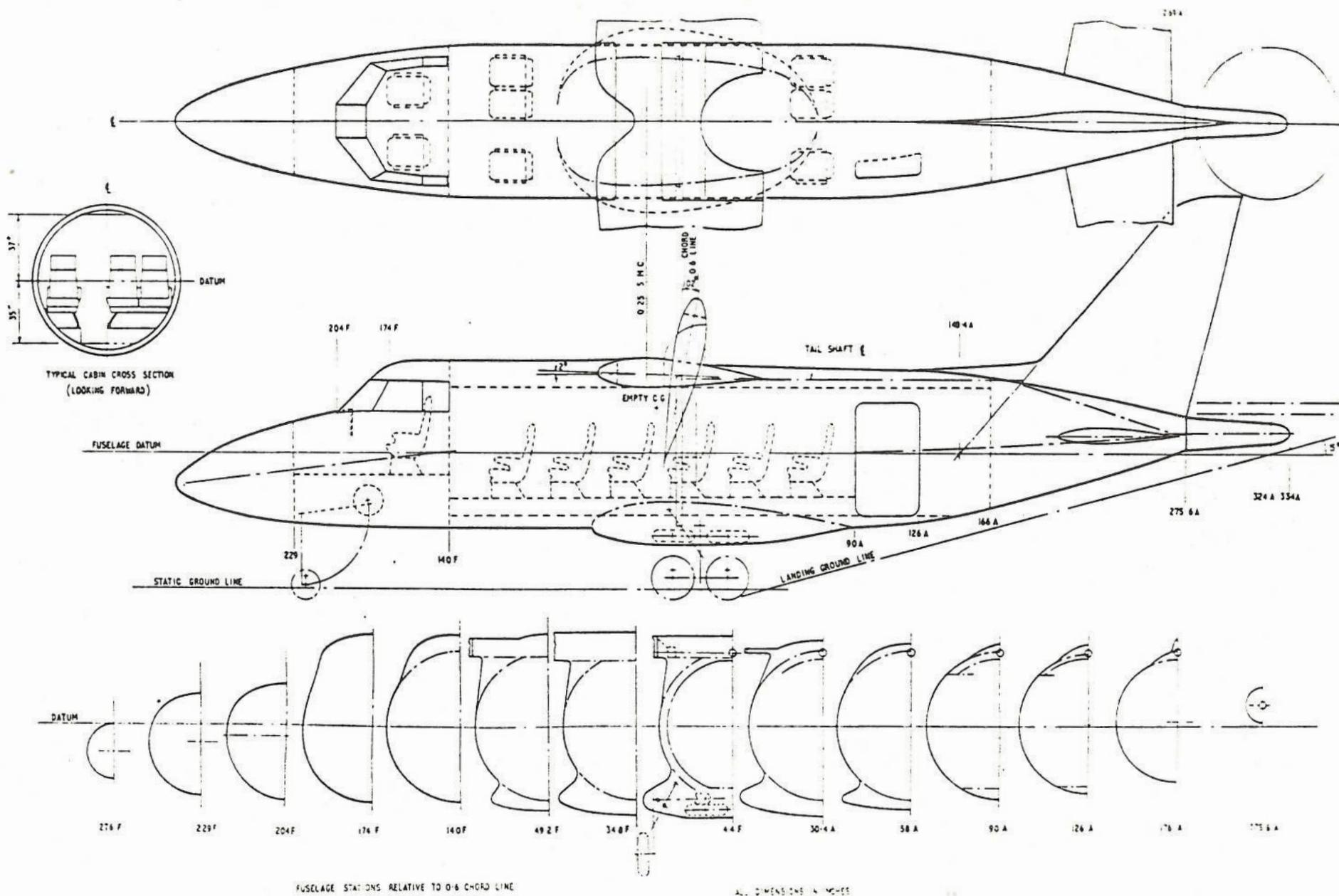


FIG. 2. FUSELAGE LAYOUT

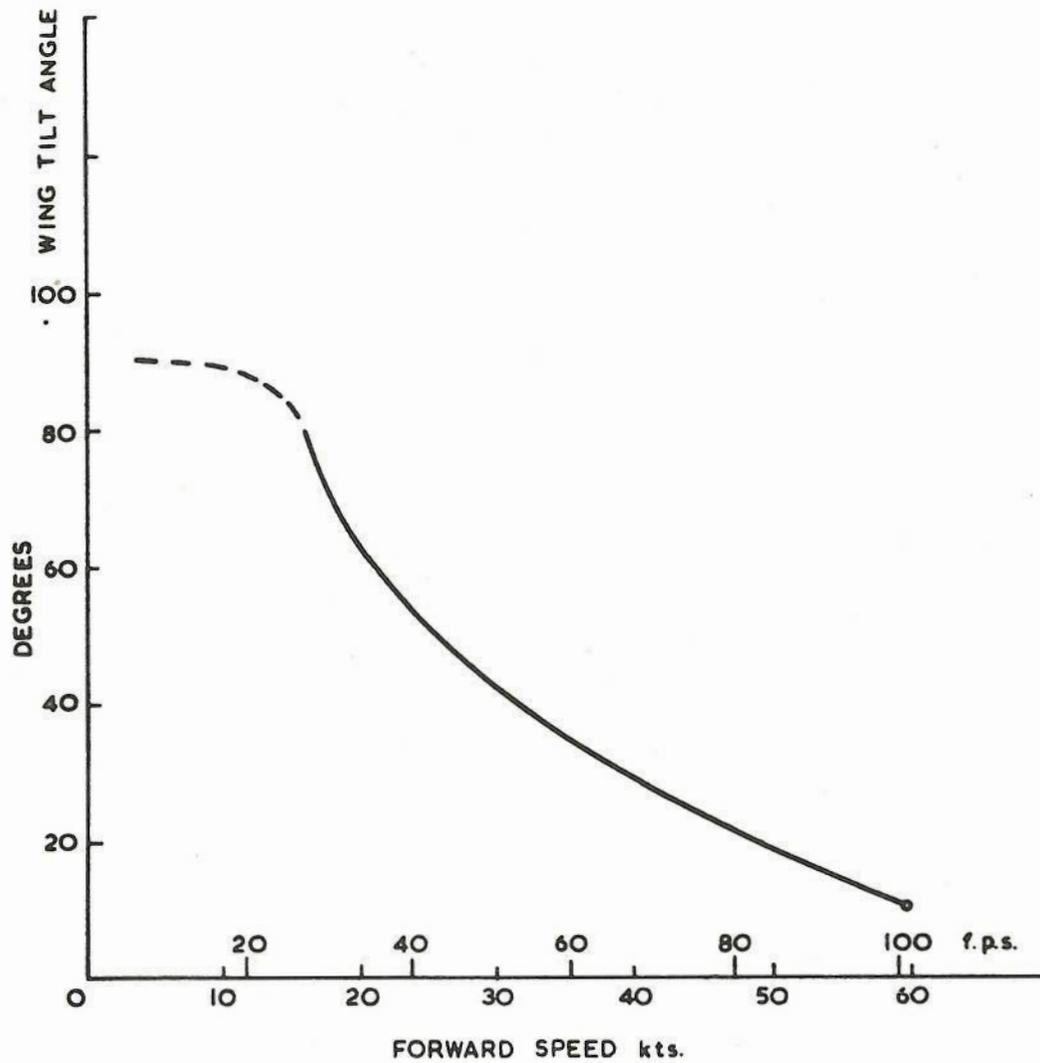


FIG. 3. VARIATION OF WING TILT ANGLE WITH TRANSITION SPEED.

