SECRET
AUSTER
HR·144·T

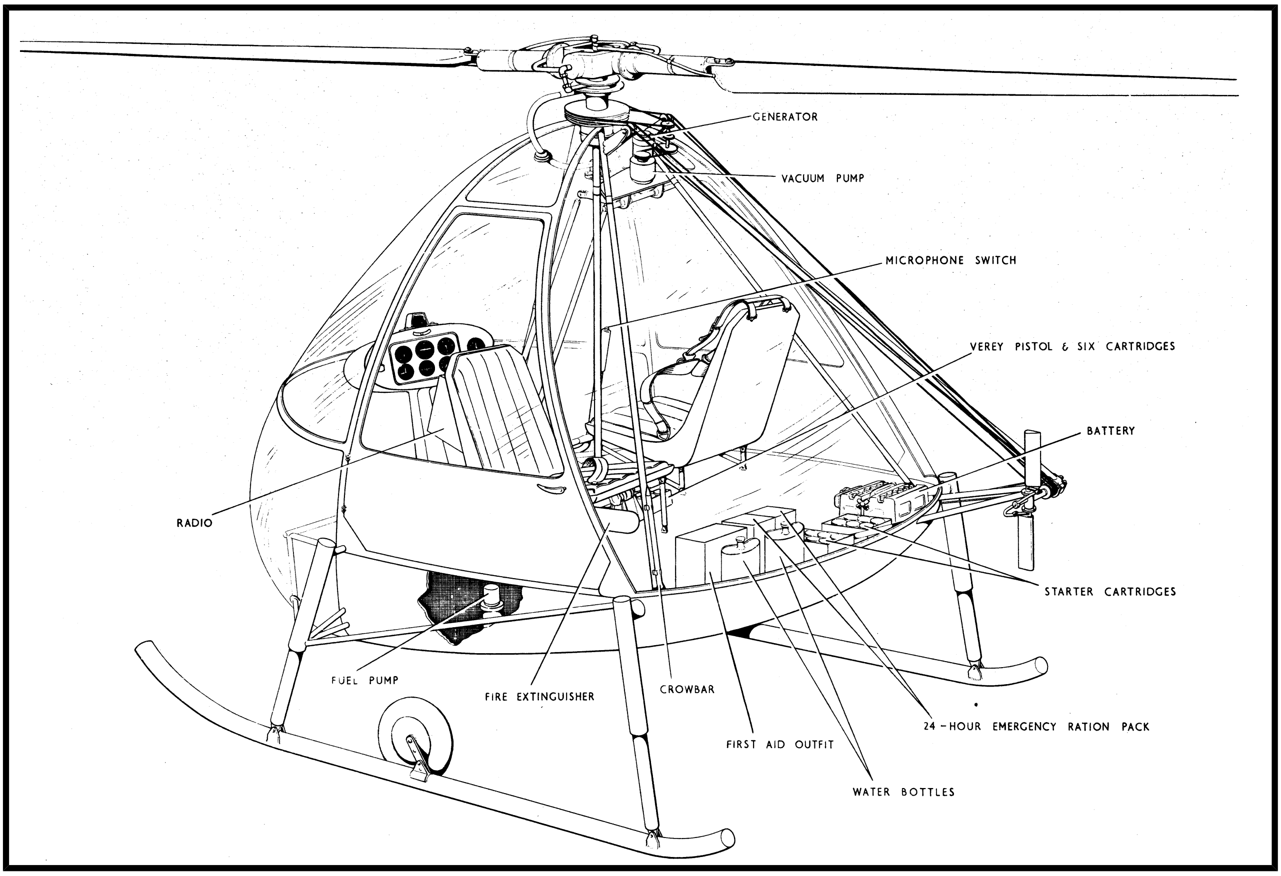
THE

# AUSTER MODEL B9

TO

# SPECIFICATION

# HR 144 T




AUSTER AIRCRAFT LTD.,
Rearsby · Leicester · England

# Introduction by International Auster Club Heritage Group

# PREFACE

The Helicopter described in this Brochure has been designed in the Auster tradition of simplicity and safety. As far as possible this design study is presented in the standard manner required by Technical Procedure Requirements, but it is inevitable that complete justification for many of the statements made cannot be included in the Brochure. This is particularly true of matters relating to the dynamic and aerodynamic design, where much is based upon original and as yet unpublished research work by members of the design team. Should amplification of such points be required, the information will be supplied on request to the Company.

In order to benefit from latest American developments before undertaking the design, visits were made to Hiller Helicopters, Inc., The American Helicopter Company, Inc. and Cessna Aircraft Company, Inc. in the United States by the Company's Chief Designer. The figures quoted to him are presented in a section devoted to the Power Unit. It should be noted that all the companies visited were unanimous in dismissing the pulsejet as impractical and inefficient in the present stage of development. In particular this was the opinion of the Chief Engineer of a Company developing pulse jet helicopters under contract the United States Government. This information confirmed the results of our own investigations and justifies the use of the ramjet in the present design.

# CONTENTS

# INTRODUCTION & PRINCIPAL FEATURES

# SECTION 1 PRINCIPAL FEATURES

In this helicopter, designed to Specification No. HR 144.T., emphasis has been placed on simplicity and safety. The achievement of simplicity is largely made possible by the use of a ramjet power plant. Small helicopters using this type of motor have already been shown to be practical, and we are pleased to find current American opinion definitely favours this motor over the pulsejet, thus supporting our own conclusions.

As a result we are enabled to tender a design for a helicopter, the performance of which under I.C.A.N. conditions is about twice that required by the Specification and much better than the Hiller Hornet. Under tropical summer conditions it is capable of 55 minutes endurance when the take off weight is limited by the altitude performance requirements.

A feature of the design is that of twin engined reliability. Unlike a machine powered by a pulsejet, a helicopter powered by ramjets can fly on one unit without excessive vibration. One unit supplies sufficient power to enable the machine to maintain height.

Engine starting in the field is very simple, being analogous to conventional cartridge starting. Any desired number of starting charges may be carried, but in the interests of economy it is hoped to provide for starting by an external motor when the helicopter is operating from base.

Fuel control is carried out on the ramjet unit in order to avoid time lag in response. The patented metering device is so designed to maintain a constant rotor speed, once the pilot has set the throttle for a given flight condition.

To reduce the possibility of fatigue, steel is the principal material used in the construction of the rotor hub and swash plate assembly. This has entailed a weight penalty, but the Company feels strongly that to use aluminium alloys in primary structure subject to fluctuating loads on a helicopter is most unwise. Even if a very lengthy fatigue test programme were embarked upon to justify the use of light alloys, the simulation of flight loads is so uncertain that an element of risk would still remain.

The patented undercarriage employs friction to absorb the energy of impact and was developed specifically for helicopter use. It is relatively cheap to manufacture, requires little attention in Service and permits emergency landings at high rates of descent without risk of damage. It can be folded for transport in less than one minute by one man releasing the friction.

The fuselage structure of welded steel tubes follows normal Auster fixed wing practice. The Perspex "bubble" doors and rear fairing are attached to a support structure which itself is bolted to the basic frame. In order to ensure long, trouble free service life, several components are based upon well-proven Auster designs. The seats, for example, are of the type already developed for the Auster A.O.P. Mk.9 and the honeycomb stabilised floor panel is also based upon early Auster developments.

It is confidently expected that the servicing required in the field will be relatively little and probably less than with Auster fixed wing aircraft.

Although a two bladed rotor is called for, the Specification does not lay down any vibration criteria; yet the use of a two bladed rotor with drag hinges would lead to vibration levels too high to be acceptable. This Company has investigated the associated problems of vibration, control forces, and stability in considerable detail. Whilst it is manifestly true that the limitations of contemporary helicopters are all due to defects in the dynamic design, the causes of these defects are often treated in general terms only. In-plane rotor vibration provides an excellent example of this attitude and of the dangers of extending it to future designs. On the Auster design, large second harmonic vibrations are suppressed at source by balancing dynamic and aerodynamic forces against each other. Similar treatment is adopted with stick forces, and with a hovering (stick free) period of 26 seconds it is confidently expected that this helicopter will set new standards in smoothness and controllability.

If positive stability is required, automatic stabilisation will be provided in the form of "rate and attitude" gyro bars. It is recommended, however, that such an innovation should be left until after service assessment

In the interests of safety, great importance is attached to obtaining adequate rotor height above the ground in order that the aircraft may be approached with the rotor running. The critical case is provided when the rotor is turning slowly and almost fully drooped. Under these conditions a tip height of 8 feet is obtained.

A penalty is paid for this high rotor position when the helicopter is mounted on an Army lorry. In order to pass under a 12' 6" high bridge the lowest encountered on main roads in Great Britain, the undercarriage must be collapsed. This operation takes less than one minute to complete. The unique steel and fibreglass blade design enables the ramjets to run at a Mach number high enough for satisfactory efficiency. As a result, the Specification performance requirements are easily met, and under I.C.A.N. conditions considerably exceeded.

If for I.C.A.N. conditions the performance is restricted to an 800 ft. per minute vertical rate of climb, then the aircraft can operate at an overload weight of 1,780 lbs. Typical roles at this take off weight are:

| | |
|---|---|
| A.0.P. | 75 minutes duration. |
| Local Reconnaissance. | 35 nautical miles radius from base. |
| Supply Carrying. | 700 lbs. pay-load over 10 nautical miles or 400 lbs. over 50 nautical miles. |
| Ambulance. | One sitting and one stretcher casualty over 60 nautical miles. |

In the ambulance role, which is not required by the Specification, the stretcher is carried athwart ships behind the two seats.

At maximum all up weight the rate of descent with both engines cut is 1,800 ft. per minute. The "high energy" rotor holds sufficient energy to give excellent "engine cut in hovering" characteristics, and the virtual abolition of a "critical height". A descent velocity of 3,000 ft. per minute can be "flared out" by a 12% drop in rotor revs.

At the time of submitting this brochure a prototype of the ram jet unit described in Section 6 is awaiting completion of the test rig whirling arm. A photograph of this unit which it is hoped to run within a few weeks is given in Fig. 1.3. The Company has also completed a mock up fuselage, shown in Fig.1.4. Although this is not required by the Ministry of Supply, for tendering it was felt that's its construction would enable amore reliable tender to be submitted.

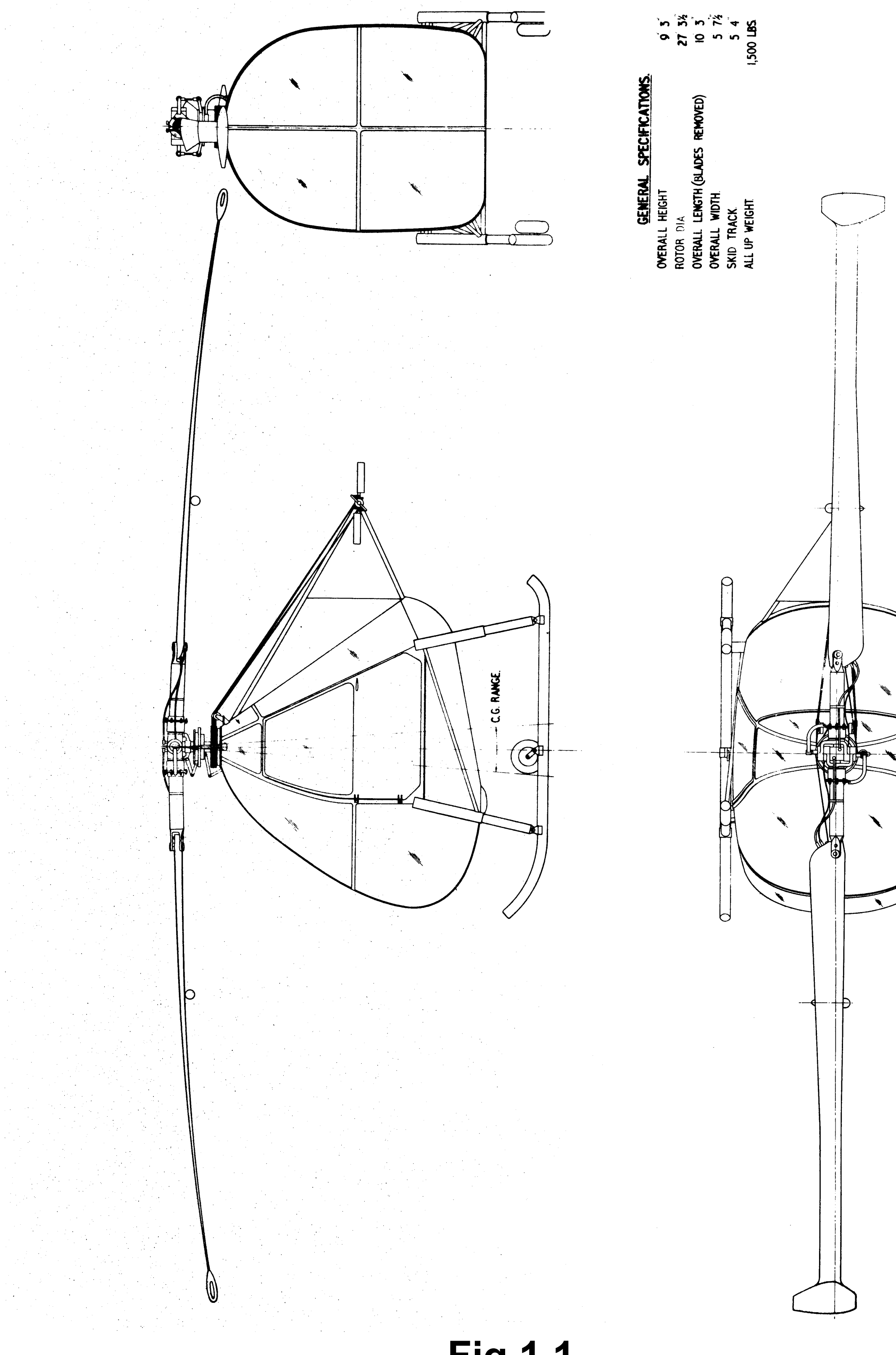

GENERAL SPECIFICATIONS.
OVERALL HEIGHT 9' 5"
ROTOR DIA 27' 3½"
OVERALL LENGTH (BLADES REMOVED) 10' 5"
OVERALL WIDTH 5' 7½"
SKID TRACK 5' 4"
ALL UP WEIGHT 1,500 LBS
C.G. RANGE

**Fig 1.1**

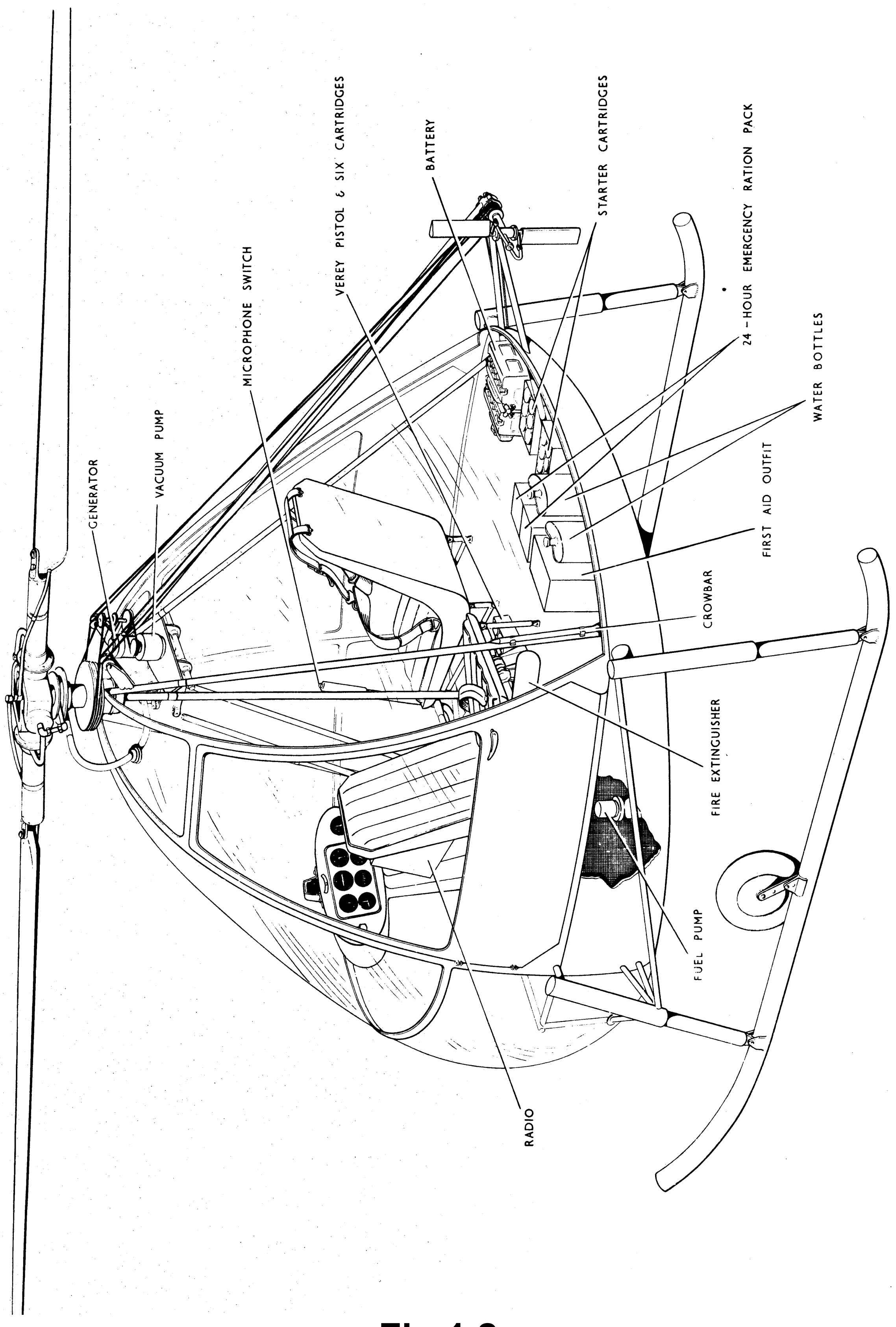

GENERATOR
VACUUM PUMP
MICROPHONE SWITCH
VEREY PISTOL & SIX CARTRIDGES
BATTERY
STARTER CARTRIDGES
24-HOUR EMERGENCY RATION PACK
WATER BOTTLES
FIRST AID OUTFIT
CROWBAR
FIRE EXTINGUISHER
FUEL PUMP
RADIO

**Fig 1.2**

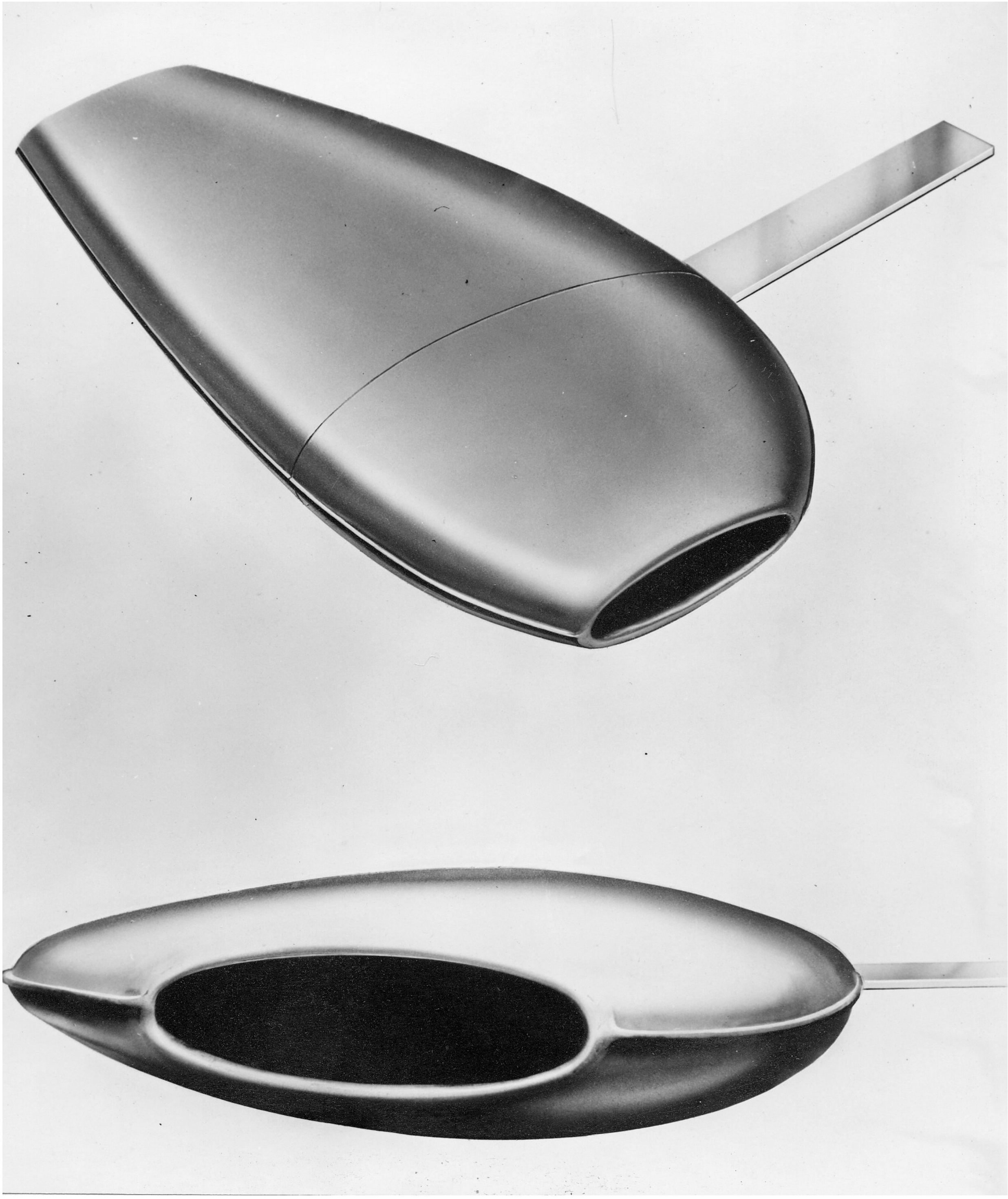


Fig 1.3

Fig 1.4

# LEADING PARTICULARS

# SECTION 2 LEADING PARTICULARS

**TABLE I. GENERAL DATA.**

| | |
|---|---|
| Name of Contractor: | Auster Aircraft, Limited. |
| Type of Aircraft: | Two-bladed single rotor two seat ramjet powered helicopter. |
| Manufacturer's Designation: | Model B9. |
| Duties of Aircraft: | Reconnaissance and general Army liaison. |
| Brief description of Aircraft | The helicopter seats a pilot and observer side by side of in a spacious bubble cockpit. Entry is by means of large hinged doors on either side of the machine. These doors are also jettisonable for emergency exit. The machine is fitted with a minimum of flying instruments, but carries an Army radio set for communication and other equipment to enable the Crew to carry out their duties efficiently.<br>The machine is powered by two tip-mounted ramjet units, and provision for dual control is made to enable crews to be familiarised with this type of helicopter. |

**DESIGN REQUIREMENTS WITH WHICH THE AIRCRAFT COMPAIBLE**

| | |
|---|---|
| M.O.S. Specification No.: | HR. 144T. |
| Derogations from this Specification: | There are no major derogations from this Specification, but such minor variations as have been thought desirable are described under the appropriate sections of this brochure,<br><br>The maximum A.U.W. of 1500 lbs., which is called for in the Specification, can be met with a fuel load of 459 lbs., which gives a greater duration than that required.<br>Performance calculations have been made, at a "Nominal All up Weight" of 1550 lbs., however, and this weight has been used throughout in all design work because it was convenient. The selection of this nominal figure does not influence the operational weight in any way. |

## TABLE IV. STRENGTH_AND STIFFNESS DATA.

| | |
|---|---|
| Maximum acceleration (flight case): | 3g (up), 1/2g (down) |
| Associated factor of safety: | 1.5 |
| Associated weight: | 1550 lbs |
| Design diving speed: | 112 knots, LA.S, |

The structure is stressed in accordance with the requirements of Section G of B.C.A.R. using the proof and ultimate factors of 1.2 and 1.5 respectively. The only loading which is not exactly defined in Section G is that due to gusts. This is found to be independent of forward speed, and the ratio of rotor thrust to all up weight due to a 35 ft./sec. sharp edged gust is plotted in Fig.2.1.against weight. It will be seen that both up and down gusts give accelerations less than the minimum values required by Section G.

Items of the blades and hub which are designed by centrifugal force are stressed for a 10% over speed condition only, because the ram jet throttle design, described in Section 6.0., effectively prevents over speeding of the rotor.

A special feature of this rotor is that the drag hinges only operate in an emergency so that the rotor must be strong enough in the drag plane to withstand all normal forces imposed upon it. Of these, by far the largest are coriolis forces caused by blade flapping. Because of the low coning angle it is found that the highest bending moments are second harmonic, contrary to normal experience. Second harmonic bending moments are independent of coning angle, being proportional to the square of the angle between the tip path and rotation planes. It is found that the critical bending moment occurs at the root pins of the blade and that the maximum permissible flapping angle relative to the shaft is 8.0º(factored). In Fig. 2.2.the envelope values of flapping amplitudes are plotted for extreme weights and C.G. positions. It is evident that, although the blade is amply strong enough for all normal flight conditions, the possibility exist of encountering a gust whilst flying at an extreme condition and, thereby, obtaining a flapping amplitude large enough to fail the blade. It is because of this possibility that the patented shear pin is fitted in the root and attachment fork, as explained in Section 3.2.

The variation of rotor thrust loading with azimuth at the maximum forward speed of 102 knots is given in Fig.2.3. . This is mainly balanced by centrifugal and inertia forces and the resultant stress fluctuation in the blade is extremely low.

Higher harmonic resonance of the blade has been provisionally investigated, but detailed calculations will be left until the prototype blade is finished and its static natural frequencies can be measured experimentally. In this connection, it should be noted that blade resonance with harmonic inputs is not dangerous with a steel blade, although undesirable.

**TABLE_V. ENGINE AND PERFORMANCE DATA.**

1. Power Required

Estimates of power required have been made in accordance with the general method given in Reference 1. The individual assumptions relating to rotor power losses, parasite drag estimates, etc., are outlined in the section of this brochure dealing with Performance (Section 11).

2. Assumed Engine Performance.

(I.C.A.N. Sea Level Conditions).

| | MINIMUM | MAXIMUM |
|---|---|---|
| Nominal circumferential velocity. | 800 ft,/sec. | 900 ft./sec. |
| Nominal Mach Number. | 0.716 | 0.805 |
| Max. gross thrust. | 45.8 lbs. | 71.0 lbs. |
| Associated specific fuel consumption. | 10.2 lb./lb./hr. | 9.0 lb./lb./hr. |

3. Calculated Speeds.

(I.C.A.N. Conditions).

| | | |
|---|---|---|
| Max. speed at Sea Level. | 82.2 knots. | 102.0 knots. |
| Max. speed at 6000 ft. | 77.9 knots. | 100.0 knots. |
| Cruising speed on 82% max thrust. | | 70 knots. |
| Max. endurance speed (67%max. thrust). | | 45 knots ,E.A.S. |

TABLE VI. CREW STATIONS LAYOUT DATA.

A drawing showing the major dimensions, view, facility of entry and exit, positioning of controls and instruments is provided in the Cockpit Layout Section of this brochure (Section 5.0.). The facilities for exit in emergency are shown in Section 9.

**TABLE VII. PRODUCTION DATA.**

1. The type of structure and the method of manufacture proposed is described in Section 4.0. of this brochure, under the heading of "Airframe Construction".

2. The remainder of the data required under Table VII is given in that Section of the brochure headed "Design and Construction Programme".

