proposal for

VTOL

VISUAL FIGHTER

IN ACCORDANCE WITH U.S
NAVY SPECIFICATION TS-140

AVRO AIRCRAFT LIMITED

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PROPOSAL FOR

VTOL

VISUAL FIGHTER

IN ACCORDANCE WITH
U.S. NAVY SPECIFICATION TS-140

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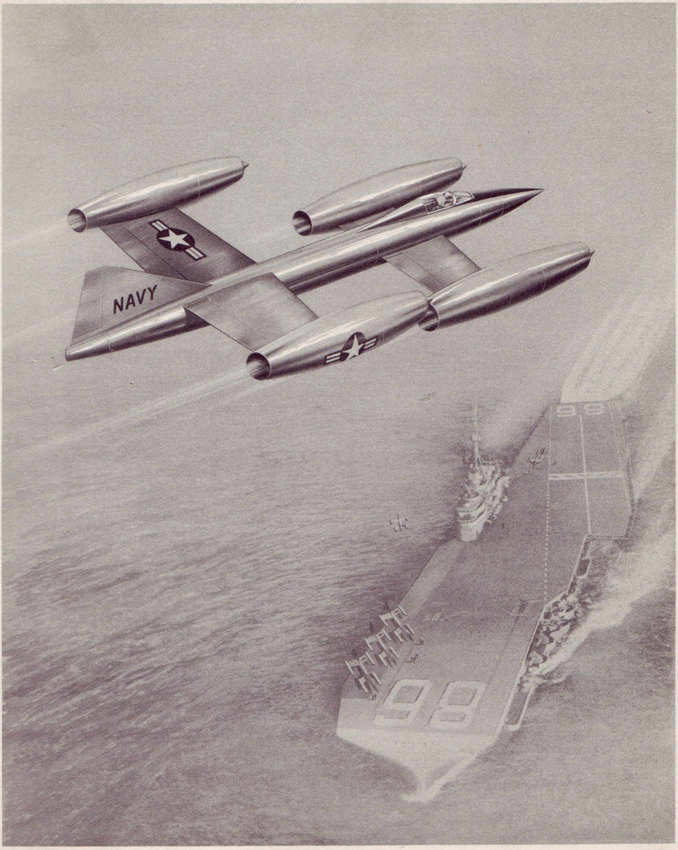
AD 36

OCTOBER 1ST 1956

AVRO AIRCRAFT LIMITED
MALTON ONTARIO

AVRO VTOL



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AVRO VTOL AIRCRAFT

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1.0 INTRODUCTION

The proposal covered in this brochure is in response to the United States Department of the Navy Specification TS-140, issued on 11th May 1956, and is in accordance with Addendum No. 7 to Spec. MIL-D-8706 (Aer). The requirements of General Specification SD-24-G were adhered to wherever applicable.

The aircraft envisaged is basically a conventional approach to an unconventional aircraft. Its basic configuration is a result of a step-by-step logical analysis of the requirements, culminating in a fighter which, by any standards, is extremely simple in concept, devoid of gadgetry, and very flexible. This latter point is deemed to be most important in an experimental aircraft - an aircraft designed to facilitate the transition from the tested and experimental article to a production aircraft without extensive and costly redesign and delay.

Radical landing techniques are not employed, so obviating any need for cockpit and control complications, thus improving reliability.

In presenting the proposal, the requirements are first of all reviewed, and the concept is discussed feature-by-feature. With the basic essentials established, it is then shown how the aircraft incorporates all the desired features. Following the discourse on the general configuration, the weight statements are given, performance is analyzed, stability and control are discussed, and an account is given of tests carried out to substantiate certain aerodynamic assumptions. Having discussed the overall aspects of the proposal, the more intimate details of the design are given, the servicing and handling problems are evaluated, and finally, a one year development programme is presented.

2.0 THE REQUIREMENTS

2.1 GENERAL

The requirements listed below are those affecting the overall design philosophy. There are many requirements specified which have little or no bearing on the basic configuration, and these are discussed in the paragraphs relevant to those requirements.

2.2 BASIC FEATURES

The aircraft, then, must have the following basic features:-

(a) High performance

(b) Vertical take-off and landing

(c) Jet Propulsion

(d) High thrust/weight ratio

(e) Low Weight

(f) Simplicity in concept

(g) Reliability

(h) Minimum size

(i) Good pilot visibility and comfort

(j) Armament provision for either 2.0" rockets or four "Sidewinder" guided missiles.

(k) Must be capable of operation from a 50 ft. square platform in a relative wind of 40 knots.

(l) Must be capable of hovering and flying at small positive and negative airspeeds without the use of auxiliary power.

(m) Must be capable of landing on a deck under adverse conditions, such as gusty air, rough seas, and 40 knot relative wind velocity.

(n) Must have space for electronic equipment, including search radar.

(o) Must be capable of expeditious handling.

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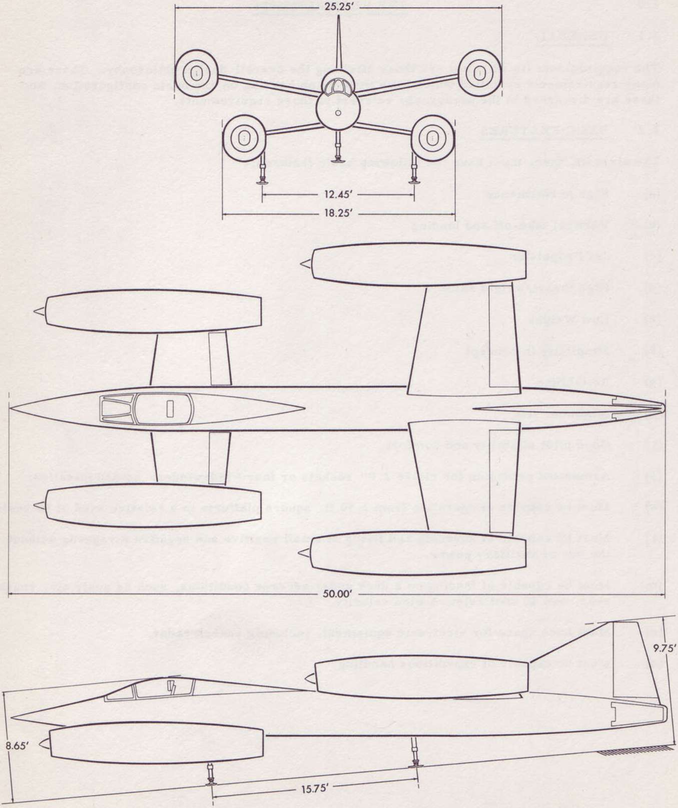

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FIG. 1 THREE VIEW GENERAL ARRANGEMENT OF VTOL AIRCRAFT

3.0 THE PROPOSAL

3.1 CONCEPT

The General Arrangement, shown in Fig. 1, is the result of an exhaustive analysis of the requirements. It is somewhat different to the more publicized types of V. T. O. L. aircraft, but it would seem to be the best compromise solution to the requirements set forth. It is possible that the requirements could have been met with a tail sitter type of aircraft, using one large engine with stabilizing ducts in the wings. The proposal described herein, however, is interesting in that the requirements are met using an entirely different approach.

The virtues and vices of contemporary V. T. O. L. aircraft were examined, and an attempt was made to incorporate all the virtues, while eliminating the vices. To accomplish this it was necessary to analyze each of the specified requirements in turn, and from this analysis, compile a statement giving the required basic essentials of the configuration.

The Analysis of Requirements, referred to above, is given in Para 3. 2, and although each individual requirement could undoubtedly be discussed at great length, the Analysis is purposely kept brief and to the point, in order that the logic may not be obscured by a mass of technicalities and argument.

The ultimate configuration developed from the above analysis, from numerous ways of meeting the requirements, and from tests, is a small flat-rising canard-type aircraft with two pairs of simple unswept wings of low aspect ratio. A Bristol Orpheus jet engine is mounted on each wing tip, the vertical thrust being derived by the use of jet-deflectors in the engine tail-pipes. The podded engines leave the fuselage free for installations of the cockpit, fuel, armament and equipment.

AVRO VTOL

3.2 ANALYSIS OF FEATURES

3.2.1 TAIL SITTER OR FLAT RISER?

TAIL SITTER HAS COMPLICATED LANDING TECHNIQUE, POOR PILOT COMFORT, DIFFICULT HANDLING AND DIFFICULTY IN LANDING UNDER ADVERSE CONDITIONS.

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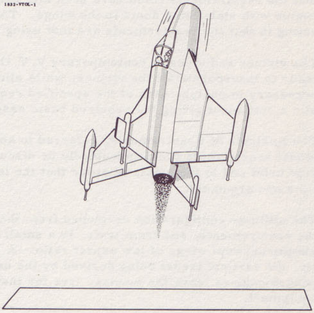
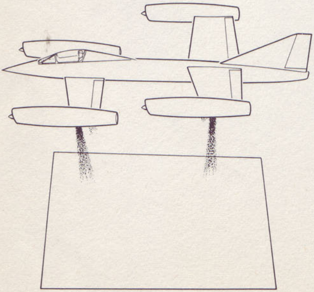


There are two basic V. T. O. L. configurations - the tail sitter and the flat riser. It would be difficult to meet all the requirements with a tail sitter - it has unconventional control, is not a particularly flexible design, and is very poor from the standpoint of the pilot's physical and psychological comfort, demanding the installation of tilting seats. Furthermore, it would probably be difficult to handle, on a rolling deck in a high wind.

FLAT RISER HAS SIMPLE LANDING TECHNIQUE, GOOD PILOT COMFORT, EASY HANDLING, AND IS NOT SEVERELY AFFECTED BY ADVERSE CONDITIONS.



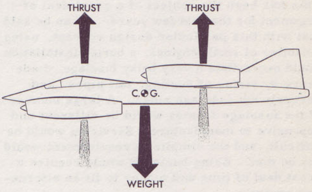
1833-VTOL-1

The flat riser sits on its platform in a near horizontal position, and takes off and lands from that position. As such, it does not require cockpit complications such as tilting seats, and the pilot can sit in a normal comfortable attitude. Being in a horizontal attitude, the aircraft could be handled easily with conventional equipment, and is not seriously affected by any adverse weather conditions. This type of aircraft could be of a very flexible and simple design.

3.2.2 LARGE ENGINES OR SMALL ENGINES?

A GIVEN AIRCRAFT POWERED BY A NUMBER OF SMALL ENGINES RATHER THAN ONE LARGE ENGINE WILL BE LIGHTER, USE LESS FUEL, AND WILL HAVE SHORTER OVERHAUL TIME.

Para. 3.2.1 has postulated that a flat riser is preferable to a tail sitter. A tail sitter can accomplish vertical take-off with one large engine, and wing ducting but the flat riser would require more complicated ducting to do the same thing. In this latter case, then, it is better to use several engines suitably placed about the aircraft C.G., than to install a large amount of ducting from the engines.



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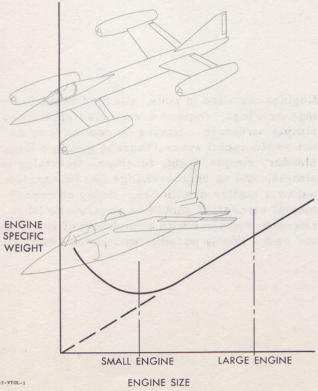

1837-VTOL-1

Another vital factor in choosing a number of small engines is that for a given thrust, several small engines are lighter than a single engine; i.e., they give the best thrust/weight ratio, and contribute to the overall weight saving. This, and the initial statement at the top of the page, are apparent from a study of the "3/2 law of turbine engine scaling", a law which can be interpreted to show that engine specific weight is directly proportional to engine diameter. An obvious advantage of multi-engines is that the thrust controls can be used directly for flight controls, thereby obviating the necessity for auxiliary ducting.

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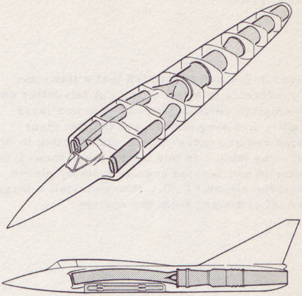
3.2.3 BURIED ENGINES OR PODDED ENGINES?

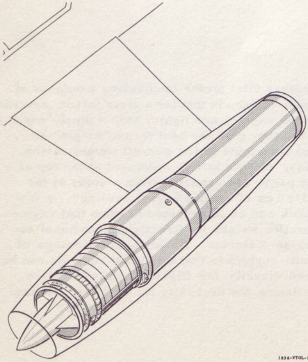
WITH THIS DESIGN, HAVING FOUR SMALL ENGINES, BURIED ENGINES WOULD BE A HEAVY, COMPLICATED, SPACE-CONSUMING, INFLEXIBLE METHOD OF INSTALLATION.

This has been the subject of a great deal of argument for the past few years. It can be said that with this particular design concept, using a number of small engines, a buried installation would necessitate a very bulky fuselage, made heavy by virtue of ducting, tail pipes, and severely cut fuselage rings. A large number of the fuselage frames would be different and expensive to manufacture. Servicing would be difficult, and the simplicity requirement would not be met. Being buried it would require a great deal of time and money to fit an alternative engine.



1835-VTOL-1

PODDED ENGINES HAVE A LIGHT, SIMPLE INSTALLATION, REQUIRING NO DUCTING, ARE SIMPLE TO SERVICE, FLEXIBLE, AND USE LITTLE SPACE.



1836-VTOL-1

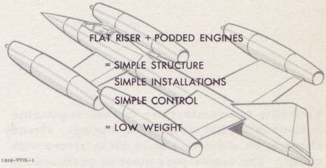
Engines installed in pods, attached to the fuselage or wings, require a very elementary and simple structure. Having no ducting they do not waste useful space, thereby giving a long, slender, simple, light, fuselage. Servicing is simple, and an engine-change can be carried out in a matter of minutes. Being an experimental aircraft it should be possible to fit alternative engines with a minimum of redesign and cost - this is possible with podded engines.

1840-VTOL-1

3.2.4 SIMPLICITY, RELIABILITY, AND WEIGHT

THESE ARE PARTLY SYNONYMOUS - ABSOLUTE SIMPLICITY LEADS TO LOW WEIGHT.

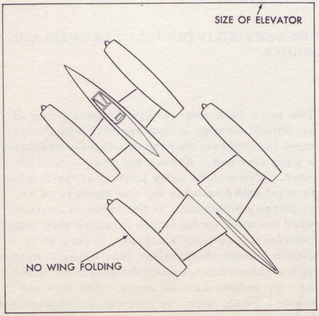
The requirement is that the aircraft shall be simple in concept, reliable, and of low weight. The foregoing paragraphs show how the requirements can be met whilst still maintaining the requirements for simplicity and lightness. All V.T.O.L. aircraft including the helicopter have a degree of unreliability in the take-off and landing phase, and it must be admitted that this proposal has a degree of unreliability during these phases. The overall aspects of this problem are more fully discussed in the Analysis of Configuration, Para. 4.2.2.



1838-VTOL-1

3.2.5 MINIMUM SIZE

TO BE OPERATED FROM 50 FT. SQUARE PLATFORM, AND SMALL ENOUGH TO BE TAKEN BELOW DECKS WITHOUT COMPLICATION OF WING FOLDING.



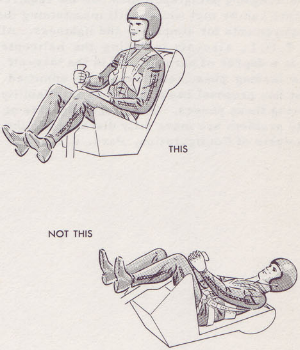
1843-VTOL-1

Size is, of course, a function of weight, and if weight is to be low, then size must be a minimum. The overall span is kept to a minimum by the use of low aspect ratio wings. Fuselage size will be determined purely by the cockpit, equipment and fuel stowage, and drag, and must be as short as possible. The size of the proposal, compared with a contemporary naval aircraft is shown in Fig. 2, and it should be noted that although wing folding is not necessary to fit the aircraft elevators, it can be accomplished if it should be necessitated by hangar stowage requirements.

3.2.6 PILOT VISIBILITY AND COMFORT

FOR VISIBILITY - NO NACELLES, FUSELAGE, OR WINGS IN LINE OF SIGHT. FOR COMFORT - NO TILTING SEATS OR COMPLICATED CONTROL.

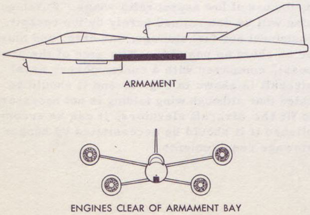
Visibility and comfort are further arguments in favor of the flat riser. The fuselage, already shown to be slender, should not interfere with vision. The engine pods must be positioned in such a way that they do not interfere with vision, and the wings also must be suitably placed. The flat riser previously proposed should be ideal for pilot comfort, both psychological and physical. It obviates the necessity for tilting seats, awkward posture, and difficult control.



1844-VTOL-1

3.2.7 INTERNAL ARMAMENT STOWAGE

ARMAMENT OF SUCH PROPORTIONS CAN ONLY BE CARRIED INTERNALLY IN FUSELAGE, AND MUST NOT HAVE INTERFERENCE FROM ENGINES.