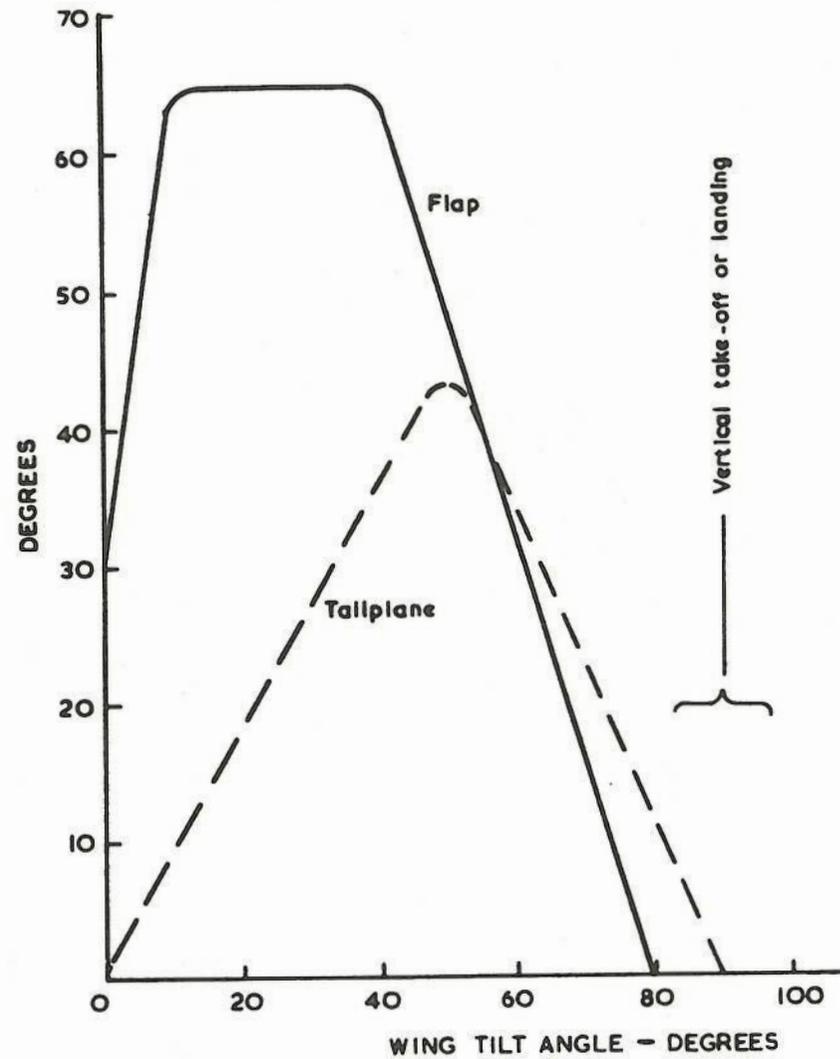


FIG. 4. VARIATION OF TAILPLANE AND FLAP ANGLES DURING WING TILT.

Cruise 285 kts. at 20,000 ft. altitude
I.S.A. conditions, max. continuous power.
No reserves except for bailed landing contingency
Passenger aircraft, operating empty weight 8865 lb.