**TABLE VIII. REPAIR AND MAINTENANCE DATA.**

The information required for this Table can be found in the following Sections of the brochure;

| | | | | Section No. |
|---|---|---|---|---|
| 1. | (a) | Man-hours and elapsed time involved in engine change unit and major component replacement. | Servicing and Maintenance | 10.0 |
| | (b) | Type of landing gear shock absorbers. | Airframe Construction | 4.0. |
| | (c) | Voltage of electrical circuits. | Equipment | 8.0. |
| | (d) | Method of catering for wear of component attachment fittings. | Airframe Construction | 4.0. |
| | (e) | Ground handling and jacking. | Servicing and Maintenance | 10.0. |
| | (f) | Details of special design features for simplification of inspection, servicing or replacement of equipment, accessories or components,. | See description of appropriate component. | |
| 2. | | G.A. of detachable units and breakdown with overall dimensions. | Servicing and Maintenance | 10.0 |

**TABLE IX DESIGN PROGRAMME**

The information required for Table IX is shown in that Section of the brochure headed "Design and Construction Programme"

## LEADING PARTICULARS.

| | |
|---|---|
| **Major Dimensions.** | |
| Overall length (rotor fore and aft). | 27,29 ft. |
| Overall length (blades detached). | 10.25 ft. |
| Overall width. | 5.63 ft. |
| Overall height (normal static height). | 9.25 ft. |
| Overall height (undercarriage folded). | 8.08 ft. |
| Rotor height above ground (rotating). | 9.ft. |
| Rotor tip height above ground (drooping). | 7.83 ft.. |
| Distance between main and tail rotor centres. | 6.6 ft. |
| **Power Units.** | |
| Type. | Ram jet. |
| Number off. | 2 |
| Size. | 19.13 ins.2 each combustion chamber area. |
| Max. rated thrust I.C.A.N Sea Level) per unit. | 71.0 lbs. each at 900 ft./sec, tip speed. |
| Corresponding specific fuel consumption. | 9.0 lb./lb./hr, |
| Nominal max. centripetal acceleration. | 2000 g. |
| Material. | Nimonic 90.. |
| **Main Rotor.** | |
| Diameter | 25.66 ft. |
| Disc area. | 516.7 ft.2 |
| Nominal disc loading. . | 3.0 lbs./ft.2 |
| Max. tip speed. | 900 ft./sec. |
| Max. rotor R.P.M. | 859 R.P.M. |
| Nominal CL BASIC.. | 0.409. |

Main Rotor Leading Particulars (continued)

| | |
|---|---|
| Effective straight taper (tip chord) | 0.4 |
| (root chord)eff | |
| Blade twist (root to tip). | 10.60 |
| Standard solidity | .025 |
| Root solidity 6R . | .05 |
| Theoretical root chord | 1.006 ft. |
| Theoretical root section. | NACA 0015. |
| Tip chord | 0.503 ft. |
| Tip section | NACA 0004. |
| Standard chord (at O.7R). | 0.503 ft. |
| Standard section. | NACA 0007.3 |
| Mean blade inertia axis position. | 0.24 chord. |
| Control advance angle. | 860 |
| g3angle. | 0 |
| Flapping pin offset. | 0 |
| Blade root end pin radial position r/R | 0.19 |
| Blade weight (ram jet to root end pin). | 43.0 lbs. (with ram jet). |
| Total first moment of mass about hub CL | 11,29 slugs ft. |
| Total second moment of mass about hub CL | 111.7 slugs ft.2 |
| Tail Rotor. | |
| Diameter. | 2 ft. |
| Disc area. | 3.142 ft.2 |
| Max. tip speed. | 400 ft./sec, |
| Max. rotor R.P.M. | 3820 R.P.M. |
| Gear ratio (main to tail rotor).. | 4.45 |
| Tail rotor drive pulleys | 2.0 ins. diameter at tail rotor, 8.9 ins, diameter at main rotor. |
| V.belt. size, 2 belts | 3/8in x 15/64 in x in. |

Tail Rotor Leading Particulars (Continued)

| | |
|---|---|
| Blade taper | 0 |
| Standard solidity | 6.7 0.106 |
| Chord | 0.167 ft. |
| Section | NACA 0025 |

# GUST LOAD ENVELOPES.

## 35 FT/SEC SHARP EDGED GUST

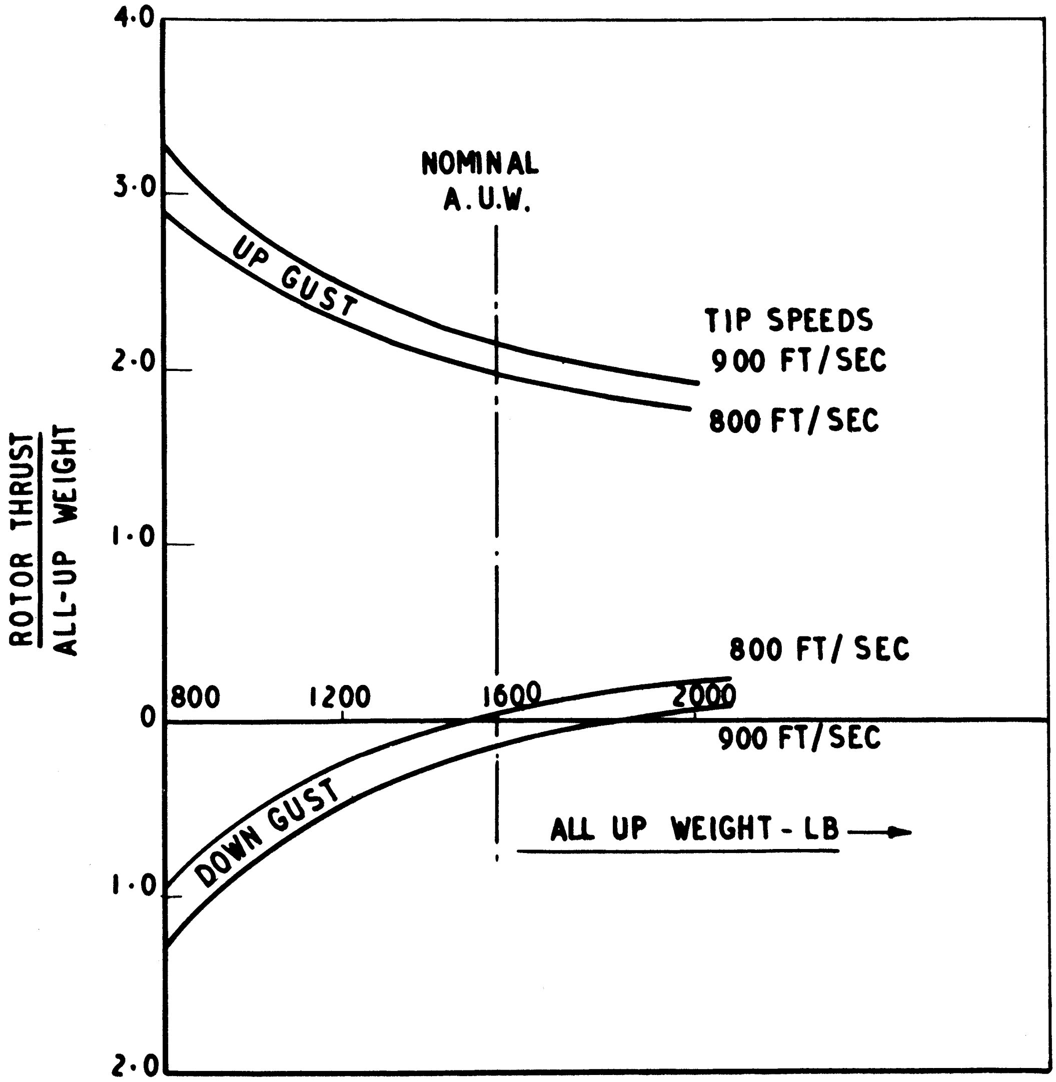



**Fig 2.1**

# VARIATION OF FLAPPING ANGLE WITH FLIGHT CONDITION

I.C.A.N. SEA LEVEL CONDITIONS
TIP SPEED = 900 FT/SEC.

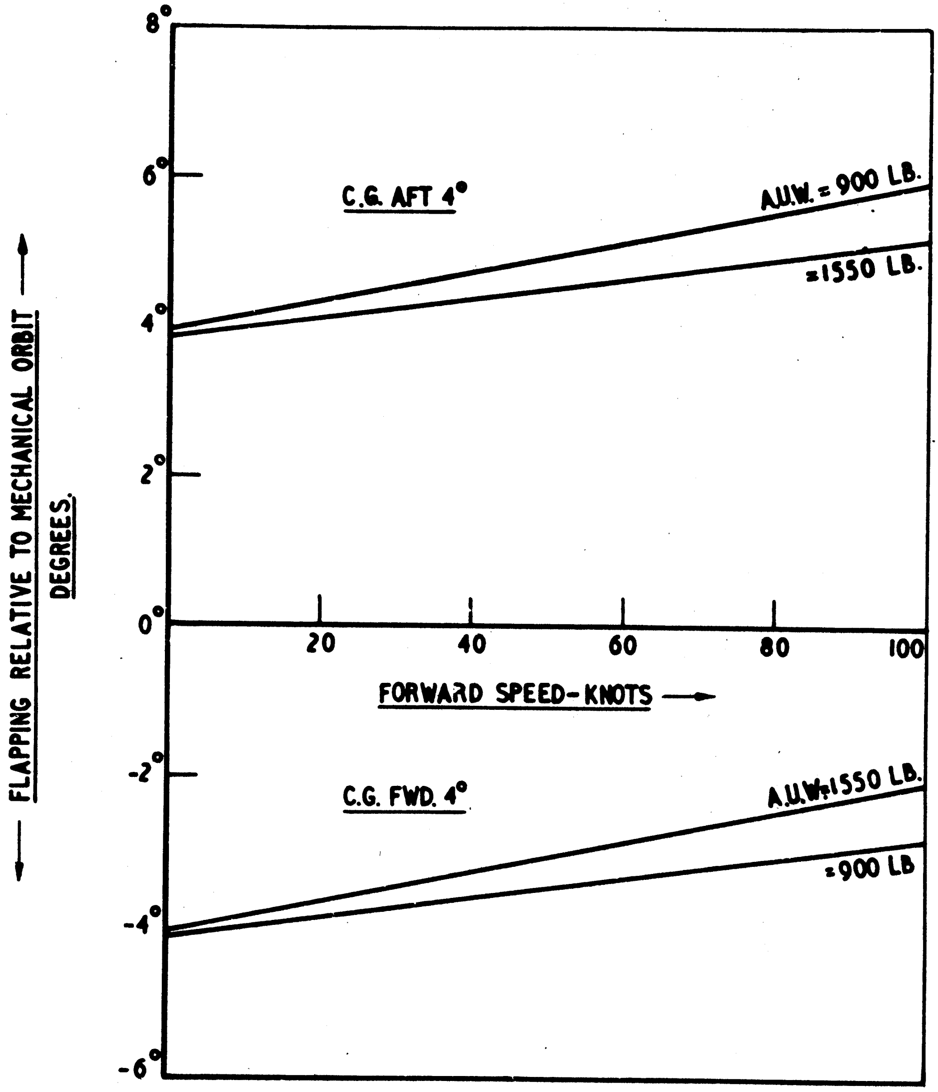



Fig 2.2

## BLADE THRUST LOADING AT MAX. FORWARD SPEED.

I.C.A.N. SEA LEVEL CONDITIONS.

NOMINAL A.U.W.

TIP SPEED $V_T$ = 900 FT/SEC.

TIP SPEED RATIO $\mu$ = 0·191

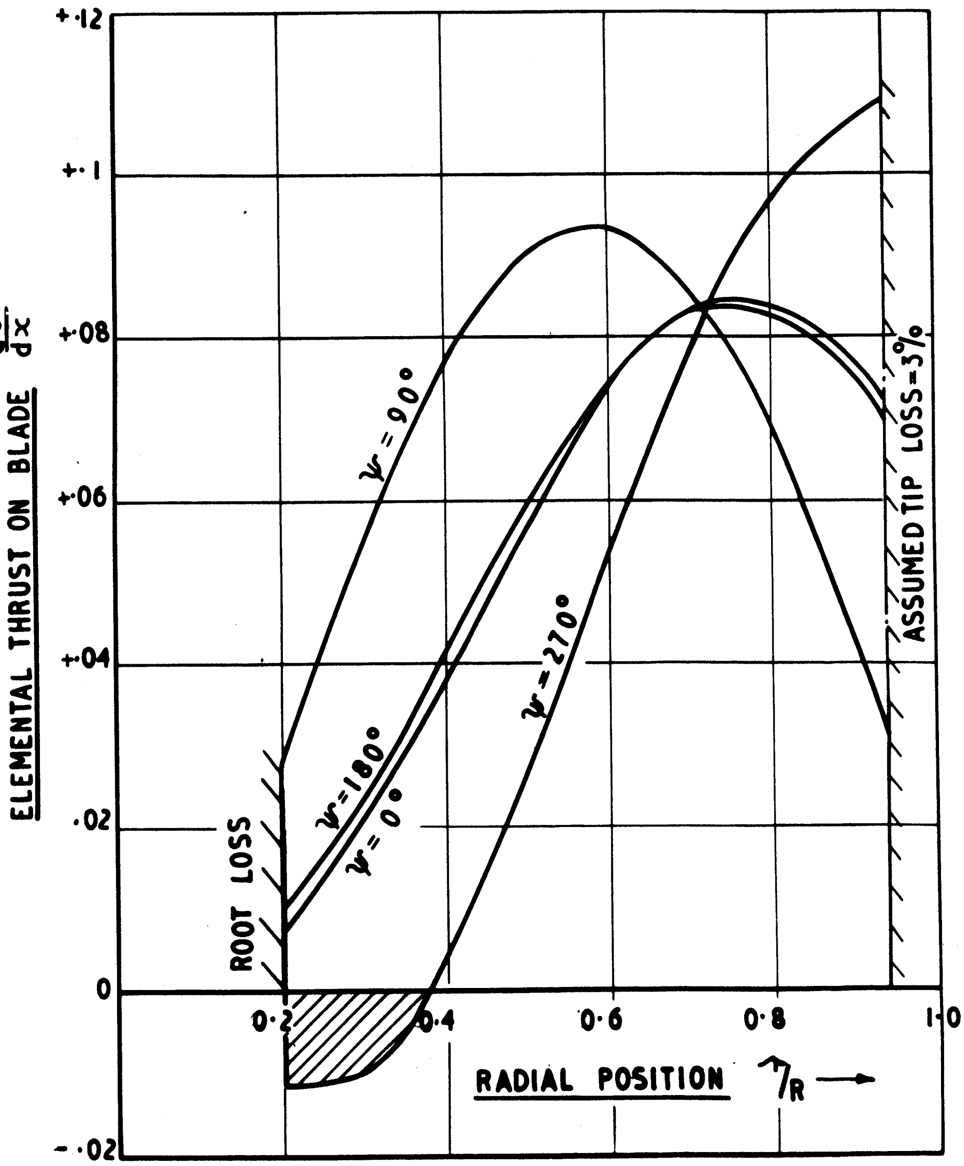



Fig 2.3

# DESIGN PRINCIPLES

# SECTION 3 DESIGN PRINCIPLES

## 3.1 CHOICE OF POWER UNIT.

In the past few years many studies have been made and published of the relative efficiencies of various possible methods of supplying power to a helicopter. The conclusion has, in all cases, been reached that a ram or pulse jet unit is most suitable when durations of less than an hour are required. The more superficial studies have shown that of the two, the pulsejet is to be preferred.

However, the use of the pulsejet presents difficulties in ensuring that it is strong enough to withstand its own centrifugal forces. In particular, it is difficult to design a satisfactory tail pipe because material added to increase its strength also increases the centrifugal loads tending to bend it. Because of this, it is found that the maximum circumferential velocity of a pulsejet is around 430 ft./sec. if mounted on the rotor tip, or 250 ft./sec. when mounted inboard at, for example, 50% of the radius.

This speed limitation means that a pulsejet works at a mechanical disadvantage compared with a ramjet. It is not, therefore, possible to compare directly their specific fuel consumptions on a thrust basis.

In view of the difficulties entailed in mounting a pulse jet at the tip of a rotor blade (unlike a ramjet, it is not possible for a pulse jet to execute feathering motion with the blade) a compromise arrangement is to mount the pulse jet inboard of the rotor hinges ( on a high off set rotor) or on a separate arm at right angles to the rotor. Mounted at 50% of the radius, such a design would require a thrust of about 160 lbs. from each unit, compared with the 45 lbs. delivered by the Auster ramjet. A comparison of the relative efficiencies is given in the following table for I.C.A.N. See Level Conditions:

**TYPE OF ENGINE**

| | RAM JET | PULSE JET |
|---|---|---|
| Thrust per unit | 45 lbs. | 162 lbs. |
| Circumferential velocity | 900 ft./sec. | 250 ft./sec. |
| Specific fuel consumption | 9.0 lb./lb./hr. | 6.0 lb./lb./hr. |
| Fuel consumed | 405.0 lbs./hr. | 971.0 lbs./hr. |

It will be seen that for equal duration, the pulsejet specific fuel consumption would have to come down to 2.5 lb./lb./hr., or that the ramjet would have to be as high as 21.6 lb./lb./hr. Moreover, no allowance has been made for the greater weight and complexity of the pulse jet installation. For these reasons, it is felt that the case made out for the ramjet is so strong as to be virtually unaffected by the precise values used for, fuel consumption in the argument.

A similar comparison can be made for a tip-mounted pulsejet with the same results. In this case, however, not only are the units heavier than a ramjet, but also the mounting (which must include torsion bearings) and the entire rotor blade have to be immensely stronger than before.

An example of this configuration is provided by the XH 26 helicopter currently under development by a leading American company for the United States Army. The maximum tip speed attainable with their design is 375ft./sec., and the pulse jet performance is quoted as follows:

| | STATIC | IN FLIGHT |
|---|---|---|
| Tip speed | 0 | 375 ft./sec. |
| Thrust | 40 lbs. | 30 lbs. |
| Present fuel consumption | 6 lb./lb./hr. | 9 lb./lb./hr. |
| Hoped for fuel consumption | - | 71/2lb./lb./hr. |

The figures were quoted to this Company's Chief Designer in confidence during a recent visit to the United States to examine American progress in light helicopters. The Chief Engineer of the American company stated that their main problems arose from inefficiencies due to yawed flow, fatigue and from high centrifugal force loads on the pulse jet unit.

He stated that the hoped for fuel consumption - if achievable at all - was still a very long way off. Severe yawed flow is an unavoidable result of low tip speeds and, as Fig. 3.1 shows, the flow may yaw as much as 20° for a pulse jet rotor, compared with 10° for a ramjet.

In both positions, as shown in Section 3.2.3, single engine flight with pulsejets is not possible because of the very high vibration level entailed. With a ramjet, the single engine vibration is barely above the threshold of feeling.

### 3.1.1. Future Developments

It is generally conceded that the intermittent ramjet currently being developed in the United States should result in the development of units with greatly reduced fuel consumption. This means that with the development of such units in this country, the standard ramjets of the Auster machine could be replaced by the more efficient intermittent type, at small cost, with a corresponding increase of both range and endurance.

A second point is that a light ramjet helicopter flying within the next year or so will materially assist the future design and construction of the large flying cranes which seem likely to be required by the Army of the future.

Finally, there is the development of supersonic ramjet powered rotors, with which several companies of the United States are reputed to be making considerable progress. For all military applications, such a rotor offers great promise, and the development of a successful ramjet powered subsonic rotor is an essential preliminary.

## VARIATION OF AIRFLOW ANGLE TO INTAKE, WITH TIP SPEED RATIO.

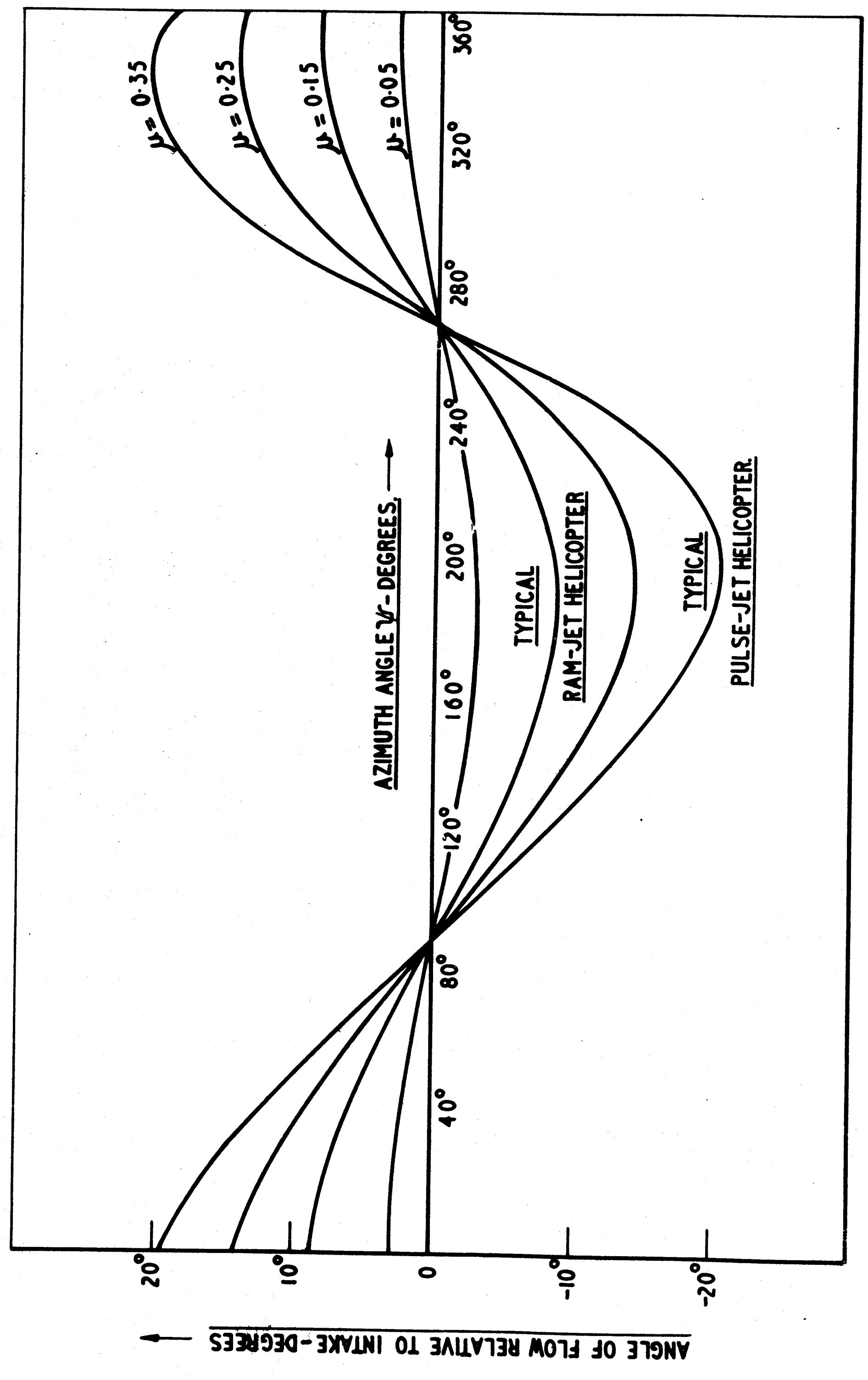



Fig 3.1

## 3.2 DYNAMIC DESIGN.

The dynamic design of a helicopter rotor system is of more importance to the success of a new aircraft than any other single aspect of design. Many helicopters so far produced have been limited in their usefulness by dynamical considerations; either on the ground in the form of "ground resonance" or by rotor vibration in the air. On the Auster design these problems have been faced in two ways. Firstly, by the choice of a configuration, which avoids the bad features of the conventional fully, articulated rotor, whilst at the same time retaining all its essential characteristics. Secondly, by the aid of a considerable programme of theoretical research into the fundamental mechanics of rotor behaviour and vibration, correlated with available flight measurements.

3.2.1. Rotor Configuration.

With the wide range of rotor speed and aircraft weights in which a tip jet helicopter can operate, it is not possible to construct a rotor blade which is strong enough to withstand the bending moments due to lift without the inclusion of some form of flapping hinge. This hinge may be of the conventional type, where both blades are hinged separately, or of the "see saw" type, with the two blades connected rigidly together. The latter type suffers from two serious handicaps;

(a) Whilst it eliminates fluctuating root bending moments due to change of blade azimuth position in forward flight, it cannot allow for steady changes of bending moment due to change of rotor speed or thrust from the design condition. This is particularly important with an aircraft, which has a disposable load of nearly 60%.

(b) See saw rotors are subject to a form of instability called "weaving" which is analogous to wing flutter. The problem is of the same order of magnitude as "ground resonance" and there is every reason to believe that its solution might require as much technical effort. The placing of concentrated masses at the blade tips is directly de stabilising in this connection.

Having accepted the desirability of flapping hinges, the problem arises of accommodating the coriolis forces caused by the blades flapping with respect to the axis of rotation. On a fully articulated rotor, this is done by the use of drag hinges, but the penalty for their use is increased complexity in the rotor, the need for highly damped oleos in the undercarriage to control "ground resonance" and, in a two bladed rotor, very severe second harmonic (twice rotor) vibration.

Coriolis forces in the blades of the Auster helicopter are easily carried by the structure in all normal flight envelope conditions. The possibility does exist, however, of encountering a severe gust at very low rotor speed, and the resulting coriolis forces would then be large enough to break the rotor. This is countered by the use of one main and one subsidiary bolt in attaching the blade to the root arm fitting. In the event of the blade being subjected to an in plane bending moment which is greater than 90% of the proof strength of the blade, then the subsidiary bolt shears and the blade moves on the main pin using it as a drag hinge and relieving the coriolis forces. This movement is noticed by the pilot as vibration, but the flying qualities of the aircraft are in no `way impaired. (An application for the patenting of this system has been filed by Auster Aircraft, Limited).

The rotor system resulting from this chain of reasoning is, therefore, an orthodox fully articulated one, except that the drag hinges `only function as such in the unusual case of flying into a severe gust at low rotor speed. The bad effects of drag hinges Ä excessive vibration and "ground resonance" are, thereby, avoided.

3.2.2. Vibration.

All rotors are subject to vibration, both vertical and in plane, and, although the rotor design avoids the worst effects, a two bladed rotor is always subject to unpleasant vibrations, which can only be reduced to acceptable levels by designing the rotor with this as a primary aim. The chief sources of vibration are briefly listed in the following paragraphs, together with the design action, which has been taken to minimise or eliminate them.

**VERTICAL VIBRATION,**

First Harmonic. (once rotor) — This is due to geometrical and dynamic differences between the two blades. Although appreciable on present day aircraft, the dimensional accuracy to be expected from the manufacturing techniques envisaged should enable this source of vibration to be considerably reduced,

Second Harmonic. — This frequency is the result of second harmonic lift
Twice rotor fluctuations on individual blades, since the maximum value of the tip speed ratio u is 0.2 on the Auster design, second harmonic lift variations will be negligibly small.

Higher Harmonics. — The two bladed rotor will theoretically be subject to vibration frequencies whose harmonic number is an even integer, but in the absence of blade resonance (which is avoided by the combination of small mass and high centrifugal forces) their presence is not detectable in a rotor of the present design.

**<u>IN-PLANE VIBRATION.</u>**

<u>First Harmonic</u>. (once rotor)

This vibration is common on conventional fully articulated rotors and is due to the blades being "out of drag track". That is, the blades are not equally spaced round the disc due to one of several possible causes, and the rotor is unbalanced. On the Auster design, the absence of drag hinges eliminates this source of vibration.

A second source is unequal thrusts from the two jet units. The extreme case of one engine out is considered below in Section 3.2.3 and is shown to be acceptable. This leads to the further conclusion that differences in drag between the two blades in a rotor have a negligible effect when it is rigid in the plane of rotation. Thus no blade tracking is required, other than a simple check to ensure an equal distribution of lift between the two blades.

<u>Second Harmonic</u>. (twice rotor)

With a two-bladed rotor, the large first harmonic force fluctuations on the blades are transmitted to thehub as a second harmonic vibration, whereas on a three— bladed rotor, they effectively vanish. This means, in general, that a two-bladed rotor has a very much higher vibration level than a three—bladed one, particularly if it has drag hinges to magnify the forces involved.

A considerable amount of research into this problem has been carried out and it has been found possible to virtually eliminate first harmonic force fluctuations on the blade (and hence second harmonics at the hub) by balancing coriolis forces against the aerodynamic ones. Since this is only possible at one condition (i.e. mean cruising speed and weight for this aircraft) a further improvement has been obtained by mounting the thrust bearing in rubber so that the fuselage is sysmic at this frequency. It is emphasized that the control of second harmonic vibration on a two—bladed rotor requires action at the Design Stage and cannot be left until Flight Trials. It is the opinion of this Company that any proposal to produce a fully articulated two bladed rotor would be most unrealistic.( (Figs. 3.2. and 3.3,).

Higher Harmonics. Third harmonic force fluctuations on the blades produce both second and fourth harmonics on the hub. For the low tip speed ratio values at which this design operates, these and higher frequencies can be regarded as negligible.

3.2.3. Vibration due to single-engined flight

In the event of the failure of one jet unit, the helicopter should continue to fly and maintain height on the remaining unit. As shown in Section 11, this is possible on a performance basis, so that the only objection to single engine flying is that of vibration. Fig. 3.4 shows the maximum single engine vibration for this design in relation to the generally agreed limits, together with the corresponding point for a typical pulsejet helicopter design. It is evident that the high tip speed and the low jet thrust associated with the ramjet enable single engined flight to be carried out without discomfort to the occupant. It follows that for maximum fuel economy, single engine flight could be a standard practice, after suitable modification.

With the lower tip speeds and higher jet thrusts of the pulse jet type of helicopter it is interesting to note that single engine flight is not a practical proposition.

3.2.4. "Ground Resonance".

It is broadly correct to say that with small" rigid" rotors the phenomenon of "ground resonance" cannot occur if there are no drag hinges Thus on the Auster design the possibility of costly development work in this field does not arise. In the event of the shear pins failing, clue to excessive in-plane loading of the blade in a gust, the rotor will then be fully articulated and "ground resonance" becomes a possibility when landing. Calculations for this system have been made, using the theory developed by Coleman, and these indicate that no instabilities occur in the possible R.P.M. range. These results are not conclusive in the present state of knowledge on the subject, but it should be remembered that the chance of encountering a severe gust at a low rotor speed, and then experiencing "ground resonance" on landing is extremely remote.

The elimination of rotor drag hinges leads to the conclusion that an orthodox highly damped oleo is not mandatory. This is well illustrated by the many American designs, which are fitted with simple welded-up skids. Nevertheless, calculations show that to meet the stringent landing requirements of B.C.A.R., Section 6, would require an undercarriage of the above type weighing about 250 lbs., even if it were allowed to collapse in the 12-ft./sec.-descent speed case. A second disadvantage with this simple skid system is that ordinary "shaft whirl" ground resonance still occurs and can grow to a dangerous amplitude if no undercarriage damping is present in the system.

The standard oleo type of aircraft undercarriage leg is still an attractive proposition, therefore, but both its first and servicing costs are relatively high, and for a helicopter, which must be folded and stowed, it has special disadvantages.