1845-VTOL-1

The nose radar precludes the possibility of armament stowage in that region. The requirement for internal stowage of missiles precludes a wing stowage. Armament pods cannot be used. The centre belly portion of the fuselage is ideal and available for the stowage of all required armament. If the forward engines were mounted on the upper fuselage they would interfere with pilot's vision. If they were mounted under the fuselage they would interfere with armament trajectory and might be "blown out" by weapon exhaust gases. The forward engines must, then, be mounted away from the fuselage.

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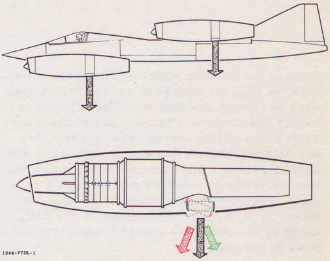
3.2.8 HOVERING

A FLAT RISER WITH A NUMBER OF SUITABLY POSITIONED PODDED ENGINES CAN, BY SIMPLE MEANS, BE MADE TO HOVER, OR TO FLY AT SMALL POSITIVE AND NEGATIVE SPEEDS.

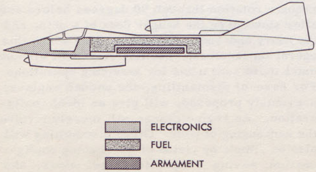
Moveable "eyelids" mounted at the jet exhaust, together with suitable throttle controls and jet deflectors, will deflect the jet in such a way that hovering or small positive or negative speeds can be accomplished. To do this with one large engine would be difficult, involving spanwise ducting for stability. The use of several small engines, each with eyelids, obviates the necessity for such complications, saves weight, and further enhances the simplicity, while affording adequate stability during the manoeuvre.



1846-VTOL-1

3.2.9 SPACE FOR ELECTRONICS, FUEL AND EQUIPMENT

MOST OF THIS, BY NECESSITY, MUST BE IN THE FUSELAGE, AND TO FACILITATE ROUTINE SERVICING, THE INSTALLATIONS SHOULD BE SIMPLE AND EFFICIENT.



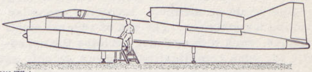
1847-VTOL-1

As previously stated the fuselage should be simple, devoid of engines and ducts. As such it would be ideal for efficient stowage of all items. Radar and associated electronics would be in the forward fuselage, all mounted for easy servicing. Fuel Cells would be in the centre fuselage - an efficient, simple position, ideal for small C.G. movement. Armament would be in the lower belly, and all general equipment could be simply and efficiently mounted.

3. 2. 10 HANDLING

CRITERIA: HANDLING EQUIPMENT SHOULD BE LIGHT, SIMPLE, SMALL, EASY TO OPERATE IN A HIGH WIND WITH A ROUGH SEA. AIRCRAFT ITSELF SHOULD BE SMALL, EASY TO SERVICE, AND EASY TO DISMANTLE.

Some of the handling equipment for a tail sitter must be high for access to cockpit. In a high wind and rough sea this would be difficult to handle and dangerous to use. A flat riser aircraft could utilize conventional present-day handling equipment, easy to design, easy and safe to operate, and some existing equipment may be easily adapted for use on such a project. The shipboard equipment proposed in Para. 11.0, being concentrated in one unit, could be economically stowed in the close confines of the ship.



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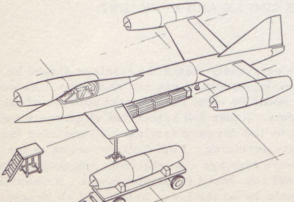

1849-VTOL-1

Some mention has already been made of aircraft size, and of how it should be small enough to be stowable without folding wings. A small aircraft will obviously be easier to handle and will take up less hangar space than a large one. Unlike the tail sitter, the flat riser will not require rotation through 90 degrees before and after stowage on the hangar deck of a ship, and the heavy and space-consuming gear which this operation requires. The flat riser will also be much more accessible for servicing operations. For ease of dismantling, the podded engines, previously proposed, will give an ideal configuration. An engine change will merely involve the exchanging of pods. Other dismantling will also be simple by virtue of the ensuing configuration, having simple fuselage, wings, and structural joints.

3.2.11 BELLY LANDING

A FLAT RISER WITH PODDED ENGINES IS WELL SUITED TO BELLY LANDING.

There is always the possibility that an emergency, such as engine failure, may arise necessitating a belly landing. If the lower surfaces of the forward engine pods are lower than the fuselage, the aircraft could glide, or fly-in, to a three-point landing on the forward engine pods and the rear fuselage, the structure at each point being designed to take the loads imposed on it by such a landing.

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The aircraft may, of course, be damaged. Unlike most other aircraft this would not render the aircraft useless for very long. The damage would, in all probability, be confined to the two forward engine pods, which could be easily and speedily replaced.

The proposed aircraft can glide or fly-in to a belly landing on one or more engines, the approach speed being 172 knots.

The configuration is well suited to ditching requirements. Unlike contemporary tail sitters it has no protuberances on the lower surface, thereby obviating the possibility of "pitching in".

4.0 BASIC CONFIGURATION

4.1 PHILOSOPHY

From the foregoing Analysis, it is considered that the aircraft should incorporate the following basic features:-

(a) Should be a flat riser

(b) Should have podded engines

(c) Should have a number of small engines

(d) Should be simple and small

(e) Should have no obstacles in the pilot's line of sight

(f) Should have an armament bay in the centre fuselage

(g) Should have no interference between engines and armament

(h) Should have jet deflectors for hovering

(i) Should have extremely simple, easy-to-service, fuselage stowage for electronics, and general equipment

(j) Should have fuselage fuel tanks

4.1.1 ENGINE POSITIONING

As mentioned in Para. 3.2.2 the engines must be so positioned that the thrust is balanced about the centre-of-gravity of the aircraft. It has already been shown in that same paragraph that a number of small engines should be used. The net result of these two parameters is that the aircraft should have one or more engines at the forward end of the aircraft, and one or more engines at the aft end.

The engines must be well separated in the spanwise direction to provide adequate roll control, and thereby obviate the necessity for installation of wing ducting. Being at some distance from the fuselage the engines must be in pods, to satisfy the requirements of Para. 3.2.3, and they would automatically satisfy the pilot's vision and armament requirements of Paras. 3.2.6 and 3.2.7.

To summarize the above, the aircraft should have several small engines, housed in pods, positioned equally about the c.g., and that these pods should be at some position away from the fuselage commensurate with pilot's vision and roll control requirements.

4.1.2 TYPE AND NUMBER OF ENGINES

The proposal is based on the use of a number of small jet engines, and the Bristol Orpheus B.Or.11 was selected as the most suitable engine to meet the requirements. It is small, has a high thrust/weight ratio, and a great deal of information is available on the engine.

It is rarely that an engine is ideally suited to a given set of requirements, and this engine is no exception. It has its faults, mainly in regard to frontal area and accessories layout, and these are more fully discussed in Para. 10.3.

The Orpheus 11 has the advantage of being a development from a series of engines. The Orpheus 3 is an already proven engine, and the Orpheus 11 is merely a development of this engine with a higher thrust and better fuel consumption. It will be available early in 1958.

The Orpheus 3 engine, currently available, would be used in the one-year development programme, to prove the principles involved in this proposal. By the time this programme is completed the Orpheus 11 will be available.

Further details of the engine choice are given in Para. 10.3.

There are, undoubtedly, other engines being developed that would be suitable for this proposal, and they would be considered when information became available. Some engine companies have already made statements on this matter, and some tentative figures have been given. Para. 6.0 shows how the performance would be affected by the use of such engines.

4.1.3 WING SHAPE AND POSITION

Having had experience with delta-wing aircraft, such a planform was considered. This design, although it had aerodynamic merit, was somewhat inflexible. It was necessary to have two engines mounted in the aft end of the fuselage, and two mounted in pods attached to a pair of forward wings.

The fuselage had to have ducts passing along most of its length to feed the engines, and as stated in Para. 3.2.3 the engines buried in the fuselage can be ruled out from the standpoints of flexibility, simplicity, and weight. If, at a later date, it is decided to install different engines a major redesign would be involved.

In view of the above it was decided to keep all four engines in pods, with each pod mounted at the end of a wing.

The next logical step was to mount the engine pods on a conventional straight-wing aircraft. An engine was mounted on each wing tip, and two other engines were mounted on the forward fuselage.

A requirement for the forward engines is that they should not interfere with the pilot's visibility, so if their fore-and-aft position must be coincident with that of the pilot, they must be mounted either on the fuselage, tucked under the cockpit, or on stub wings. The fuselage mounting is not satisfactory as it interferes with armament, and is inflexible, so they must be pod-mounted at the ends of stub wings.

An analysis of asymmetric flight conditions was carried out, and this determined the distance between engines. This, together with the C.G. position demanded that the main wing be moved aft, and the forward stub wings be moved forward.

These wing movements were of such a magnitude that the conventional tailplane became redundant, and the aircraft became a canard.

4.1.4 FUSELAGE SHAPE

With all engines and ancillary ducting taken out of the fuselage, it merely becomes a container for pilot, electronics, armament, fuel and equipment.

There is nothing which demands that its cross-section be other than the ideal circular shape, tapering at front and rear from a cylinder.

A high performance aircraft should have high fineness ratio, and this can be achieved. The nose shape is determined by radar requirements, and the pilot's position is determined by visibility and centre-of-gravity requirements.

The obvious position for armament is the centre under-belly, a position ideally suited to speedy and simple arming of the aircraft. The flat riser sitting, as it does, in a fairly horizontal attitude, could be loaded with armament, using conventional gear, or the tender vehicle described in Para. 11.4. Furthermore, keeping in mind that this is an experimental aircraft, it would be a comparatively simple matter to fit alternative armament or reconnaissance equipment - a larger depth requirement would involve either a change to the fuel cells or a ventral bulge.

Having placed the vital essentials, the radar, the pilot, the fuel, and the armament, there is still adequate space left for electronics and equipment.

The electronics are an integral part of the nose radar and the armament, so for overall efficiency the electronic pack should be close to both of these items. One other requirement is that the electronics should be ideally placed for quick servicing. All these requirements can be met by placing the electronics behind the pilot in the fuselage.

The general equipment, such as oxygen, pneumatic, hydraulic, air conditioning, and so on, has two basic requirements in common: (a) it should be capable of being serviced easily and quickly and (b) for low weight it should be placed as close as possible to the items connected with that equipment. These suggest that the equipment should be in two basic packs: one near the cockpit, and one further aft. By suitable adjustment, the fuel tanks and equipment bays can be placed to give maximum overall efficiency.

4.2 DESIGN ANALYSIS

4.2.1 GENERAL

Paragraph 3.2 showed how the requirements could best be met, giving the basic essentials of the aircraft. Paragraph 4.0 shows the logic behind the choice of configuration of the individual basic elements, i.e., wings, engines and fuselage.

Fig. 1 shows the general arrangement of the proposed aircraft and the following paragraphs are an analysis of that configuration. The more intimate details of the various component parts are given in Para. 10.0.

4.2.2 ANALYSIS OF CONFIGURATION

The aircraft is a canard with two pairs of simple unswept wings of low thickness and aspect ratio. A Bristol Orpheus jet engine is mounted on each wing tip, being housed in a detachable pod. The fuselage has a high fineness ratio, and a circular cross-section.

The following are the main characteristics of the configuration:-

(a) Small overall size. To illustrate this aspect, reference should be made to Fig. 2, in which the proposed aircraft is compared, in size, with a typical present-day fighter, namely the McDonnell F2 H "Banshee". The aircraft is, indeed, small enough to be taken below decks without recourse to wing-folding.

(b) Low height. The maximum height from the deck to the nose radar equipment is 6 ft., and as the aircraft sits in a tail-down attitude, all or most of the servicing can be done without the use of stepladders. The height from the deck to the canopy rail is approximately 7 ft. 2 in. The maximum height of the aircraft, to the tip of the tail fin, is approximately 9 ft. 9 in. This excludes any small additional height that may result from the use of lifting devices placed under the landing pads, but in any event the aircraft will be some 7 ft. under the maximum height requirement.

(c) Simplicity of Concept. This can be split into three parts: overall layout, structure, and equipment.

The overall layout is of the utmost simplicity, having a conventional fuselage, conventional tail fin, and conventional unswept wings. The positioning of these main components, with the pod-mounted engines at the wing tips is ideal for an experimental aircraft. Alternative engines could be fitted, new wing shapes could be investigated, and fuselage changes could be carried out with a minimum of redesign, cost and time, thereby leading to a quick transition from the experimental aircraft to the production aircraft. Another facet of the above is that the aircraft could be easily broken down for quick and efficient manufacture, and in service, any damaged parts could be quickly removed and replaced by new parts.

Such a simple overall layout lends itself naturally to a simple structure. Most of the frames are circular, and fuselage stressing should be comparatively simple. The simple frames, together with the conventional non-tilting pilot's seat and complete absence of duct-work and inlets all lead to a reduction in weight and in engineering man-hours. Also, the lofting will be comparatively simple due to the absence of irregular shapes. The above observations are suitably illustrated in the Inboard Profile, Fig. 3.

The unswept wing has conventional spar-and-rib construction and as the fuel is completely housed in the fuselage there are not complications in this respect. There is no undercarriage cut-out, and this again leads to simplified stressing and design, and to saving of weight.

The undercarriage has no wheels. Pads are attached to the ends of conventional oleo legs, and the operating mechanism is simple.

The engine pods are rigidly, but simply, attached to the wing, and are of conventional monocoque construction - a system of frames and skin. The engine mounting is simple and the only complications are those portions of the pod affected by the jet deflection mechanism, and undercarriage attachment.

All the equipment and armament is efficiently stowed in the fuselage. The simplicity of the arrangement can be seen in Fig. 3 and Fig. 61, and is discussed in some detail in Para. 10.9, Para. 10.10, and Para. 10.11. To illustrate the simplicity of concept, it is pertinent to repeat here some of the statements made in those paragraphs.

The AN/APS-67 Search Range Radar is completely housed in the detachable nose fuselage, forward of the cockpit.

The AFCS Ex-16 equipment is housed in the equipment bay immediately adjacent to the armament it serves.

The AN/ARN-21 and AN/ARC-52 for navigation and communication, (to be replaced later by AN/ASQ-19), and all automatic stabilizing equipment is housed in the detachable, non-structural aft canopy region, again adjacent to the parts they serve in the cockpit.

The rest of the equipment is housed in the two main equipment bays of the centre fuselage, and again, that equipment is so placed that it is as close as possible to the components served by that equipment. A typical instance of this is the air conditioning equipment. This is situated in the forward equipment bay, immediately adjacent to the parts requiring air-conditioning.

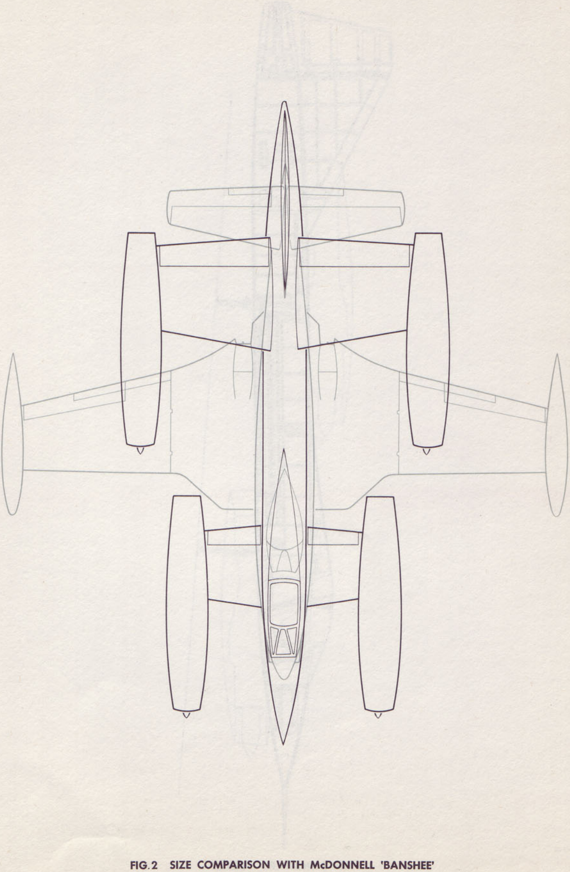

FIG. 2 SIZE COMPARISON WITH McDONNELL 'BANSHEE'

AVRO VTOL

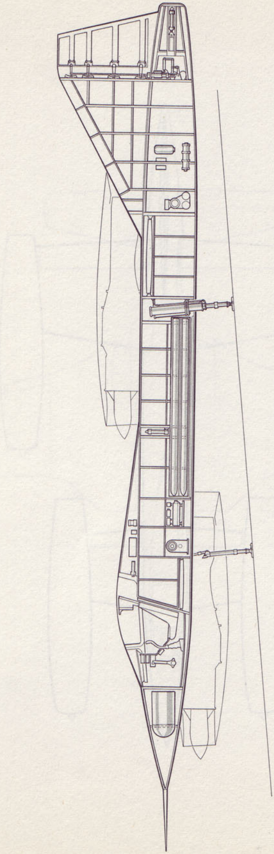

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FIG. 3 INBOARD PROFILE

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The armament is housed in the accessible under-belly of the centre-fuselage. This is an ideal position for loading and unloading the armament and also for firing it.