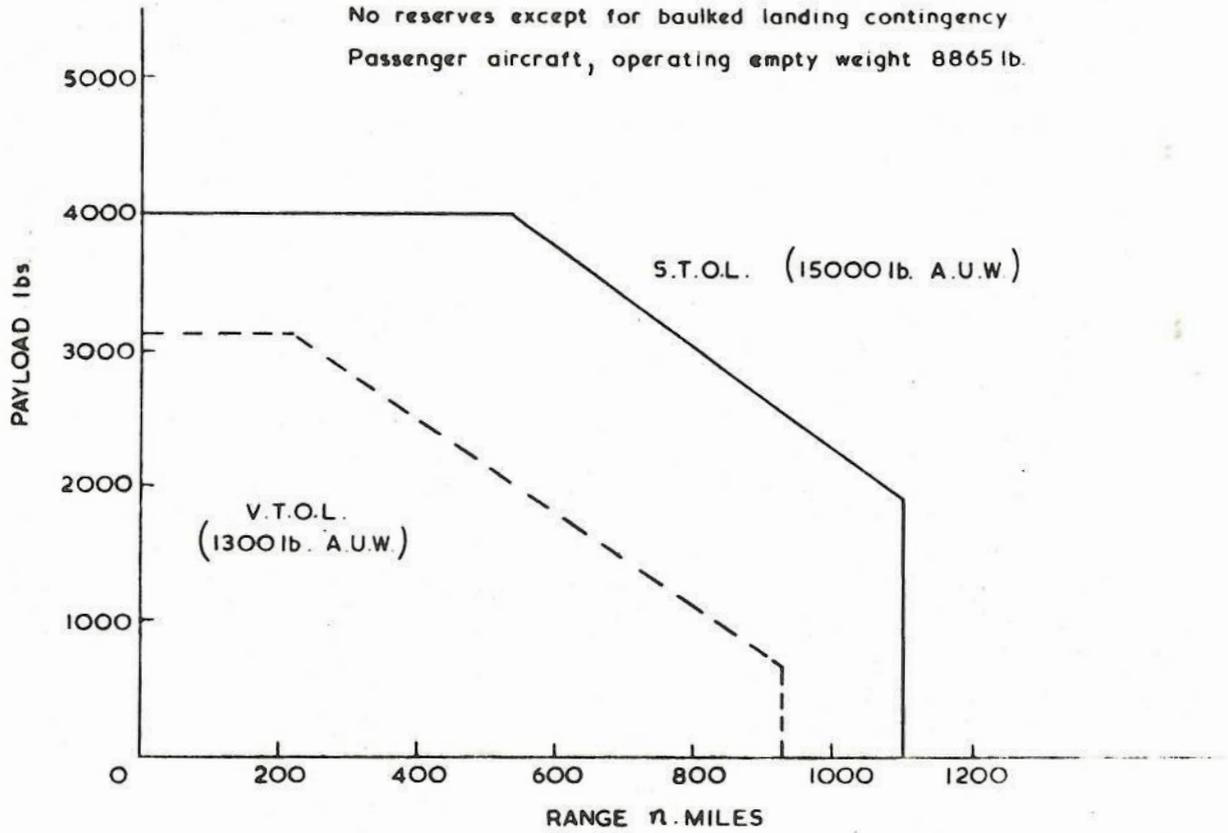


FIG. 5. PAYLOAD RANGE DIAGRAM

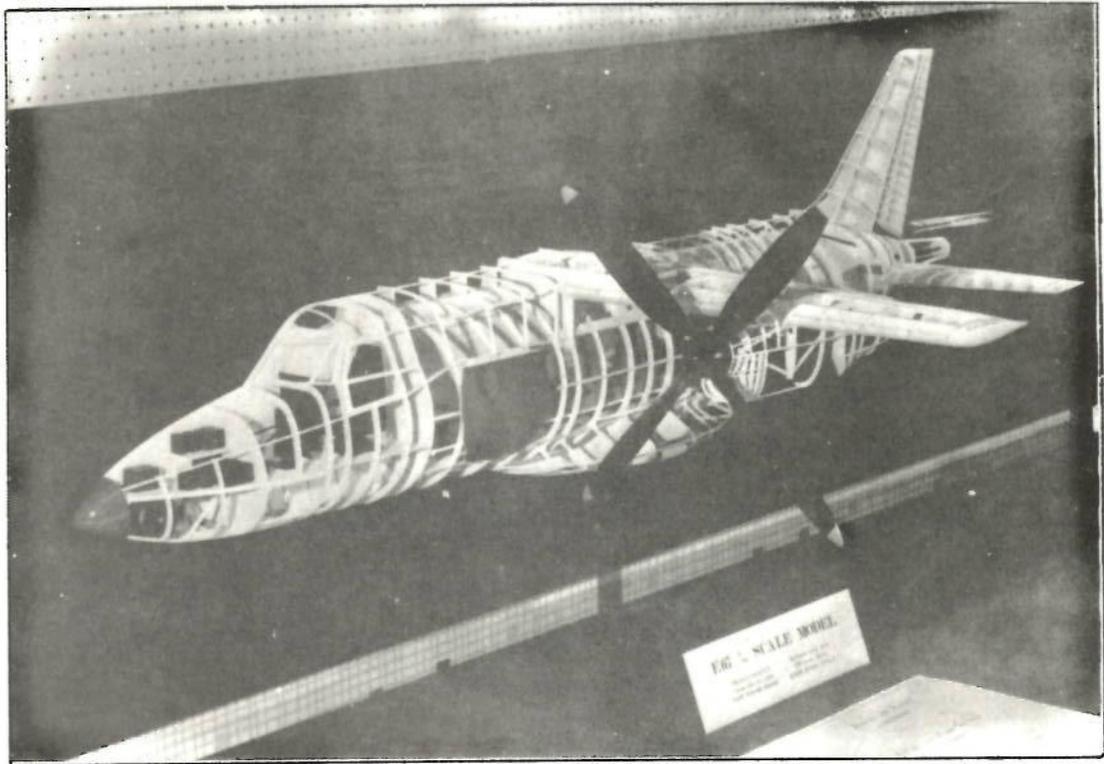


FIG. 6. PHOTOGRAPH OF CUT AWAY MODEL OF E67

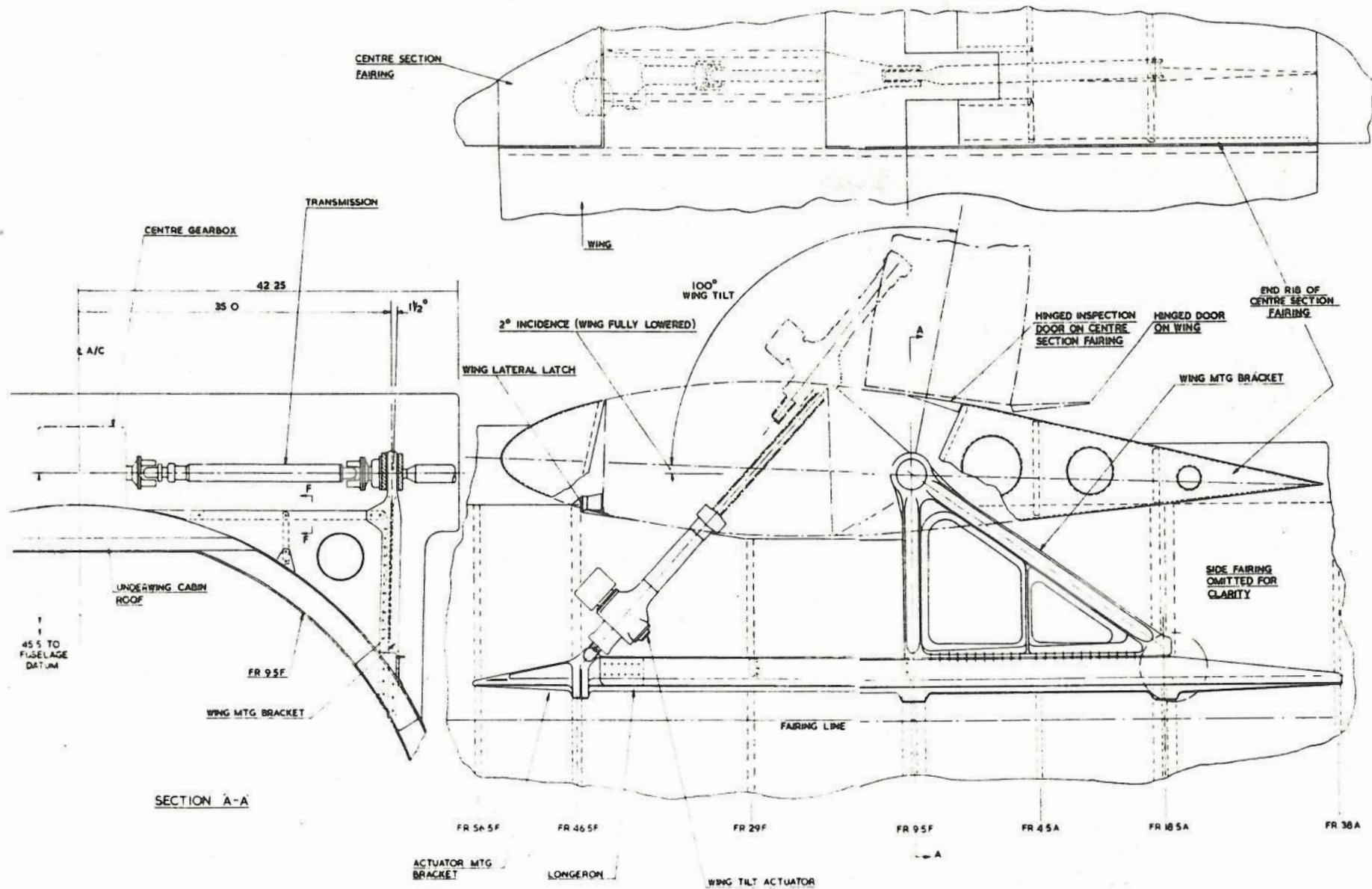


FIG. 7. STRUCTURE AND MECHANISM OF WING TILT