## VARIATION OF DRAG AND CORIOLIS FORCE PER BLADE WITH AZIMUTH ANGLE.

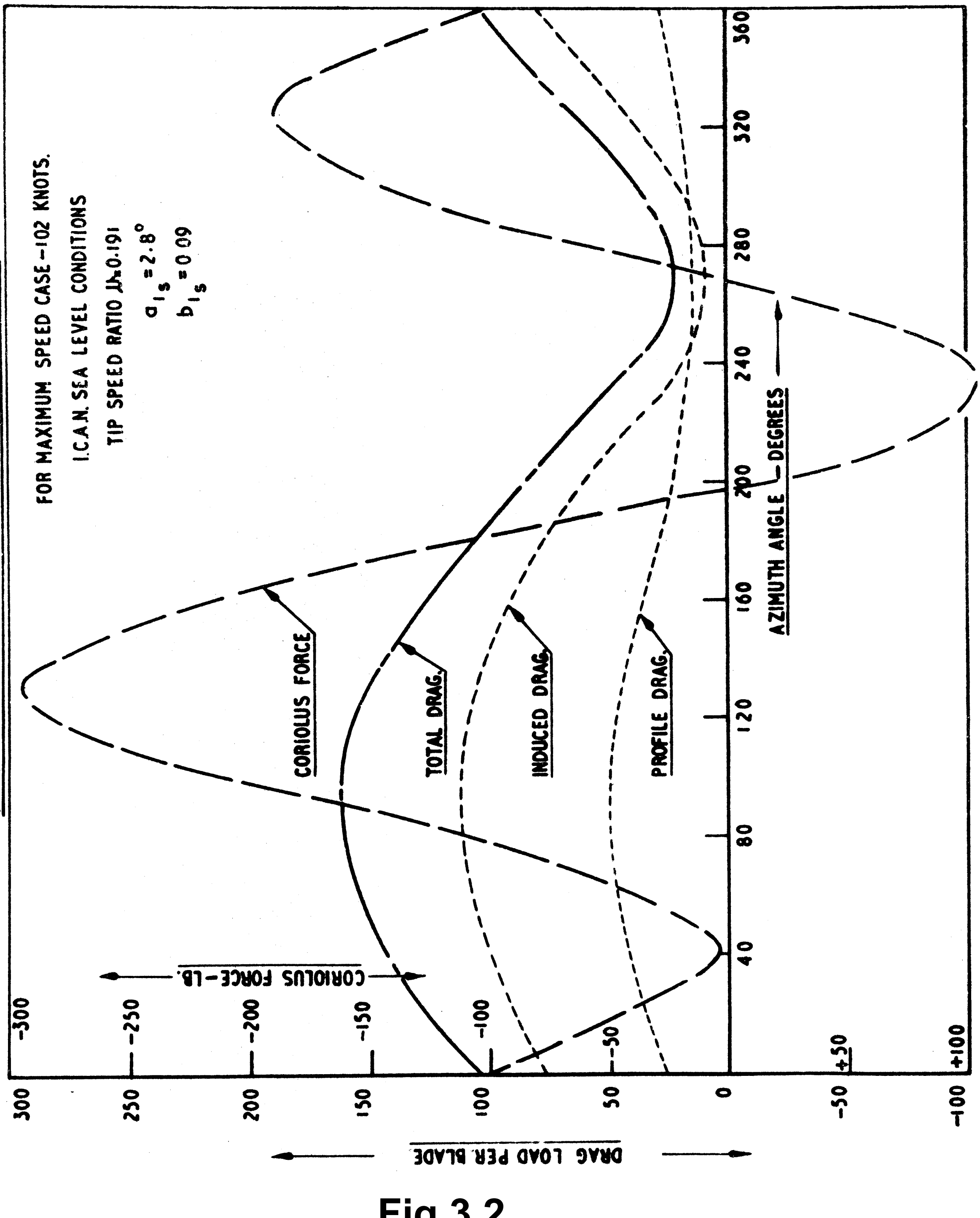



Fig 3.2

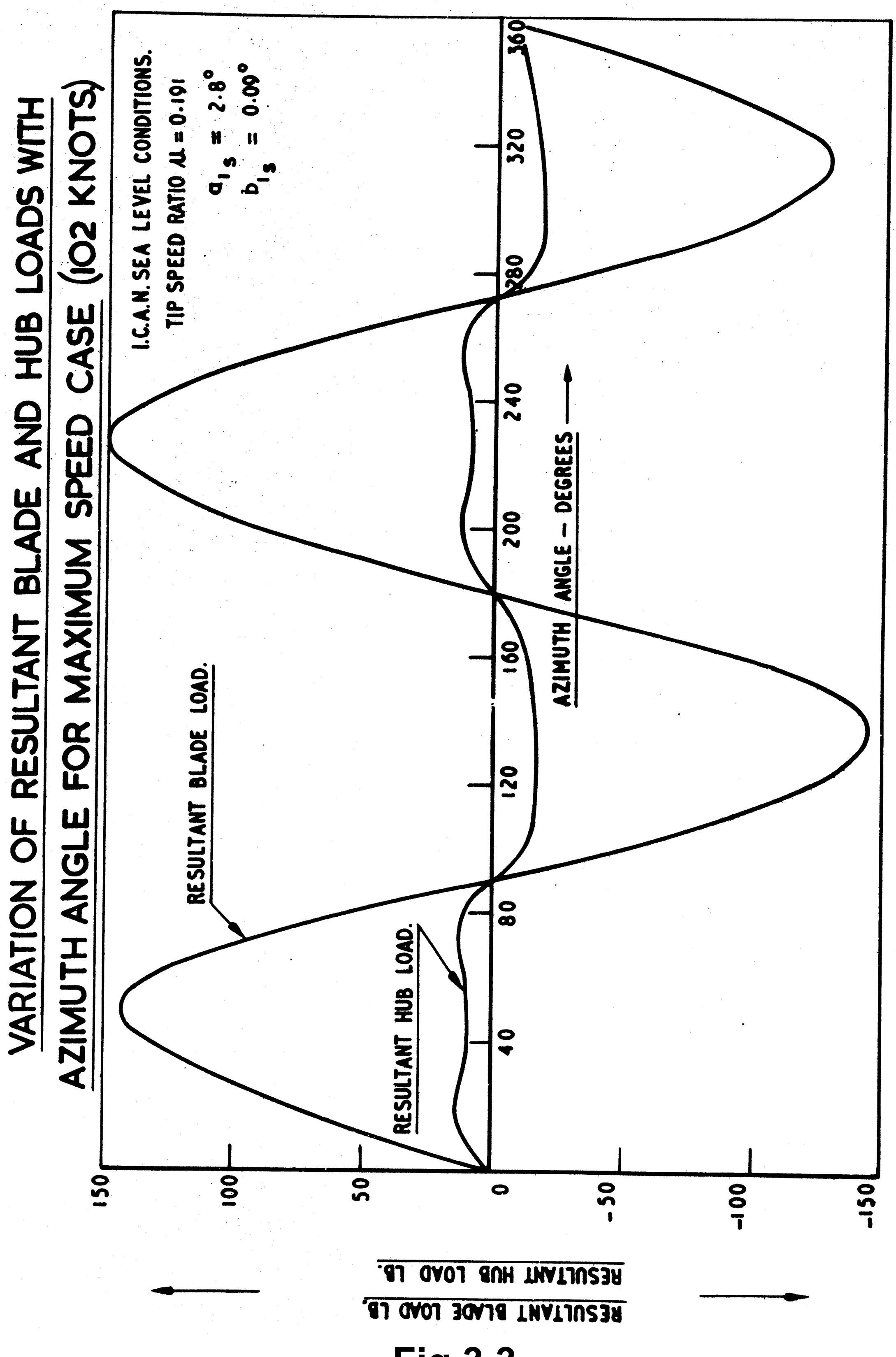

VARIATION OF RESULTANT BLADE AND HUB LOADS WITH
AZIMUTH ANGLE FOR MAXIMUM SPEED CASE (102 KNOTS)
I.C.A.N. SEA LEVEL CONDITIONS.
TIP SPEED RATIO $\mu$ = 0·191
$a_{1_s}$ = 2·8°
$b_{1_s}$ = 0·00°
RESULTANT BLADE LOAD.
RESULTANT HUB LOAD.
AZIMUTH ANGLE — DEGREES
40
80
120
160
200
240
280
320
360
150
100
50
0
-50
-100
-150
RESULTANT HUB LOAD LB.
RESULTANT BLADE LOAD LB.

Fig 3.3

# VIBRATION DUE TO FAILURE OF ONE TIP PROPULSIVE UNIT.

(WITHOUT DRAG HINGES)

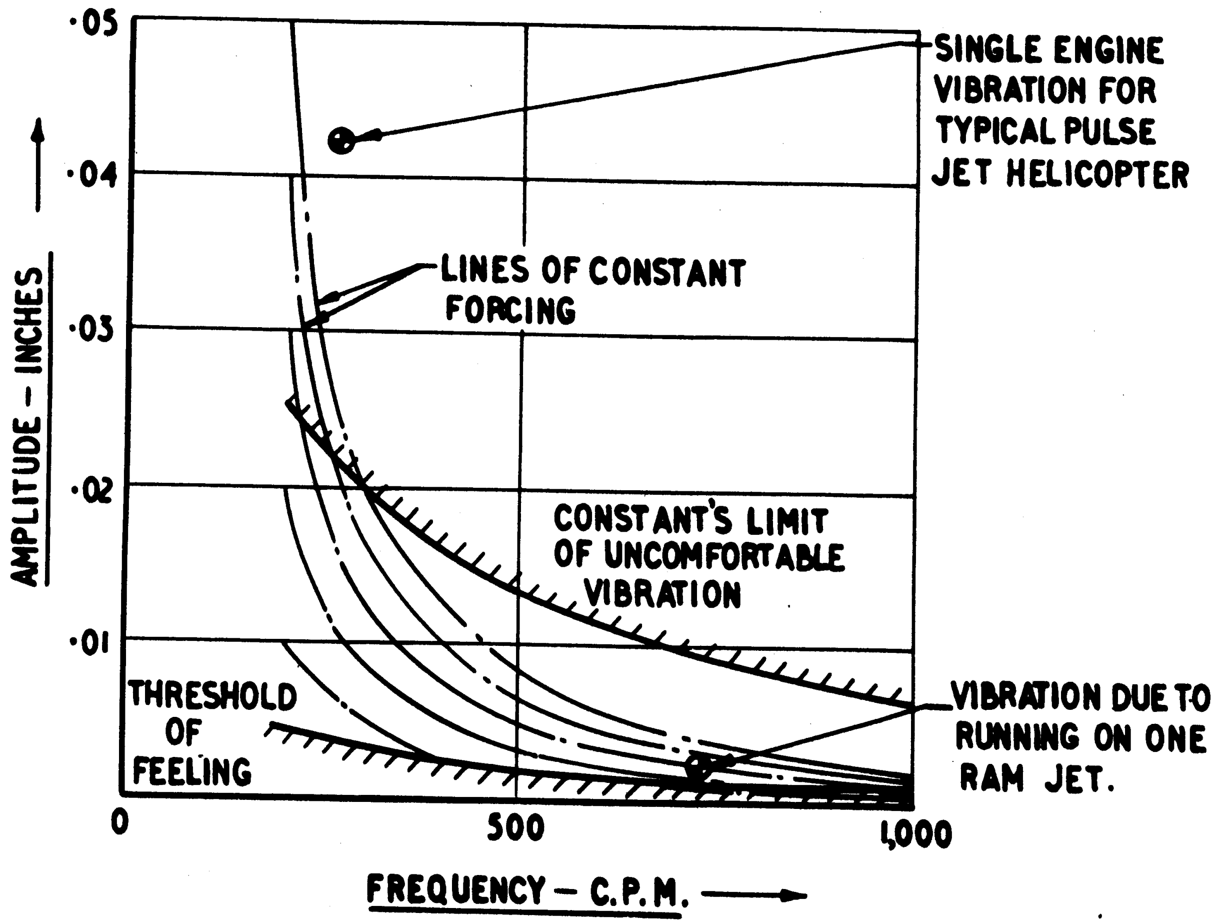



Fig 3.4

## 3.3. AERODYNAMIC DESIGN.

3.3. 1. General Aerodynamic Design.

Viewed aerodynamically, the Auster helicopter is essentially a ram jet powered rotor, with the necessary controls running from the hub to the pilot. Because of the number of auxiliaries fitted, coupled with the rotary seal and bearing friction, a small tail rotor is found to be preferable to an inclined hinge rudder.

The parasitic drag of the fuselage is naturally high, being 116.3 lbs. at 100 ft/sec, but since the estimated top speeds are well above the Specification requirements, no special attempts have been made to reduce drag. Fig.3.5.gives a typical breakdown of power required by the rotor plotted against forward speed. At the cruising speed of 70 knots required by the 3pecification, it is evident that the parasitic power component is a relatively small proportion of the total power required. It follows that there is no need f or extensive streamlining of the fuselage.

In Fig.3.6 .the rotor "gust clearance chart" is given. The blade deflection parameter $\Delta\ \beta/\lambda G$ MAX is plotted against the tip speed ratio for the four principal azimuth positions.

$\Delta\ \beta$ MAX is the maximum angular flapping deflection of the blade from its equilibrium position, in a sharp edged gust, the sign being positive when the blade flaps against the direction of the gust.

$\lambda G$ is the inflow ratio of the gust (gust velocity divided by the rotor tip speed) and has a positive sign for a down gust. Both $\Delta\ \beta$ max and $\lambda$ G are angles.

A gust deflection chart of this nature is a complete statement of blade motions in gusts f or the design, under any operating conditions. The worst deflection under normal operating conditions occurs at 800 ft/sec. tip speed, in a 35 ft./sec. Sharp edged gust, when the blade can reach a maximum deflection of 2.80° from its equilibrium position.
Even with zero coning, this is well clear of the structure, but since a coning angle of about 0.8° is usual, the droop stops have been built to give a droop of 2° below the horizontal.

Controls are of conventional pattern, and the ranges are specified in the Table below. Centralising springs are fitted to the cyclic stick to allow the pilot to trim out loads, and an adjustable friction nut enables the collective pitch stick to be stiffened up or locked as required. Control force feedback from the rotor is minimised by correct balancing of the blades in production and careful control of the blade inertia axis, but this does not affect blade interchangeability.

## TABLE OF CONTROL RANGES

| | |
|---|---|
| Collective Pitch Range | 120 |
| Mm. root angle | 7,50 |
| Max. root angle | 19.5ø |
| | |
| Cyclic Pitch | |
| Fore and aft | +/- 6° |
| Lateral | +/- 6° |
| | |
| Total Blade Angles | |
| Min. root angle | 1.5° |
| Max root angle | 25.5° |

In Figs, 3.7 and 3.8 the rotor control angles to trim for typical conditions of flight are plotted, for vertical and forward flight respectively. It is from consideration of the envelope values of these angles that the control ranges and the Table have been determined.

3.3.2. Rotor Design,

The aerodynamic design of the rotor is the key to the performance and efficiency of the Auster design. The fundamental requirement is that the ramjet shall be allowed to run as fast as possible whilst avoiding shock waves or adverse compressibility effects at the blade tip. This is achieved by a tip section thickness of 4%, which has critical blade Mach Numbers of

| | | |
|---|---|---|
| $M_{CRIT}$ | 0.925 for $C_L =$ | 0 |
| $M_{CRIT}$ | 0.84 for $C_L$ | 0.3 |

The provision of adequate bending strength at this thickness leads to a section, which is almost entirely solid steel spar at the tip and, therefore, a heavy blade. This combination of high tip speed and high weight leads to a rotor, which has many attractive features. The most obvious of these is the large amount of kinetic energy, which can be stored in the rotor and, in Fig. 3.9; it is shown that for rates of descent as high as 50 ft./sec., a complete flare-out landing can be achieved with only a 12% reduction in Rotor R.P.M.

The second feature of' the high tip speed is that the maximum tip speed ratio, u= 0.2. This is very much lower than the maximum values currently achieved, and since fluctuating loads in the rotor blade are proportional to u. or $u^2$ this results in a much lower overall vibration level. (The other extreme is presented by a pulse jet powered rotor with which the low tip speed would result in unusually high fluctuating loads). The small fluctuating loads result in low vibration levels in the structure, freedom from stall in the retreating blade and good overall control characteristics. This is clearly shown by Fig. 3.10 where it is evident that even at top speed the maximum blade angle of attack is less than $9^0$

The design of the rotor blade falls into two parts, ensuring freedom from compressibility effects and blade stall on the one hand, and designing for maximum efficiency (that is, minimum power loss) on the other. Fig.3.11 gives a typical plot of the governing parameters along the blade in the case of I.C.A.N. Sea Level Conditions. It is seen that the maximum lift coefficient, which occurs about halfway along the blade, less than CL = 0.6. Thus, stalling is not encountered under any flight condition envisaged in the Specification. The critical Mach number of the blade is also seen to be higher than the actual Mach Number, and it was found in practice that the maintenance of this requirement in all flight conditions governed the blade thickness and tip speed. For example, if it were decided for some reason to increase blade thickness, this would lead to lower maximum tip speeds and poorer performance.

The results of this and other similar calculations are embodied in Fig.3.12, which gives the envelope of permissible tip speed against aircraft forward speed for I.CA.N. Conditions. For completeness, the simple Hafner criterion (Reference 2), sometimes used for preliminary project purposes, is also plotted. It will be observed that, although a tip speed of 900 ft./sec. could be used for hovering and forward flight at low speeds, it must be dropped for higher speeds if the compressibility effects are to be avoided. Retreating blade stall occurs at a relatively low tip speed and for practical purposes, this phenomenon can be neglected in the Auster design,

A similar flight envelope is given in Fig.3.13 for Tropical Summer Conditions, and here the blade stalling occurs at slightly higher R.P.M. Because of the higher speed of sound, a greater tip speed can be used for forward flight and performance calculations have been carried out on the assumption that 900 ft./sec. is suitable for all normal tropical conditions.

# BREAKDOWN OF POWER REQUIRED AT I.C.A.N. SEA LEVEL.

A.U.W.= 1550 LB. ASSUMED FOR PARASITE DRAG ESTIMATE

$V_T$ = 800 FT/SEC

$e_{HOV}$ = 0·8

$D_{100}$ = 116·3 LB

p = 3·0 LB/FT²

$\sigma_{.7}$ = 025

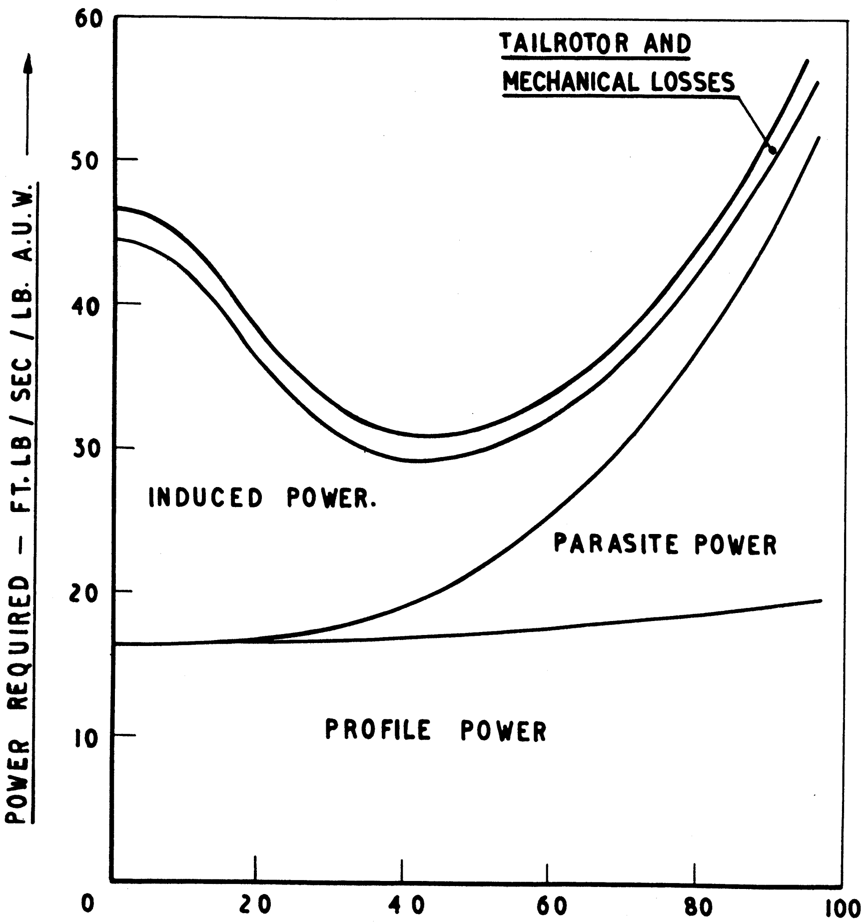



Fig 3.5

# GUST CLEARANCE CHART.

(SHARP EDGED GUSTS)

$\gamma = 2{\cdot}42 \qquad \delta_3 = 0$

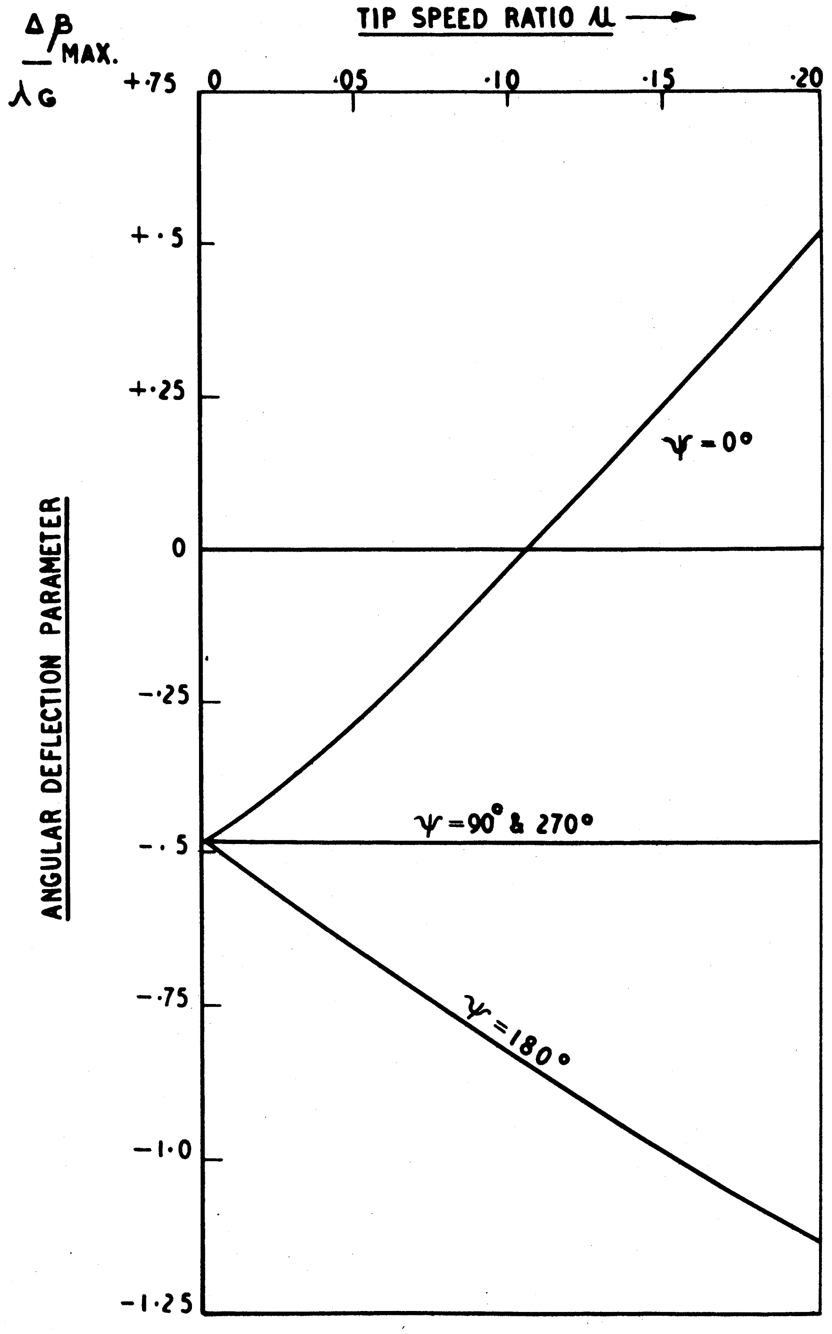



Fig 3.6

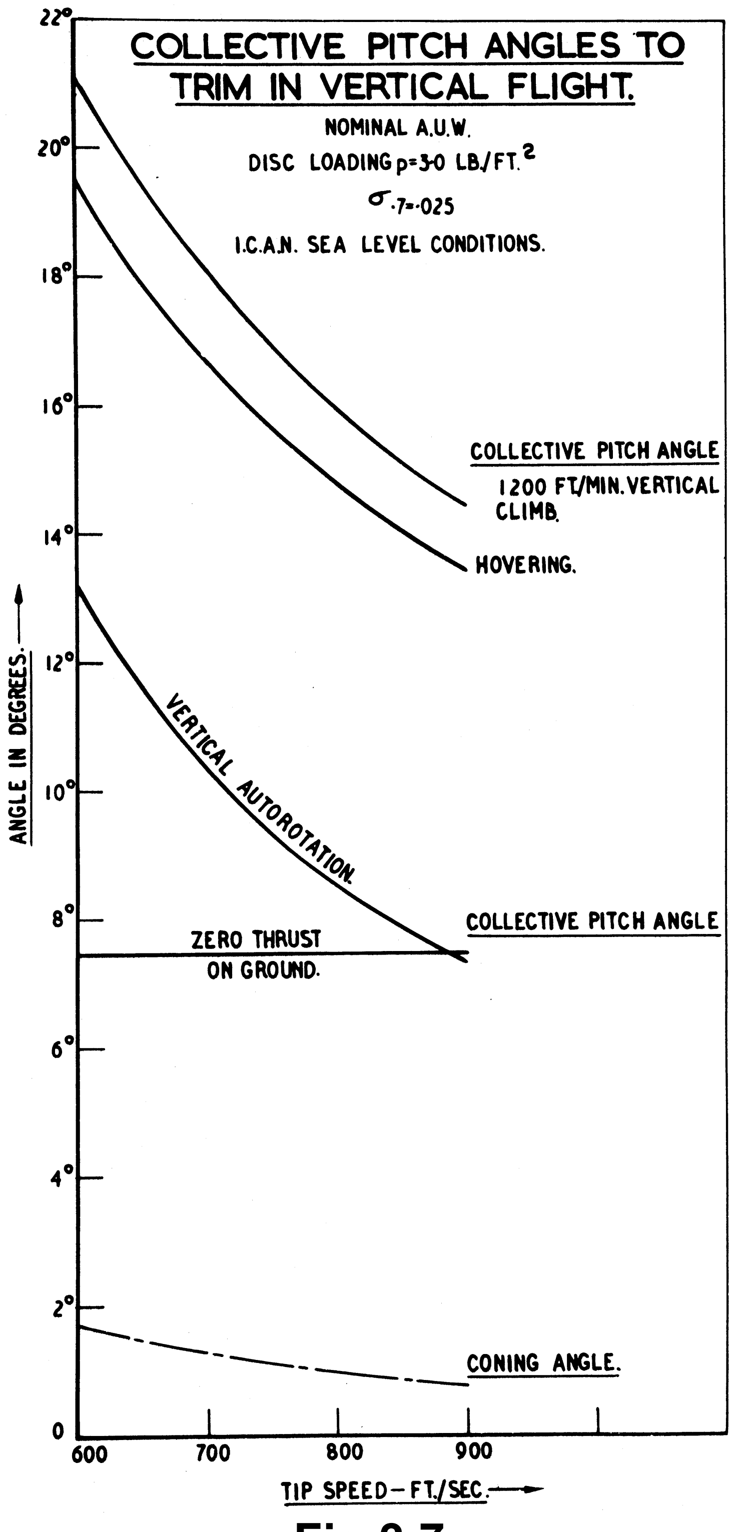

COLLECTIVE PITCH ANGLES TO TRIM IN VERTICAL FLIGHT.
NOMINAL A.U.W.
DISC LOADING p=3·0 LB./FT.$^2$
$\sigma$ .7=.025
I.C.A.N. SEA LEVEL CONDITIONS.
22°
20°
18°
16°
14°
12°
10°
8°
6°
4°
2°
0
ANGLE IN DEGREES.
COLLECTIVE PITCH ANGLE
1200 FT/MIN. VERTICAL CLIMB.
HOVERING.
VERTICAL AUTOROTATION.
COLLECTIVE PITCH ANGLE
ZERO THRUST ON GROUND.
CONING ANGLE.
600
700
800
900
TIP SPEED – FT./SEC.

Fig 3.7

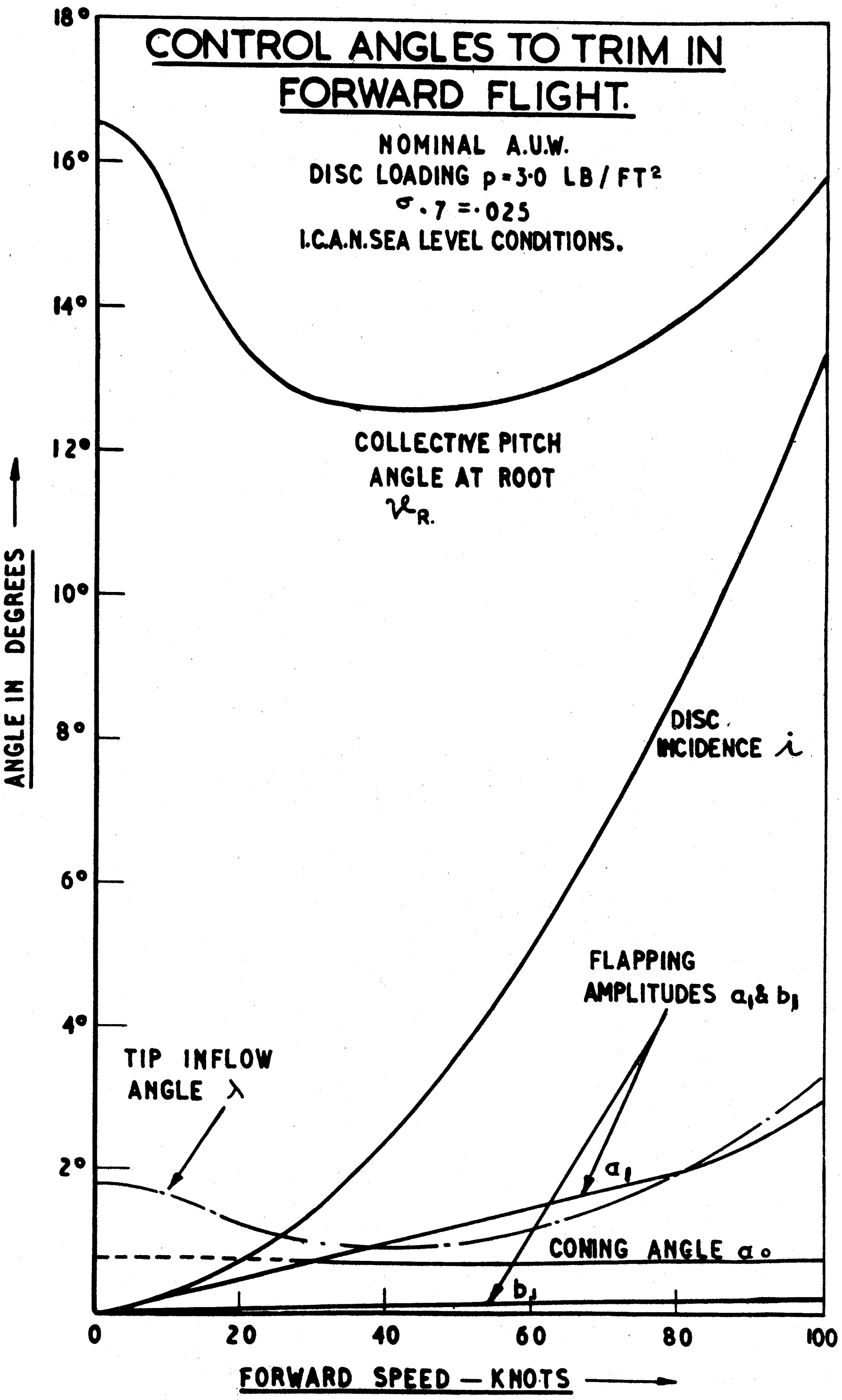

CONTROL ANGLES TO TRIM IN FORWARD FLIGHT.
NOMINAL A.U.W.
DISC LOADING p = 3·0 LB/FT²
σ .7 = ·025
I.C.A.N. SEA LEVEL CONDITIONS.
COLLECTIVE PITCH ANGLE AT ROOT $\vartheta_R$.
DISC INCIDENCE $i$
FLAPPING AMPLITUDES $a_1$ & $b_1$
TIP INFLOW ANGLE λ
$a_1$
CONING ANGLE $a_0$
$b_1$
ANGLE IN DEGREES
FORWARD SPEED — KNOTS
18°
16°
14°
12°
10°
8°
6°
4°
2°
0
20
40
60
80
100

Fig 3.8

# FLARE OUT PERFORMANCE AT I.C.A.N. SEA LEVEL

VARIATION OF VELOCITY AFTER FLARE-OUT WITH DROP IN ROTOR REVS. & INITIAL DESCENT VELOCITY.