The equipment and armament are stowed, in a very efficient, simple manner, and this leads to easy installation, simple servicing, and minimum weight.

(d) Pilot Visibility. The only obstructions to the pilot's vision are the engine pods, but these are placed so far away from the fuselage that their effect is negligible. Apart from that, the visibility is excellent, the aircraft having a short slender nose giving good forward visibility, and no interference whatsoever to downward vision.

(e) Pilot Comfort. His physical comfort is a function of two things - his posture, and the number and position of the instruments and controls to be operated. His pychological comfort is again a function of two things - posture, and complication of control. The flat riser does not necessitate the pilot sitting on his back. He is in a conventional sitting position. The instruments and controls are little different to those of a conventional aircraft, and, by virtue of the automatic devices, they have very little complication. The pilot is comfortable both physically and pyschologically throughout the duration of the flight.

(f) Podded Engine Installation. The reasons behind this installation have been given in preceding Paragraphs 4.1.1 and 4.1.2. The full details of the installation are given in Paragraph 10.4. The installation is flexible in that alternative engines can easily be fitted, and leads to a light, duct-free, simple structure, the engines being so positioned that they are not affected by the weapon exhausts.

(g) Speed Brakes. These are fitted to the aft end of the fuselage, and are of the petal type. They are simply operated from jacks housed inside the fuselage, and are further discussed in the section on Stability and Control.

(h) Reliability. Paragraph 3.2.4 states that in one particular phase of flight the proposed aircraft has a degree of unreliability, but it is pointed out that this is common to all vertical take-off aircraft. The unreliable phase is that between initial take-off and transition and between transition and landing. Present day V.T.O. aircraft must make a crash landing in the event of engine failure during this period, but reliability is a function of time, and as the time element is relatively short - less than a minute in fact - the overall reliability is not seriously impaired. It should be pointed out that after transition, the aircraft can fly normally on one or more engines and, as stated in Paragraph 3.2.11, it can if necessary make a very successful belly landing.

Other aspects of reliability are a function of simplicity. Simple controls, simple installations, simple and easily-accessible equipment installations, all lead to a well maintained and reliable aircraft, and it has already been shown how the proposed aircraft meets these requirements.

5.0 WEIGHT AND C. G.

5.1 WEIGHT AND C. G. STATEMENT

The weight statement conforms to the requirements as laid down in the General Specification SD-24-G, Paragraph 3. 1. 3, and is given in Table I.

The centre-of-gravity statements for the aircraft are in accordance with the requirements of the General Specification SK-24-G, Paragraph 3. 1. 4, and are given in Table II.

5.2 DETAILED WEIGHTS

The weight data presented in Tables I and II are derived from a combination of empirical and statistical methods. The principal references are:-

(1) Wing Weight Equation by Ivan H. Driggs (U. S. Navy) as presented in the Journal of the Royal Aeronautical Society of March 1950.

(2) Fuselage Weight Equation by Ivan H. Driggs - See Ref. 1.

(3) Fuselage and Tail Weight Equations by F. Grinstead as presented in the (U. K.) Royal Aircraft Establishment Report Number Structures 24.

(4) Undercarriage Weight Prediction by M. E. Burt and E. L. Ripley as presented in the (U. K.) Royal Aircraft Establishment Report Number Structures 80.

(5) Rand Corporation Report R-171 (U. S. A.) - Bomber Fuselage Weight Prediction Equation.

(6) Additional unpublished data available to Avro Aircraft Limited.

The weight of the mainplane was derived independently by two methods:

(1) From Driggs equation (Ref. 1) by assuming a stress factor along the "mean" line and

(2) by estimating weights from a set of preliminary drawings. The two methods gave an exact agreement.

The wing weight obtained in this manner is deemed to be conservative both in the manner in which Driggs' equation was used, and in the manner in which the preliminary schemes were treated. The use of a point from the "mean" stress factor line is thought to be conservative because on the CF-105, we have been able to design a wing which has a stress factor above the "good design" line. In addition, in the preliminary schemes, certain weight saving features were not incorporated in order that the weight estimate be properly conservative.

The basic airload is supported almost equally between the two lifting surfaces, and because of this, the weight of the main plane can be substantially less than that for a conventional tailed airplane.

The weight of the nose plane was taken directly from the weight of the main plane, Driggs' formula being applied to reduce the weight for the smaller span and area . In addition, this weight was checked against one calculated from preliminary stress calculations.

The fuselage weight was determined by deriving values from Rand (Ref. 5), Driggs (Ref. 2), and Grinstead (Ref. 3). These values were then checked with a weight calculated from a preliminary structural layout.

The fuselage weight will be less than one would normally expect for a conventionally-tailed airplane because the basic airloads are transferred to the fuselage from two planes almost equally, instead of from one.

The tail weight was calculated from a unit weight per square foot derived from Grinstead's report (Ref. 3) augmented by other unpublished data on high speed aircraft (Ref. 6).

The undercarriage weight was based on the method of Burt and Ripley (Ref. 4) with sheels and tires replaced by pads. This method was supplemented by calculations from preliminary drawings.

The nacelle weight is derived from an assumed structural layout required to house the engine and accessory equipment.

The engine weight is based on data presented in a Bristol Brochure for the Orpheus 3 and on an Advance Performance Folder for the Orpheus 11. The weight includes allowances for oil tanks and piping, fuel filters and fire bulkheads. The jet pipe weight per lineal foot is taken directly from the Orpheus 3 brochure. The starter weight assumes an air turbine starter plus installation and ducting weights. The weight of the jet deflectors is based on an investigation made by Orenda Engines Limited.

System weights are based on the data available from the CF-100 and CF-105 aircraft. Instrument weights contain a ten percent provision for expansion. The hydraulics and pneumatics systems have been assumed similar to the CF-100 and CF-105 with the power supply and piping reduced to suit the aircraft size and system load. The electrics were calculated in a similar manner, each circuit having been examined and compared with the CF-100 and CF-105. The power supply and wiring has been reduced to suit the size of aircraft and the system load. The flying control system has been estimated assuming a hydraulic power boost with artificial feel.

The weights presented are estimates and are, of course, believed to be correct. In addition, where judgement has been required, estimates tending to be conservative have been made.

5.3 C.G. POSITION

Due to their importance in the proper operation of a flat riser the C G figures, shown in Table II have been very carefully estimated. The moment of the engines' thrust about the C.G. must be zero in the take-off configuration, and in addition to this, the normal requirements must be met, the C.G. position being so located in flight as to give adequate stability and control.

TABLE I - WEIGHT STATEMENT

I(a) WEIGHT EMPTY (GUARANTEED) LBS				10,280
Wing Group - Main Plane			720	
- Nose Plane			350	
Tail Group			200	
Body Group			1000	
Alighting Gear Group			180	
Flight Controls Group			200	
Nacelle Group			1200	
Propulsion Group			4930	
Main Engine Installation		3900		
Exhaust System		700		
Jet Pipes	200			
Jet Deflectors	500			
Fuel System		190		
Engine Controls		20		
Starter System		100		
Instruments and Navigational Equipment Group			35	
Hydraulic and Pneumatic Group			260	
Electrical Group			325	
Electronics Group			400	
Armament Group			200	
Furnishings and Equipment Group			130	
Air-Conditioning and Anti-Icing Group			150	
I(b) USEFUL LOAD LBS				5,970
Crew - Pilot			215	
Fuel			5000	
Oil			30	
Armament			700	
Oxygen Equipment			5	
Miscellaneous Equipment			20	
I(c) SUMMARY LBS				
Weight Empty				10,280
Useful Load				5,970
GROSS WEIGHT LBS				16,250

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TABLE II - CENTRE-OF-GRAVITY LOCATIONS

NOTE: M. A. C. refers to the Mean Aerodynamic Chord of the Main Plane. *

NORMAL GROSS WEIGHT - Basic Flight Configuration

Forward of Leading Edge of M. A. C.	108. 8% M. A. C.
Below Leading Edge of M. A. C.	19. 7 ins.

NORMAL GROSS WEIGHT - Landing Configuration

Forward of Leading Edge of M. A. C.	109. 8% M. A. C.
Below Leading Edge of M. A. C.	19. 9 ins.

EXTREME FORWARD POSITION POSSIBLE IN FLIGHT - Landing Gear Down

Forward of Leading Edge of M. A. C.	117. 6% M. A. C.
Below Leading Edge of M. A. C.	18. 8 ins.

EXTREME REARWARD POSITION POSSIBLE IN FLIGHT - Landing Gear Up

Forward of Leading Edge of M. A. C.	105. 1% M. A. C.
Below Leading Edge of M. A. C.	19. 0 ins.

* Leading Edge of M. A. C. of Main Wing is 30 ft. aft of Fuselage Datum. Length of M. A. C. is 7. 82 ft. Extreme Range of C. G. is 20. 8 ft. to 21. 8 ft. from Fuselage Datum.

6.0 PERFORMANCE

6.1 GENERAL

The weight of VTOL aircraft plays an even more prominent role than it does for conventional aircraft. For instance a VTOL aircraft designed for 1.1 thrust/weight ratio would have its primary role eliminated by a 10% increase in the design take-off weight. On the other hand a conventional jet aircraft could still take-off to clear 50 feet with approximately a 25% increase in take-off distance. Therefore, in order to cater for possible weight increases and/or worsened installed power plant efficiencies, a reserve of about 10% in thrust/weight ratio is kept in hand. This can be obtained if necessary by water-methanol injection including the extra weight of fluid and equipment.

Minimizing the weight indicates fixed engine installation intake and outlet geometries, necessitating a compromise over a wide speed range (M=0 - 2.0). Detail design at a later stage may show improvement in certain variable geometries (or the equivalent) in that the fuel weight saved to accomplish the required mission, may override the fixed weight penalty.

Minimizing the weight also indicates the elimination of afterburning with its inherent high rate of fuel flows. Fortunately the large relative thrust required at sea level static conditions for VTOL aircraft, leads to the possibility of achieving supersonic flight up to M=2 without afterburning. This possibility is further enhanced through overspeeding the engine at high speeds. This not only provides better net thrust but achieves better engine matching, thus minimizing the spillage drag. However, the necessity is still required of tending to optimize the thrust-drag configuration for the M=2 case with some fuel penalty during the subsonic and transonic portions of the mission.

The performance presentation following, is based on the Bristol Orpheus 11 engine with sea level static thrust rating of 5,760 lb., and with engine overspeeding capability of 10% at M=2. With this engine the Avro VTOL proposal falls short of the specification but is considered a good interim aircraft. The project has two deficiencies. First, it lacks the sea level static thrust to allow sufficient fuel to meet the required mission profile. Secondly, it has insufficient excess thrust to accelerate to M=2 in the required time. For these reasons, the mission profile is reduced from a M=2 combat to M=1.75 combat with decreased loiter time.

A development of the Bristol Orpheus 11 promises a 21% thrust boost with extremely little weight increase. This increased power should successfully meet the performance specification in all aspects but one. This is the time of 3.5 minutes to 60,000 feet. Buffet requirements require an acceleration to 1.5 M.N. prior to climbing to 60,000 feet. However, the energy height obtained in 3.5 minutes will be equivalent to M=.9 at 60,000 feet.

It is believed that performance in excess of the specification could be achieved by the proposed configuration if an engine was available with the attributes of the Bristol Orpheus engines plus a considerably improved thrust to frontal area ratio. It appears that such an engine is within the present state of the art.

6.2 PERFORMANCE DATA

The following is a summary of the performance data and its derivation used to obtain the subsequent performance figures.

6.2.1 AIRCRAFT DRAG

6.2.1.1 Profile Drag at Zero Lift

The subsonic drag is based on data obtained from the R.Ae.S. data sheets. A 25% drag increase

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is added to the fuselage and engine pods and a 15% drag increase is added to the wing, controls and fin. These allowances are for manufacturing limitations such as rivets, joints roughness, protuberances and leakage. Table 3 then gives the subsonic profile drag breakdown based on the nominal wing area of 145 sq. ft. The Reynolds Number corresponds to .9M at 40,000 feet.

TABLE 3

SUBSONIC PROFILE DRAG AT ZERO LIFT BREAKDOWN

Aircraft Part	C_{D_o}	% Total C_{D_o}
Fuselage (incl. canopy)	.01008	29.9
Engine pods	.01120	33.1
Wing	.00660	19.5
Controls	.00298	8.8
Fin	.00294	8.7
TOTAL	.0338	

To appreciate the above figures more, the following observations are made:

- Aerodynamic cleanness (drag based on wetted area) = .00344
- Cleanness ratio (profile drag/friction drag) = 1.45

The supersonic drag is estimated on the C_{D_π} system in a similar manner to the subsonic case. Time prevents a full scale investigation of drag by the supersonic 'area rule'. However, a check point at 1.35 M by the area rule is given in the appendix. This substantiates the present estimate. Table 4 then gives the supersonic profile drag breakdown at 2.0 M, based on the nominal wing area of 145 sq. ft. The Reynolds number corresponds to 2.0 M at 60,000 feet.

The wave drag of the fuselage and engine pods is based on Fraenkel's work, [2,3] and the main-plane, control and fin wave drags come from a collection of drag data by Merchant [4]. For the canopy drag, see reference [5]. The skin friction allowance is based on turbulent boundary layers in compressible flow as ascertained by Cope [6].

TABLE 4

2.0 M PROFILE DRAG AT ZERO LIFT BREAKDOWN

Aircraft Part	C_{D_o}	% Total C_{D_o}
Fuselage	.01401	19.9
Canopy	.00420	6.0
Engine Pods	.02707	38.5
Wing	.00871	12.4
Controls	.00342	4.8
Fin	.00371	5.3
Interference	.00918	13.1
TOTAL	.0703	

See figure 4 for the curve of total C_{D_o} versus Mach Number.

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AVRO VTOL

1913-VTOL-1-1

FIG. 4 PROFILE DRAG AT ZERO LIFT VERSUS MACH NUMBER

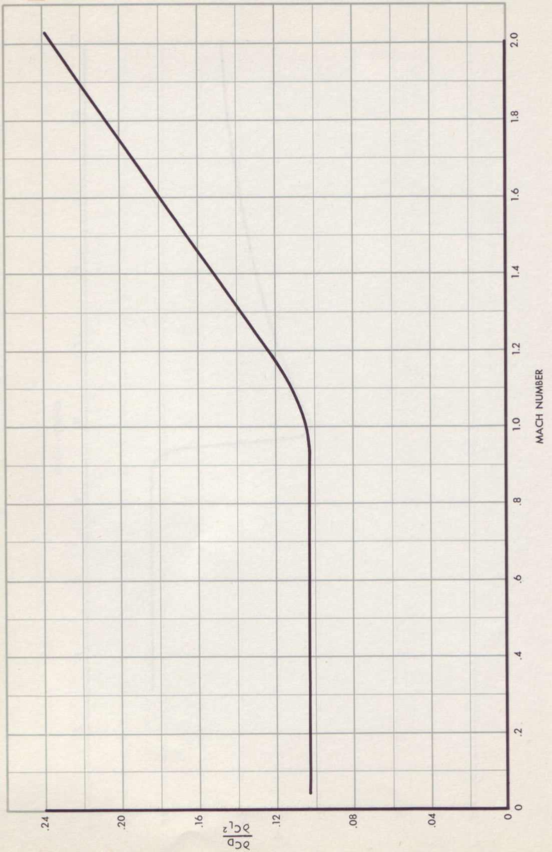

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FIG. 5 DRAG DUE TO LIFT VERSUS MACH NUMBER

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6.2.1.2 Drag due to Lift

The subsonic drag due to lift is based on a basic drag efficiency of wing and control of 1.43 (i.e. $C_{D_i} = 1.43\, C_L^2 / \pi\, AR$. This basic drag is influenced by the engine pods by the relation:

$$C_{D_i \text{ with pods}} = C_{D_i \text{ without}} \times \frac{1}{1 + \frac{\Delta \text{span}}{\text{span}}}$$

. Further the lift is nearly equally distributed between wing and control. Thus based on the wing nominal area of 145 sq. ft. the subsonic $d\,C_D / d\,C_L^2 = .103$.

Since the leading edge of straight wings is supersonic at supersonic speeds $dC_D/dC_L^2 = 1/C_{L_\alpha}$. For the derivation of C_{d_α}, see Paragraph 7.0 on Stability. Figure 5 then presents the curve dC_D / dC_L^2 versus Mach Number.

6.2.2 MAXIMUM LIFT AND BUFFET

Nearly half the weight of the aircraft is supported in flight by the control surface which is less than half the size of the wing. Fortunately the aircraft being of the canard type, the elevator on the control surface will improve the lift capabilities of the control.

Test data [11] on a thin straight, low aspect ratio wing with centre mounted tip nacelles gives a $C_{L_{Max}}$ of .8. An increment in C_L of .24 due to trailing edge flaps is determined from Reference 12. On referring the lift coefficient to the nominal wing area of 145 sq. ft., $C_{L_{Max}}$ becomes equal to 1.08.

Only actual flight tests will determine the buffet boundaries of the aircraft. However, on the basis of References 9, 10, constant $C_{L_{Buffet}}$ equal to .6 is conservatively chosen.

6.2.3 INSTALLED ENGINE DATA

6.2.3.1 Performance of Deflected Nozzle

Discussion of the deflected nozzle is detailed in paragraph 9.0. It is only noted here, that examination of outside references, and results of testing being done by Orenda Engines, show that a 95% thrust coefficient can be achieved.