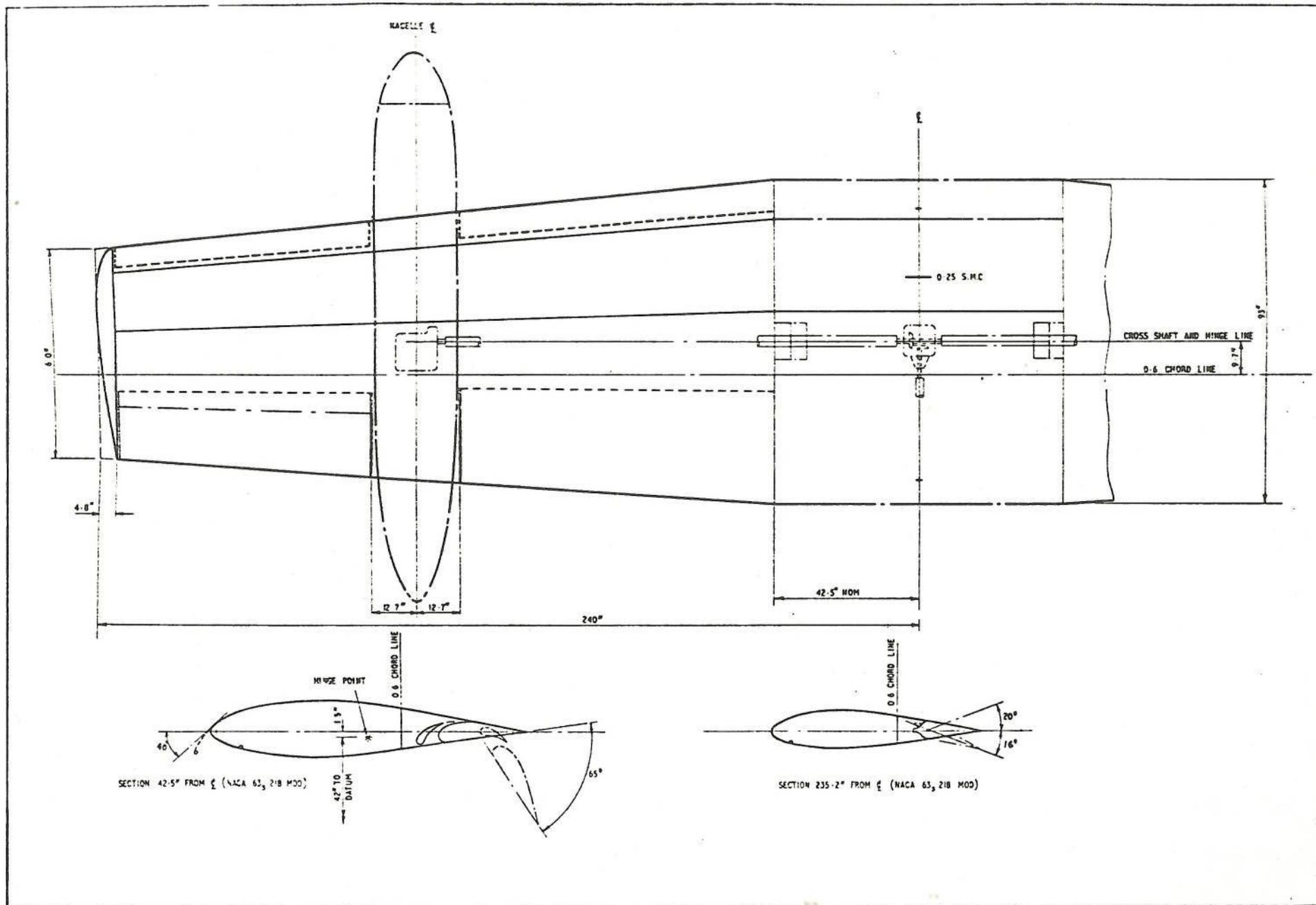


FIG. 8. LAYOUT OF WING

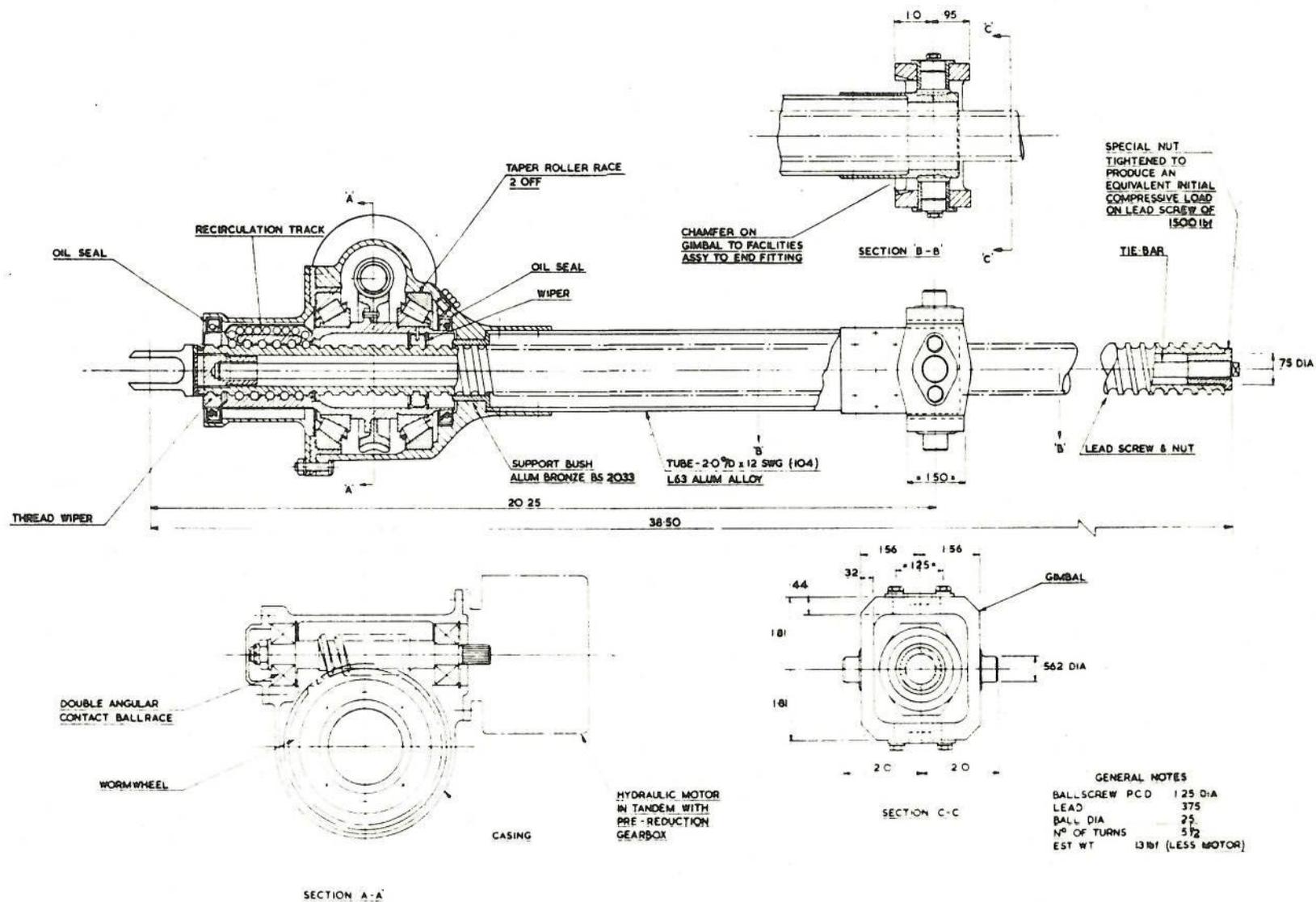


FIG. 10. WING TILT ACTUATOR.

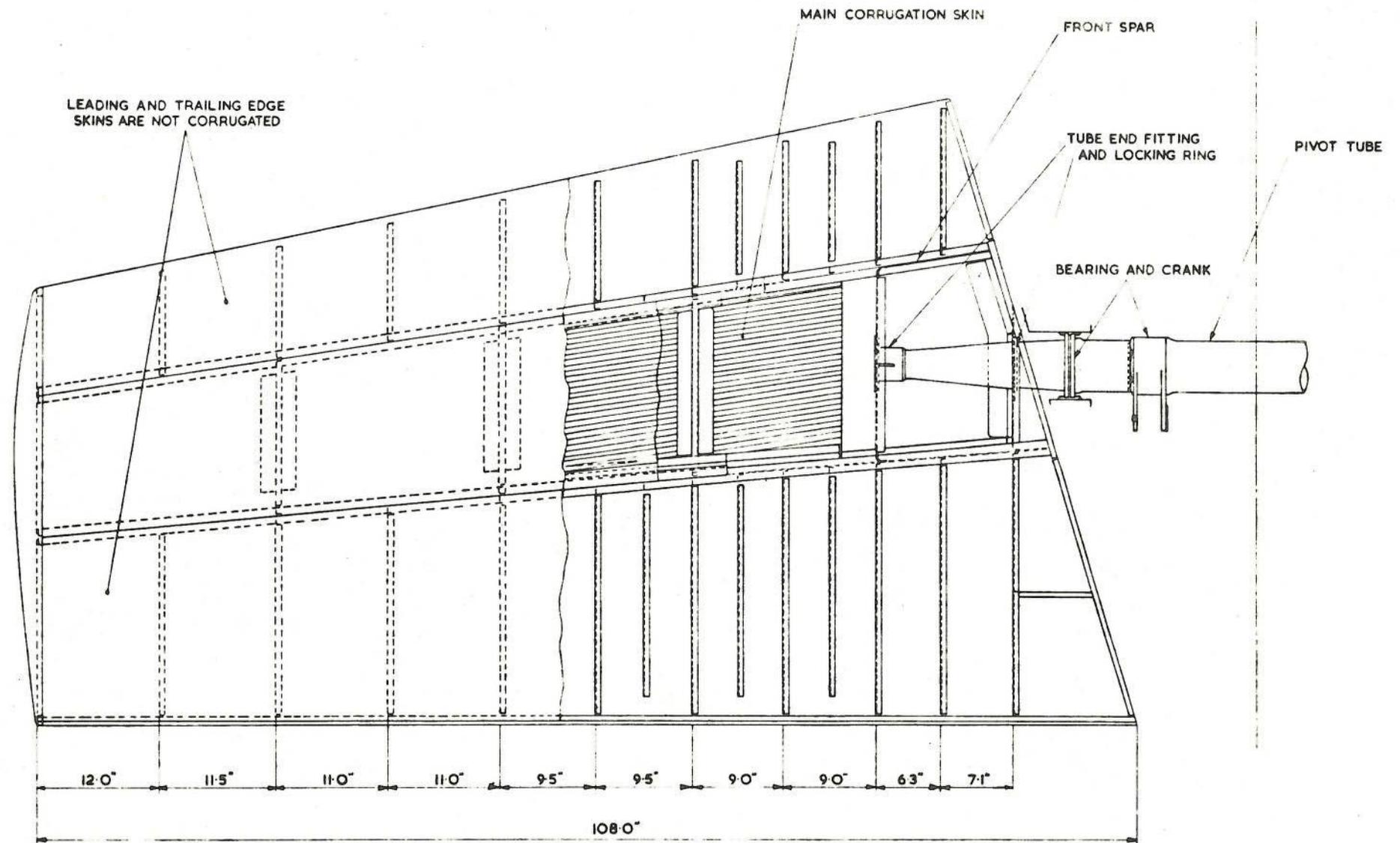


FIG. 11. TAILPLANE STRUCTURE.