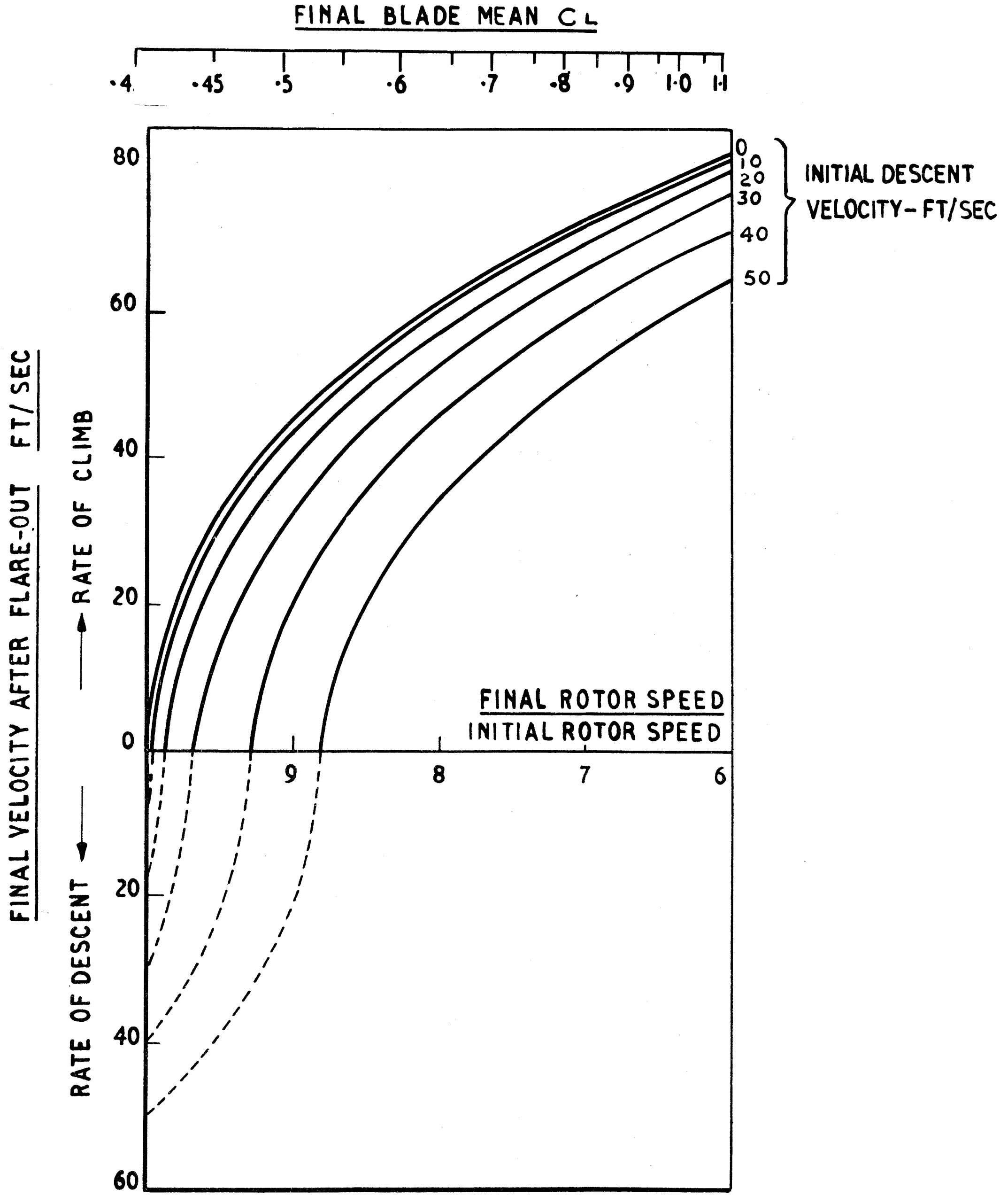



**Fig 3.9**

## ANGLE OF ATTACK DISTRIBUTION AT 102 KTS. ($\mu$=0·191)

TIP SPEED = 900 FT/SEC.
I.C.A.N. SEA LEVEL CONDITIONS.

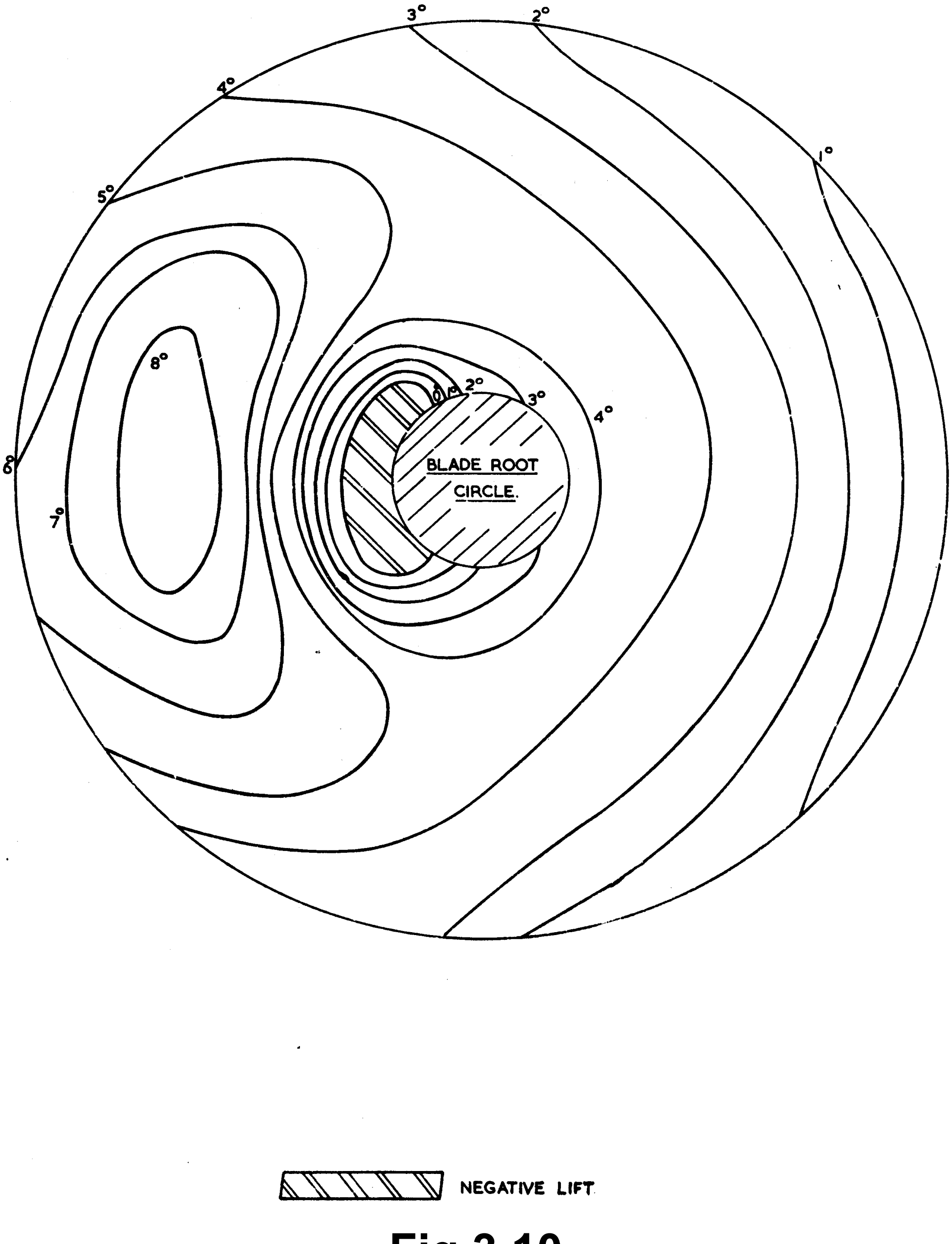



**Fig 3.10**

# TYPICAL STRIP THEORY ANALYSIS OF BLADE.

HOVERING AT I.C.A.N. SEA LEVEL

ROOT PITCH ANGLE $\vartheta_R = 14{\cdot}77°$

BLADE TWIST $\vartheta_T = 10{\cdot}6°$

TIP SPEED $V_T = 800$ FT/SEC

$p = 3{\cdot}0$ LB/FT$^2$

$\sigma_R = {\cdot}05$ $\sigma_{\cdot 7} = {\cdot}025$

NOMINAL A.U.W.

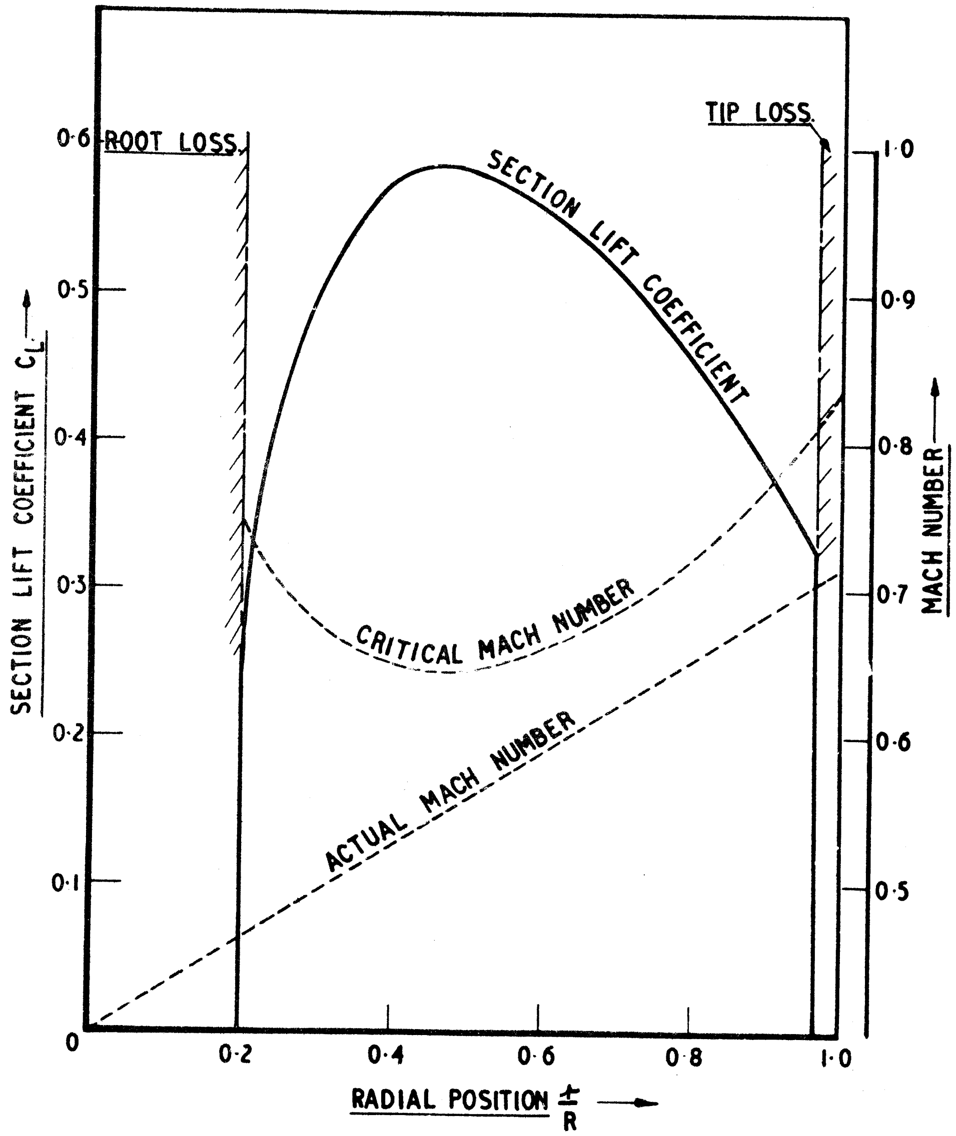



Fig 3.11

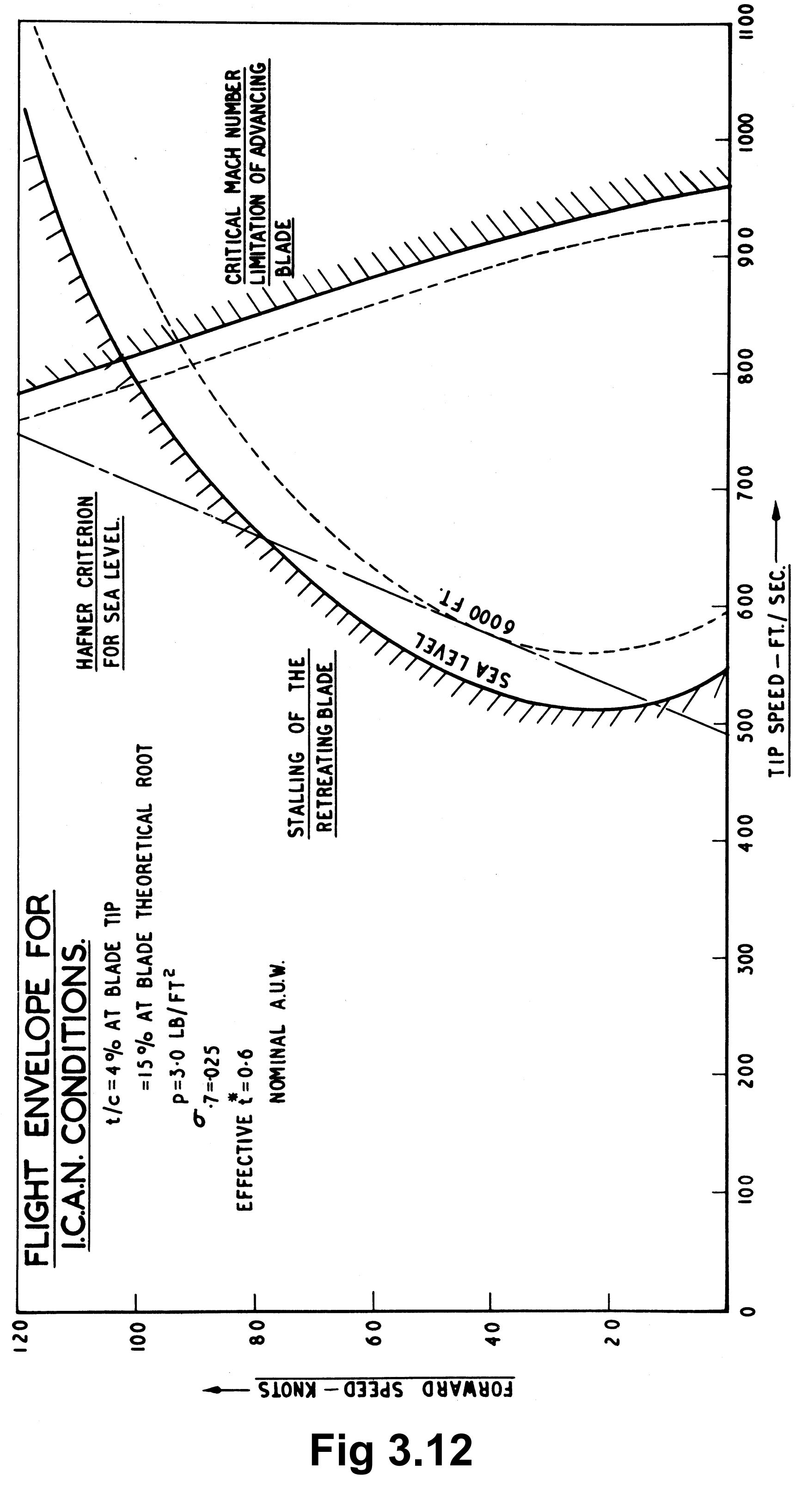

FLIGHT ENVELOPE FOR
I.C.A.N. CONDITIONS.
t/c = 4% AT BLADE TIP
= 15% AT BLADE THEORETICAL ROOT
p = 3·0 LB/FT²
σ.7 = 025
EFFECTIVE t* = 0·6
NOMINAL A.U.W.
CRITICAL MACH NUMBER
LIMITATION OF ADVANCING
BLADE
HAFNER CRITERION
FOR SEA LEVEL.
STALLING OF THE
RETREATING BLADE
6000 FT.
SEA LEVEL
TIP SPEED — FT./SEC.
0
100
200
300
400
500
600
700
800
900
1000
1100
FORWARD SPEED — KNOTS
20
40
60
80
100
120

**Fig 3.12**

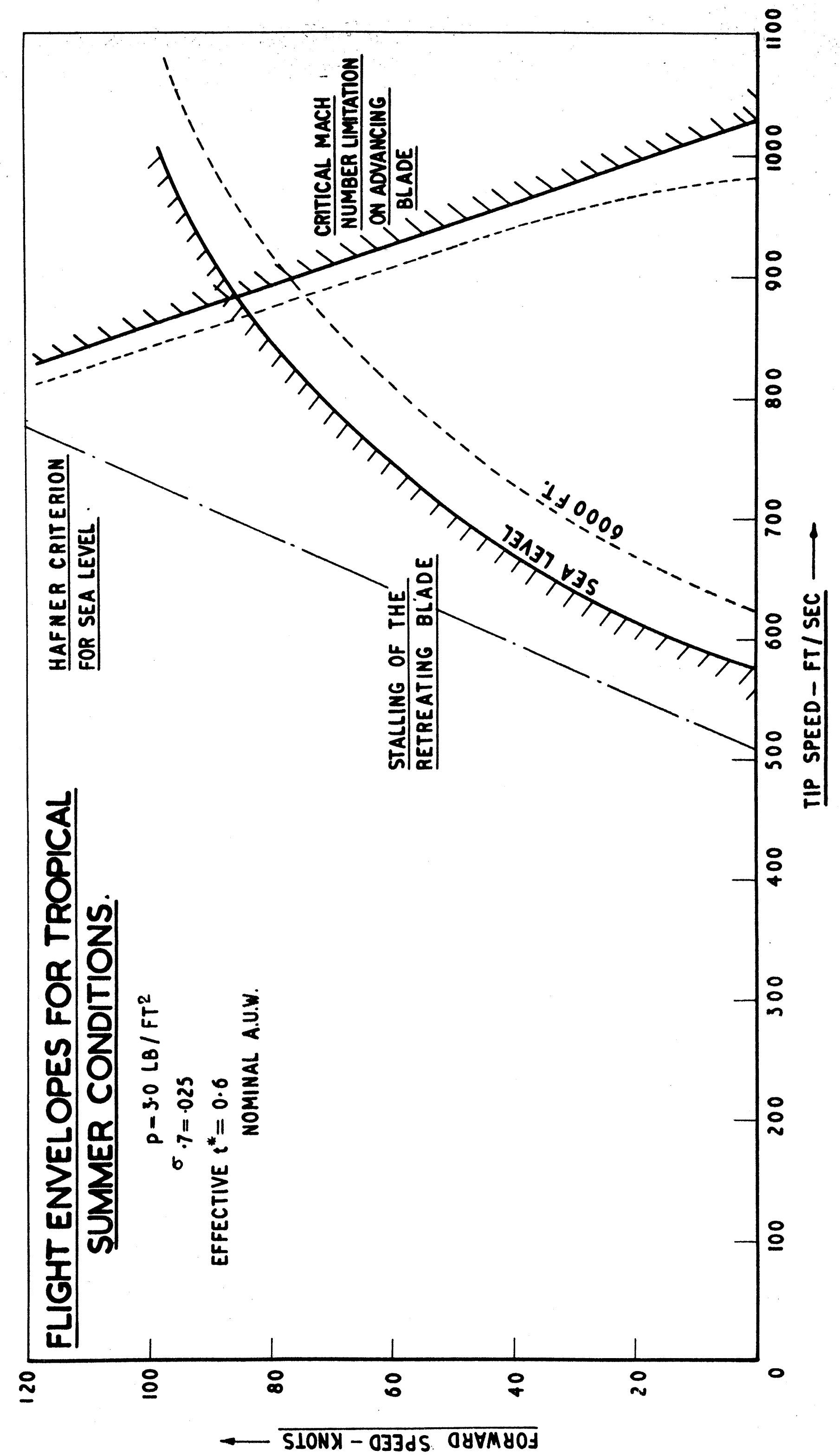

FLIGHT ENVELOPES FOR TROPICAL SUMMER CONDITIONS.
p = 3·0 LB/FT²
σ ·7 = ·025
EFFECTIVE t* = 0·6
NOMINAL A.U.W.
HAFNER CRITERION FOR SEA LEVEL
STALLING OF THE RETREATING BLADE
CRITICAL MACH NUMBER LIMITATION ON ADVANCING BLADE
SEA LEVEL
6000 FT.
TIP SPEED — FT/SEC
FORWARD SPEED – KNOTS
0
100
200
300
400
500
600
700
800
900
1000
1100
20
40
60
80
100
120

Fig 3.13

# AIRFRAME CONSTRUCTION

# SECTION 4 AIRFRAME CONSTRUCTION

The airframe has been designed throughout for cheapness of manufacture, coupled with the rugged construction, which is a feature of the Company's fixed wing A.O.P. aircraft. A welded steel tube structure has been established in the past as the most satisfactory one for an aircraft, which has to operate in the field for long periods. Steel is also used for all components, including controls, which are subject to fluctuating loads greater than 2% of ultimate.

The weight penalty is not excessive, and is held to be justifiable when the consequences of a fatigue failure are considered and to lessen the frequency of fatigue examinations.

## 4.1 FUSELAGE - STRUCTURE.

The basic structure is in the form of three steel tubes, which form a pyramid. At the apex of the pyramid, these tubes are welded to the thrust plate, in which the hub assembly is located, whilst at the base, the tubes join a partially braced box like structure of tubes, as shown in Fig.4.1. The four main uprights of this box are the four main undercarriage legs, the front two of which are integral with the structure.

The tail rotor is supported by a further three struts, which are made sufficiently robust for ground handling. The convenient position of these struts enables the rear of the aircraft to be lifted by one man for such operations as lowering the wheels or extending the undercarriage legs.

The fuselage structure is clearly depicted in Fig.4.1 The simplicity of the basic structure and its similarity to existing Auster fixed wing fuselages means that the manufacture of the prototypes can be undertaken with a minimum of jigging and, even in production quantities, the amount of tooling required would not be excessive.

The fuel tank is not a load-carrying member, and is quickly detachable. The standard 100-gallon tank is of aluminium alloy sheet and has no double curvature. The separation of the tank from the structure means that special tanks can be provided for special roles, which might be required in the future. An example is the provision of a bulletproof tank for aircraft, which might have to operate under fire.

In accordance with the Company's policy of producing a truly flexible design, the fuselage width is several inches wider than is strictly necessary to fair round the crew. This results in greater ease of getting into and out of the cockpit, and also enables bulky freight to be carried behind the seats. Although ambulance work is not foreseen in the Specification, it was felt that the ability to carry one sitting and, with slight modification, one stretcher patient would be of value provided it has no adverse effect on other aspects of the design.

The cockpit floor is a sandwich structure of light alloy skins with light alloy honeycomb core bonded in. This floor panel, which supports the seats, etc., is bolted to the welded steel framework and is readily removable for replacement in the event of damage.

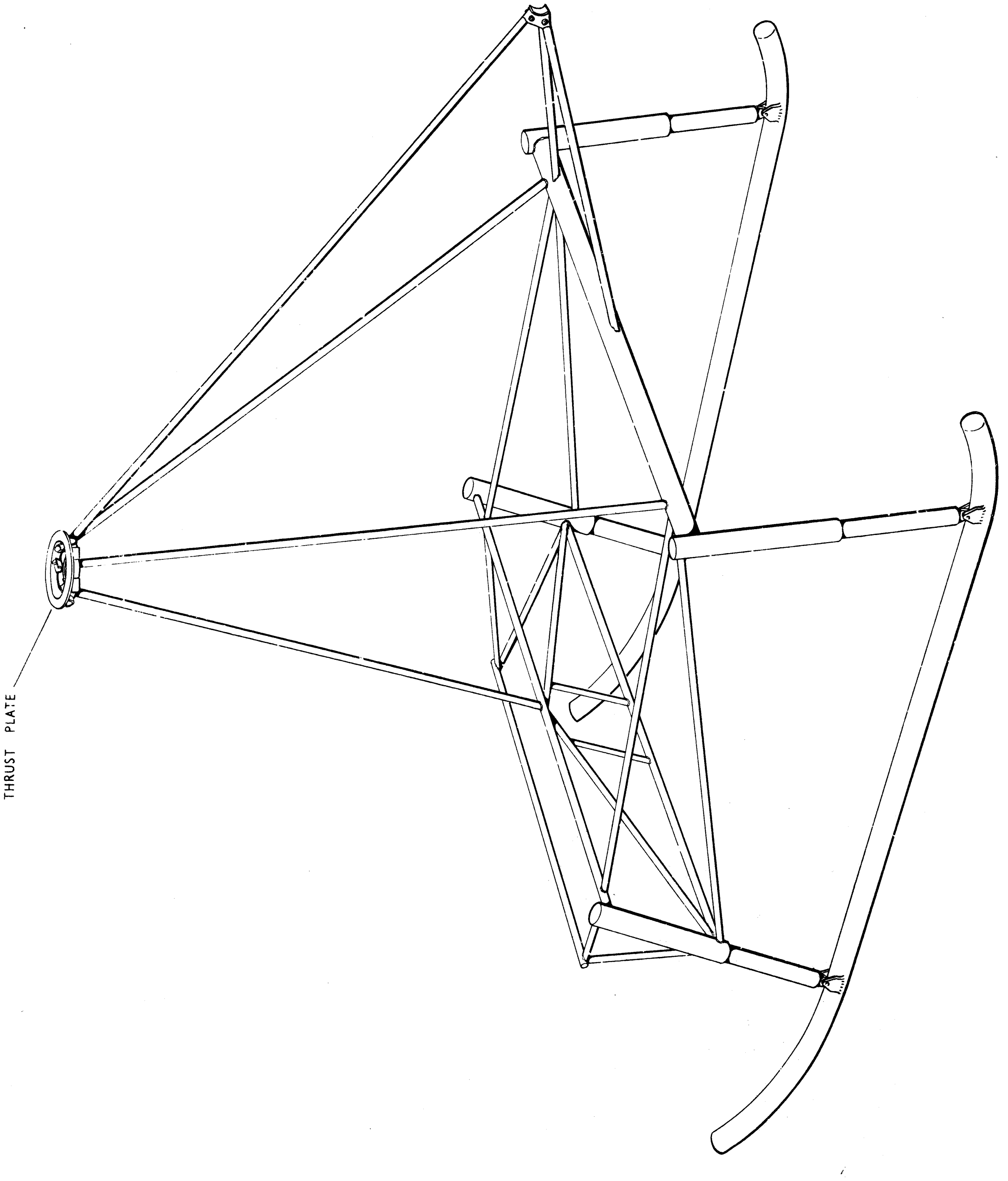



**Fig 4.1**

**4.2. FUSELAGE FAIRINGS.**

The bubble fairing enclosing the crew carries only those loads due to air forces and its own weight. It is divided into three main sections and can be entirely removed without affecting the airworthiness of the helicopter. In order of removal, the component parts are:

Side doors. - single curvature.

These large doors can be jettisoned for emergency exit or detached for roles where this is desirable. The door comprises a large window; part of which slides open, and a light alloy panel beneath. The light alloy framework is self—supporting.

Rear fairing. - Single curvature.

The rear fairing is wrapped into position, and can therefore be stored flat. It comprises a single sheet of Perspex.

Windscreen - Double curvature.

The large windscreen is made in four segments for cheapness and gives a field of view, which is superior to the minimum recommendations of the Helicopter Cockpit Layout Committee, without the need for side blisters. The complete windscreen is attached to the basic structure at three points and is self-supporting by virtue of a light frame, which reinforces its edge and also carries the side door hinges.

There are two small areas of single curvature transparency above the side doors, which are not detachable, and a moulded sheet, which forms the rear dome. It is proposed to use tinted Perspex for these in order to provide the crew with some relief from sun glare, unless the reduction in visibility is objected to.

The possibility of making the windscreen with single curvature has been considered, but it was found impossible to meet the bird impact stressing case without prohibitive weight increases.

## 4.3. MAIN_ROTOR BLADE.

The structural design of the rotor is governed by the conflicting requirements of large centrifugal forces caused by the ramjets and their high tip speed demands on the one hand, and by the necessity for minimum section thickness on the other. To meet these requirements, a steel spar was found to be essential. As Fig.4.2 shows, this is built up by placing fore—and—aft spacers between two 12 s.w.g plates, joining them together by bonding and riveting. Fibreglass fairings make up the aerodynamic envelope of the blade, and transmit air loads to the spar. The principal reason for using this material is its low stiffness (about one twelfth that of steel) which implies that to carry the same stress as steel (in an integral structure) it would need to be twelve times as far from the neutral axis. This property makes fibreglass an ideal material for trailing edge fairings, but its mechanical properties and ease of fabrication are also attractive. This Company has been carrying out pilot experiments with this material for some time on items such as fairings, pilots' seats, etc.

The blades are mass-balanced during manufacture so that the mean effective inertia axis position lies in front of the blade aerodynamic centre line. This is done to avoid flutter and minimize objectionable stick forces.

The hole at the root end (by means of which the blade is attached to the root arm) is provided with a replaceable bush.