A further 4% reduction inherent in the multi-engine configuration flat take-off technique is made. This allowance is for controlling the aircraft and is discussed in paragraph 7.

6.2.3.2 Performance of Fixed Conical Nose Inlet

The inlet total mass flow ratio and internal contraction ratio is of primary importance at low subsonic speeds. (References 13, 14, 15, 16 & 17.) To minimize the lip separation losses the inlet will be run at sea level static at a mass flow ratio of .64 and a contraction ratio of 15%. Such a combination gives a total pressure recovery of .955 going to .985 at .6 M found from bellmouth tests [13, 14].

The shock geometry and its attendent physical characteristics follow from the following considerations.

Region I (see sketch on Figure 6) is assumed undisturbed by the aircraft at supersonic speeds, but to have a 'lead-up loss' due to flow distortion of 1% between .6 M and 1.0 M [24]. Below .6 M this loss is overshadowed by lip separation losses.

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In the supersonic regime when the conical nose shock is attached, Region II is assumed conical and the physical characteristics are found from the theory of Taylor-Macoll [18] Because the flow is not constant in radial planes (the terminal normal shock wave will be slightly curved) an average for the flow quantities is assumed between those existing on the cone surface and immediately behind the conical shock wave [19]. These averaged quantities are then used assuming an essentially two-dimensional terminal normal shock.

It is necessary to keep the cone surface mach number less than 1.33 [20] to prevent separation of the turbulent boundary layer on the surface by the normal shock wave. This requires a cone half-angle not less than 28° at 2.0 M. Fortunately optimizing the intake on the basis of thrust less additive drag [21] results in a cone half-angle 31.1°. This gives a surface Mach Number of 1.21. An allowance of 1% pressure recovery is made for the turbulent boundary layer build-up on the conical centre-body disturbed, as it is, by the normal shock wave. The area contained within the momentum thickness of the boundary layer is approximately 1.5 sq. in. giving a total pressure loss of .5% which is conservatively doubled because of shock - boundary layer interaction. The lip boundary layer is bled through the bypass.

A further allowance due to expected distortion level at the compressor face [22, 23] is made giving the final pressure recovery prediction shown on Figure 6.

6.2.3.3 Performance of Fixed Divergent Shroud Ejector

The ejector configuration is optimized for the 2.0M condition. The data is obtained experimentally from Reference [25]. The established configuration has a diameter ratio of 1.45 and a spacing ratio of 1.06 with an optimum bypass flow of 14%. This leads to a net thrust boost of 18%. Unfortunately there will be approximately a reduction of 14% from the convergent nozzle thrust at M=1.0 since the jet would be overexpanded.

6.2.3.4 Installed Engine Performance Summary

The installed engine performance is summarized in Table 5 at sea level static and at 2.0 M at 60,000 feet.

TABLE 5

Bristol Orpheus II Installed Net Thrust

Correction	Installed Net Thrust			
	Sea Level Static		2.0M, 60,000 Feet	
	ΔF_n	F_n	ΔF_n	F_n
Bare engine (std. day)		5760		2500
Navy tropical day (89.6°F)	-460	5300	-	
Pressure Recovery				
- S.L.S., H_c/H_o = 95.5%	-400	4900		
- 2.0M, 60,000 ft., H_c/H_o = 89.8%			-310	2190
Deflection nozzle	-240	4660	-	
Controlling A/C in VTOL phase	-190	4470 *	-	
10% overspeeding	-		270	2460
Ejector nozzle	-		440	2900
Spillage drag			-320	2580

$$* \text{ Design take-off wt.} = \frac{4470 \times 4}{1.1} = 16,250 \text{ lb.}$$

1840-VTOL-1

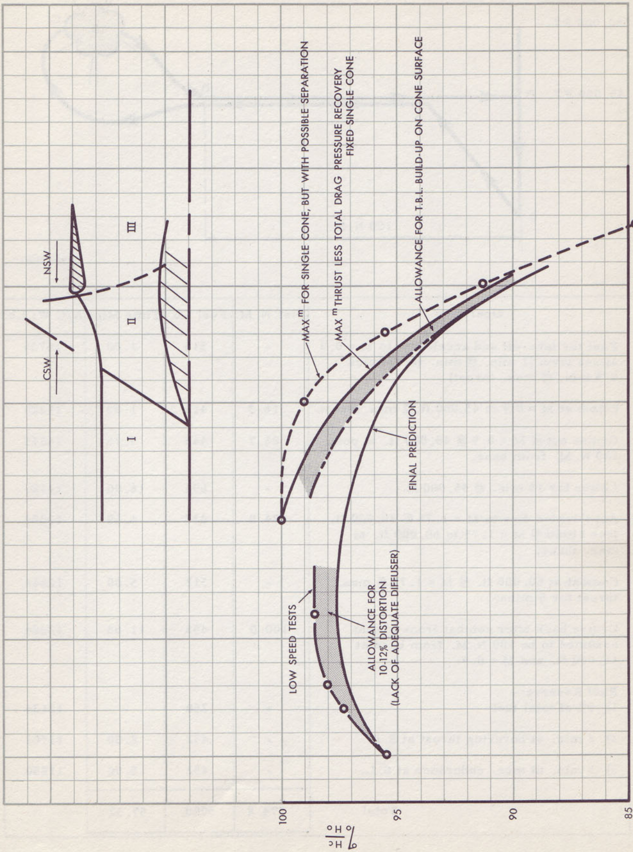

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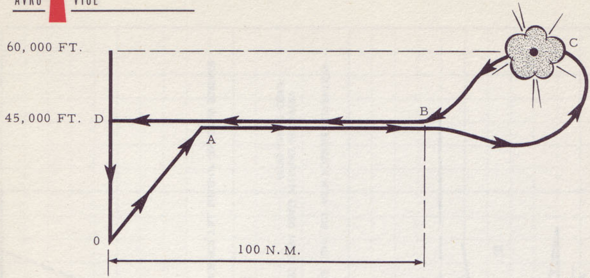
FIG. 6 31.1° SINGLE FIXED CONE PRESSURE RECOVERY

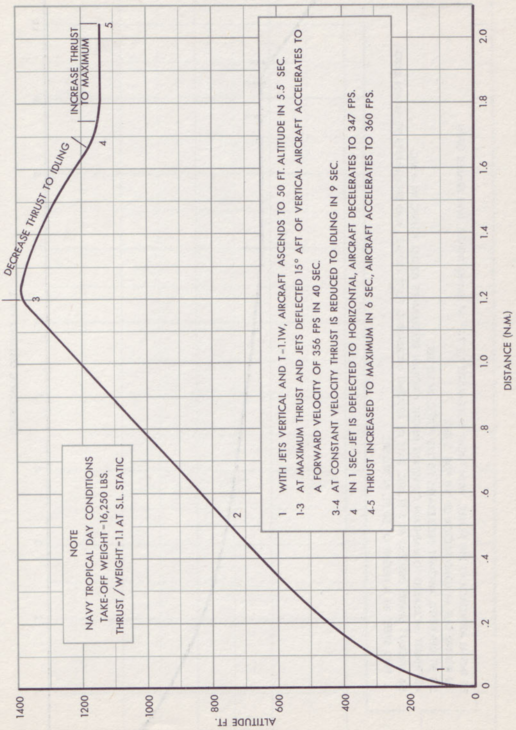


1912-VTOL-1

	Operation	Dist. N. M.	Fuel Lb.	Time Mins	A/C Wt. Lb.
O	Fuel for take-off and accelerate to M = 0. 9 at sea level (1 min, @ min. thrust plus 0. 5 min. @ max. thrust)	-	515	1. 50	15735
O-A	Climb at M = 0.9 to 45,000 ft. @ max. thrust.	16. 3	415	1. 83	15320
A-B	Cruise out @ M = 0. 9 @ 45, 000 ft. to point 100 N. M. from base.	83. 7	445	9. 75	14875
B	Loiter for 16 min. @ 45, 000 ft.	-	694	16. 00	14181
B-C	Accelerated dive to M = 1. 75 @ 36, 000 ft. then climb @ M = 1. 75 to 60, 000 ft. at max. thrust.	74. 8	827	4. 59	13354
C*	Combat at 60, 000 ft. @ M = 1. 75 @ max. thrust for 5 mins.	-	512	5. 00	12842
B-D	Cruise back after combat from a point assumed to be 100 N. M. from base at 45, 000 ft. and M = 0. 9.	100. 0	458	11. 65	12384
O	Fuel Reserve:-				
	1) 5% of total fuel	-	250	-	12134
	2) 2 min. at hovering thrust at S. L.	-	432	2. 00	11702
	3) 5 min. at max. endurance at S. L.	-	452	5. 00	11250
	Total	274. 8	5000	57. 32	

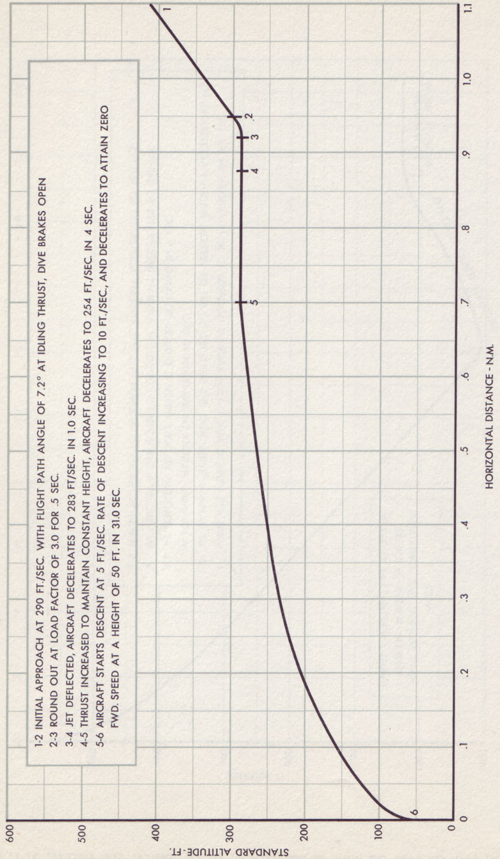
* Armament assumed to be carried throughout mission.

FIG. 7 COMBAT MISSION PROFILE



1925-VTOL-1-1

FIG. 8 FLIGHT PATH ON TAKE-OFF WITH JET DEFLECTOR 15° AFT OF VERTICAL AND AIRCRAFT AT 6° TO HORIZONTAL

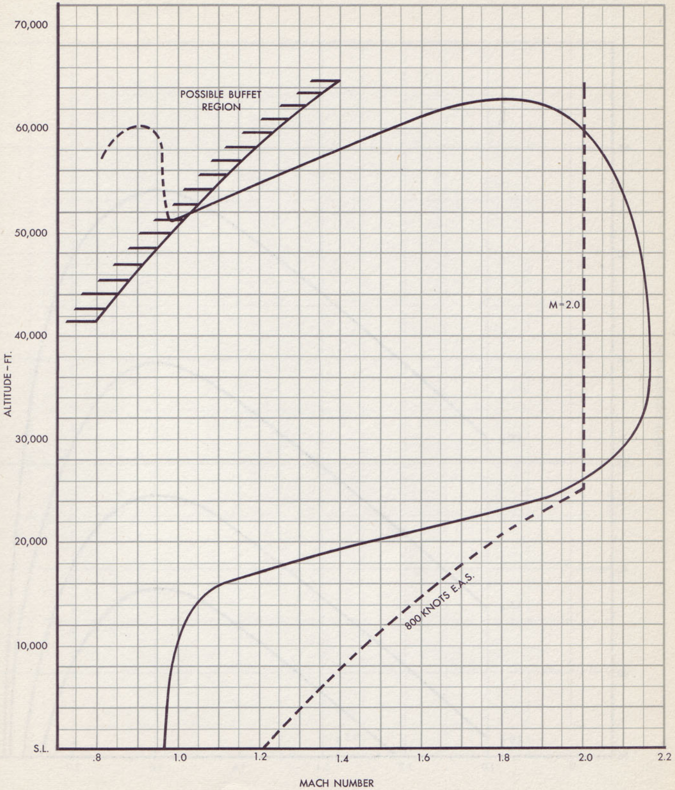


1930-VTOL-1-1

FIG. 9 FLT. PATH DURING APPR. AND LDG. WITH JET DEFLECTED 15° FWD. OF VERTICAL AND AIRCRAFT AT 6° TO HOR'L.

1840-VTOL-1

AVRO VTOL

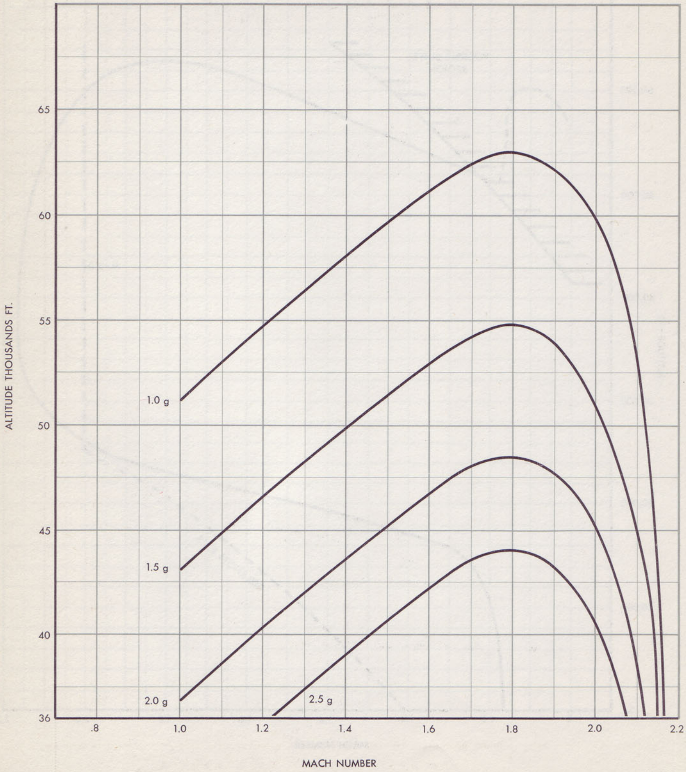


1926-VTOL-1-1

FIG.10 MAXIMUM LEVEL SPEEDS AT COMBAT WEIGHT

1828-VTOL-1

AVRO VTOL



1923-VTOL-1-1

FIG. 11 POWER LIMITED MANOUVERABILITY

1840-VTOL-1

AVRO VTOL

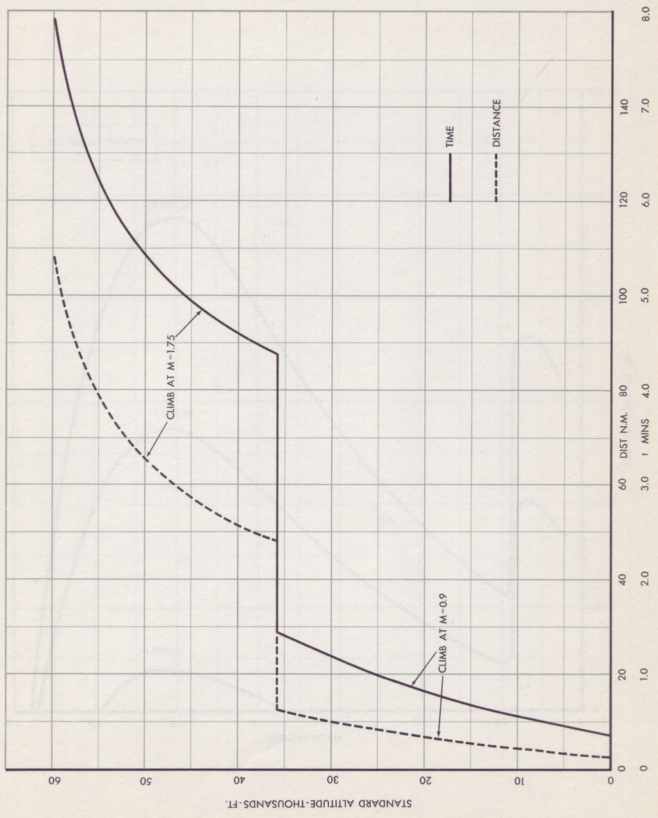


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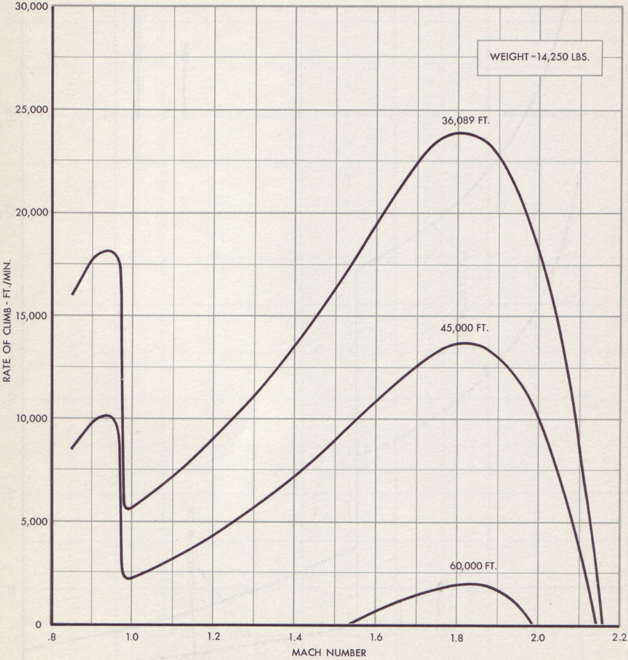
FIG. 12 TIME AND DISTANCE TO ALTITUDE FROM TAKE-OFF

AVRO VTOL



1921-VTOL-1-1

FIG. 13 RATE OF CLIMB VS CLIMB SPEED

1840-VTOL-1

6. 3. PERFORMANCE SUMMARY

TABLE 6

LOADING AND PERFORMANCE

(I. C. A. O. Standard Atmosphere except where otherwise stated)

WEIGHT:

Take-off gross weight - - -	- 16, 250 lb.
Operational weight empty - -	- 11, 250 lb.
Combat weight (40% fuel gone) -	- 14, 250 lb.
Landing weight (with reserve fuel and armament) -	- 12, 384 lb.
T/W ratio; sea level at 89. 6°F and T. O. gross weight	- 1. 1

SPEED:

Maximum level speed at combat weight:

(a) At sea level - - -	- 635 Kts.
(b) At 60, 000 ft. - - -	- 1146 Kts.