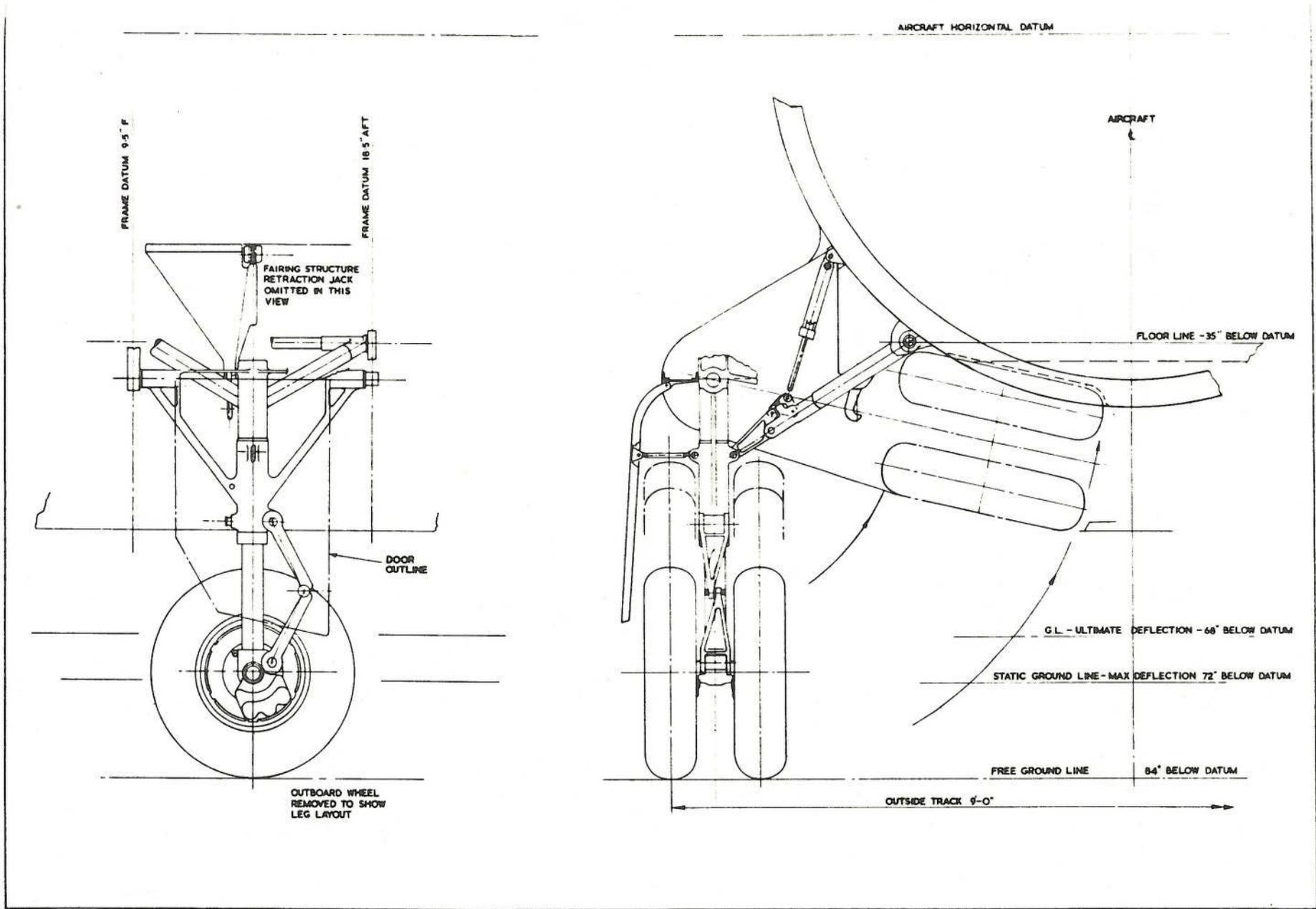


FIG. 12. MAIN UNDERCARRIAGE UNIT

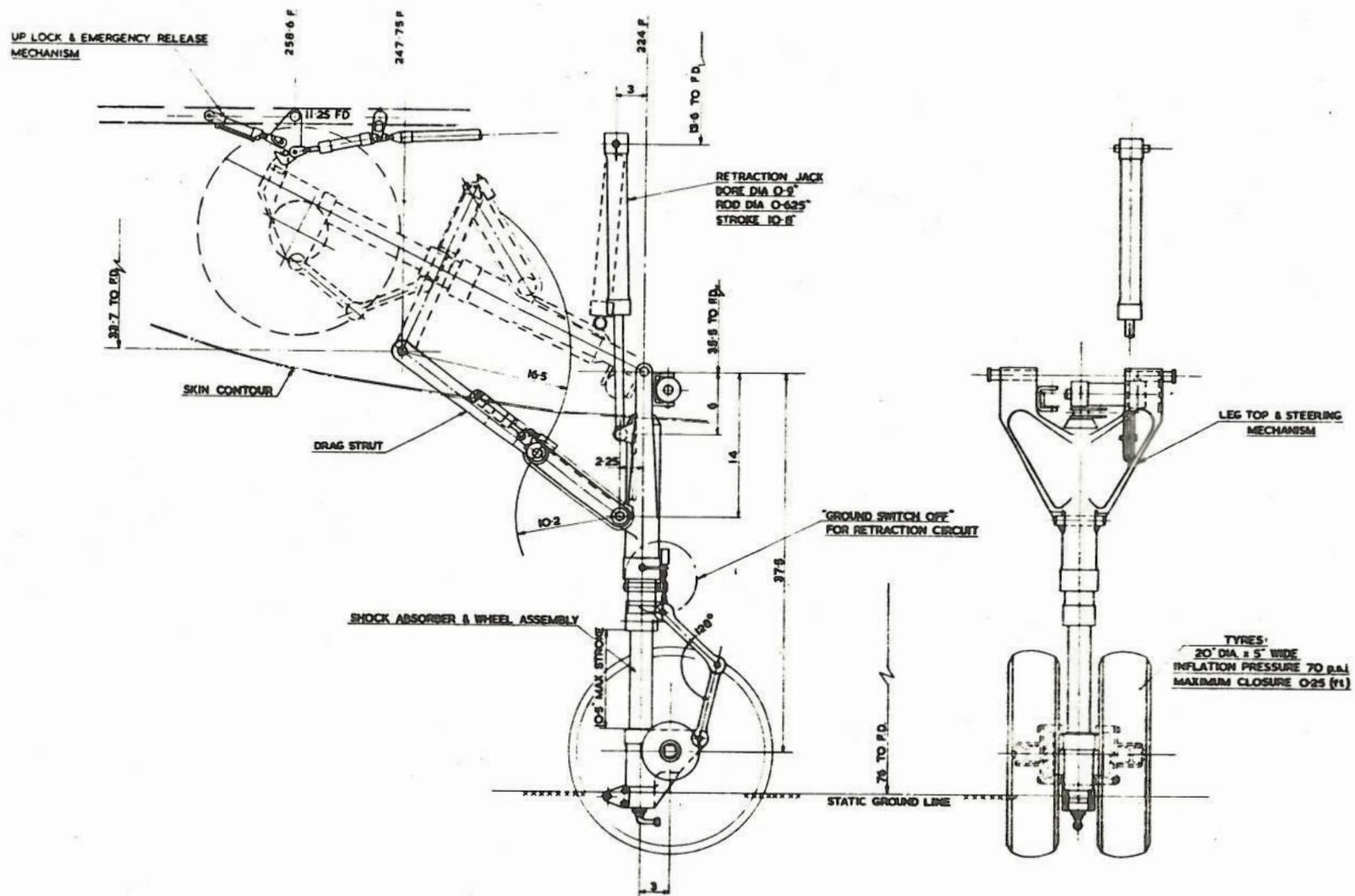


FIG. 13. NOSE UNDERCARRIAGE UNIT.