Most of the manufacturing processes involved in the production of the blades are within the scope of the Company's existing facilities, and arrangements can readily be made to cope with the few exceptions

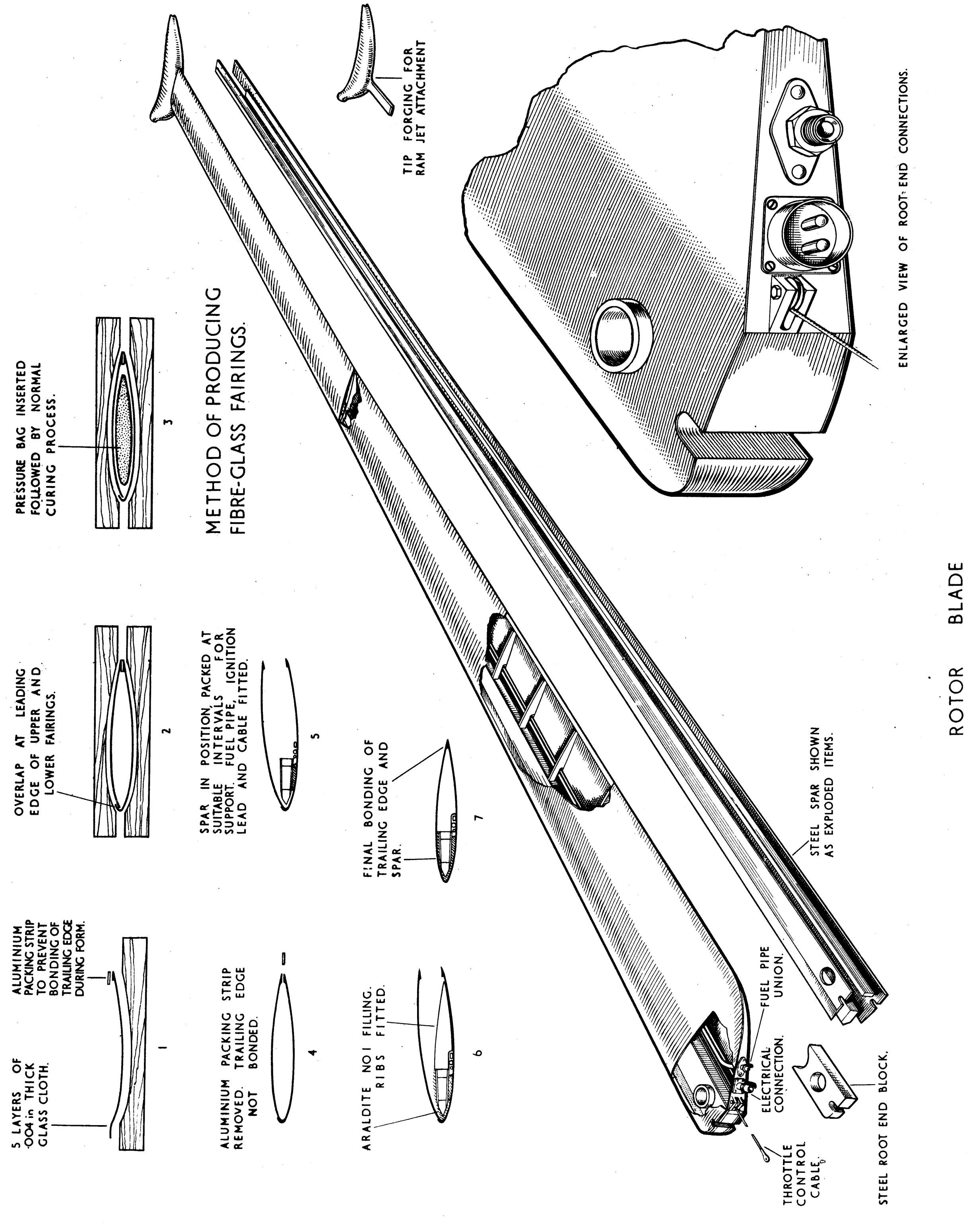

METHOD OF PRODUCING FIBRE-GLASS FAIRINGS.
5 LAYERS OF ·004 in THICK GLASS CLOTH.
ALUMINIUM PACKING STRIP TO PREVENT BONDING OF TRAILING EDGE DURING FORM.
1
OVERLAP AT LEADING EDGE OF UPPER AND LOWER FAIRINGS.
2
PRESSURE BAG INSERTED FOLLOWED BY NORMAL CURING PROCESS.
3
ALUMINIUM PACKING STRIP REMOVED. TRAILING EDGE NOT BONDED.
4
SPAR IN POSITION, PACKED AT SUITABLE INTERVALS FOR SUPPORT. FUEL PIPE, IGNITION LEAD AND CABLE FITTED.
5
ARALDITE NO I FILLING. RIBS FITTED.
6
FINAL BONDING OF TRAILING EDGE AND SPAR.
7
TIP FORGING FOR RAM JET ATTACHMENT
ENLARGED VIEW OF ROOT-END CONNECTIONS.
FUEL PIPE UNION.
ELECTRICAL CONNECTION.
THROTTLE CONTROL CABLE.
STEEL ROOT END BLOCK.
STEEL SPAR SHOWN AS EXPLODED ITEMS.
ROTOR BLADE

**Fig 4.2 (a)**

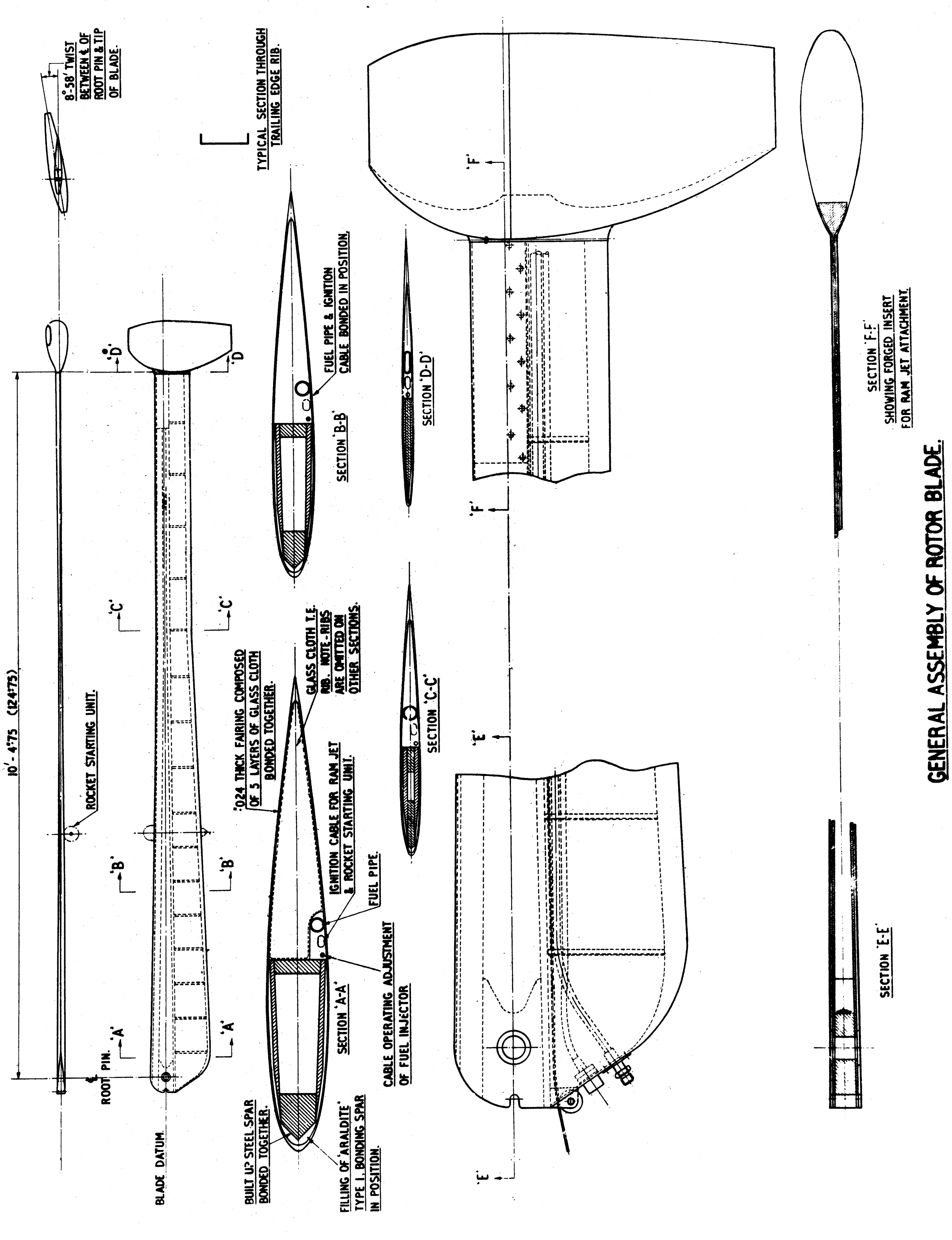

GENERAL ASSEMBLY OF ROTOR BLADE.
8°-58' TWIST BETWEEN ℄ OF ROOT PIN & TIP OF BLADE.
TYPICAL SECTION THROUGH TRAILING EDGE RIB.
10'-4·75 (124·75)
ROCKET STARTING UNIT.
℄ ROOT PIN.
BLADE DATUM.
'A'
'B'
'C'
'D'
'E'
'F'
FUEL PIPE & IGNITION CABLE BONDED IN POSITION.
SECTION 'B-B'
SECTION 'D-D'
SECTION 'C-C'
SECTION 'A-A'
SECTION 'E-E'
SECTION 'F-F' SHOWING FORGED INSERT FOR RAM JET ATTACHMENT.
·024 THICK FAIRING COMPOSED OF 5 LAYERS OF GLASS CLOTH BONDED TOGETHER.
GLASS CLOTH T.E. RIB. NOTE - RIBS ARE OMITTED ON OTHER SECTIONS.
IGNITION CABLE FOR RAM JET & ROCKET STARTING UNIT.
FUEL PIPE.
CABLE OPERATING ADJUSTMENT OF FUEL INJECTOR
BUILT UP STEEL SPAR BONDED TOGETHER.
FILLING OF 'ARALDITE' TYPE I. BONDING SPAR IN POSITION.

Fig 4.2 (b)

## 4.4. BLADE ROOT ARMS.

The blade root arms are relatively simple components, the function of which is to transmit the centrifugal loads and in plane bending moments from the blade to the hub whilst permitting feathering by the pilot.

At the hub end of the blade root arm is a forged fork end, through which passes the flapping hinge pin (this forming the attachment to the hub). At the outboard end of the forged fork end is milled a fixing for the laminated torque bar, which consists of a pack of several thin stainless steel links. The great tensile strength of the torque bar resists the centrifugal loads, whilst its low tensional rigidity permits the easy variation of blade pitch.

The outer end of the torque bar is bolted to the blade attachment fork end which, in order to transmit the in-plane bending, is bolted to the tubular sleeve. The inboard end of the sleeve is attached to a flanged collar which, in turn, is bolted to the flange of the housing around the two taper-roller bearings which help to feed the in-plane bending moments into the forged fork end. The pitch change arm, which transmits the control movements from the swash-plate via a push rod is a simple tubular part with a welded-in end plate, by means of which it is fastened to the flange of the bearing housing.

Replaceable bushes are provided at all points where wear can be expected to take place.

All manufacturing processes are conventional and within the scope of the firm's existing facilities.

The root arms are depicted in Figs. 4.3 and 4.5.

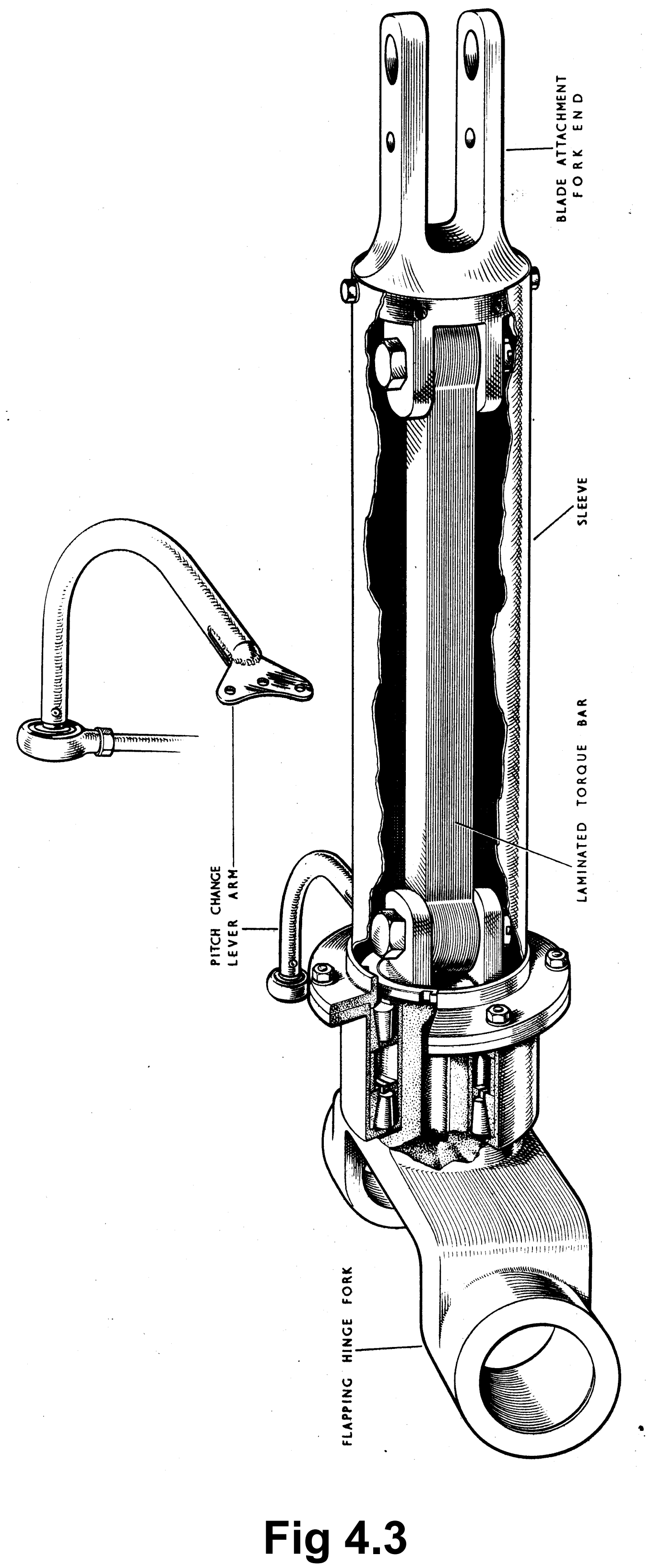

BLADE ATTACHMENT
FORK END
SLEEVE
LAMINATED TORQUE BAR
PITCH CHANGE
LEVER ARM
FLAPPING HINGE FORK

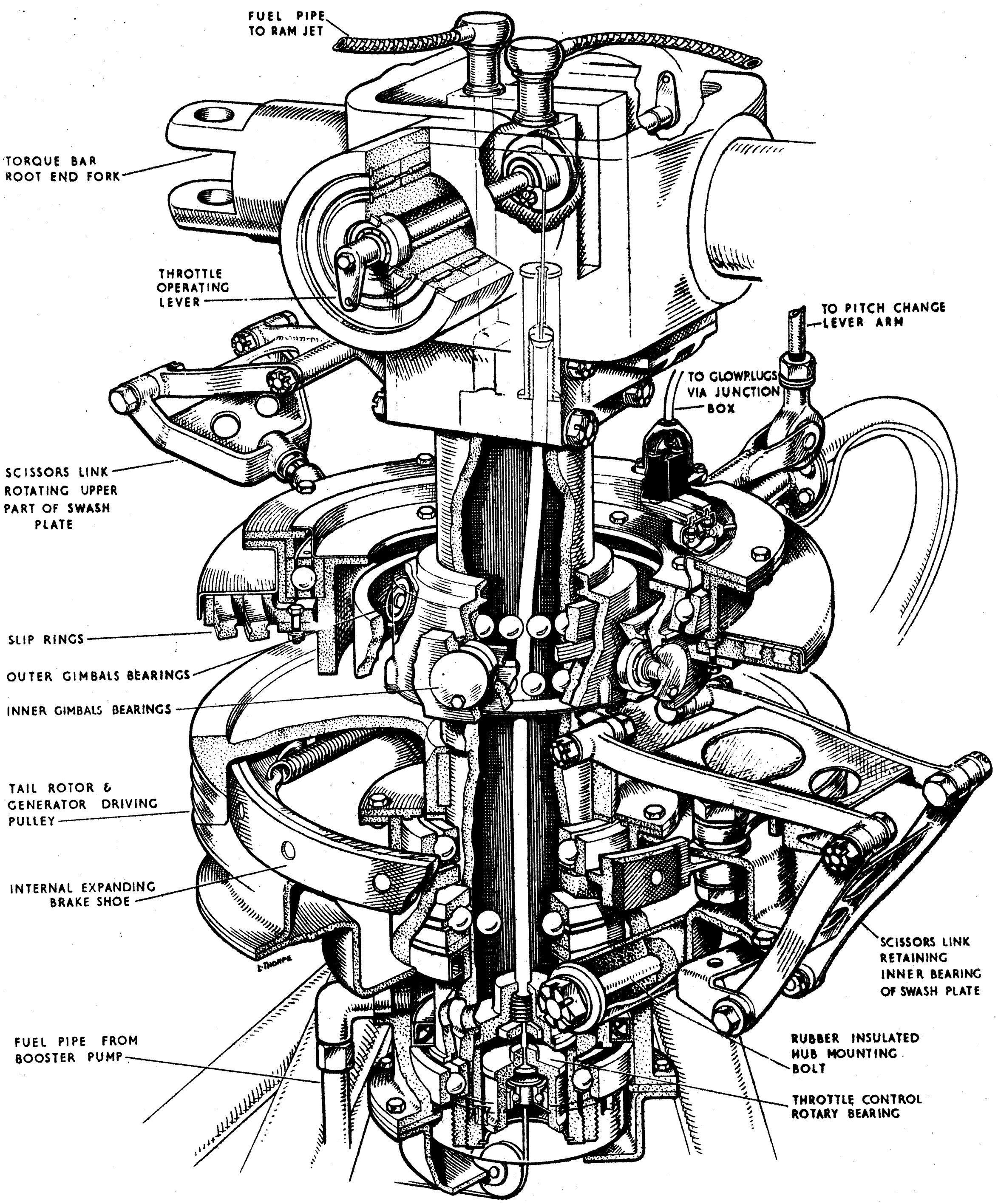
**Fig 4.3**

## *4.5.* Hub Assembly

The hub assembly which is illustrated in Fig.4.4, is a fairly compact unit mounted directly on to the thrust plate at the apex of the basic fuselage structure. The actual attachment takes the form of rubber insulated mounting bolts, designed to restrict the vibrations reaching the fuselage. The hub shaft is a substantial machined item running in standard ball races, which are contained, in a flanged machined housing attached to the thrust plate. At the bottom end of the bearing housing, immediately above the lower ball race, a rotary seal is provided so that fuel entering the bearing housing from the fuel feed pipe can be fed through transfer ports into the rotor shaft, and thence through flexible pipes at the top of the shaft into the rotor blades. Also passing up the centre of the rotor shaft is the throttle control cable, which passes through a guide tube to a lay shaft housed in the centre of the flapping hinge pin at the top of the rotor hub. The guide tube referred to forms a fuel-tight housing for the cable so that the cable can be readily changed without disturbing any fuel-tight seals.

Immediately above the fuselage thrust plate, a large diameter machined disc is keyed to the rotor shaft. The inner surface of the outer flange of this disc is used as a friction surface for the rotor brake, which is of the internal expanding shoe type. The outer surface of the flange is provided with three machined grooves for the V-belts driving the tail rotor and the generator-driving pulley.

On that portion of the main rotor shaft which lies above the V-belt pulley, a machined bearing housing forming the inner part of the swash plate is free to slide up and down, whilst being prevented from rotating by a set of scissor links connecting it to the fuselage thrust plate. Vertical motion of this housing adjusts the collective, pitch of the rotor blades through push rods to the blade root arms. In the sliding bearing housing are mounted the inner gimbals bearings engaging with the intermediate gimbals ring. This latter part, which is a machined item, carries the outer gimbals bearings, which engage with the outer-flanged machined ring of the swash plate assembly. This outer flanged ring, in turn, carries an insulated annulus into which are embedded slip rings through which passes the electric current for the glow plugs and starter cartridge firing. A ball race, fitted around the flanged outer ring and immediately above the slip ring assembly, is in contact with the rotating portion of the swash plate to which is mounted the attachment for the push-rods to the pitch change lever arms. This portion of the swash plate is caused to rotate with the rotors by a set of scissor links connecting it to projections from the rotor shaft, which also form the droop stops. The gooseneck, by means of which the pilot’s control forces are transmitted to the swash plate, is a simple tubular part bolted to an extension from the non-rotating flanged outer ring of the swash plate assembly. The electrical cables, carrying current to the slip rings, are attached to the side of the gooseneck and are attached to the slip ring terminals in an insulated junction box. All the manufacturing processes involved in the production of the hub assembly are straightforward in character and are within the scope of this Company's existing facilities.




MAIN ROTOR HUB ASSEMBLY

**Fig 4.4**

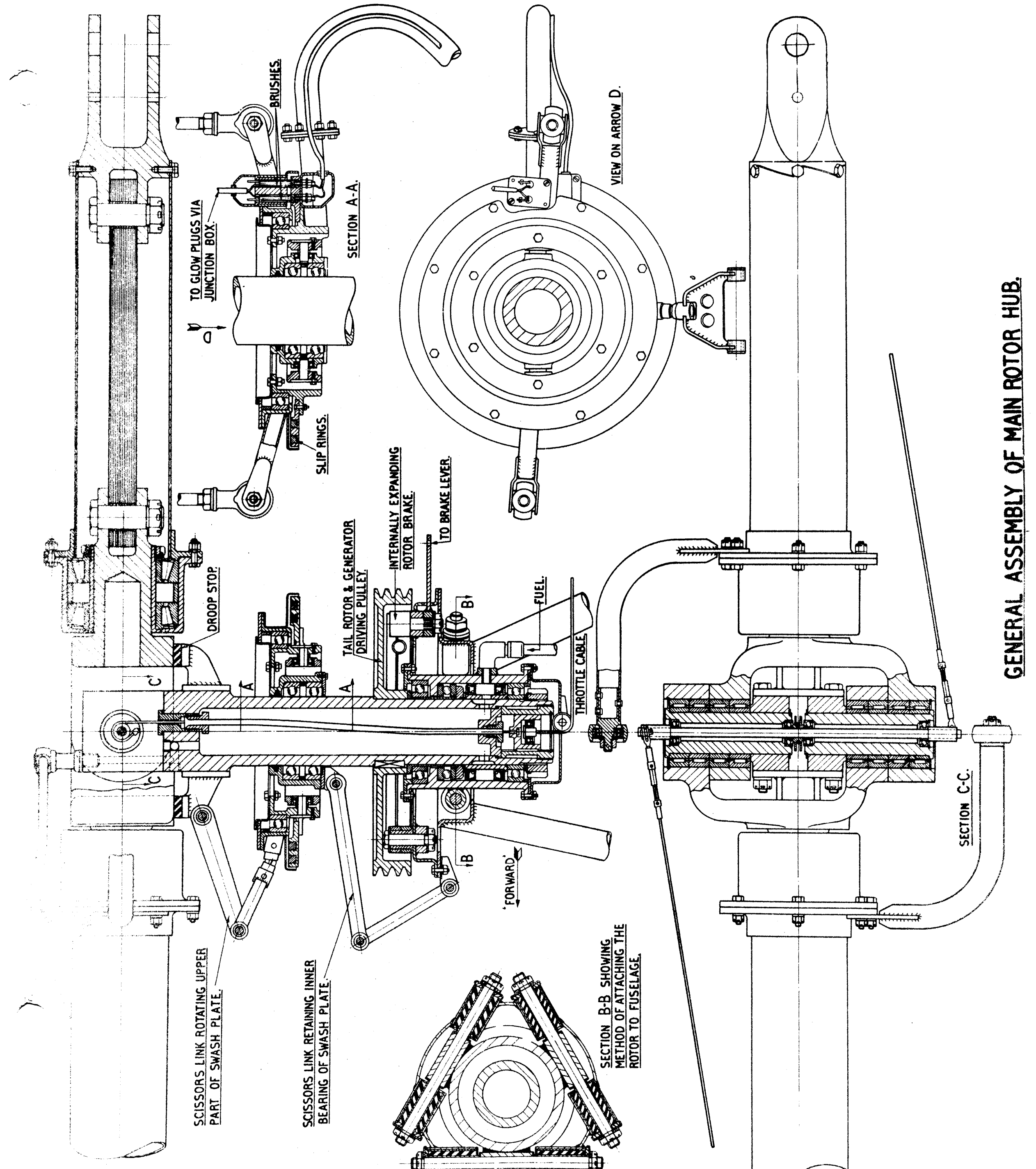

GENERAL ASSEMBLY OF MAIN ROTOR HUB.
VIEW ON ARROW D.
BRUSHES.
TO GLOW PLUGS VIA JUNCTION BOX.
SECTION A-A.
SLIP RINGS.
INTERNALLY EXPANDING ROTOR BRAKE.
TO BRAKE LEVER.
TAIL ROTOR & GENERATOR DRIVING PULLEY.
DROOP STOP.
FUEL.
THROTTLE CABLE.
'FORWARD'
SECTION C-C.
SECTION B-B SHOWING METHOD OF ATTACHING THE ROTOR TO FUSELAGE.
SCISSORS LINK RETAINING INNER BEARING OF SWASH PLATE.
SCISSORS LINK ROTATING UPPER PART OF SWASH PLATE.

Fig 4.5

## 4.6. CONTROLS.

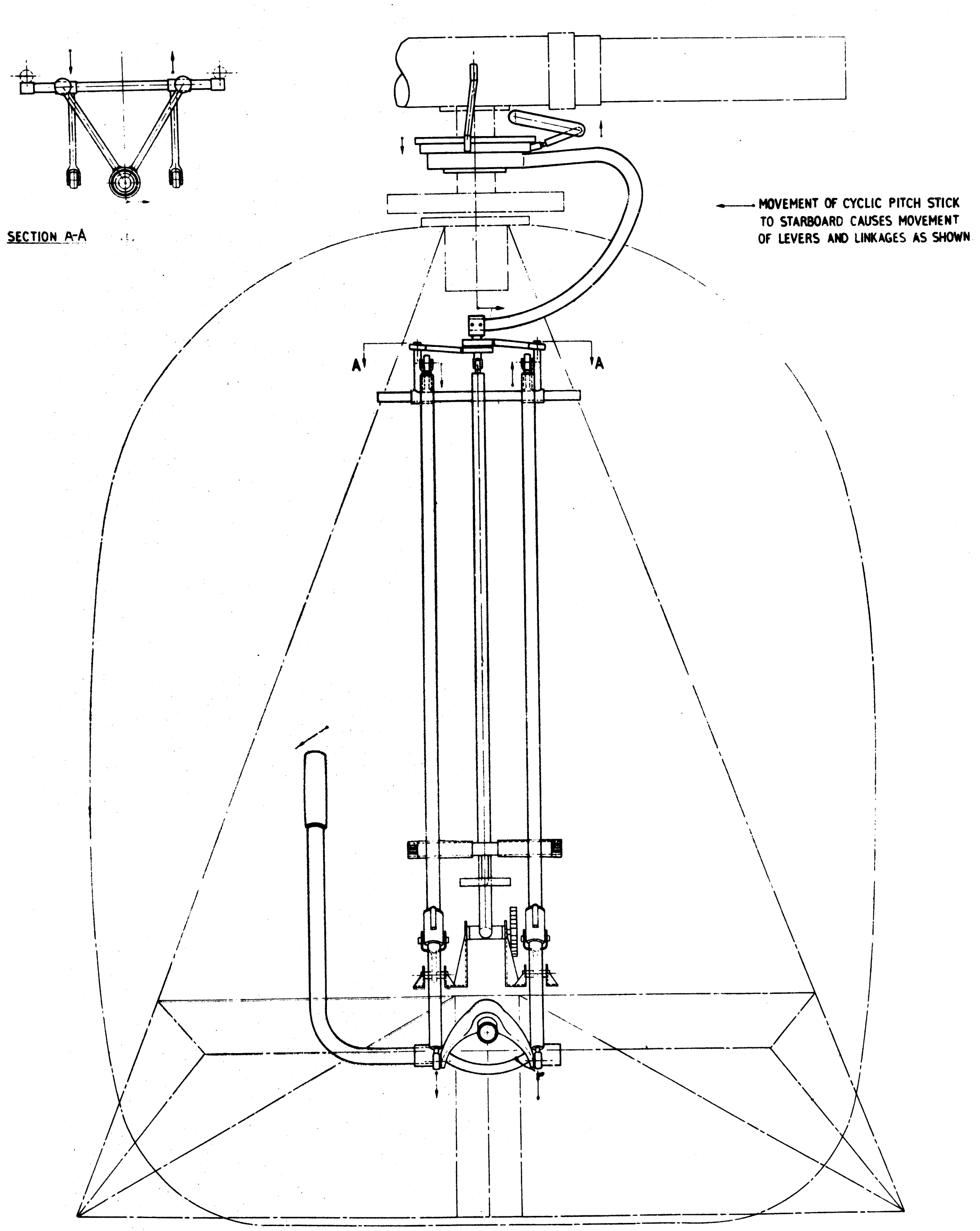
Conventional controls are employed with ranges as defined in Section *3.3.1* Movements of the cyclic pitch stick are transmitted to a swash-plate of conventional design underneath the rotor hub. This linkage is direct, via two push-pull struts, and only two bell cranks are used, in parallel, to transmit motion to the swash-plate swan-neck. Centralising bias is provided by two arms, biased by rubber bushes, as shown in Fig.4.6. No bias trimmer is provided. If the pilot wishes to trim the stick in a particular position, he presses the bias-shift lever on the end of the collective pitch stick, This releases the friction grip of the bias arms on the control tubes and allows them to move to a neutral position, where they again grip the tubes as soon as the bias-shift is released. Thus, in practice, trimming is achieved at any stick position simply by pressing the bias-shift lever. A form of this system is already in use in the United States and is regarded as superior to a hand-operated trim wheel.

As shown in Fig.4.6, the collective pitch control comprises a simple tube running vertically between the pitch change lever and the swash-plate swan-neck. A conventional friction nut is provided for locating the stick in any desired position.

Collective pitch stick loads are principally caused by propeller moment feedback from the rotor, and it is usual to balance this out by a spring, which works with the stick through a linkage system. At the time of submitting this brochure, the Company has not completed this aspect of the design investigation. It is hoped that a plain spring, working against the stick, will provided sufficient for this small rotor when used in conjunction with the friction nut. Should these expectations not be realised, a conventional linkage unit will, of course, be used.

The tail rotor collective pitch is governed by a simple bell crank and push-rod which is operated from the rudder pedals by wire cable. This control circuit is entirely orthodox, (Fig. 4.8).

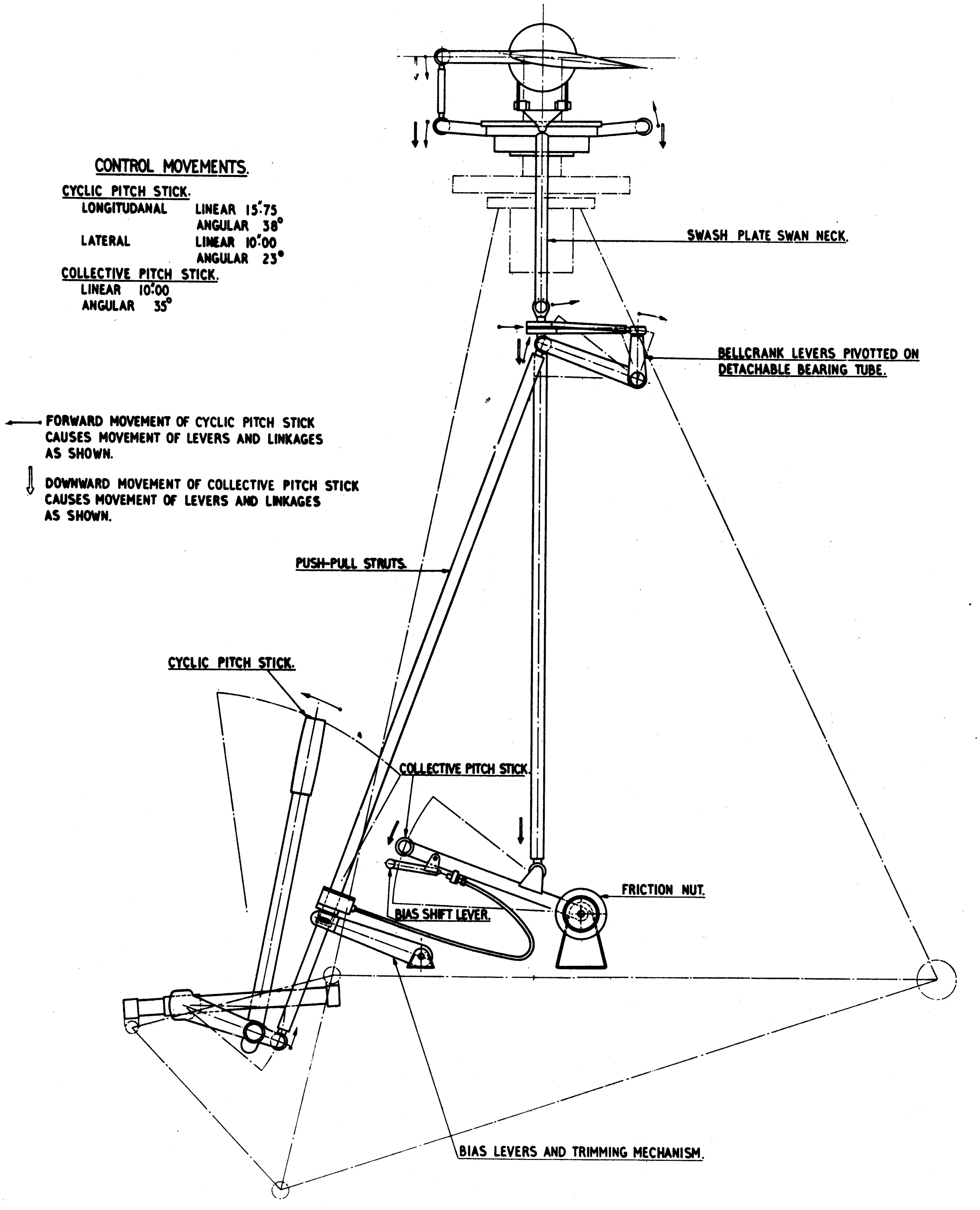
The only engine control is the “throttle”. This is a flexible wire cable with an adjustable tension spring in parallel at the collective pitch stick, which runs from the pilot’s twist grip to the metering piston in the ramjet. The general principle of this system is explained in Section 6.0.




GENERAL ASSEMBLY OF CONTROL SYSTEM

FRONT VIEW.