CEILING:

Combat ceiling, (rate of climb 500 ft/min), at combat weight at 1. 75 M. N. - - -	- 62, 000 ft.

RATE OF CLIMB:

Steady rate of climb at combat weight:

(a) At sea level at 0. 9 M. N. - -	- 55, 600 ft/min.
(b) At 45, 000 ft. at 1. 75 M. N. - -	- 13, 200 ft/min.

TIME TO HEIGHT:

Time to attain 60, 000 ft. and 1. 75 M. N. from T. O.	- 7. 94 mins.

ACCELERATION:

Time to accelerate in level flight at 40, 000 ft. from M = 1. 0 to M = 1. 75 at combat weight - -	- 3. 21 mins.

MANOEUVRABILITY:

Combat load factor at combat weight at 45, 000 feet and 1. 75 M. N.	2. 34

MISSION:

For a 100 N. M. combat radius of action with initial climb at 0. 9 M. N., cruise out at 0. 9 M. N. at 45, 000 feet, accelerate to 1. 75 M. N., climb to 60, 000 feet at 1. 75 M. N., combat at 60, 000 feet at 1. 75 M. N. for 5 Min., cruise back at 0. 9 M. N. at 45, 000 feet, requisite fuel reserve on landing, then loiter time over station	16 Mins.

AVRO VTOL

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1840-VTOL-1

7.0 STABILITY AND CONTROL

7.1 GENERAL

Paragraphs 7.1.1 to 7.1.3 are intended to describe, in broad outline, the essentials of stability and control on the proposed aircraft. For the more technical aspects of the problem reference should be made to paragraph 7.2 onwards.

7.1.1 HORIZONTAL FLIGHT

For horizontal flight a conventional control layout is used consisting of control column for pitch and roll, and rudder pedals for yaw control. From these points there is a direct mechancial linkage to hydraulic jacks operating the control surfaces. Engine thrust control is provided through a bank of four independent throttle levers.

7.1.2 HOVERING FLIGHT

For hovering flight and transition an automatic stabilizing system is provided. In this phase, control is obtained by monitoring, automatically, the individual engine outputs in such a manner as to automatically maintain the aircraft attitude selected by the pilot. The pilot will be able to select this attitude in pitch or roll through his conventional control column, and this control will permit the pilot to execute horizontal translation in any desired direction. Rudder pedals will provide yaw control by connecting them to a mechanism in the engine pods. This mechanism will suitably tilt the thrust vector of two or more engines with respect to airframe, thereby giving the required control. The automatic stabilizing system will provide damping of any resulting oscillation in roll, pitch, yaw, and in vertical translation, and will limit their amplitudes to a desirable level.

7.1.3 TRANSITION

Transition to or from conventional flight is obtained by tilting the vertical thrust vector by 15 degrees aft. In this way a forward acceleration component will be created and forward velocity will start to build up. When the forward velocity exceeds stalling speed by a safe margin the aerodynamic (non-automatic) controls take over, and the engine thrust is rotated to a position parallel to the aircraft datum. A switching device is provided to disengage the automatic engine controls. The reverse procedure is employed for transition to hovering flight.

Controls required for the rotation of thrust are integrated with the throttle control used in hovering flight. When the automatic system is switched off at the end of transition, thrust control is achieved through the conventional bank of individual throttle levers.

7.1.4 CONTROL SYSTEM

A block diagram of the control system is shown in Fig. 14. The details of the control system and the stability analysis are given in the subsequent paragraphs. The proposed design of the automatic control system for hovering flight, coupled with a "flat riser" configuration results in very stable behaviour of the aircraft in gusty air, during take-off or landing, and provides the pilot with very effective controls at a small expense of overall thrust.

7.2 DESIGN CONSIDERATIONS

7.2.1 CHOICE OF JET DEFLECTION

Preliminary investigation of a vertical take-off and landing aircraft indicated that there were two basic methods of supporting the aircraft by the engine thrust, viz:

1828-VTOL-1

(a) the "tail sitting" method, in which an aircraft takes off and lands with its fuselage axis vertical.

(b) the jet deflection method, in which the line of action of the engine thrust is rotated to a position normal to the fuselage axis.

The pros and cons of these two methods are discussed in Paragraph 3.2.1 where it is postulated that the flat riser is the logical choice to meet the specified requirements.

The practicability of a vertical take-off and landing aircraft will depend very largely on finding a satisfactory solution to the problems of hovering flight and transition from hovering to normal flight. An investigation indicated that these problems have simple solutions only when Method (b) is adopted. This fact, together with the advantages shown in the comparison of the two methods, led to the decision to use jet deflection to support the aircraft. Mechanical methods of deflecting the line of action of the jet thrust were compared with the use of rotating engines, but the latter method was found to be impractical. Therefore, the whole design was based on the concept of using fixed engines with a mechanical device for deflecting the jets through an angle of approximately 90° so that the aircraft could hover with its fuselage axis horizontal, control moments in pitch and roll being obtained by differential engine thrusts.

7.2.2 ENGINE LAYOUT

Satisfactory hovering performance will depend on the rate at which a change in attitude can be obtained, so that the aircraft can be returned to its original attitude after being disturbed by a gust. The equations of dynamic motion show that the largest rotational accelerations are obtained when the ratio $\frac{\text{Control Moment}}{\text{Moment of Inertia}}$ is a maximum, and as both the Moment of Inertia and the Control Moment are functions of the distance apart of the engines a criteria is obtained which permits optimum engine positions to be determined. This criteria was used to obtain the engine positions, a check showing that the resultant control moments available were many times larger than any disturbing moment due to gusts was likely to be.

7.2.3 HOVERING CONTROL

To avoid a complicated control system in hovering flight it was decided that all manoeuvring accelerations in a horizontal plane would be obtained by pitching and/or rolling the aircraft so that a resolved component of its thrust would provide the necessary force. Pitch and roll control moments are provided by differential variation of the engine thrusts, but as this gives no control moment about the yaw axis a method of deflecting the jets $\pm 5^{\circ}$ laterally from the vertical has been provided. Vertical velocity is controlled by varying the total engine thrust.

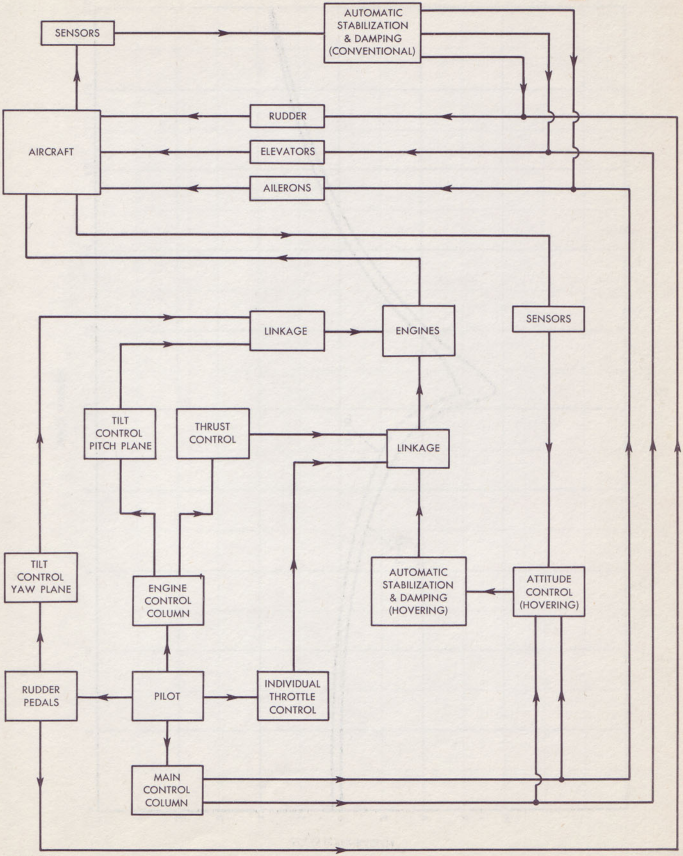
The above controls are satisfactory for manoeuvring, but do not provide sufficient longitudinal force for accelerating after take-off and decelerating before landing. Therefore, provision is made to deflect the jets to fixed positions $\pm 15^{\circ}$ from the vertical position.

7.2.4 AERODYNAMIC CONSIDERATIONS

The aircraft was designed around the engines, the prime consideration being that the centre of gravity of the fully loaded aircraft would fall midway between the lines of action of the deflected jets of the forward and aft engines, so that maximum thrust on all engines would be available for take-off. Planform and wing sections, etc., were determined by the necessity to have a statically stable aircraft which will cruise subsonically and yet be capable of M = 2 for short periods.

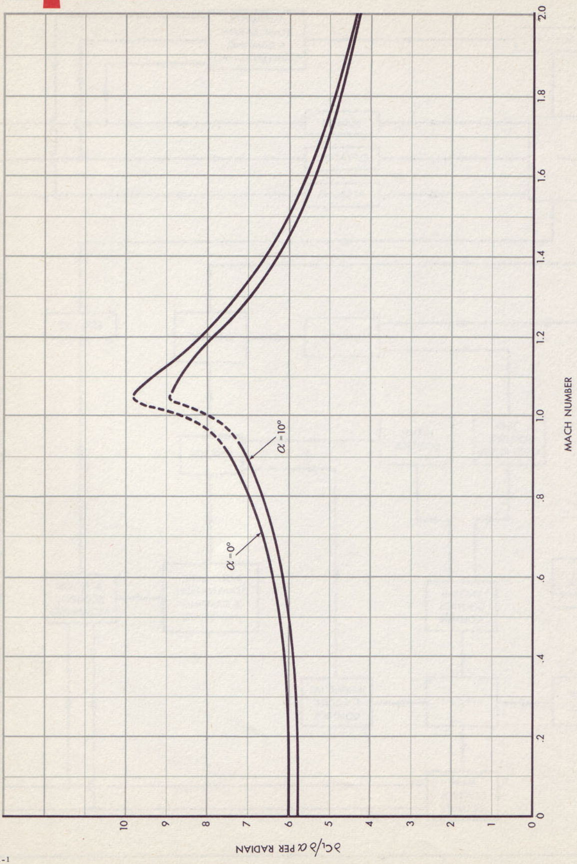
The use of a canard configuration was dictated by static stability considerations. This meant a compromise in the lateral engine positions, the front engines moving inboard and the rear

AVRO VTOL



1878-VTOL-1

FIG. 14 SCHEMATIC DIAGRAM OF CONTROL SYSTEM



1866-VTOL-1-1

FIG. 15 VARIATION OF THE LIFT CURVE SLOPE WITH MACH NUMBER

engines moving outboard so that there would be little change in the ratio $\frac{\text{Control Moment}}{\text{Moment of Inertia}}$ in roll.

7.3 AERODYNAMIC DERIVATIVES

7.3.1 GENERAL

The aerodynamic derivatives are presented in Figs. 15 to 29. They were estimated using references 1 to 15 and derived methods based on these references. All induced effects between the nose plane and the rear wing were based on the method of reference 1. Compressibility effects, when not otherwise available, were obtained by the use of the subsonic and supersonic similarity laws.

The derivatives were based on a rigid aircraft structure. To permit the first order effects of flexibility to be determined control reversal speeds for the elevators and ailerons are given in Figs. 19 and 23. No estimate has been made of the effects of fuselage flexibility on the longitudinal and lateral derivatives.

All non-dimensional aerodynamic data is based on the following reference area and lengths:-

$S = 145 \text{ ft.}^2$

$\bar{c} = 7.82 \text{ ft.}$

$b = 18.67 \text{ ft.}$

7.3.2 LONGITUDINAL DERIVATIVES

The longitudinal derivatives are presented in Figs. 15 to 18 and Figs. 20 and 21. Wing flexibility effects on the controls are given in Figs. 19 and 23, and are introduced through the relationship

$$C_\delta \text{ (elastic)} = C_\delta \text{ (rigid)} \left[1 - \left(\frac{V}{V_r}\right)^2\right]$$

The lift curve slope (Fig. 15) and aerodynamic centre position (Fig. 16) were derived using References 1 and 2. The lift curve slope is larger than normally found on aircraft with a similar aspect ratio; this is because the nose plane lift is additive to that of the main wing. The aerodynamic centre movement with Mach Number is gradual throughout the transonic region and should give a smooth subsonic - supersonic transition in flight.

Lift and pitching power due to the nose plane controls (elevators) and the main plane controls (elevons) are given in Figs. 17, 18, 20, and 21. These derivatives were obtained using references 3 to 6, induced effects between the nose and main plane being taken from reference 1.

Control reversal speeds (Figs. 19 and 23) were estimated from references 7 and 8, the methods being modified at supersonic speeds to conform to the change in pressure distribution over the wings.

7.3.3 LATERAL DERIVATIVES

The lateral derivatires are presented in Figs. 22 and 24 to 27. Aileron reversal speeds are given in Fig. 23, and are used in a similar manner to the longitudinal control reversal speeds (Section 7.2.2).

No estimate has so far been made of the rudder reversal speed, but it is thought that a reasonable assumption would be to use the symmetrical deflection curves of Fig 23 as a preliminary value.

The aileron rolling moment (Fig. 22) and rudder yawing moments (Fig. 24) were estimated using references 3 to 6.

The sideslip derivatives C_{Y_β} (Fig. 25), C_{n_β} (Fig. 12), and C_{ℓ_β} (Fig. 27) were calculated using references 9 to 13 for basic data and the nose-plane-wing interference effects of reference 1. Compressibility effects in the subsonic region were obtained from similarity laws.

7.3.4 DAMPING DERIVATIVES

The only damping derivatives estimated are C_{m_q} (Fig. 28) and C_{ℓ_p} (Fig. 29). These derivatives were based on references 9 and 14 with interference effects from reference 1.

7.4 FLIGHT PERFORMANCE IN NORMAL FLIGHT

7.4.1 GENERAL

The flight performance curves are based on a representative weight of 14,000 lb. The results are derived from the information in Figs. 15 to 29 and do not make any allowance for fuselage flexibility. Allowance has been made for wing flexibility by the control reversal curves. All results are estimated for four heights, viz: Sea Level, 20,000 ft., 40,000 ft., and 60,000 ft.

7.4.2 LONGITUDINAL PERFORMANCE

The longitudinal performance curves are given as elevator angle to trim, elevator angle per 'g', elevon angle per 'g' and maximum normal acceleration coefficients available throughout the flight envelope.

Analysis indicated a value of C_{m_0} =-0.0084 when M = 0. This will be reduced by compressibility but for analytical purposes has been assumed to be constant at all Mach Numbers. This assumption makes the longitudinal results slightly conservative.

The elevator angle to trim curve (Fig. 30) indicates that control movements are free from discontinuities and, prior to the compressibility rise, in the stable sense. The change in sign of $\frac{\partial \delta e}{\partial M}$ at the beginning of the transonic region is normal to all aircraft and, therefore, must be accepted. The large elevator angles required between M = 1.0 and M = 1.2 at sea level are largely due to the nose plane torsional flexibility in this region, the flight envelope approaching the aileron reversal speeds. It is thought that this fact is not important in flight, because there is ample control available to attain the design envelope values of normal acceleration in this region. Should the slope of this curve be thought undesirable it can be reduced by increasing the nose plane torsional stiffness.