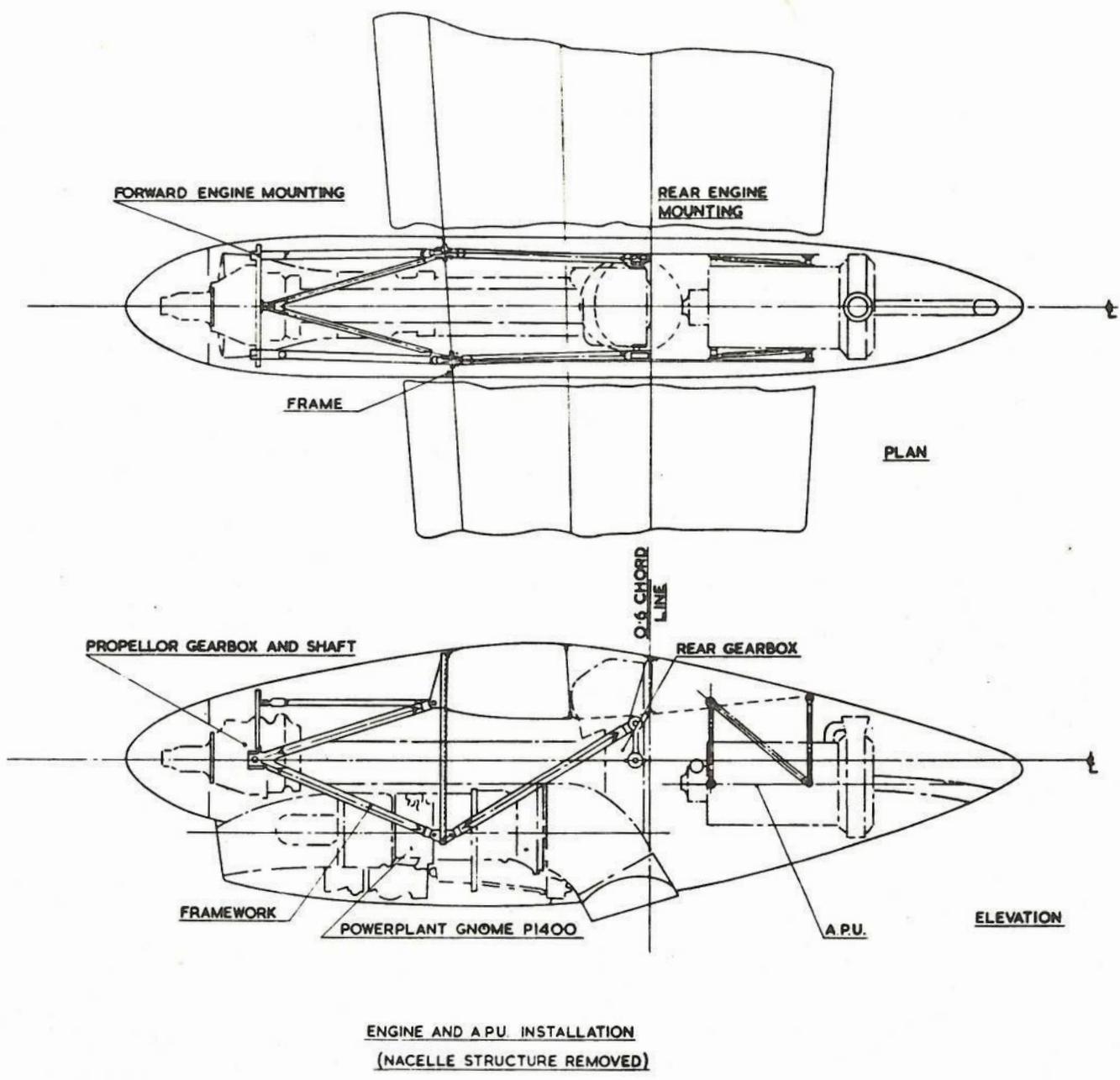


FIG. 14. POWERPLANT AND NACELLE

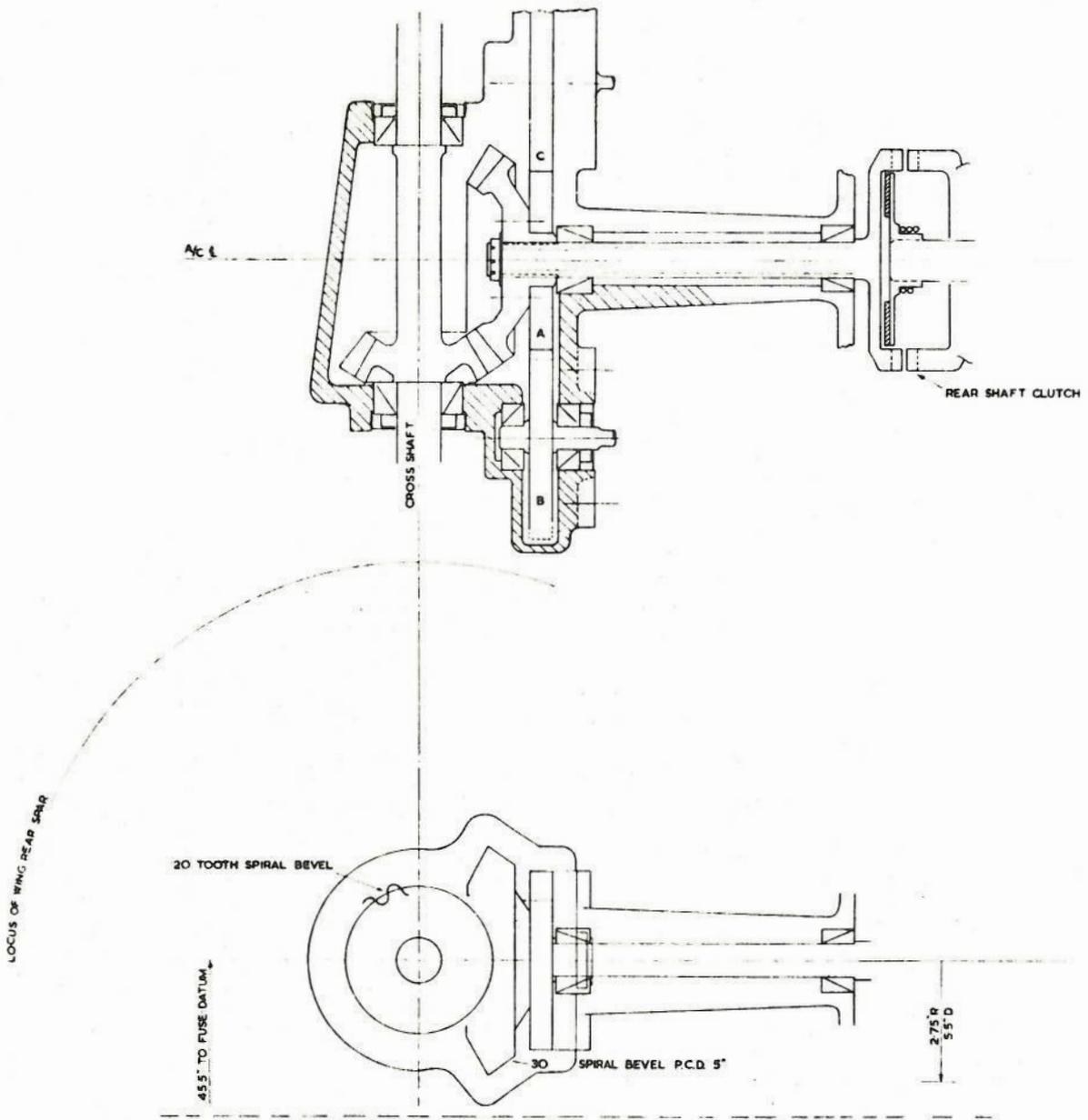


FIG. 15. CENTRE GEARBOX

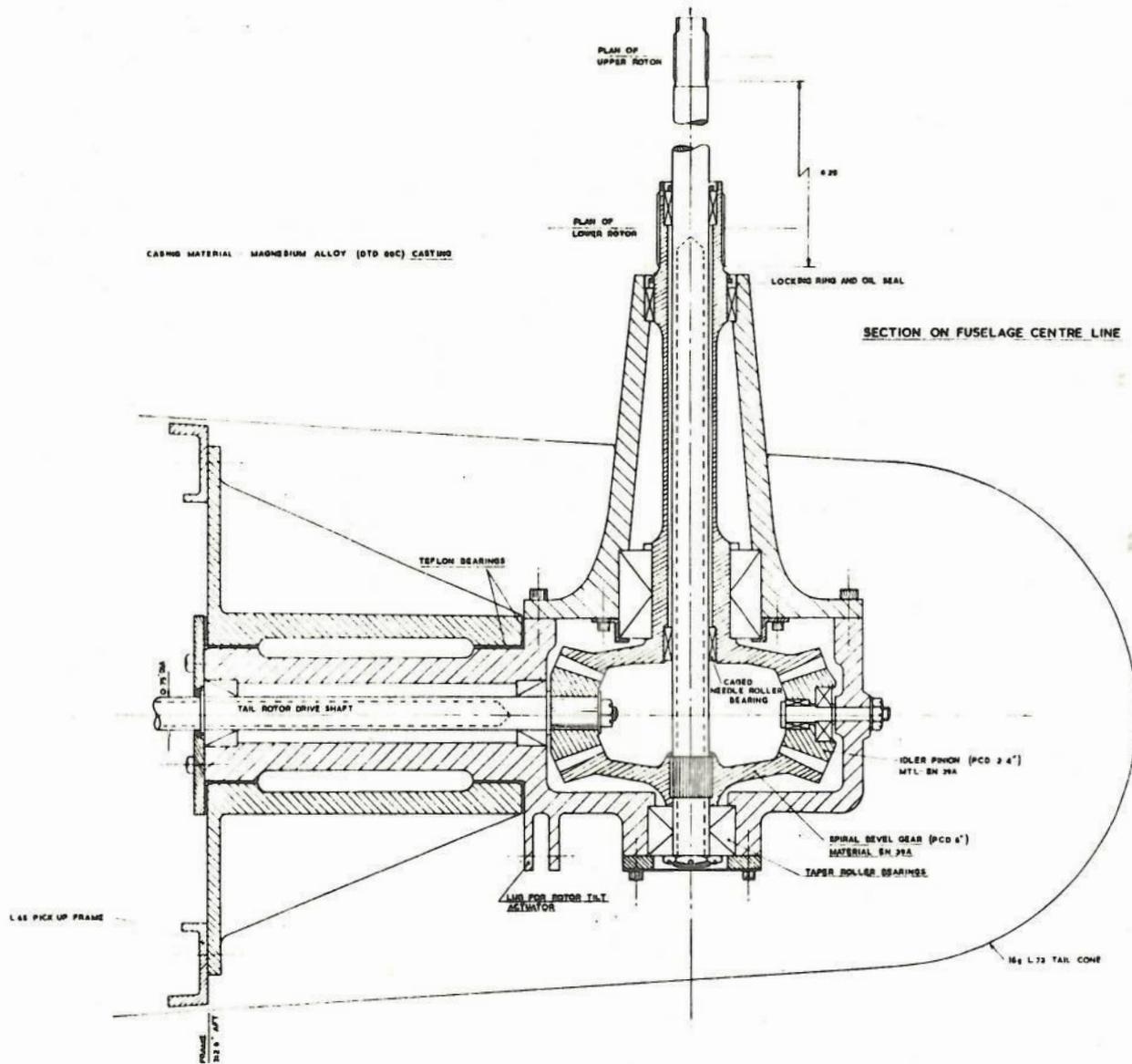


FIG. 16. TAIL ROTOR GEARBOX