**Fig 4.6 (a)**




GENERAL ASSEMBLY OF CONTROL SYSTEM

SIDE VIEW.

**Fig 4.6 (b)**

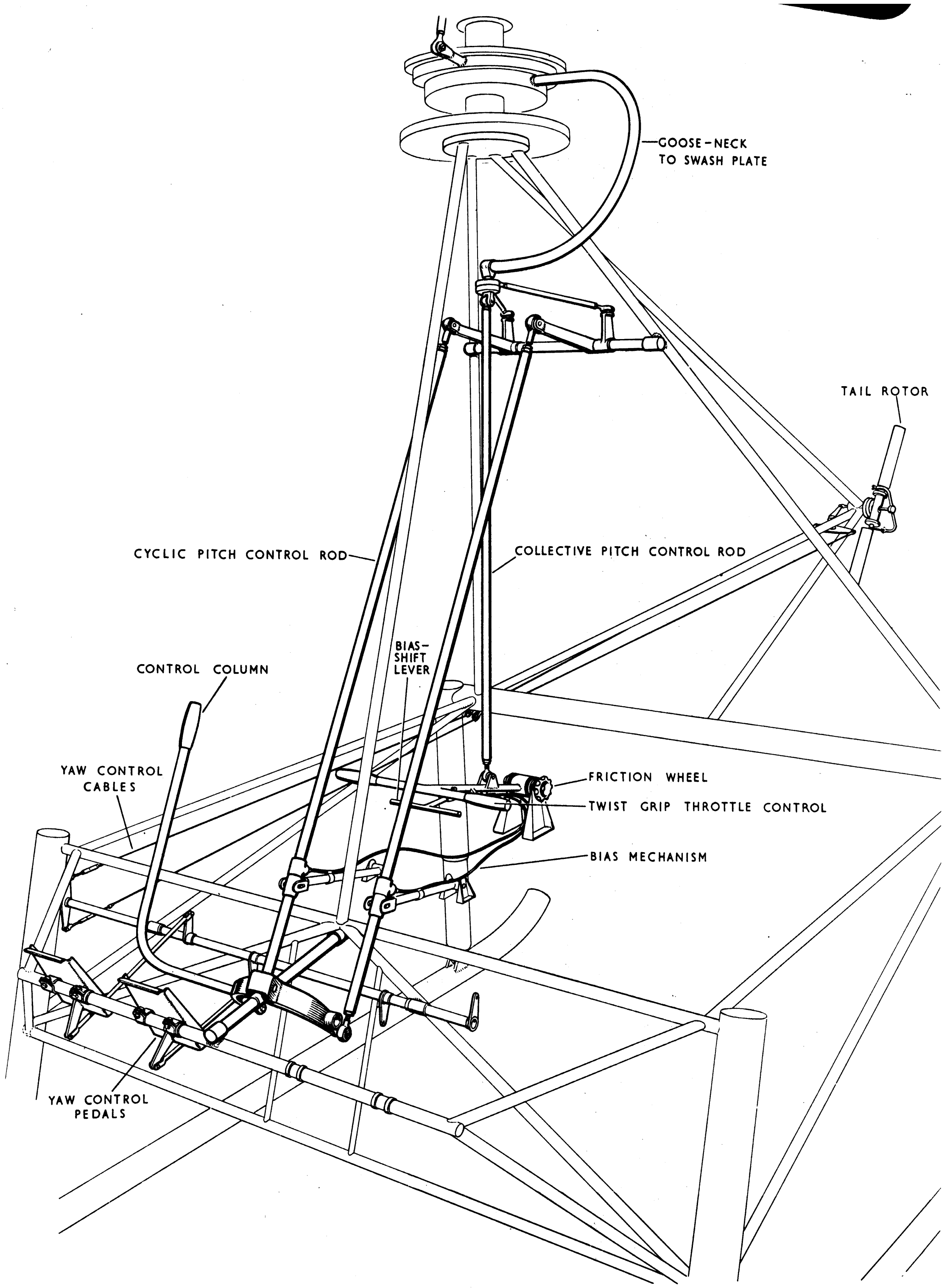

GOOSE-NECK
TO SWASH PLATE
TAIL ROTOR
CYCLIC PITCH CONTROL ROD
COLLECTIVE PITCH CONTROL ROD
BIAS-
SHIFT
LEVER
CONTROL COLUMN
YAW CONTROL
CABLES
FRICTION WHEEL
TWIST GRIP THROTTLE CONTROL
BIAS MECHANISM
YAW CONTROL
PEDALS

**Fig 4.7**

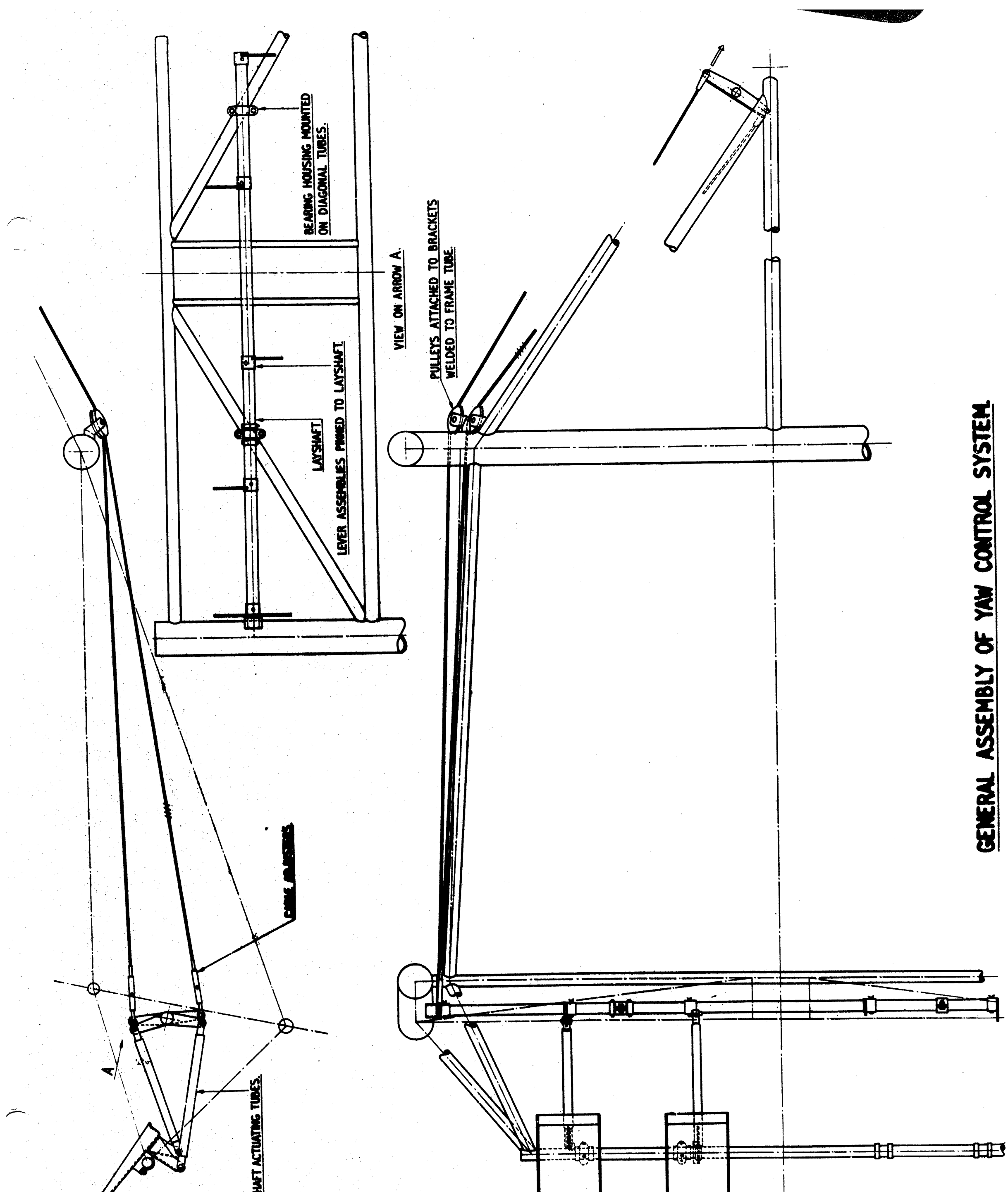

BEARING HOUSING MOUNTED ON DIAGONAL TUBES.
VIEW ON ARROW A.
PULLEYS ATTACHED TO BRACKETS WELDED TO FRAME TUBE.
LAYSHAFT
LEVER ASSEMBLIES PINNED TO LAYSHAFT.
LAYSHAFT ACTUATING TUBES.
A
GENERAL ASSEMBLY OF YAW CONTROL SYSTEM.

Fig 4.8

## 4.7. TAIL ROTOR AND DRIVE.

Although a tail rotor is not strictly necessary in a "torque less" rotor design, American experience has shown it to be desirable if satisfactory manoeuvrability is to be obtained under all conditions. On the present design it is designed to give a turn of 180° against torque in three seconds, corresponding to a rate of turn of 120° per second. This entails a maximum thrust of 25 lbs., so that no flapping hinges are required on the tail rotor, the blade structure being well able to carry the thrust bending moments.

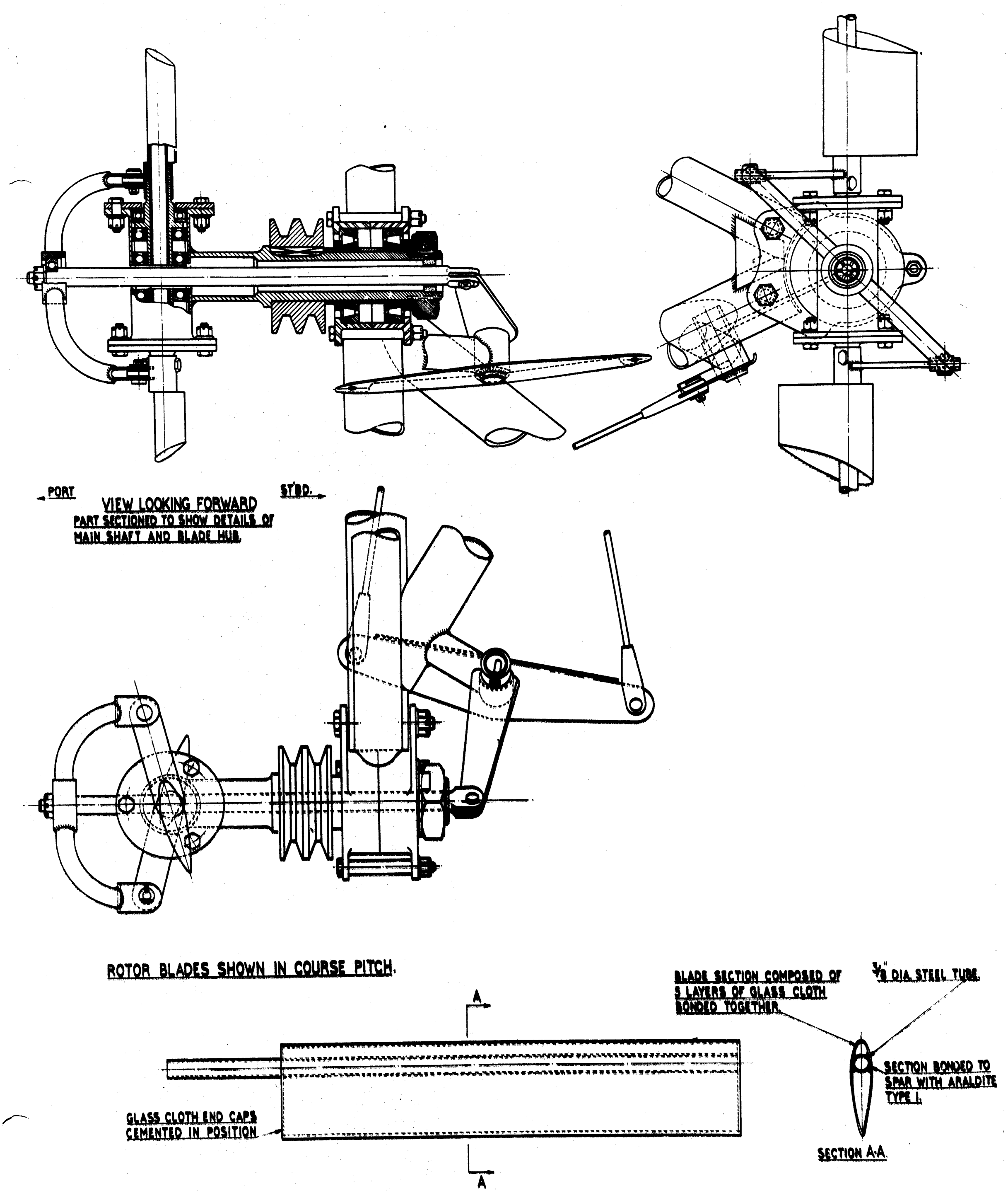
The blade comprises a single fibreglass fairing bonded round a steel tube. The inboard end of this steel spar carries the torsion bearings. As shown in Fig.4.9, three standard bearings are used; two journal bearings to carry the blade thrust moment and a ball race to carry centrifugal forces (ball thrust race). Blade pitch is governed by a simple spider, actuated by the rudder pedals through a bell crank.

The tail rotor bearings are of the taper-roller type, and are held in a split housing for ease of removal. The tail rotor drive pulleys are of 2 in, diameter and are keyed to the shaft.

The twin belt drive to the tail rotor passes from the main pulley, over a pair of twin guiding pulleys, to a single 2 in. double belt pulley at the tail rotor. The normal power requirement is less than half a horsepower, but in sudden manoeuvres, transient power loadings of two horse-power are obtained. For normal flight, one belt is sufficient, so that the use of two provides a measure of duplication in the system.

No difficulty is anticipated with "whipping" at the low speeds associated with a helicopter of this nature, since the belt is entirely in the wake of the bluff fuselage in forward flight. Should the need arise, a light fibre-glass fairing can be provided to cover the belts

In the unlikely event of both belts failing, the aircraft fuselage would normally rotate, and to prevent this, a fabric fin has been provided by fastening fabric to two of the tail rotor support struts as shown in the drawings. Since the fin is inclined to the rotor downwash in hovering, a lateral force is generated which opposes rotation of the fuselage. This is only exactly correct in the design case of maximum all up weight and, under other conditions, the fuselage would rotate slowly. A slow rotation of this nature would not be dangerous and can always be stopped by flying the aircraft forward.




GENERAL ASSEMBLY OF TAIL ROTOR.

**Fig 4.9 (b)**

**4.8. UNDERCARRIAGE.**

The undercarriage is a simple unit for which patent applications have been made by this Company. In operation, a circular brake lining is held on to the inner tube by a compression rubber spring when the leg is closing and the kinetic energy of the aircraft is absorbed in friction. There is no resistance to the return stroke, which is achieved by a light return spring when next the machine is airborne.

The normal working stroke (for a 6 ft./sec. landing velocity) is *5* inches. In the event of the aircraft landing at 12 ft./sec., the spring plate is pushed out from the top of the leg, enabling a total stroke of 20 inches to be achieved. The spring plate is self-re-setting during the next take-off or by manual extension of the leg, (Fig. 4.10).

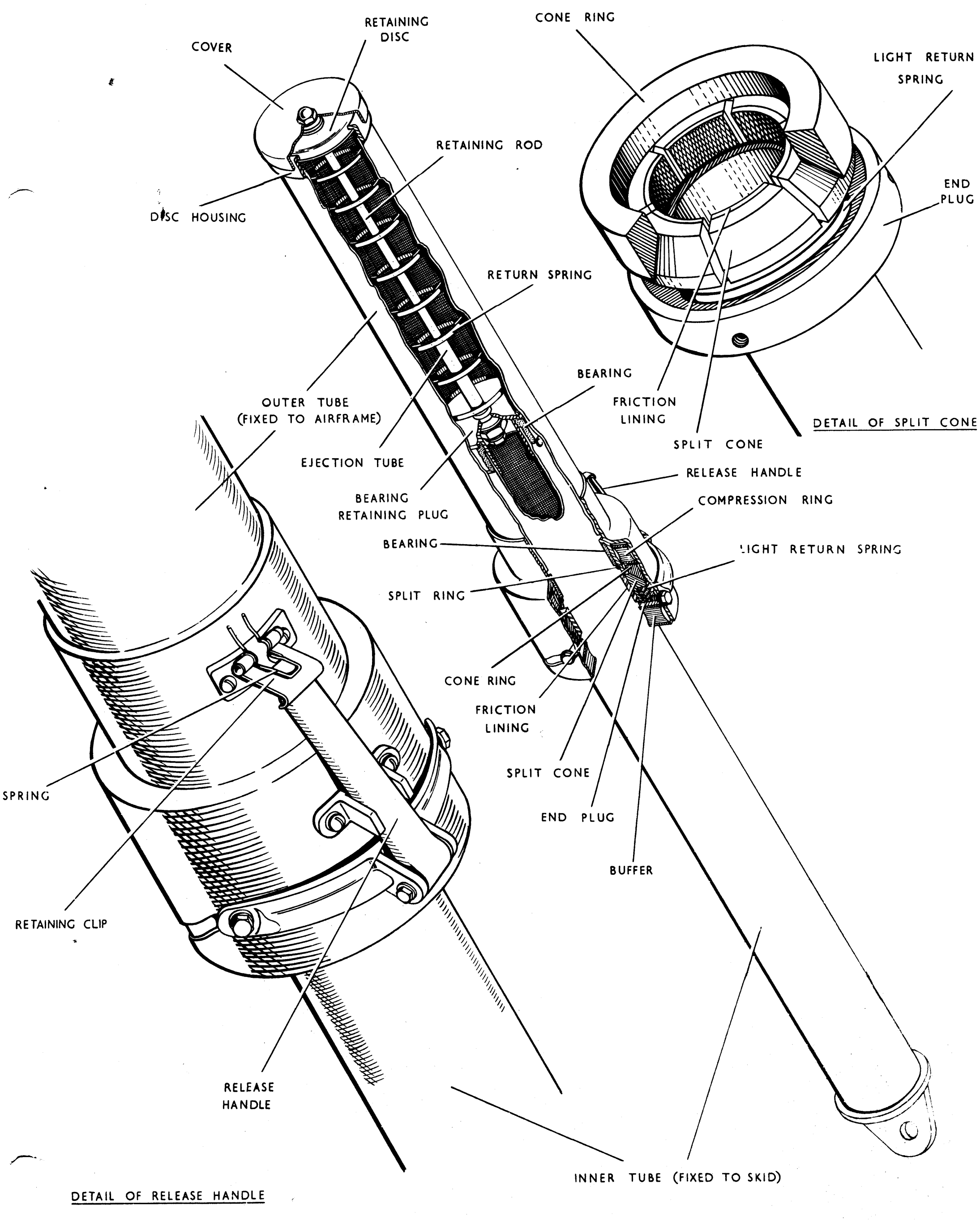
Other advantages are:

(a) The undercarriage is made up from two standard steel tubes; there is no need for the fine tolerances associated with pressure seals.

(b) The construction is both simple and robust. Should the need arise, a leg could be dismantled and re-assembled in the field by an Army mechanic,

(c) Irrespective of the descent velocity, the undercarriage reaction has always the same value, thus giving considerable savings in structure weight.

(d) The leg needs no routine servicing, but .is provided with replaceable bushes to cope with normal wear at the attachment points.




UNDERCARRIAGE LEG

**Fig 4.10**

# COCKPIT LAYOUT

# SECTION 5 COCKPIT LAYOUTS

Side-by-side seating is provided with the left-hand seat having alternative positions of forward facing for dual control, or rearward facing for observation purposes. Due to the upright seating position, it has been found that adjustable seats are not necessary, thus adding to the simplicity of construction. The seats used are identical to those of the Auster Mark 9 and are manufactured from stabilised fibreglass. Visibility is superior to the minimum recommendations of the Cockpit Layout Committee - Helicopter Panel.

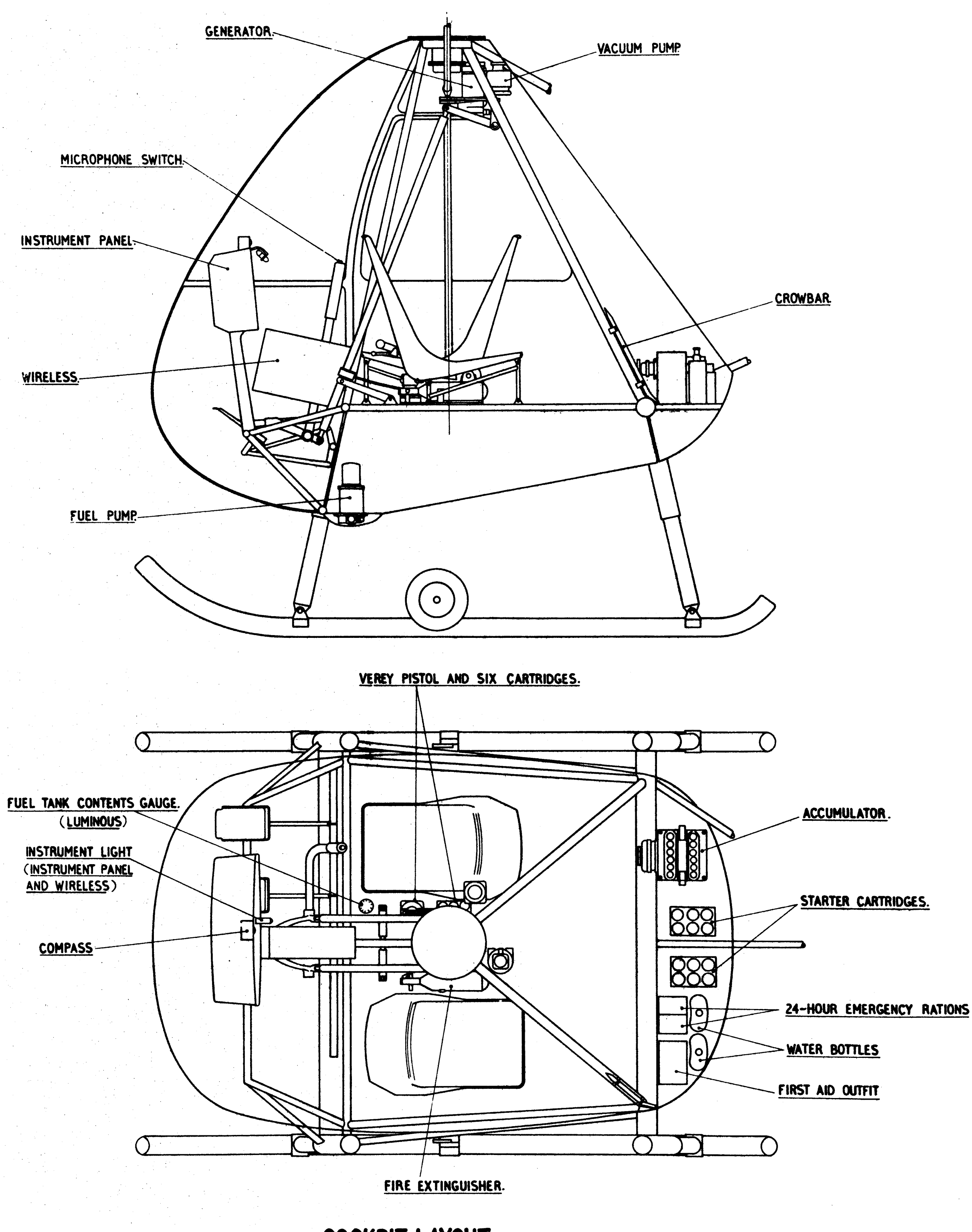
The cyclic pitch stick pivots on the centre-line of the aircraft, thus simplifying the control run and providing a simply plug-in socket for a dual control stick. A central collective pitch control is provided with twist grips for operating the fuel-metering unit in the ramjet. Conventional friction nuts are provided on these items for locating the controls in the desired positions. The tail rotor collective pitch is operated by the rudder pedals and, again, dual control is easily fitted. Further details of these controls will be found in Section 4.6.

The wireless set is mounted, on the centre-line of the aircraft between the instrument panel and the main floor. In this position, it is readily accessible to both crewmembers in the forward facing position, and can also be reached by the crewmember in the rearward facing position, if required. The “press-to-speak” switch is placed on the cyclic pitch control.

The instrument panel is in a near-central position, offset slightly to starboard, and is, therefore, convenient for both pilots when dual control is fitted. The fuel tank contents gauge, verey pistol, cartridges and fire extinguisher are mounted between the seats on either side of the collective pitch control and are accessible to both crew members.

For ease of entry and the carrying of alternative or additional payloads, the aircraft has been kept a few inches wider than is strictly necessary for fairing round the crew, and the main floor aft of the seats has been kept completely clear of equipment. In addition, the crewmember’s seat is easily removable, thus providing even more space should this be required for a bulky payload. It will be noted that the large doors provide access along the whole length of the main floor. Typical payload weights and further information on this subject are provided in Section 1 of the brochure.

In general, it can be stated that the visibility, primary flight controls, display of instruments, etc. conform to the requirements of the Cockpit Layout Committee - Helicopter Panel. Additional information on the proposed layout is given in the drawing provided in this Section, Fig. 5.1.




COCKPIT LAYOUT

**Fig 5.1**

# POWER UNITS

# SECTION 6 POWER UNITS

The ramjet designed for this helicopter is of the simple pitot-entry type, with a combustion chamber area of 19.13 ins.$^2$. Conventional baffles and burners are attached to the "horn" on the inboard face of the unit. (Fig.6.1.)

The use of a "horn" to support the ducting, combined with the cross- section shown, enables all bending moments to be carried in the horn whilst the skins carry mainly pure tension.

The horn is attached to the blade by means of an insert between the two steel blade spars, into which the centrifugal loads are diffused. Thus the horn and baffle seats are integral with the rotor blade and cannot be detached in the field.

For servicing the unit, the ducting is divided laterally into two halves, which can be slid off the horn fore and aft. The burners, baffles and fuel control units are then readily accessible and can be removed and replaced when desired.

The fuel control unit is directly controlled by a flexible wire cable running down the blade aft of the spar, to a central lay shaft, thence down the shaft centre, through a rotary bearing to the pilot's twist grip. The control is therefore positive and avoids the manifold disadvantages of solenoids or remote pressure control.

Fuel pressure is supplied by its own centrifugal forces, once the electrically driven booster pump has lifted it to rotor level. Centrifugal force on the metering piston is used to give constant speed control of the rotor in the following manner. As Fig.6.2.shows diagrammatically, the metering piston position is fixed by the pilot's twist grip, biased by a spring, which holds the piston open against the centrifugal force tending to close it. If the rotor speed increases, so will the centrifugal force, thus forcing the piston outwards and reducing the fuel flow until the equilibrium speed is again reached. The torsion spring in the pilot's twist grip can be rotated by the bias control knob, so that it can be used to fix rotor R.P.M. Once the system is "biased" to a given R.P.M., the pilot's attention to rotor speed can be relaxed.

## 6. 1. PERFORMANCE ESTIMATES.

In preparing performance estimates for the preliminary stages of this design, it was decided to use curves of achieved performance in preference to calculated ones, in order to avoid any possibility of optimistic assumptions. These are presented In Fig. 6.3 and are due to the MARQUARDT Aircraft Company, who built units fitted to several American helicopters,

## 6.2. STARTING.

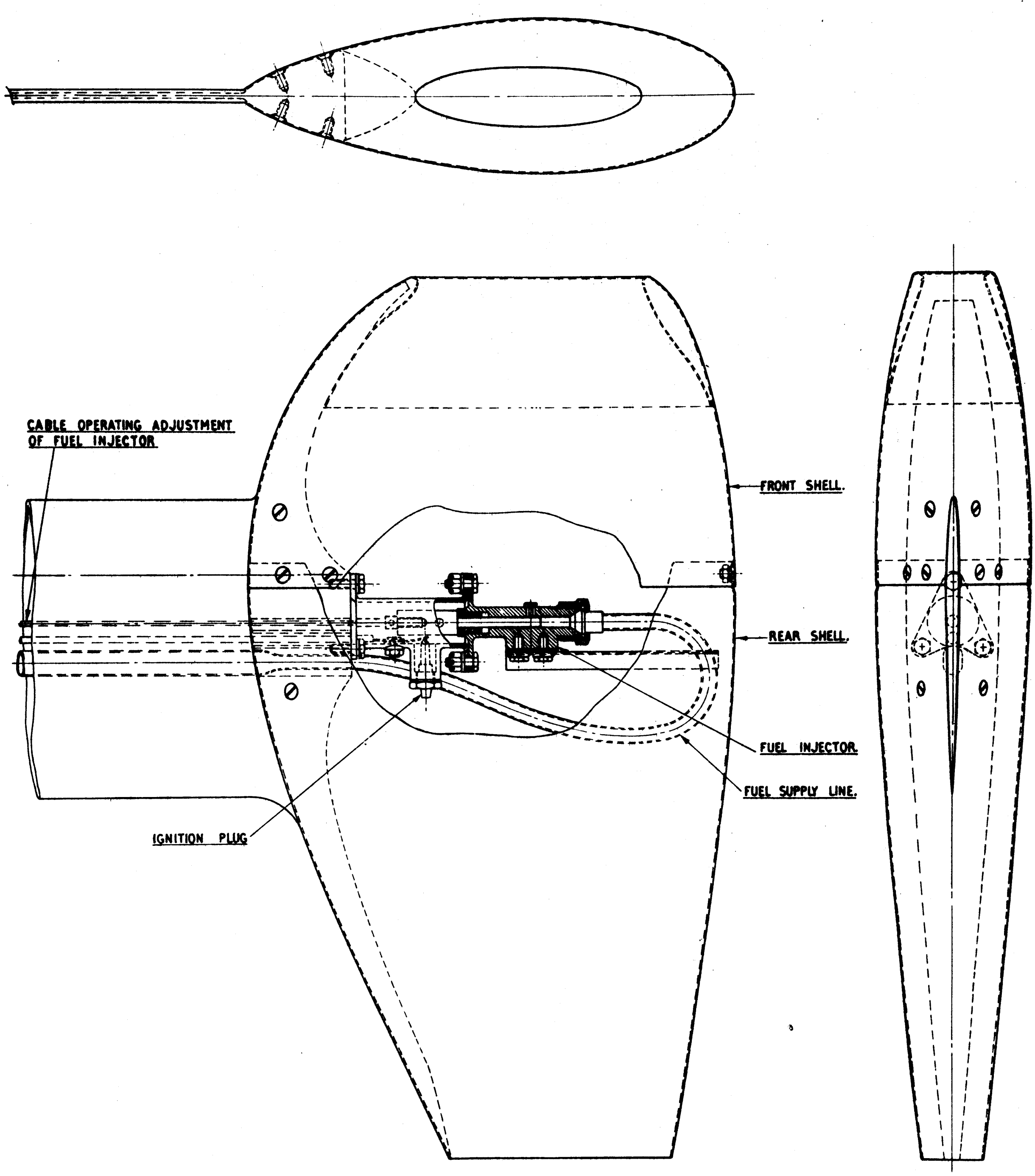
As Fig 6.3.shows, the ramjets ‘cannot be successfully started until a rotational Mach number of about 0.35 has been reached. When allowances for the rotor profile power loss are made, the minimum tip speed for starting is found to be 400 ft./sec.