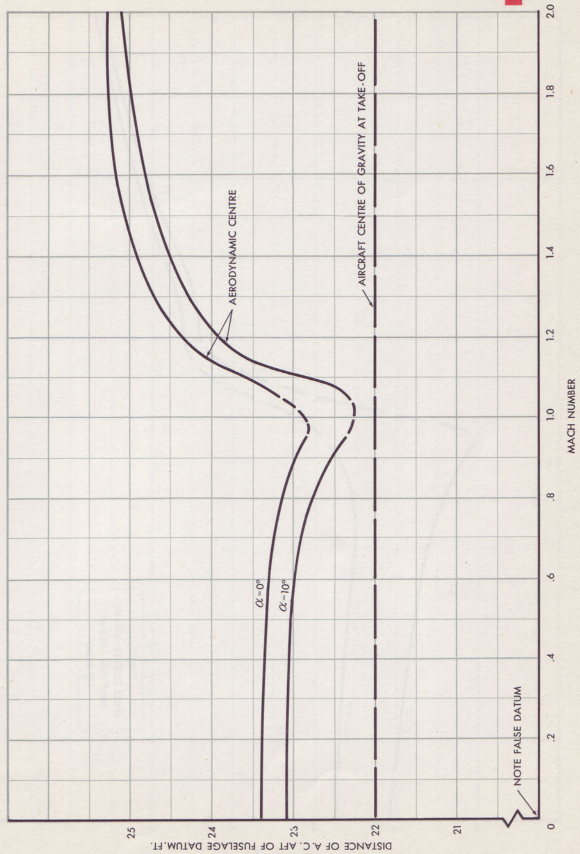
The elevator angle per 'g' (Fig. 31) and elevon angle per 'g' (Fig. 32) curves indicate the control power available in manoeuvres. These curves have been used to derive the available maximum normal acceleration coefficient curves of Fig. 33 which indicate that the only appreciable limitation to the flight envelope is at 60,000 ft. Fig. 33 has been based on the assumptions that:-

(a) $C_{L_{max.}}$ = 1.10

(b) Buffeting Effects may be neglected.

(c) There is no hinge moment limitation.

1840-VTOL-1



1865-VTOL-1-1

FIG. 16 MOVEMENT OF THE AERODYNAMIC CENTRE WITH MACH NUMBER

AVRO VTOL

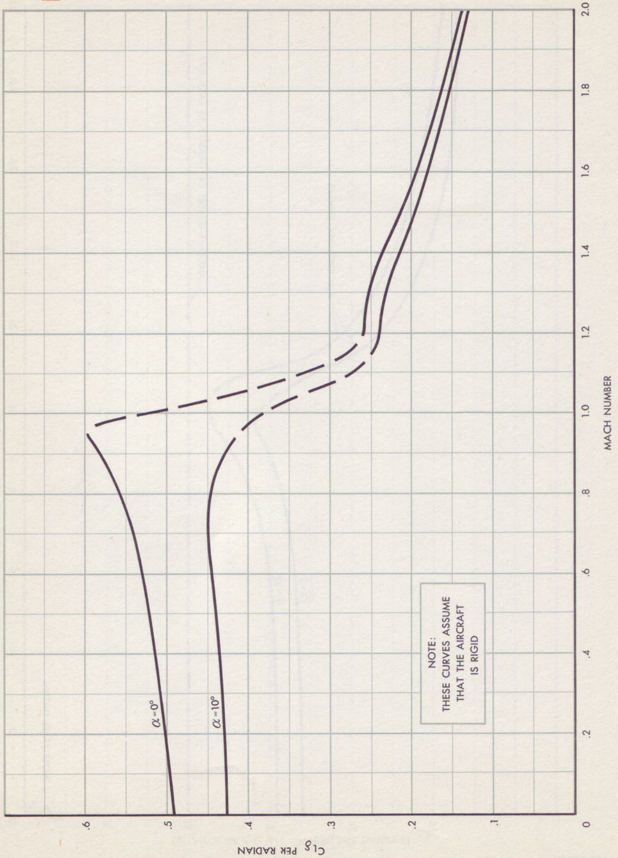


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FIG. 17 ELEVATOR LIFTING POWER $C_{L_{\delta_e}}$ VS. M

AVRO VTOL

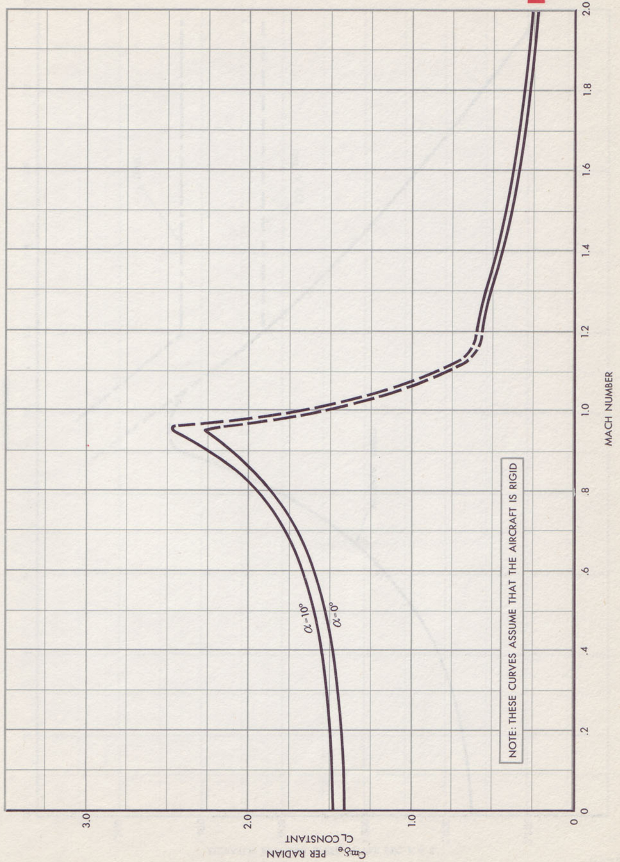

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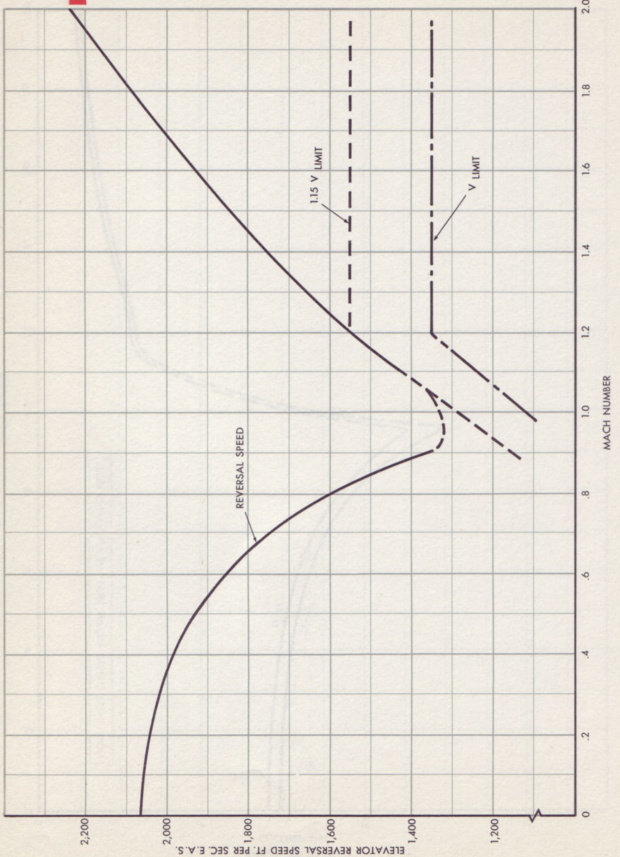
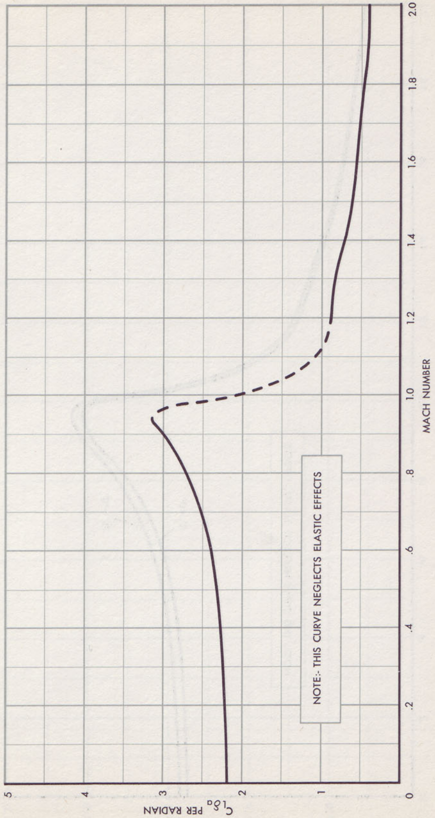
FIG. 18 ELEVATOR PITCHING POWER $C_{m\delta_e}$ VS. M



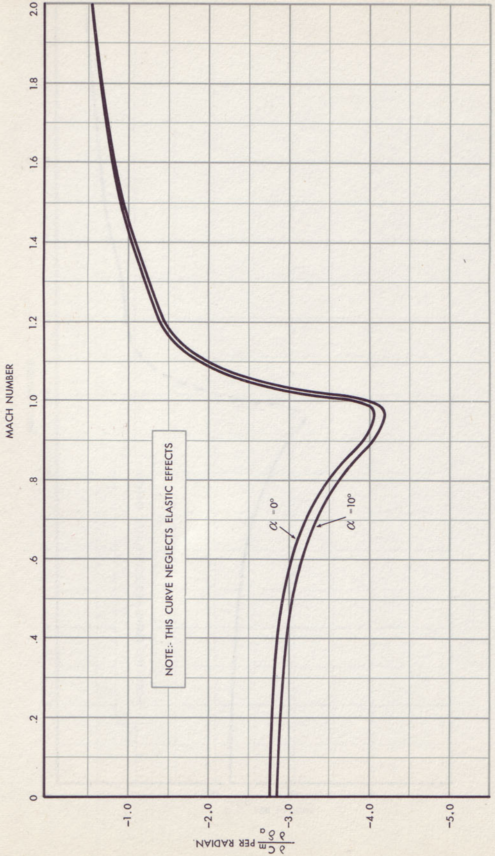
1874-VTOL-1

FIG. 19 ELEVATOR REVERSAL SPEED V_R (E.A.S.) VS. M.



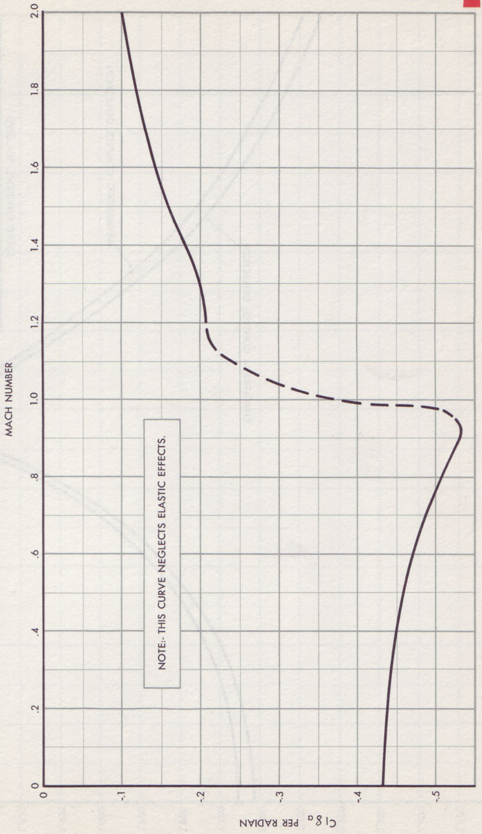
1887-VTOL-1-1

FIG. 20 EFFECT OF SYMMETRIC DEFLECTION OF THE MAIN WING CONTROL SURFACES ON THE LIFT $C_{L_{\delta_a}}$ VS. M.



1894-VTOL-1-1

FIG. 21 PITCHING POWER OF THE MAIN WING CONTROL SURFACES DEFLECTED SYMMETRICALLY $C_{m\delta_a}$ VS. M



1886-VTOL-1-1

FIG. 22 ROLLING MOMENT DUE TO ASYMMETRICAL DEFLECTION OF THE MAIN WING CONTROL SURFACES $C_{l\delta_a}$ VS. M

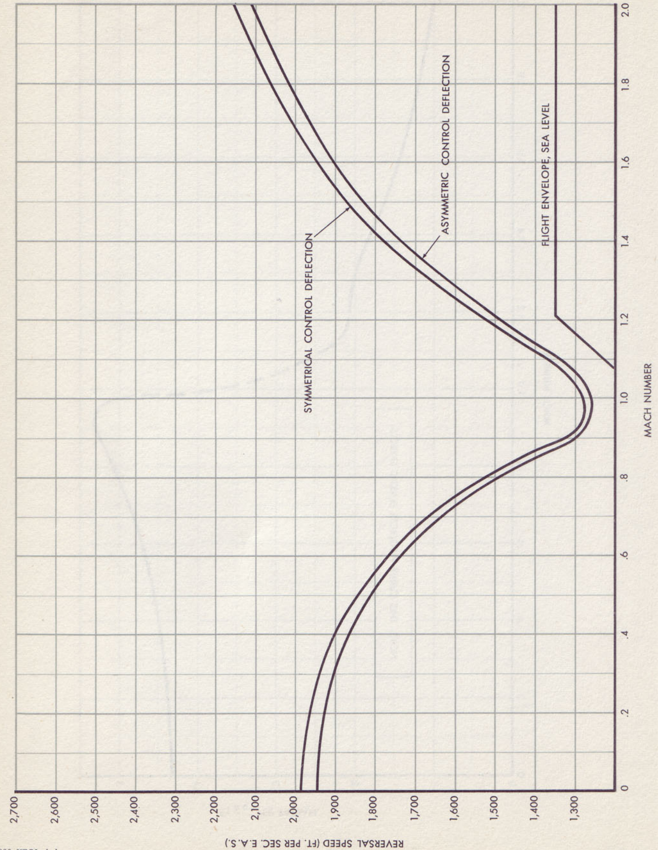


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FIG. 23 REVERSAL SPEEDS FOR CONTROLS ON THE MAIN WING V_R (E.A.S.) VS. M.

AVRO VTOL

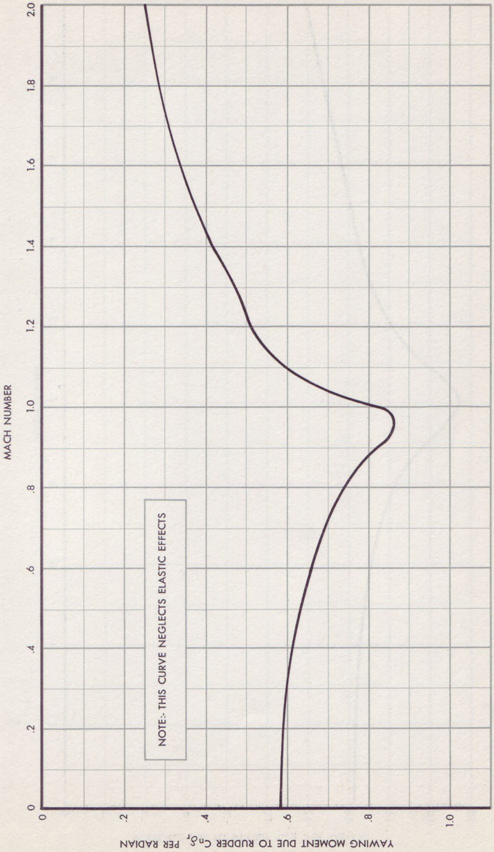

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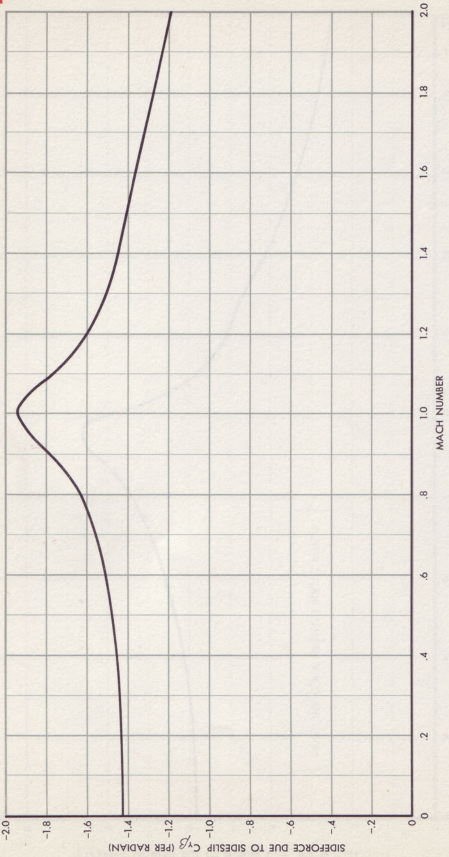
FIG. 24 YAWING MOMENT DUE TO RUDDER $C_{n\delta_r}$ VS. M

AVRO VTOL



1900-VTOL-1-1

FIG. 25 SIDE FORCE DUE TO SIDESLIP C_{Y_β} VS. M (PER RADIAN)

1840-VTOL-1

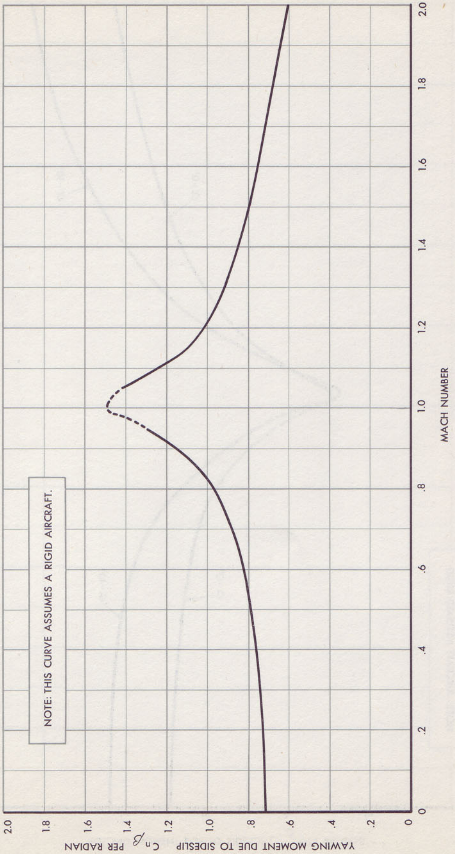


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FIG. 26 YAWING MOMENT DUE TO SIDESLIP C_{n_β} PER RADIAN