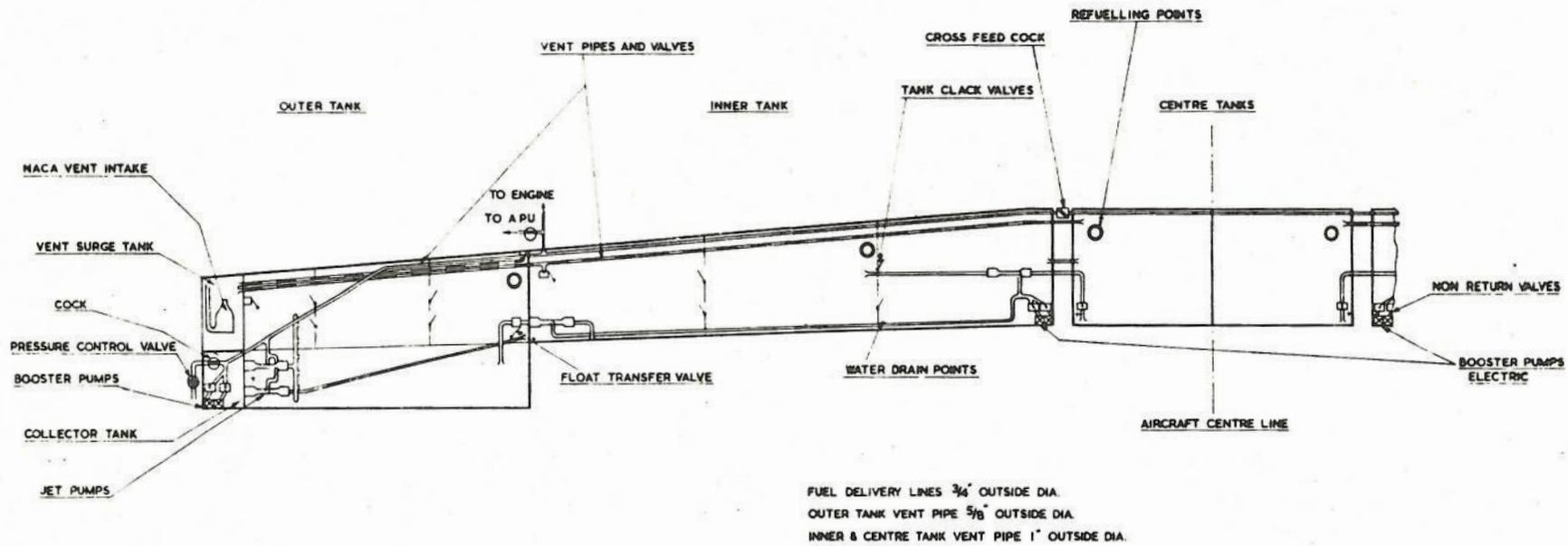


FIG. 17. FUEL SYSTEM

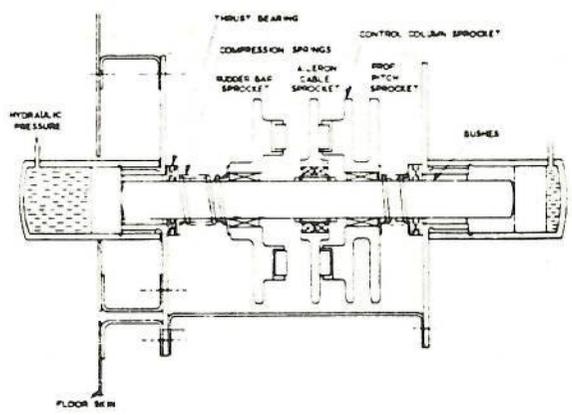
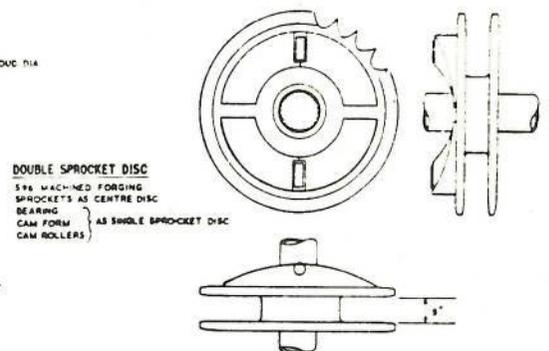
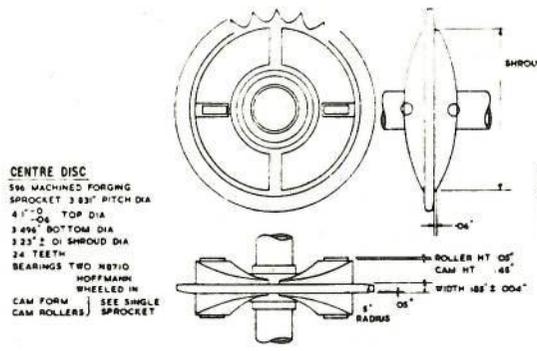
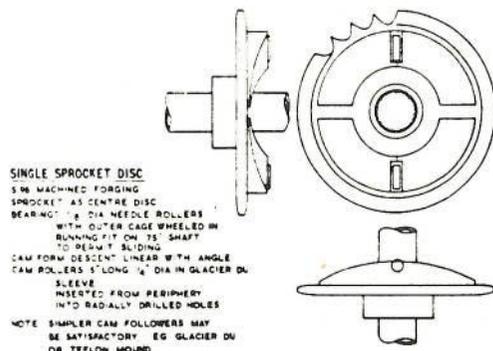