The conventional method of using a small petrol engine to spin the rotor is strongly at variance with the basic concept of an ultra-light weight helicopter. On ‘the Auster design, it is proposed to use what are in effect small rocket units to accelerate the rotor, for which the propel lent is provided in the form of pressed powder cartridges.

It is envisaged that each ‘starting unit will be attached to its blade 6.5 ft. from the hub centre, as shown on the aircraft G.A. (Fig.1.1.) and the blade G.A. (Fig. 4.2b.) For starting, the old cartridges are withdrawn from the front of the units and replaced by new ones stored in the aircraft. The base cap is then screwed home and at his convenience, the pilot switches on the fuel boost pump and then fires the units from the cockpit. When the rotor tip speed has risen above 400 ft./sec. (300 R.P.M.) he opens his twist grip throttle and again presses the firing button. Since both starter and ram jet igniters are on the same circuit, this starts the ramjets.

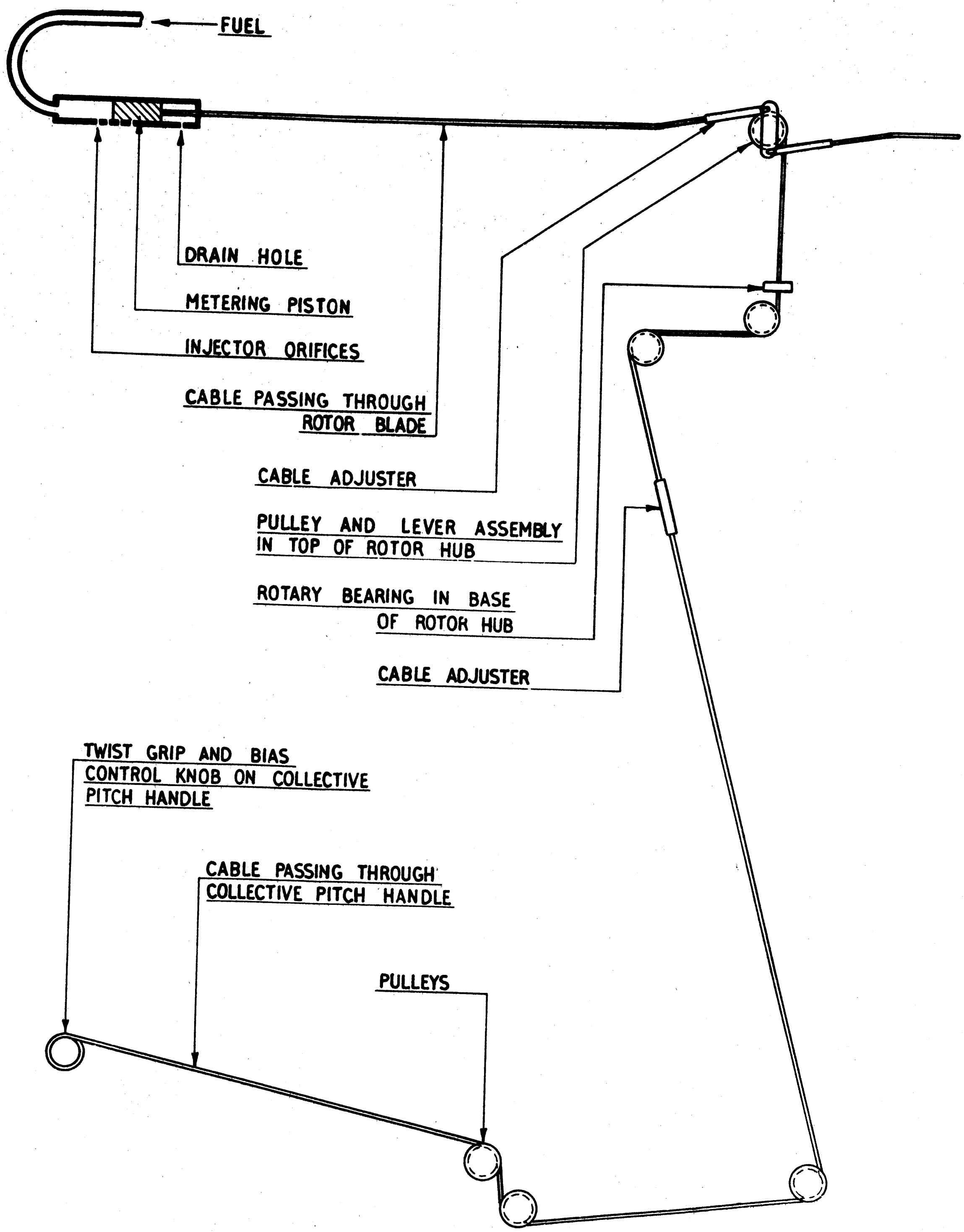
In the unlikely event of only one starter unit firing, considerable vibration would soon develop due to the increasing difference between charge weights on each blade. In such a case, the pilot would apply the rotor brake and stop the rotor. The number of starts which the pilot can make unaided in this manner is limited only by the number of starter cartridges carried in the aircraft.

When operating from an established base, it is hoped that starting will be carried out from an external starting motor, and provision for a power pick-up from such a motor is made. This provision should effect a worthwhile economy in cartridge costs and ease supply problems.




GENERAL ASSEMBLY OF RAM JET.

**Fig 6.1**




DIAGRAMMATIC SKETCH OF FUEL CONTROL SYSTEM

**Fig 6.2**

# NON-DIMENSIONAL. RAM-JET PERFORMANCE.

CURVES OF ACHIEVED PERFORMANCE USED
FOR CALCULATION OF POWER AVAILABLE.

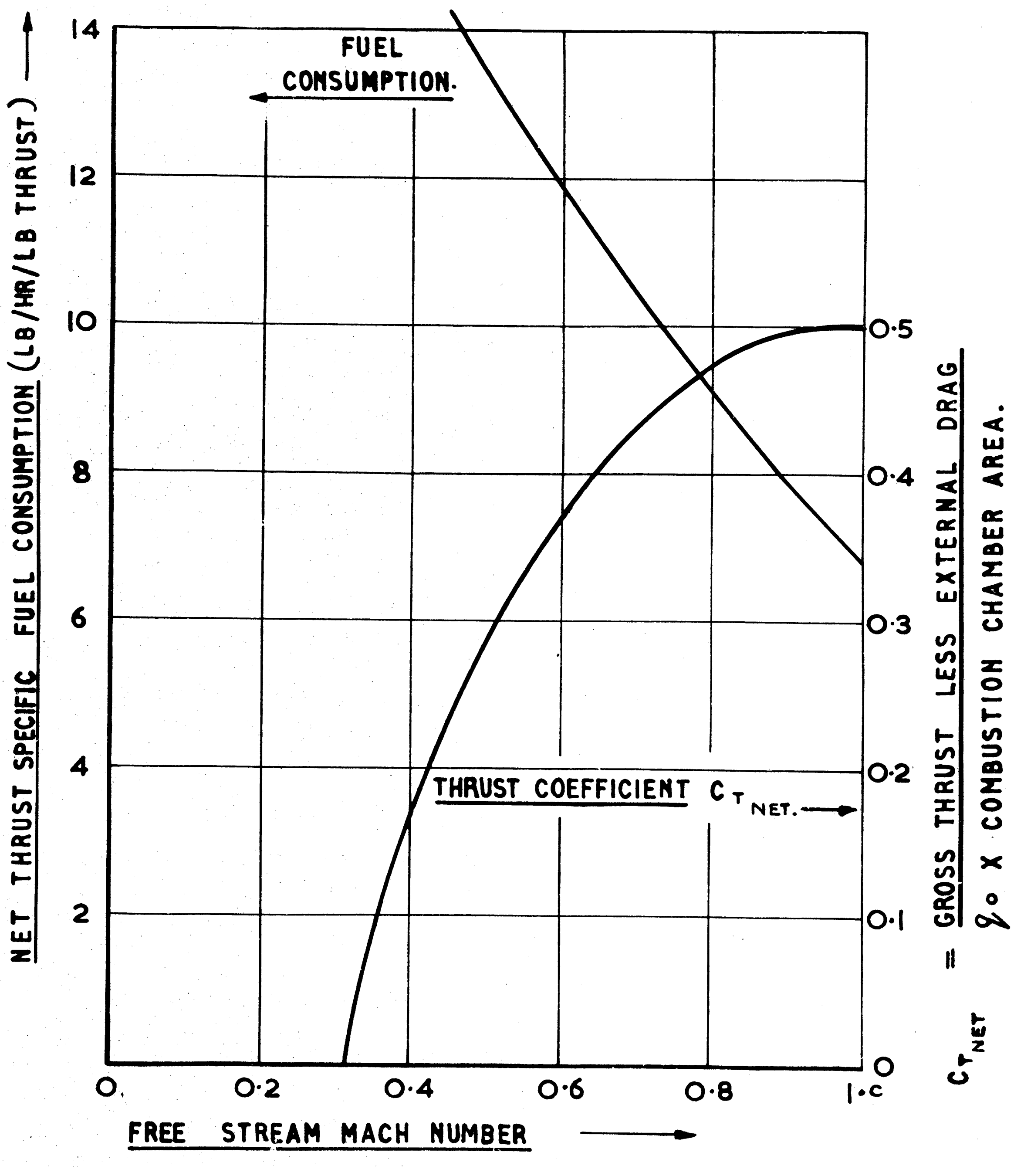



Fig 6.3

# FUEL SYSTEM

# SECTION 7 FUEL SYSTEM

The 100 gallon fuel tank is mounted Inside a framework of steel tubes and is separated from the crew seats by the double-skinned honeycomb stabilised floor of the ‘cockpit. An electronically driven fuel boost pump, mounted under the tank sump, lifts the fuel to the rotary seal in the hub, and thence via flexible pipes to union fittings on the root ends of the blades. Inside the blades it is piped to the tips, the pressure building up towards the tip because of the strong centrifugal force field, A maximum fuel pressure of 4000 lbs/in$^2$, can be achieved at the tip, and this has to be balanced by the centrifugal force acting on the metering unit piston. In practice the piston not only balances the fuel load, but has an additional mass in order to obtain a constant speed rotor, as explained in Section 6.0.

At the metering unit the fuel escapes into the rawjet via Orifices, and is burnt after mixing has taken place. A variation of the conventional orifice and anvil burner is used.

Two filters are used in the system, one at the boost pump, and a finer one inside the blade root end fairing. Both filters must be inspected and cleaned periodically.

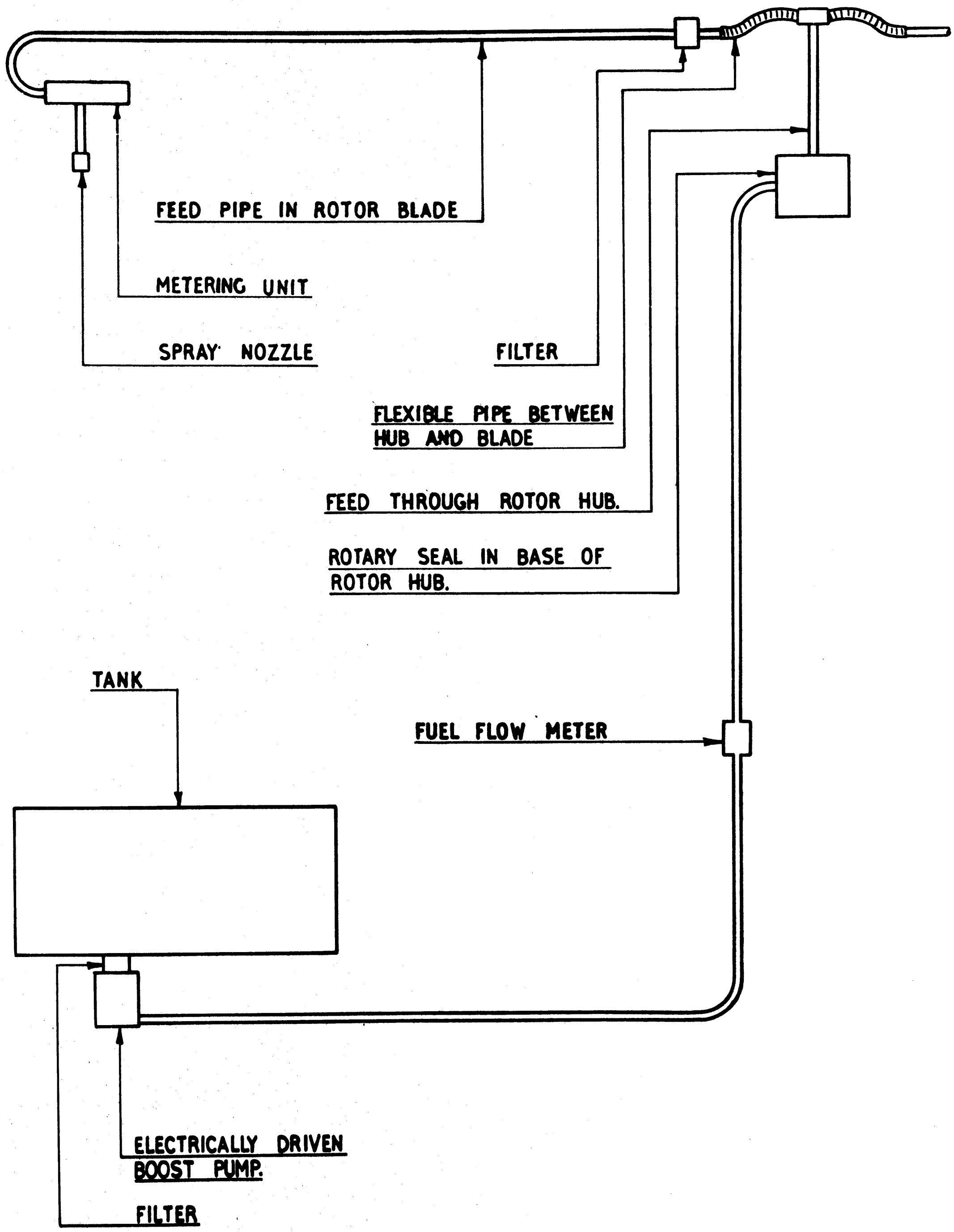
The fuel tank is readily accessible and can be quickly removed for replacement. In designing the attachments, it has been borne in mind that the Army may desire alternative and rapidly interchangeable tanks. Possible tank variations are:

(a)Standard 100 gallon tank.

(b)Bullet-proof 70 gallon tank

(c)Double compartment tank.

A fuel contents gauge is built into the top of the tank so as to project through the floor by the collective pitch stick, A fuel flow meter is mounted on the instrument panel and is in the main fuel supply circuit, as illustrated in the diagrammatic sketch of the fuel system, Fig.7.1.




DIAGRAMMATIC SKETCH OF FUEL SYSTEM

**Fig 7.1**

# EQUIPMENT

# SECTION 8 EQUIPMENT

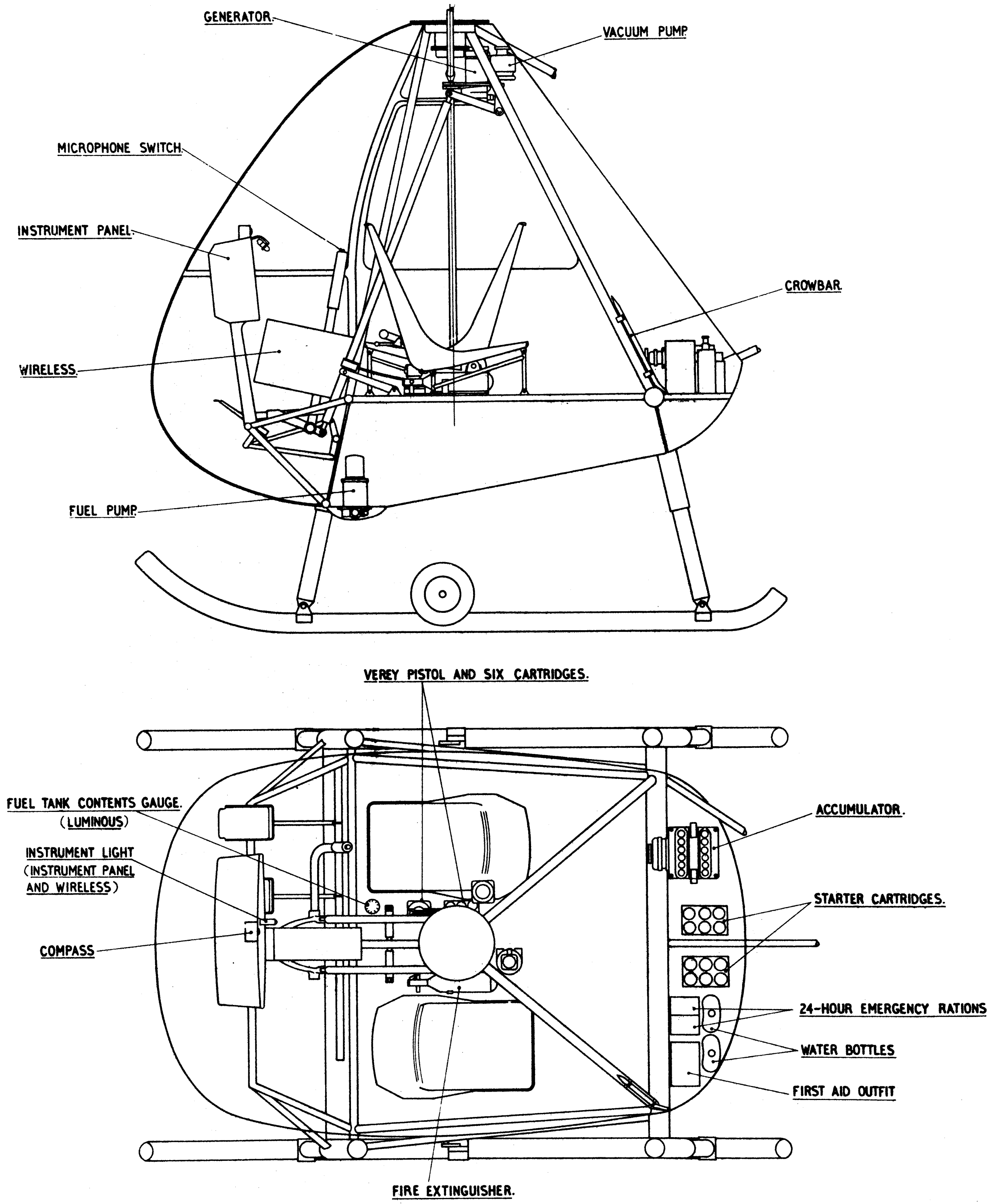
The equipment installed conforms to that called for in the Specification and has been kept as simple as possible. It can be split into' three main groups, the first of these being the electrical section. A 150 watt, 12 volt generator is mounted adjacent to the rotor hub, an accumulator on the rear fairing, a fuel pump is positioned in the forward part of the fuel tank sump and a wireless set is mounted on a cradle between the instrument panel and the main floor, on the centre-line of the aircraft. The "press-to-speak" switch is located on the cyclic pitch lever.

Storages, the second group, consist of a Verey pistol with six cartridges and a fire-extinguisher, all of which are mounted between the pilot and the crew member so as to be readily accessible to both. The first-aid box and crowbar are located on the port rear side of the aircraft, in such a position that they are accessible either inside the aircraft or from outside by breaking the rear Perspex panel. Mounted on the rear fairing are twelve starter cartridges (sufficient for six starts), crewmembers' water bottles and 24. hour emergency ration packs.

In the third group are the instruments, most of which are mounted on the instrument panel. The following flight instruments are incorporated and are displayed in such a position that they conform to the requirements of the Cockpit Layout Committee - Helicopter Panel: direction indicator, artificial horizon, vertical speed indicator, altimeter, air speed indicator, rotor tachometer and cross level. A fuel flow meter, starter button and fuel pump switch are also provided, the latter being automatically switched on when the protective flap of the starter button is raised. An E.2 type compass is mounted on top of the instrument panel'. A fuel gauge is positioned on the forward part of the floor and a vacuum pump for operating the suction instruments is mounted adjacent to' the rotor hub. Since no oil supply is available for the standard vacuum pump, an Auster designed centrifugal pump is proposed.

Fig. 8.1. Illustrates these dispositions.

Fig. 1.2. Is an artist's impression of the helicopter and shows the equipment installed




COCKPIT LAYOUT

**Fig 8.1**

# EMERGENCY EXITS

# SECTION 9 EMERGENCY EXITS

For normal entry and exit large forward opening doors have been provided, giving access along the whole length of the main floor.

For emergency exit, a simple door jettison mechanism is provided which is operated by pulling down and pushing forward a lever situated on the top door hinge. This action frees the hinge lugs from the locating pins, and the door is then free to fall away.

The emergency exit facilities are illustration in Fig. 9.1.

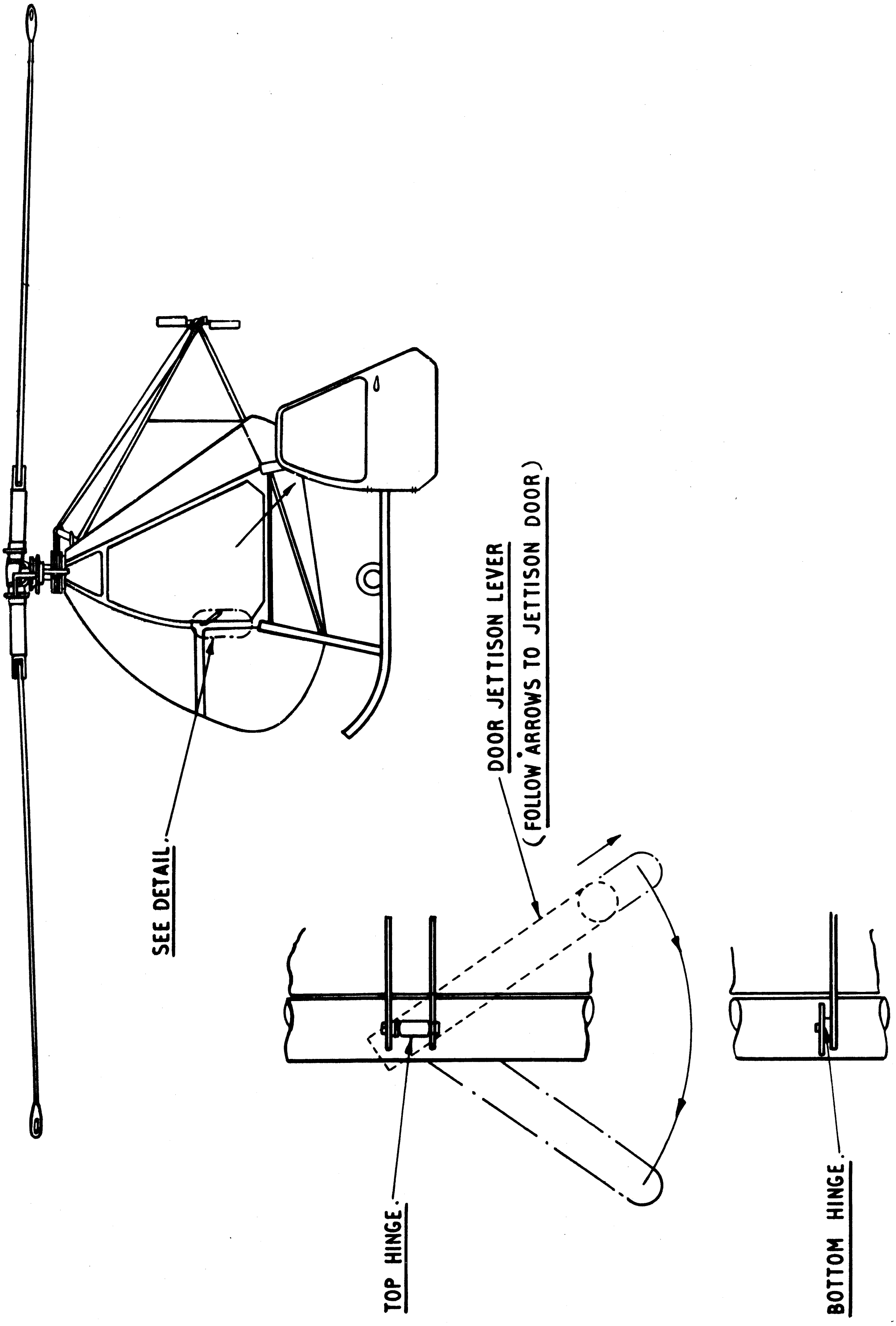

SEE DETAIL.
DOOR JETTISON LEVER
(FOLLOW ARROWS TO JETTISON DOOR)
TOP HINGE.
BOTTOM HINGE.

Fig 9.1

# SERVICING & MAINTENANCE

# SECTION 10 SERVICING & MAINTENANCE

## 10.1. Servicing

Field servicing of this helicopter is limited to the carrying out of the following visual examinations at suitable intervals.

Checking that ignition plugs and fuel filters are clean and that the throttle cable and pitch changing mechanism are free.

Inspection of the drive to the tail rotor and generator. V-belts can be replaced if wear is apparent, but as the belts normally transmit only 25% of their rated capacity, it is not expected that more than one belt change will take place during the life of the aircraft. In order to remove the V-belt from the tail rotor drive, it is necessary to remove one main rotor blade. To remove the generator drive belts it is necessary to detach one root arm in addition. The procedure for removing a main rotor arm is described in paragraph 10.2.2 of this Section.

Inspection of the tail rotor control cables and pitch change is limited to ensuring that the parts are free and clean

Inside the fuselage, it is only necessary to ensure that the stick and foot controls are fully operative. Access to the instruments is provided by a “hinge-down” panel, which brings all connections within easy reach of a mechanic seated in the machine. The vacuum supply relief valve is also readily accessible.

All parts of the electrical system are readily accessible for inspection or removal. The accumulator is reached through the starboard door and can be easily “topped-up”. The radio is quickly removable from its stowage between the crewmembers and can be taken to a workshop f or servicing

## 10.2. Maintenance.

In order to give a fair picture of the ease of servicing operations on this helicopter, the breakdown of the main components is dealt with in some detail below. The times for dismantling and re-assembly are set out in the Table at the end of this section.

10.2.1. Main Rotor.

Piping, electric and control wires entering the blade are easily broken at the root end joints. The rotor blade can then be removed by withdrawing the 1-inch diameter pin which passes through the blade attachment fork,

Maintenance of the rotor blade is confined to visual inspection of the skinned surfaces. The ram jet shell is detached by removing the bolts attaching the unit to the beam, enabling the metering unit and spray gear to be removed for inspection.

10.2.2, Rotor Hub Assembly

To dismantle this assembly, the V-belts are released and the fuel pipe disconnected at the union against the bearing housing. The throttle cable is broken at the adjuster situated near the apex of the rotor tripod. The gooseneck, controlling the pitch changing mechanism, is released by the removal of bolts at the junction near the slip ring. At the same time, the electric cable running along the gooseneck is disconnected at the plug on the end of the cable. Finally, the bolts attaching the brake mechanism plate to the thrust plate are removed.

The hub assembly is then ready f or removal, upon the withdrawal of the three bolts connecting the anti-vibration bushes to the rotor spindle.

After the removal o the throttle control pin from the centre of the flapping pin, the circlips, locating the flapping pins and bearing, can be extracted and the flapping pins withdrawn. ‘This action separates the hub assembly into three components; spindle assembly and two blade root arms.

10.2.3. Tail rotor.

With the removal of’ the pin connecting the clevis fitting to the tail rotor spindle and the two bolts that secure the bearing housing to the fuselage tube, the tail rotor assembly can be withdrawn.

The driving pulley and bearing housing is withdrawn upon removal of the locknuts on the end of the tail rotor spindle.

10,2.4. Fuselage

The fuel tank is free to drop away upon releasing the two bolts holding the tank straps.

Shock Absorbing Struts The legs can be removed after jacking the aircraft on the transverse members and removing the skids by releasing the friction damper pressure and pushing each leg out through the top of its outer tube.

Ground Handing Folding the undercarriage legs, and the use of ground handling wheels, are illustrated in Figs. 10.1 and 10.2.

## SERVICING AND MAINTENANCE

Time required to change major components.

| COMPONENT | | MAN HRS. | ELAPSED TIME |
|---|---|---|---|
| Engine Change. | Remove | 10minutes | 10 minutes |
| | Fit | 12 " | 12 |
| | | 22 " | 22 |
| Rotor Blade Change (Complete with Ram Jet). | Remove | 5 " | 5 |
| | Fit | 10 " | 10 |
| | | 15 " | 15 |
| Rotor Blade Change (including transfer of Ram Jet to new Blade). | Remove | 15 " | 15 |
| | Fit | 2 " | 22 |
| | | 37 " | 37 |
| Rotor Head Change (including Brake). | Remove | 1.30hours. | 1.00hour., |
| | Fit | 3.00 " | 2.00 |
| | | 4.30 " | 3.00 |
| Fuel Tank Change, | Remove | 7minutes | 5 minutes |
| | Fit | 10 " | '8 |
| | | 17 " | 13 |
| Tail Rotor. | Remove | 5 " | 5 |
| | Fit | 8 " | 8 |
| | | 13 " | 13 |
| - Skid (including trestling time). | Remove | 5 U | 5 |
| | Fit | 8 " | 8 |
| | | 13 " | 13 |
| Rear Leg — complete. | Remove | 7 " | 7 9 |
| | Fit | 10 " | 10 |
| | | 17 " | 17 |
| Tail Rotor Belt 'Change | Remove | 1 " | 1 |
| | Fit | 1.00 hour. | 1.00 hour. |

Fig 10.1 (a)

Fig 10.1 (b)

Fig 10.1 (c)

Fig 10.1 (d)

Fig 10.2 (a)

Fig 10.2 (b)

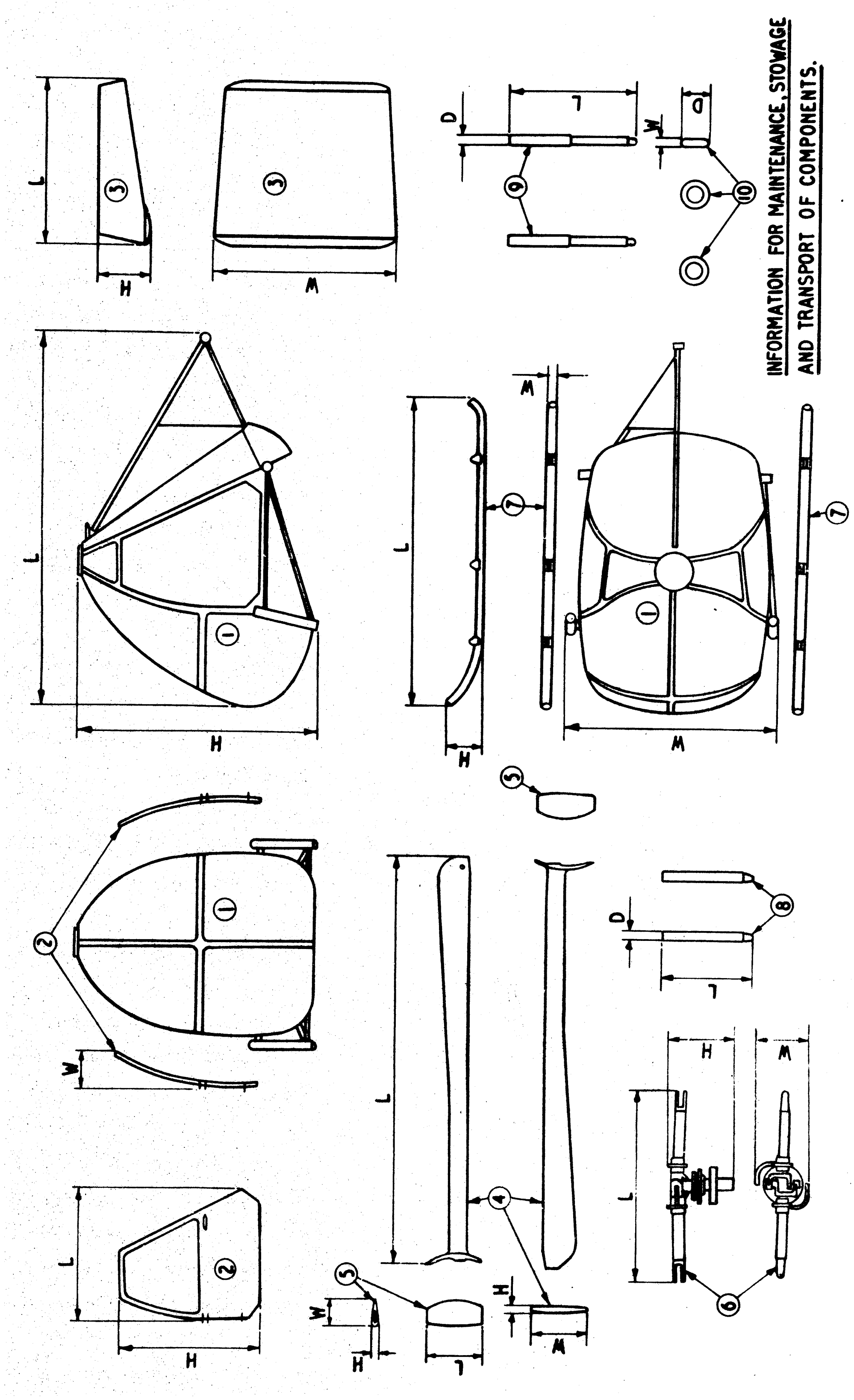

INFORMATION FOR MAINTENANCE, STOWAGE AND TRANSPORT OF COMPONENTS.