AVRO VTOL

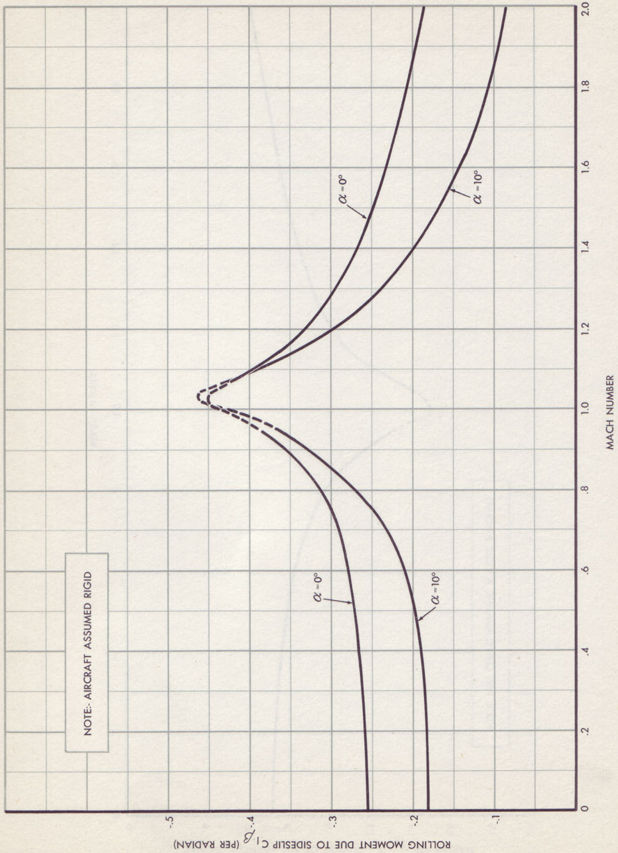

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FIG. 27 ROLLING MOMENT DUE TO SIDESLIP C_{l_β} VS MACH NUMBER

1840-VTOL-1

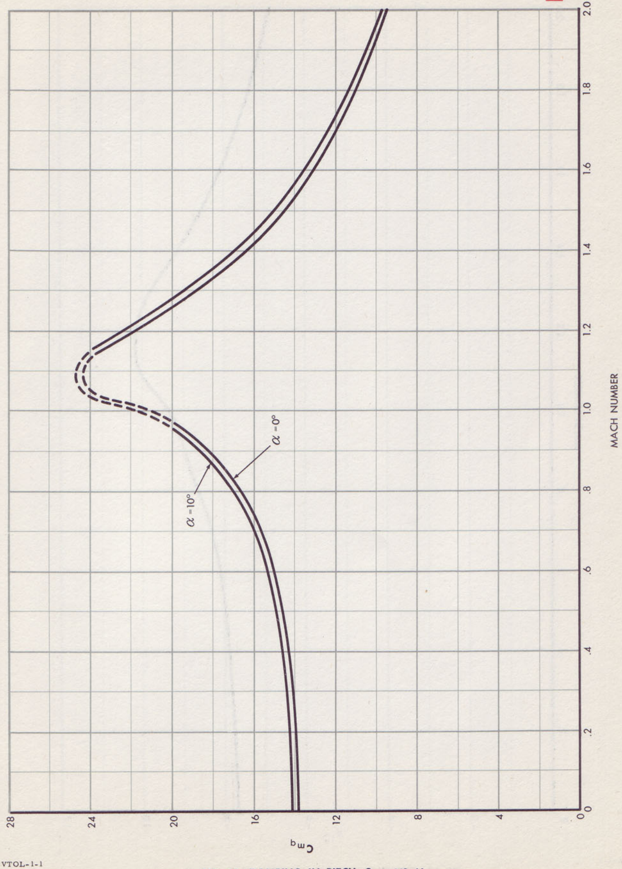

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FIG. 28 DAMPING IN PITCH C_{m_q} VS. M.

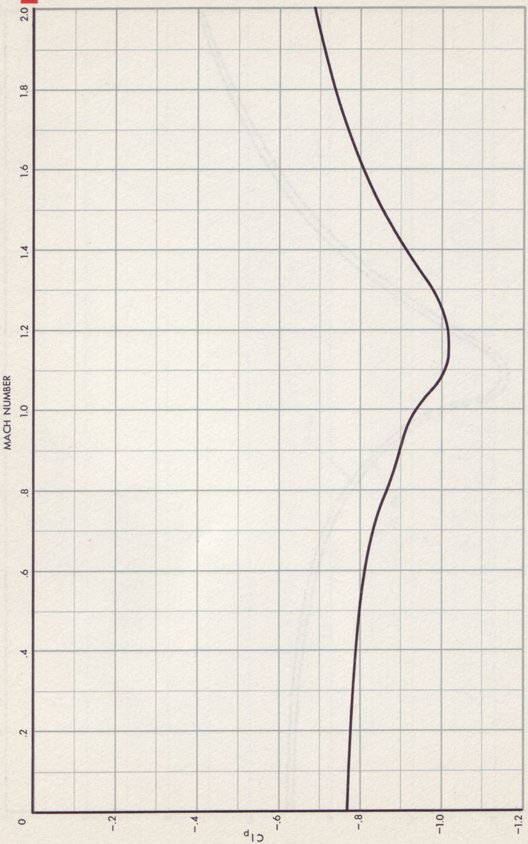

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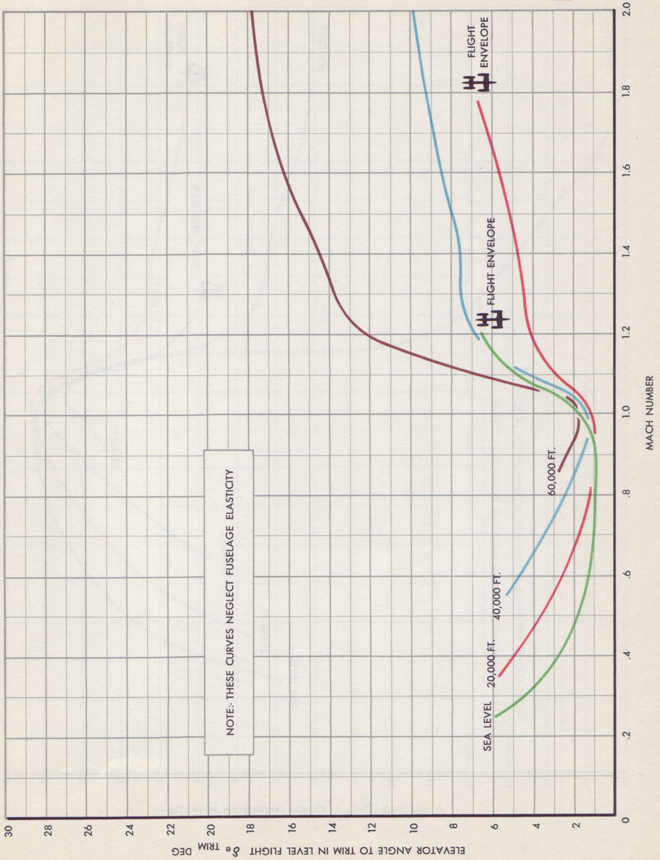
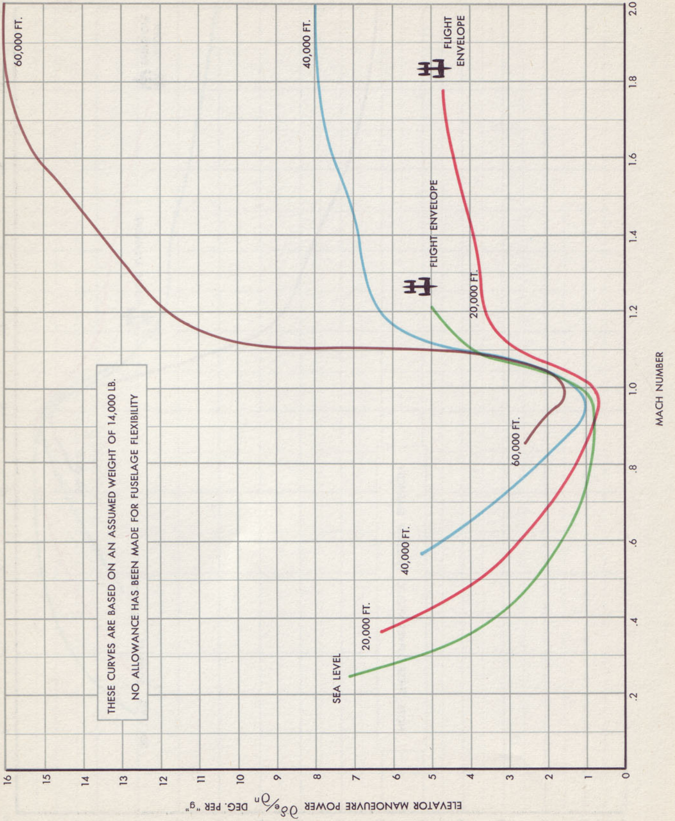
FIG. 29 DAMPING IN ROLL COEFFICIENT C_{l_p} VS. M.

AVRO VTOL



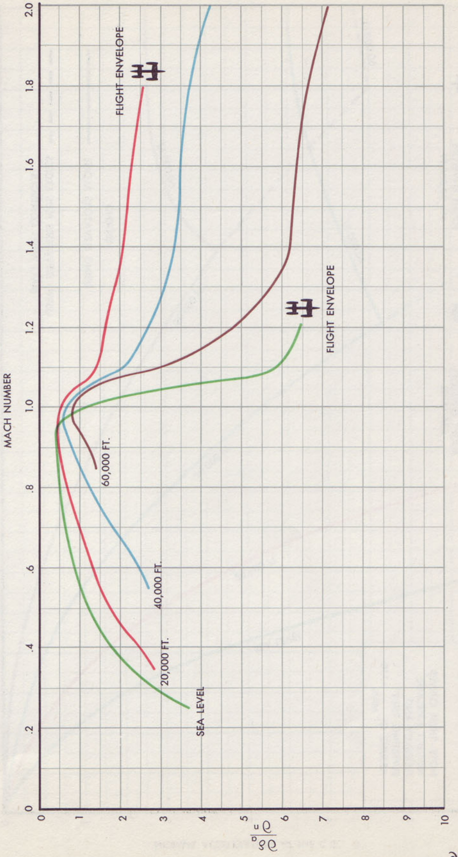
1896-VTOL-1-1

FIG. 30 ELEVATOR ANGLE TO TRIM IN LEVEL FLIGHT $\bar{n} = 1.0$ $\delta_{e_{TRIM}}$ VS. M.



1897-VTOL-1

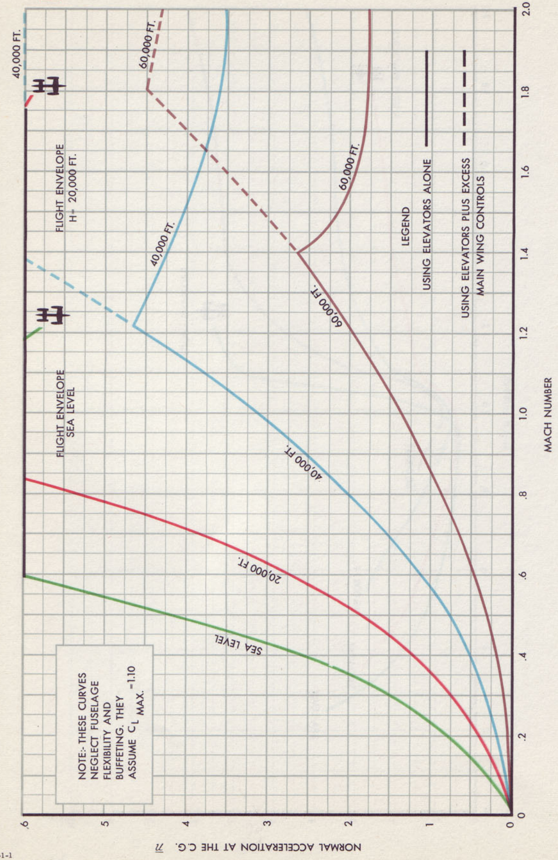
FIG. 31 MANOEUVRE POWER OF THE ELEVATOR $\partial\delta_e/\partial n$ VS. M



1891-VTOL-1-1

FIG. 32 MANEOUVRE POWER OF MAIN WING CONTROLS WHEN DEFLECTED SYMMETRICALLY $\frac{\partial \delta_a}{\partial n}$ VS. M

AVRO VTOL



1892-VTOL-1-1

FIG. 33 AVAILABLE NORMAL ACCELERATION AT THE C. G. $\bar{n}$ VS. M.

1840-VTOL-1

AVRO VTOL

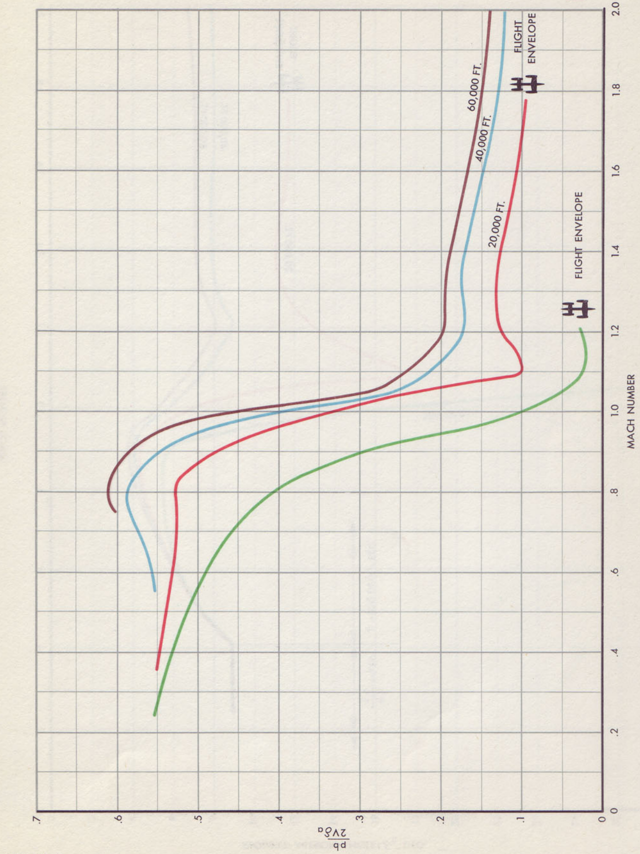


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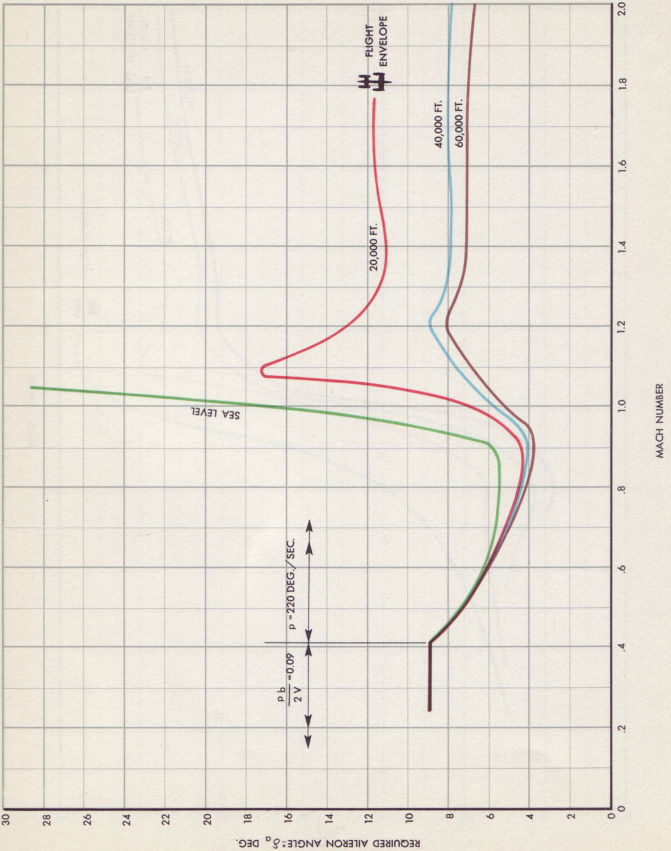
FIG. 34 AILERON ROLLING PERFORMANCE $\frac{pb}{2V\delta_a}$ VS. M

AVRO VTOL



1899-VTOL-1-1

FIG. 35 AILERON ANGLE TO SATISFY ROLLING REQUIREMENTS

(d) Control is by elevator application, but if this is insufficient, symmetrical application of the ailerons (elevons) may be used. Elevon application is restricted to the amounts not required for roll at the design condition.