FIG. 18. AILERON ROLL-YAW INTERCONNECTION MECHANISM

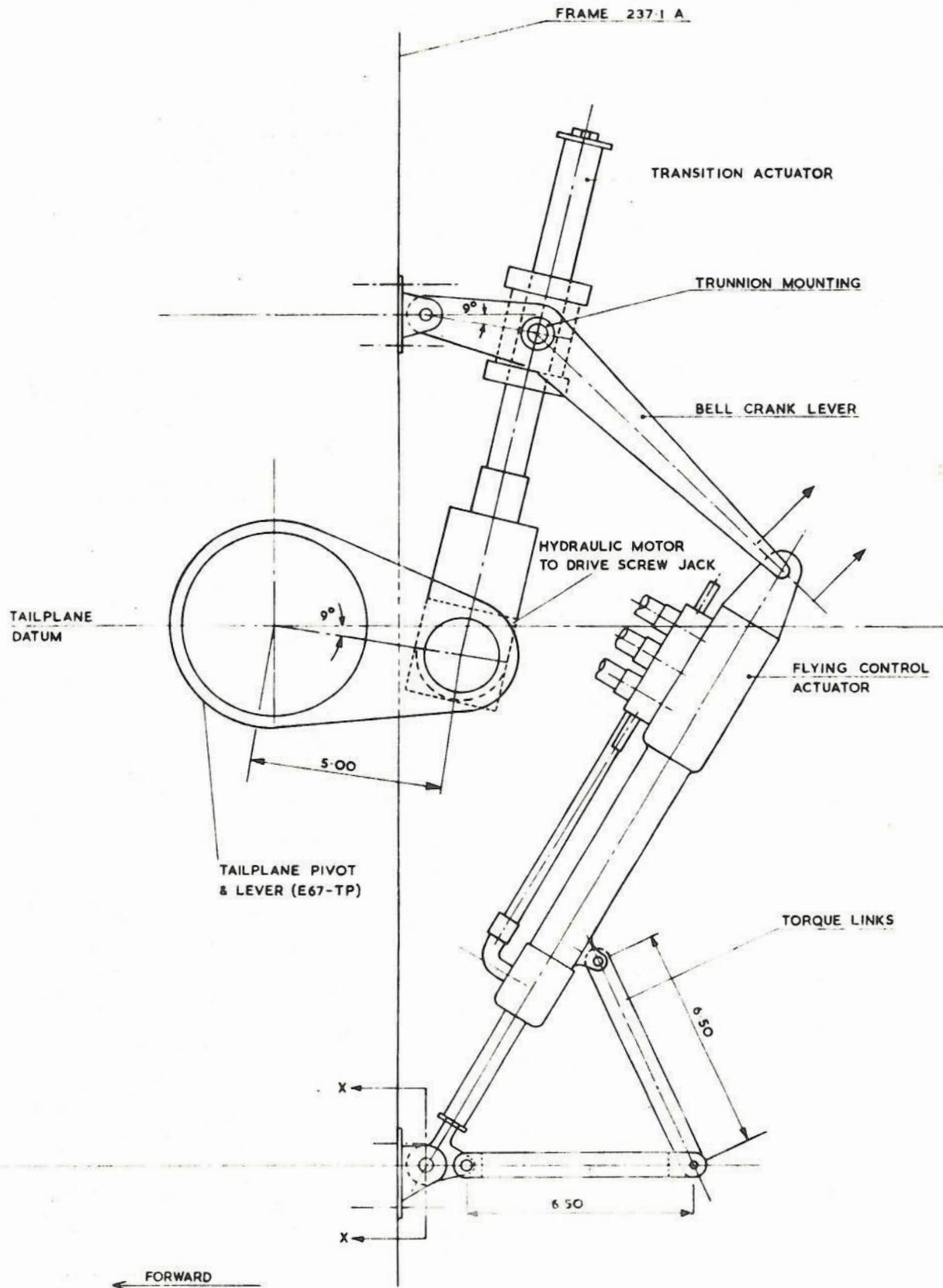


FIG. 19. TAILPLANE ACTUATION MECHANISM

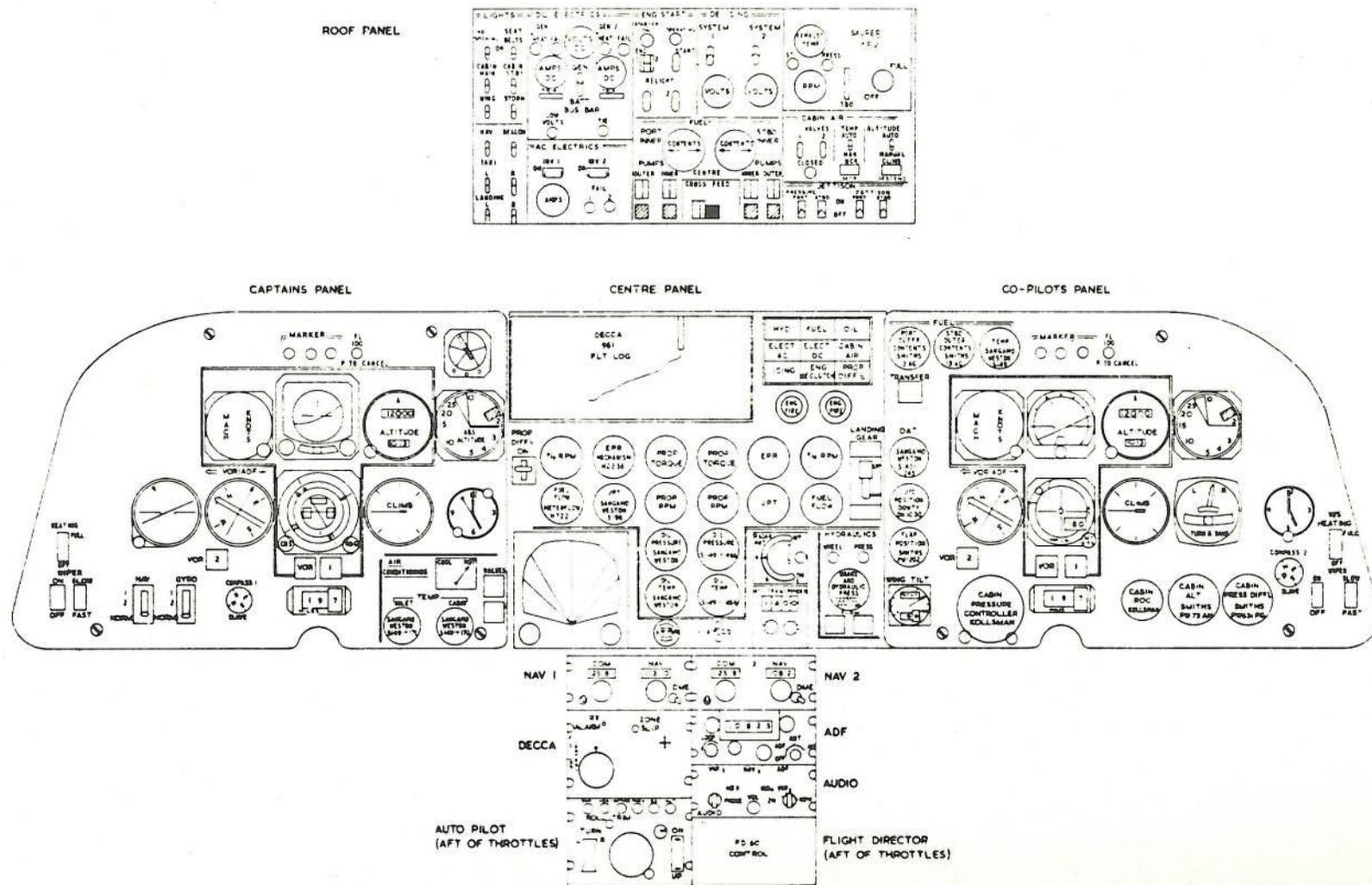


FIG. 20. PILOTS INSTRUMENT PANEL LAYOUT.

(MAX. CONTINUOUS CRUISE POWER)

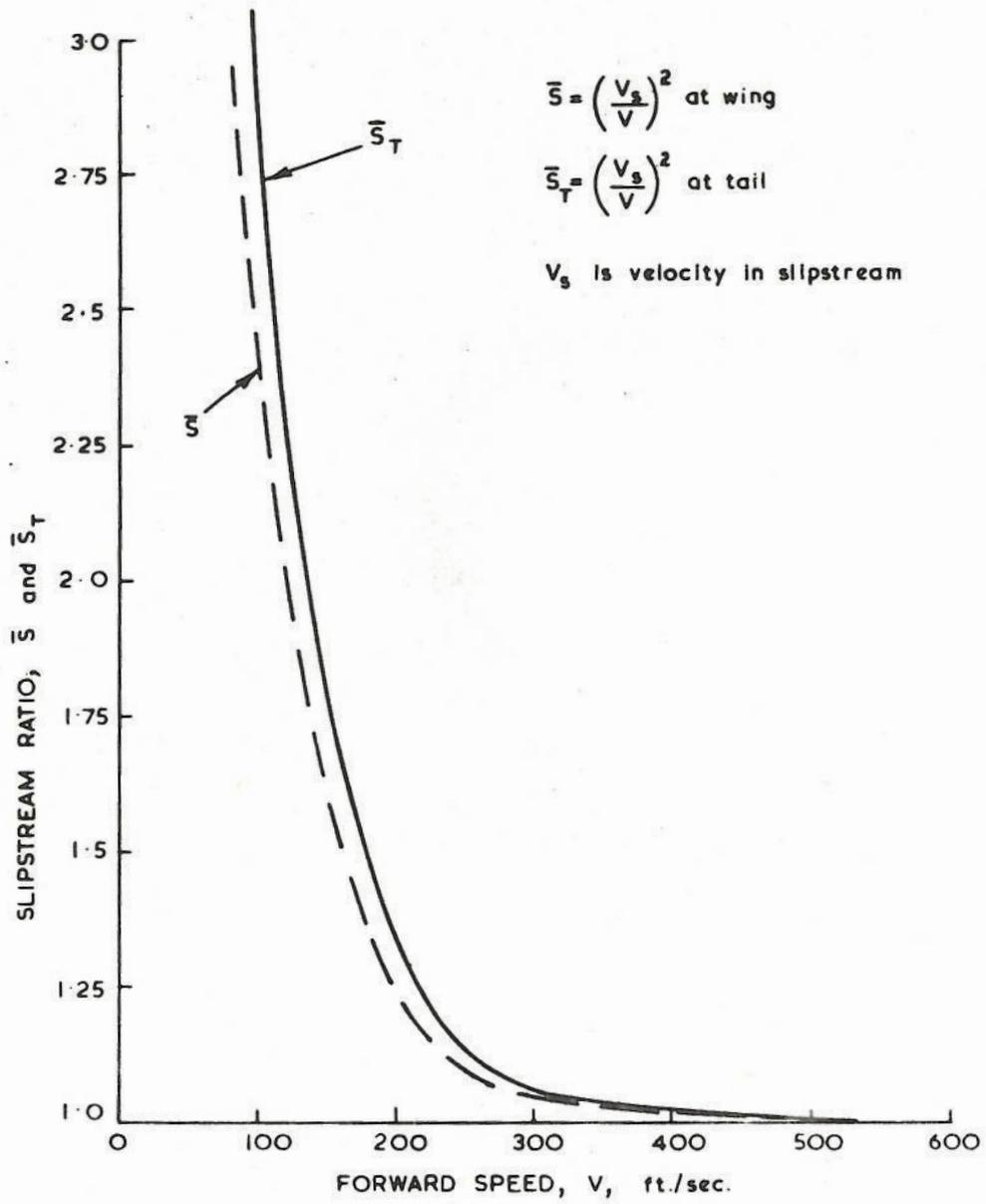


FIG. 21. SLIPSTREAM EFFECTS.