**Fig 10.3**

| | FUSE-LAGE | DOORS | FUEL TANK | ROTOR BLADE | RAM JET UNIT | HUB ASSY. | SKID | U/C LEG FORWARD | U/C LEG REAR | WHEEL |
|---|---|---|---|---|---|---|---|---|---|---|
| ITEM NO. | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 |
| LENGTH 'L' | 120" | 44" | 53" | $130\frac{1}{2}$" | 18" | 61" | 100" | 34" | $42\frac{1}{2}$" | - |
| HEIGHT 'H' | 77" | 46" | 18" | 2" | 3" | 22" | 13" | - | - | - |
| WIDTH 'W' | 67" | 11" | 59" | 18" | 10" | $17\frac{1}{2}$" | $3\frac{3}{4}$" | - | - | 4" |
| DIAMETER 'D' | - | - | - | - | - | - | - | 3" | 5" | 10" |

# PERFORMANCE

# SECTION II PERFORMANCE ESTIMATES

The performance method used for making the estimates given in this Section has been published in Reference 1. This method has been developed largely in the light of practical experience of achieved performance, and gives realistic figures for contemporary helicopters. In view of this, it is considered that the estimates prepared for the Auster design can be safely met, particularly since the glass-fibre rotor blade will have a finish which is superior to that achieved on present day designs.

The drag polar which has been used is plotted in Fig. 11.1 and is based on leading edge transition to turbulent flow in the boundary layer at a Reynolds Number of $1.0 \times 10^6$ This is a severe assumption insofar as transition normally occurs at between 20% to 30% on a rotor blade of this type, but 'to 'some extent this is offset by the drag of surface irregularities which are unavoidable on all production surfaces.

Using the non-dimensional method of Reference 1, curves of power required per pound of aircraft weight have been calculated f or the nominal all up weight of 1550 lbs. when the disc loading is 3 lbs./sq.ft. The estimated rates of climb and decent, and forward speed have all be based on this nominal figure. Since the aircraft weight is reduced by nearly 500 lbs. when the fuel is consumed, it follows that the mean performance will be considerably in excess of the figures quoted. A complete carpet of performance against all up weight has not been produced f or this brochure, however, because its accuracy depends to a great extent on the predictions of ram jet performance. For the purposes of this brochure, the ram jet performance curves are statistical ones, based on achieved American results and are not suitable for use in a complete performance statement.

Vertical rates of climb and descent for both I.C.A.N. and Tropical Conditions are plotted in Fig. 11.2. It will be observed that the single-engine performance, in vertical flight, differs little from autorotation with both engines stopped. This is due to the influence of the vortex ring state, with its attendant high-induced power loss. Despite the high rates of decent, Fig. 3,9.shows that the kinetic energy of the rotor is ample to achieve a complete flare-out followed by touchdown at zero velocity.

Rates of climb and descent at 45 knots equivalent air speed are plotted in Fig. 11.3, At 10,000 ft. in Tropical Conditions, a rate of climb of 800 ft./min. can be achieved. This is considerably in excess of the 100 ft./min. required by the Specification. The rates of descent are also well within the values specified, and it should be noted in this respect that the ram jet rotor is superior to a pulse jet driven one because of the low drag coefficient of the units when running cold, and because it can be easily re-started in flight after a period of autorotation.

A plot of disposable load against endurance and range is given in Fig. 11.4. This curve makes no allowance for the reduction in disc loading as fuel is burnt during the sortie, so that rather higher figures will be expected in practice, the actual increase depending on the nature of the mission.

The principal performance figures are summarised in the following Tables.

# PERFORMANCE SUMMARY

Nominal Design All-Up-Weight - 1550 lbs.

| | I.C.A.N. | Tropical Summer. |
|---|---|---|
| 1. Level Speeds. | | |
| Maximum Speeds at 900 ft,/sec. Tip Speed. | | |
| Sea Level — knots | 102 | 99 |
| 6,000 ft. — knots | 100 | 96 |
| Maximum Cruising Speeds at 800 ft./sec. Tip Speed. | | |
| Sea Level — knots | 82 | 78 |
| 6,000 ft. — knots | 78 | 73 |
| Maximum Cruising Speeds at 800 ft./sec. | | |
| Tip Speed, | | |
| Sea Level - knots | 82 | 78 |
| 6,000 ft. — knots | 78 | *73* |
| 2. Vertical Rates of Climb. | | |
| Maximum vertical rate of climb — ft./min. | | |
| Sea Level | 1900 | 1300 |
| 6,000 ft. | 910 | 0 |
| 3.Maximum Rates of Climb. | | |
| Best rate of climb — ft./min. | | |
| Sea Level at 45 knots E.A.S. | 2076 | 1680 |
| 6,000 ft. at 45 knots *E.A.S.* | 1512 | 1134 |
| Minimum Vertical Rate of Descent — ft./min, | | |
| Sea Level ) . | 1818 | 1824 |
| 6,000 ft. 3 Full autorotation | 1830 | 1842 |
| 4. Ceilings (Out of Ground Cushion). | | |
| Maximum ceilings — ft. | 22000 | 18600 |
| Maximum Hovering Ceilings — ft. | 15000 | 6000 |
| 5. One Engine Performance. | | |
| Best Rate of Climb — ft./min. | | |
| Sea Level at 45 knots E.A.S. | -48 | -219 |
| 6,000 ft. at 45 knots E.A.S. | —288 | -471 |

| | I.C.A.N, | Tropical Summer |
|---|---|---|
| Vertical Rate of Descent — ft/mm. | | |
| Sea Level | 2280 | 2530 |
| 6,000 ft. | 2670 | 2860 |

N.B. The All Up Weight rapidly decreases from the take-off value, so that the aircraft can maintain height 'or climb on one engine. To illustrate this, single-engine performance is given below at a typical landing weight of 1,100 lbs.

6. One Engine Performance at Landing Weight <u>1,100 lbs</u>.

| | | |
|---|---|---|
| Best rate of climb — ft./min. | | |
| Sea Level at 45 knots E.A.S. | 420 | 93 |
| 6,000 ft. at *45* knots E.A.S. | 324 | 0 |
| Vertical rate of descent - ft./min. | | |
| Sea Level | 1160 | 822 |
| 6,000 ft. | 696 | 408 |

## VARIATION OF ASSUMED DRAG POLAR WITH BASIC LIFT COEFFICIENT.

NACA 4–DIGIT SECTION

t/c = 7·3%

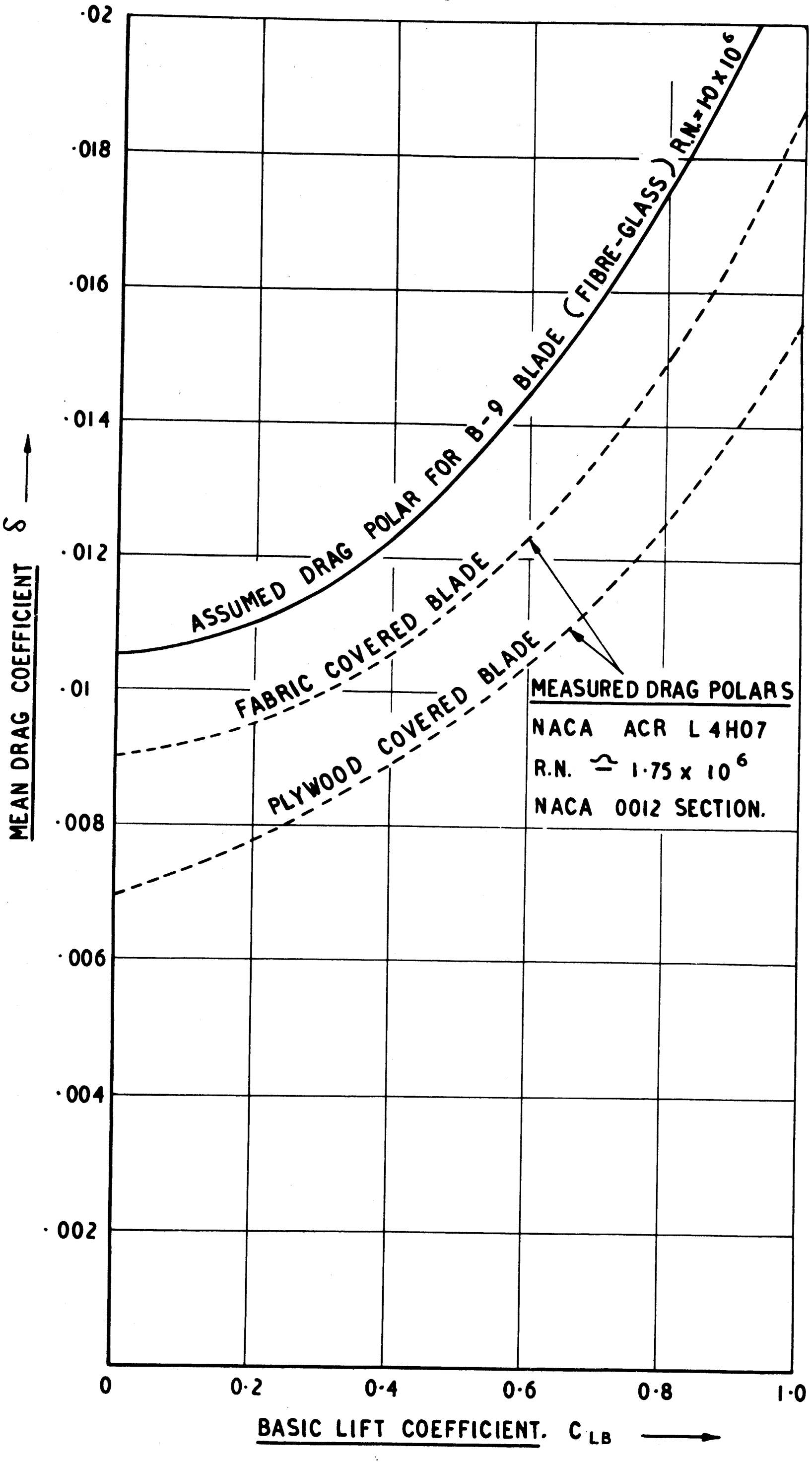
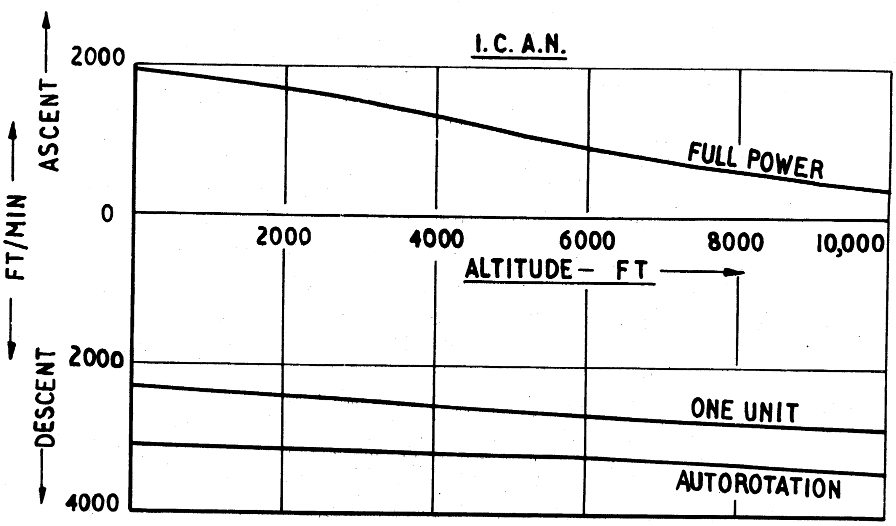



**Fig 11.1**

# VERTICAL RATES OF CLIMB AND DESCENT.

ALL UP WEIGHT = 1550 LB.
TIP SPEED $V_T$ = 900 FT/SEC
DISC LOADING $p$ = 3·0 LB/FT$^2$
ROTOR SOLIDITY $\sigma_{0\cdot7}$ = 0·025
EFFECTIVE BLADE TAPER $t^*$ = 0·60



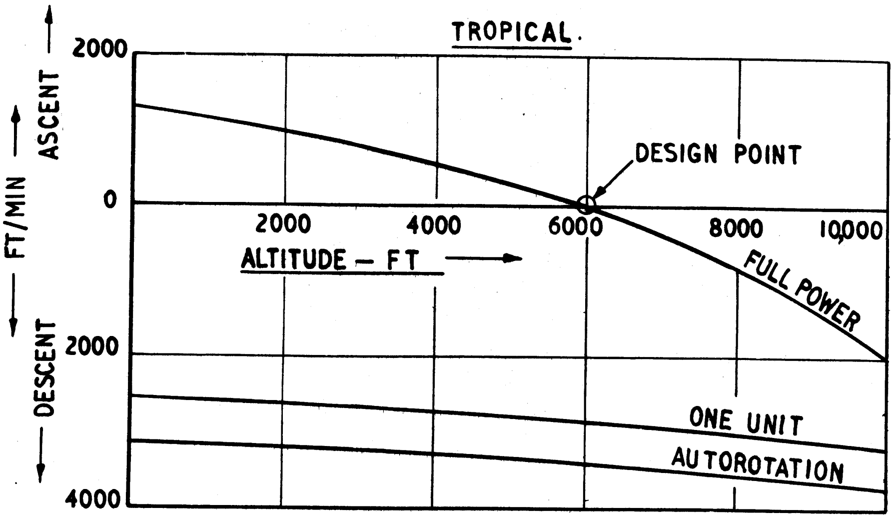




**Fig 11.2**

# RATES OF CLIMB AND DESCENT.

## AT 45 KNOTS E.A.S.

ALL UP WEIGHT = 1550 LB

TIP SPEED $V_T$ = 900 FT / SEC

DISC LOADING P = 3·0 LB/FT²

ROTOR SOLIDITY $\sigma$ 0·7 = 0·025

EFFECTIVE BLADE TAPER $t^*$ = 0·60

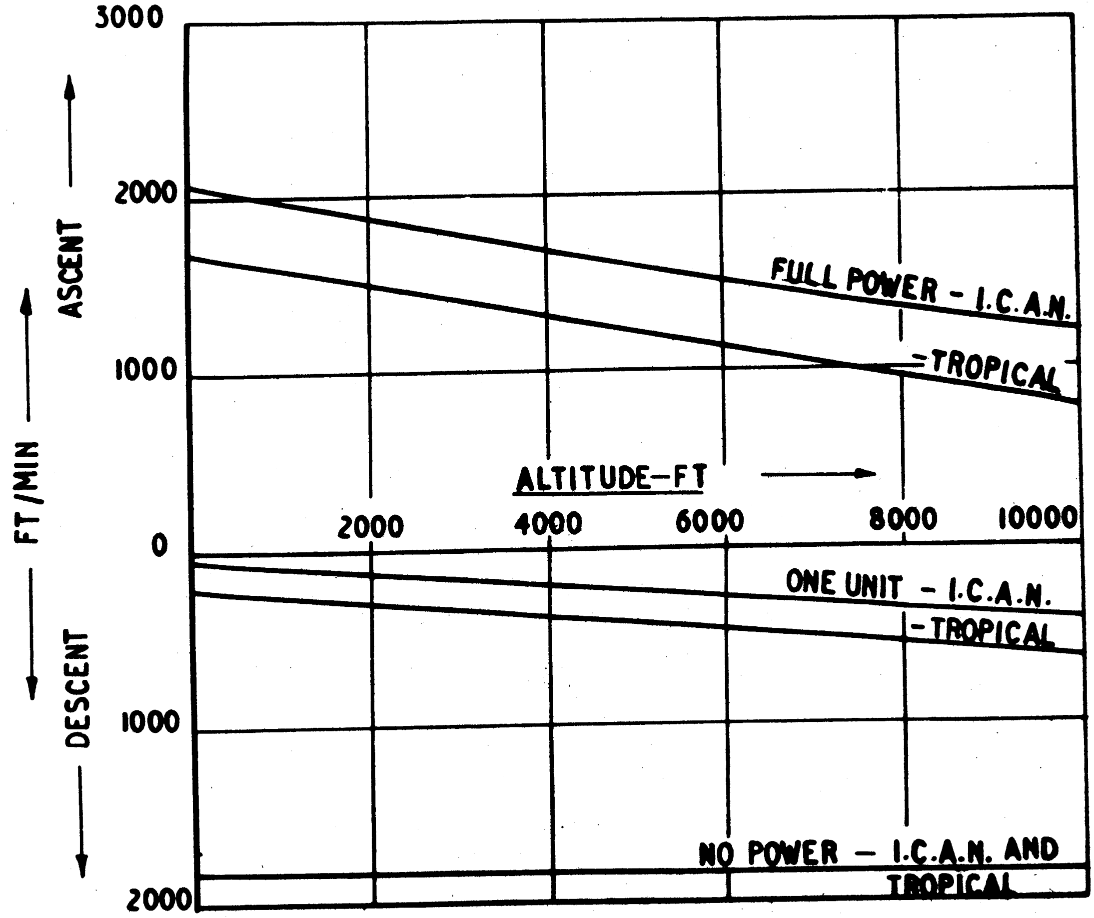



Fig 11.3

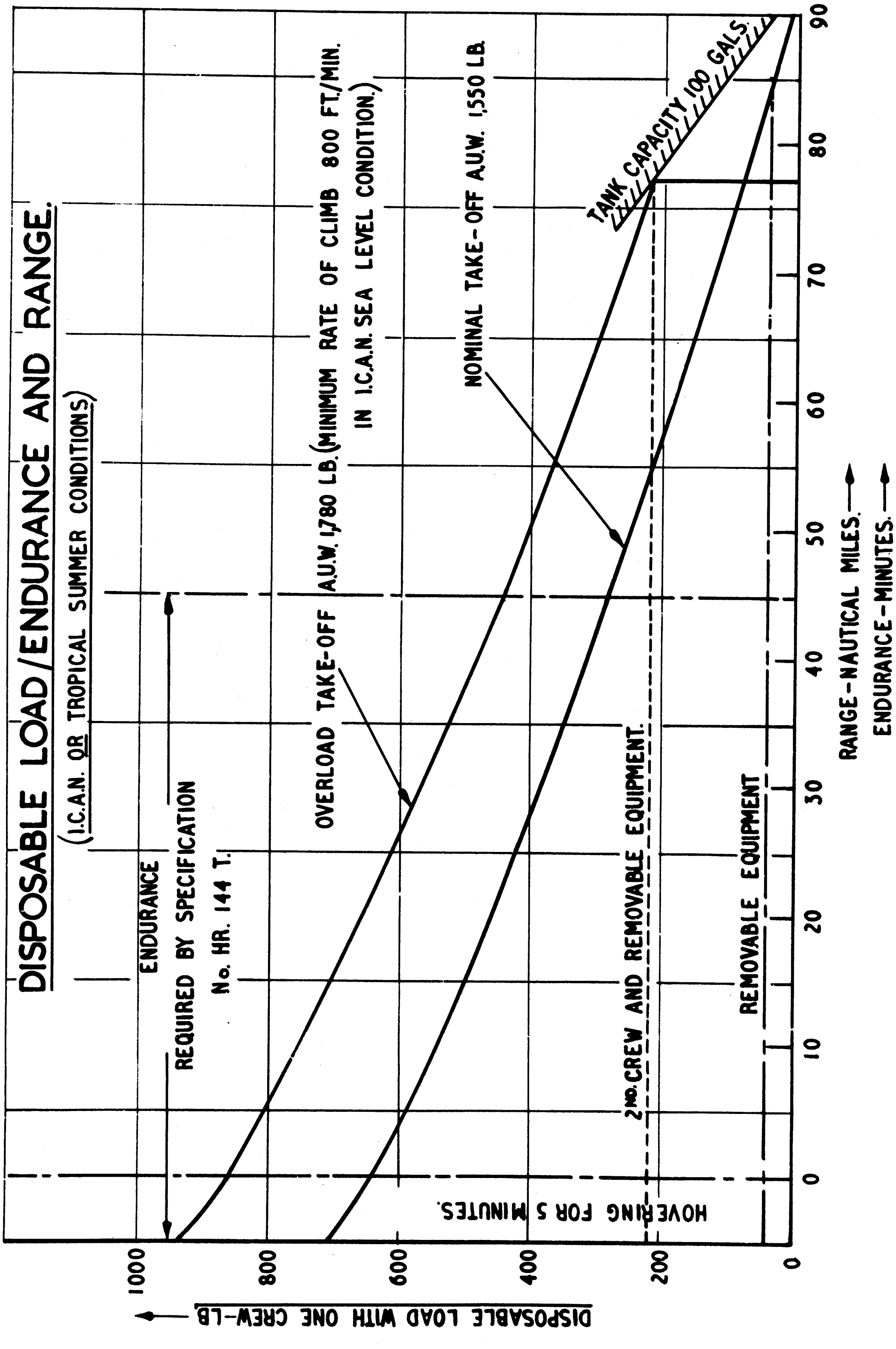

DISPOSABLE LOAD/ENDURANCE AND RANGE.
(I.C.A.N. OR TROPICAL SUMMER CONDITIONS)
ENDURANCE
REQUIRED BY SPECIFICATION
No. HR. 144 T.
OVERLOAD TAKE-OFF A.U.W. 1,780 LB. (MINIMUM RATE OF CLIMB 800 FT./MIN.
IN I.C.A.N. SEA LEVEL CONDITION.)
NOMINAL TAKE-OFF A.U.W. 1,550 LB.
TANK CAPACITY 100 GALS.
2ND. CREW AND REMOVABLE EQUIPMENT.
REMOVABLE EQUIPMENT
HOVERING FOR 5 MINUTES.
0
10
20
30
40
50
60
70
80
90
RANGE-NAUTICAL MILES.
ENDURANCE-MINUTES.
1000
800
600
400
200
0
DISPOSABLE LOAD WITH ONE CREW-LB

Fig 11.4

# WEIGHT ANALYSIS

# SECTION 12 WEIGHT ANALYSIS

| | lbs. | lbs. |
|---|---|---|
| Fuselage structure (including floorboard) | 51.0 | |
| Fuselage detachable fairings | 66.0 | |
| Fabric (Fin) ‘ | 2.0 | |
| Undercarriage | 80.0 | |
| TOTAL STRUCTURE | | 199.0 |
| Main rotor blades | 72.0 | |
| Main rotor hub | 113.0 | |
| Tail rotor complete | 5.5 | |
| Tail rotor and auxiliary drives | 5.0 | |
| TOTAL LIFTING_AND PROPULSIVE SYSTEM | | 195.5 |
| Starting units and charges | 20.0 | |
| Ham jot units | 12,0 | |
| Control cables and throttle | 4.~0 | |
| TOTAL POWER PLANT | | 36.0 |
| 100 gallon fuel tank | 42.0’ | |
| Fuel pump | 7.0 | |
| Fuel piping | 1.5 | |
| TOTAL FUEL SYSTEM | | 50.5 |
| Flying Controls | 23.5 | |
| Electrics | 43.0 | |
| Vacuum System | 3.0 | |
| TOTAL SERVICES ‘ | | 69.5 |
| Fire extinguisher | 5.5 | |
| Crowbar | 1,0 | |
| First—aid outfit | 20 | |
| Signal pistol and cartridges | 4.0 | |
| Seats, cushions and harness | 30,0 | |
| Instruments | 23,0 | |
| Miscellaneous equipment | 5.0 | |
| TOTAL FIXED EQUIPMENT | | 70.5 |
| Contingency | | 20.0 |
| Tare weight | | 641.0 |
| Removable Equipment | | 40.0 |
| Basic operationally equipped weight | | 681.0 |
| Fuel | ‘ 459.0 | |
| TOTAL WEIGHT (less crew) ‘ | | 1140.0 |
| Crew | | 360.0 |
| ALL-UP-WEIGHT | | 1500.0 |

# GROUND EQUIPMENT

# SECTION 13 GROUND EQUIPMENT

The basic simplicity of the design eliminates the need for special tools for servicing this helicopter.

The only special equipment required is as follows:

Canopy, rotor head and pitot head covers.

The machine can be manhandled on to trestles when necessary.

13.1. Mechanical Starter.

Because of the cost of rocket starting, it is felt that for airfield and base use a mechanical starter might be required.

The suggested design consists 'basically of a small internal combustion engine and a fluid flywheel mounted on a trolley. This drives a power head. The power head is clipped to the fuselage and power is transmitted to the tail rotor belts.

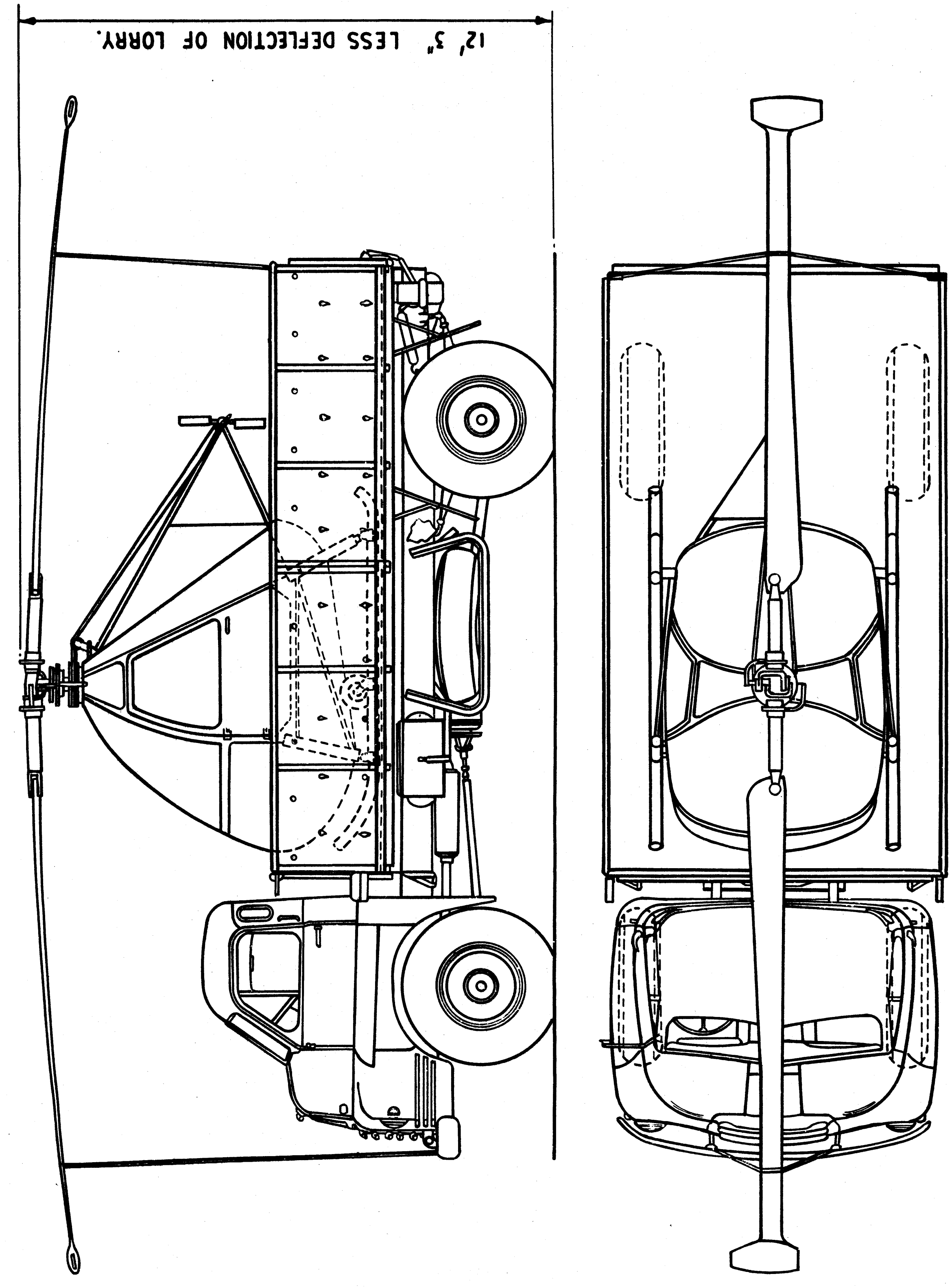



**Fig 13.1**

# DESIGN & CONSTRUCTION PROGRAMME

# SECTION 14 DESIGN AND CONSTRUCTION PROGRAMME

**TABLE VII**

1. A full description of the type of structure and the method of manufacture proposed is given in Section 4.0 headed “Airframe Construction”.
2. (a) The estimated man—hours necessary to manufacture the helicopter in production quantities of 100 off = 1800.
   (b)The estimated rate of delivery of production quantities of 100 = 8 per month.
3 .(a)The estimated man-hours to manufacture prototype jigs and tools = 13,900.
   (b)The estimated man—hours necessary to manufacture production jigs and tools, additional to *3(a)* = 42,000.

TABLE IX,

1. The estimated work for draughtsmen to complete the design as a complete prototype 850 man-weeks, including allowances for leave and sickness.
2. The estimated additional gross man-weeks necessary to productionise the drawings = 475.
3. The estimated time in weeks from the receipt of the I.T.P. to-
   (a) the completion of the design = 45.
   and (b) the first flight of the first prototype = 60.
4. The number of additional draughtsmen required = 5.
5. (a) The estimated man-hours necessary to design prototype jigs and tools = 1,550.
   (b)The estimated man-hours necessary to design production jigs and tools (additional to (a)) = 4,670.