The curves indicate that at supersonic flight above 20,000 ft., elevon application may be required to assist the elevators in manoeuvres.

Fuselage flexibility in general will tend to reduce the manoeuvre margins, and therefore, the predicted attainable normal accelerations may be expected to be conservative. They may, however, be restricted by buffeting in the transonic and supersonic speed ranges, although the wing planforms and sections chosen are expected to keep this to a minimum.

7.4.3 LATERAL PERFORMANCE

Fig. 34 presents curves of the non-dimensional rolling power of the ailerons, $\frac{pb}{2V\delta_a}$. This figure has been used to derive the aileron angle requirements to satisfy the specification performance, the results being plotted in Fig. 35.

The results indicate that apart from supersonic conditions at sea level, the aircraft has an excess of aileron power. The shortage at sea level is due entirely to the wing torsional flexibility, so will be rectified very quickly as height is increased. It is not known how important a 220°/second rate of roll is at these low altitudes at supersonic speeds, but should it be required, it can be obtained by increasing the main wing torsional stiffness.

7.5 STABILIZATION SYSTEM FOR HOVERING FLIGHT

7.5.1 GENERAL

One of the major problems in the design of V.T.O.L. aircraft has arisen through the need for positive control of the aircraft during hovering, and during transition from hovering to normal flight. This control is achieved by setting the following design objectives:

(a) The provision of positive control of the aircraft, particularly during the initial and terminal sections of the flight, by suitably combining the aircraft geometry and a damping system which will maintain the aircraft in commanded attitude, in the face of severe disturbances present during operational use of the aircraft.

(b) To provide a control system using conventional controls in a manner which reduces the demands made on the pilot, and makes use of his natural skill. This reduces the pilot training period and tends to minimize pilot errors.

7.5.2 NEED FOR AUTOMATIC STABILIZATION

During hovering flight the aircraft has no inherent stability and the only control forces consist of moments and forces produced by the engines. Conventional aircraft controls provide rolling rate proportional to stick force in the lateral direction and normal acceleration proportional to stick force in the longitudinal direction. For controlling a V.T.O.L. aircraft the pilot could be provided with stick control of pitch and roll by suitable connections to the engines, and yaw control by connecting the rudder bars to the engines.

As is shown later in the root loci plots, an unstable loop is created, when the pitch loop is closed through a feedback device indicating position.

The roll axis, the yaw axis, and the z axis (height plane), all exhibit the same characteristics. This unstable condition is common to all force displacement type systems having zero damping. This is the essential feature to be contended with by any stabilizing system for a V.T.O.L.

aircraft. Although neutrally unstable systems can be controlled manually under some circumstances, the accuracy with which the pitch and roll axis must be controlled, dictates an automatic control system, particularly when the system must be manoeuvred in a gusty environment. For this reason, automatic control of pitch and roll is considered essential and automatic control of height desirable.

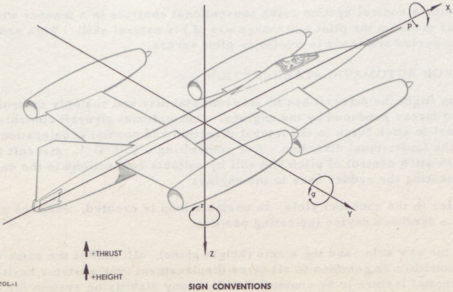
7.5.3 TYPE OF COMMAND SIGNALS

The control system investigated is one which automatically maintains constant attitude in pitch and roll and constant rate of ascent and descent as commanded by the pilot, in spite of large gusts and disturbances caused by ground effects on the engines. Stick forces and deflections, proportional to command attitudes in pitch and roll, were chosen as the command signals. A desirable method of providing height controls seems to be to use an engine control column. The engine control column would control thrust tilt and total thrust or rate of change of altitude on the same column. An investigation should be conducted to determine the best combination of motions. When the control is returned to zero the aircraft will maintain constant altitude when an automatic height control is included.

7.5.4 BASIC SYSTEM

Two basic control schemes were considered to achieve the stability required. The first method proposed, used the nose engines to control pitch, and the main plane engines to control roll attitudes, with a throttle control used to control all engines simultaneously for altitude commands. The second scheme was based on the use of all engines to control the pitch and roll attitudes, and all engines to control the height. Since the first method required greater changes in engine thrust per engine to control a given disturbance, the second method is used. Fig. 36 shows the "Schematic Diagram for Stabilization System". This diagram shows that for positive pitch commands the thrust of the nose engines is increased and the thrust of the main plane engines is decreased. A positive roll command reduces thrust on the starboard engines and increases thrust on the port engines. A positive rate of change of height command increases thrust on all engines. The resultant commanded thrust on the nose starboard engine is the sum of the pitch and throttle commands minus the roll command. The vertical gyro provides position sensing from the roll and pitch gimbels, and the position indicator provides height information. The rate gyros and the integrating accelerometer provide rate damping signals.

7.5.5 STABILITY CONSIDERATIONS



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SIGN CONVENTIONS

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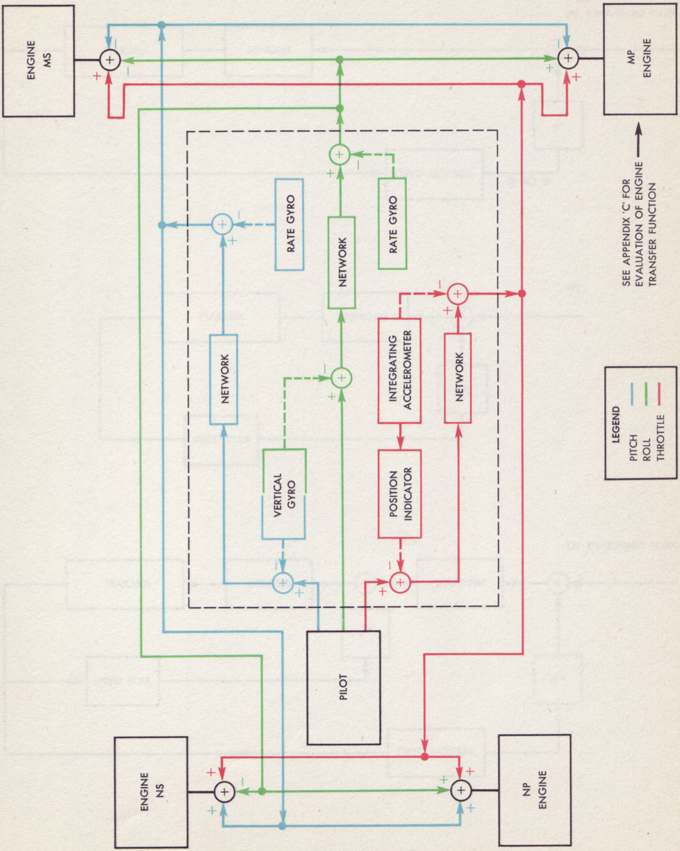

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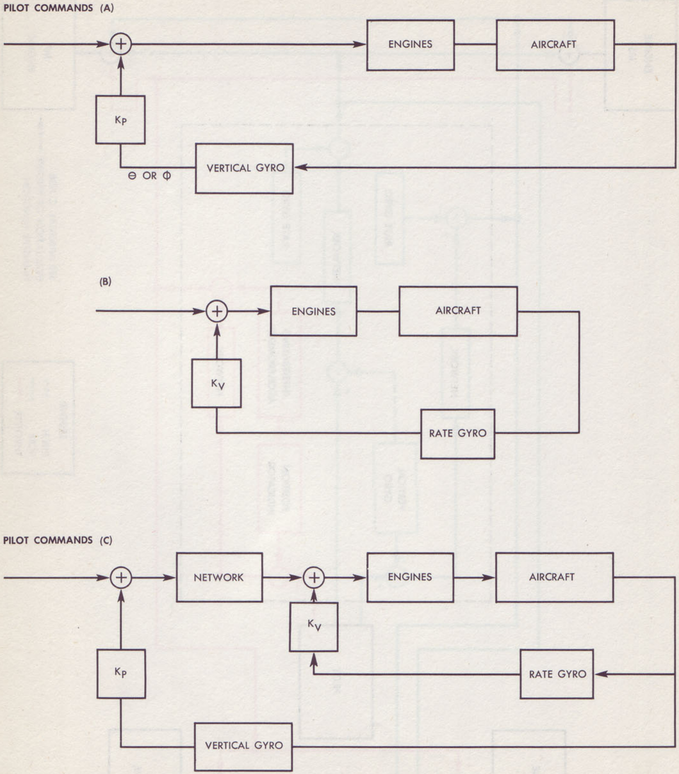
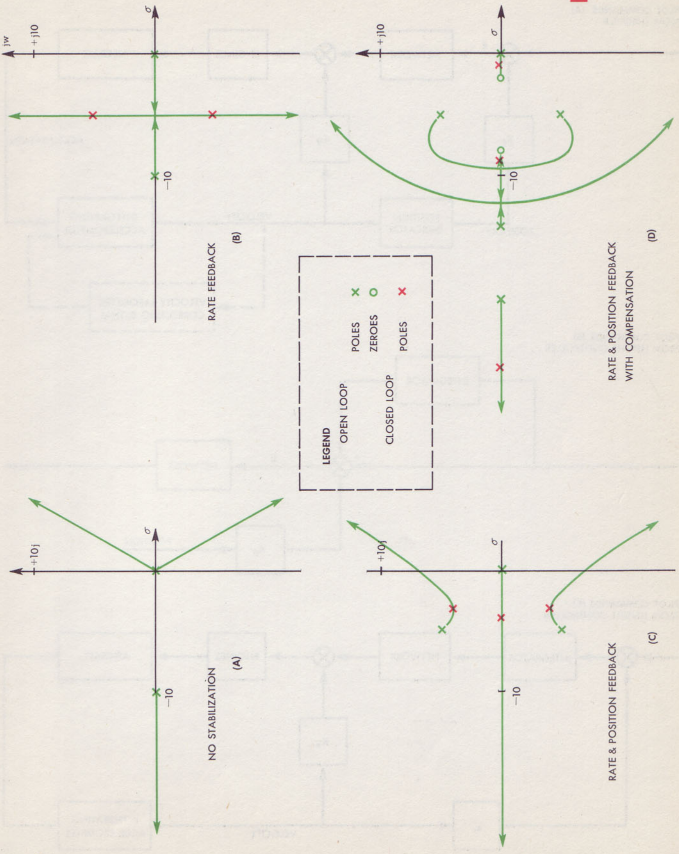
FIG. 36 STABILIZATION SYSTEM SCHEMATIC DIAGRAM



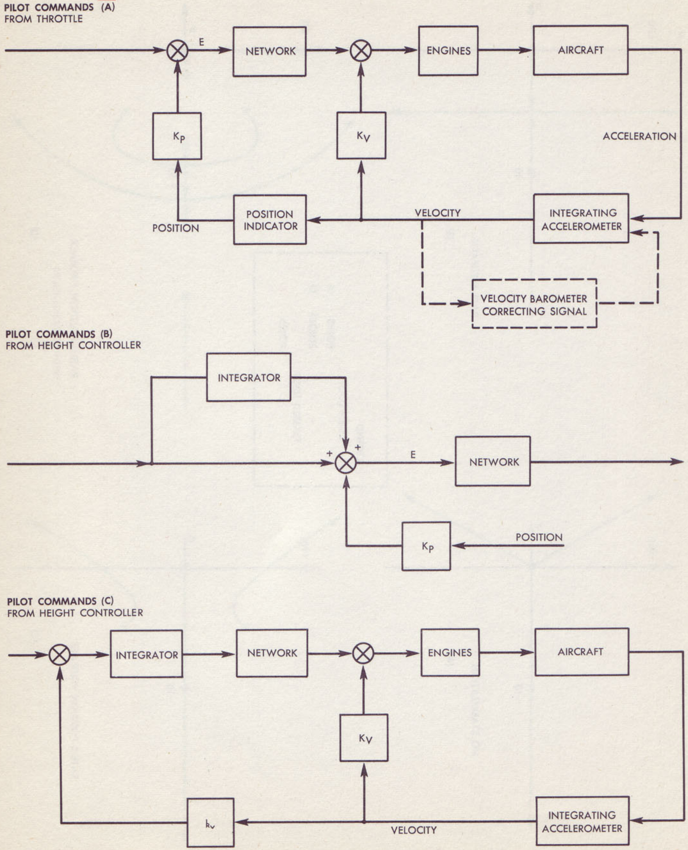
1908-VTOL-1

FIG. 37 PITCH AND ROLL STABILIZATION



1910-VTOL-1

FIG. 38 ROOT LOCI PLOTS OF THE STABILIZATION SYSTEM FOR HOVERING FLIGHT



1909-VTOL-1

FIG. 39 Z AXIS STABILIZATION

1840-VTOL-1

The diagram shows the conventions adopted for the investigations of providing automatic stabilization. The basic property of the aircraft, which determines the stabilization system required, is that the aircraft is neutrally stable. The geometric considerations for stability have been discussed in another section, however, the essential features that affect the control system are that: (a) the resultant thrust vector passes through the centre of gravity so that undesirable torques are not produced;and

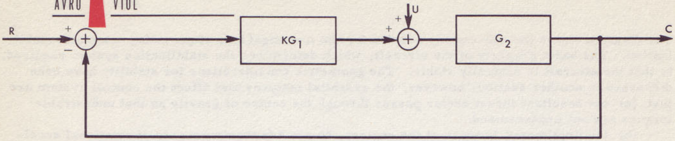
(b) the "optimum" location of the engines, to provide maximum rate of rotational acceleration from thrust available for control, has been accomplished.

When the geometry of the aircraft was established, the stabilization system was developed. The system provides damped positional control of pitch and roll, which is considered essential, and for the system investigated, height control was provided assuming the use of an integrating accelerometer. The method of obtaining velocity and positional information requires further investigation.

The "Pitch and Roll Stabilization Diagram", together with the "Root Loci Plots for Stabilization System for Hovering Flight", describe the stabilization system. These are simplified root loci plots showing first order effects only. However, for small displacements these simplifications provide realistic data without obscuring the essential features with second order effects. Fig. 37 (a) of the block diagram shows the engine, aircraft and vertical gyro loop closed to give a simple positional control device, and Fig. 38 (a) of the root loci plots shows that this system is inherently unstable, because the roots move directly into the right hand plane. This indicates the need for artificial damping because of the neutrally stable aircraft. Fig. 38 (b) shows the addition of a damping signal and the inner loop stability. A value of loop gain which gives a damping factor of $\zeta = .707$ fixes the closed loop poles. Fig. 38 (c) of the root loci plots shows the overall system stability with inner loop damping and no compensation network. The plot shows that for a useful range of gains the roots remain in the left hand plane and, therefore, the system is stable. This produces a very simple yet very effective stabilizing system. Fig. 38 (d) of the root loci plots indicates that with an appropriate compensation network a closed loop gain of approximately fifteen times that feasible for the simple system without compensation can be obtained. This increased loop gain would result in faster response and smaller errors at the expense of greater commanded thrust variations. The simple system is considered more than adequate for the stabilization required and,at the same time minimizes the demands made on the engines. Fig. 39 shows the "Z axis Stabilization". Fig. 39 (a) of this diagram shows that this loop has the same stability features as the other two loops. The three loops (pitch, roll and throttle) can all be adjusted to give similar types of response by the appropriate selection of system gains. Fig. 39 (a) shows the loop required for automatic height control and requires commanded height signals. Fig. 39 (b) shows one possible way that the throttle commands could be modified to provide a system that changes height at a given rate, as commanded by the pilot, and maintains a given height with disturbances. Fig. 39 (c) shows an alternative method of obtaining a position stabilized system which commands rates. The stability considerations of this configuration are very similar to those of the system previously discussed and are such that the root loci plots are directly applicable.

An additional modification to this system that has been considered, and should be evaluated, is the possibility of incorporating a velocity barometer to provide long term corrections to the integrating accelerometer. This may be required to compensate for vertical velocities existing during the transitions. This correcting signal could be applied to the system through a long time constant,in such a way as to slowly change the velocity feedback signal at a rate proportional to the error between integrating accelerometer velocity output,and the velocity barometer output.

Investigations of the proposed stabilization system were conducted on an analogue computer. The system simulated was that shown in the schematic diagram but without the compensation networks. In order to discuss the system response the following terms are defined:-



R = REFERENCE INPUT

C = CONTROLLED VARIABLE

U = DISTURBANCE

SERVO RESPONSE = C/R

REGULATOR RESPONSE = C/U

1905-VTOL-1

SIMULATED STABILIZATION SYSTEM

R - Reference Input
C - Controlled Variable
U - Disturbance
Servo Response = $\frac{C}{R}$
Regulator Response = $\frac{C}{U}$

Two analogue graphs are included, see Figs. 40 and 41. The first one is entitled "Response of Aircraft to Short and Long Term Disturbances". This record shows the system regulator response. The records are in sequence across the page. The left hand record shows a disturbance applied in the pitch phase only, beginning with a long term disturbance, then a short term disturbance. The fifteen hundred ft. lb. disturbance was applied as a torque to the aircraft. This is a very severe case corresponding to the moment produced by a 40 knot gust on the nose plane at stall incidence with the mainplane shielded. It can only occur if the aircraft is taking off near the nose of a carrier, and the noseplane is subjected to a vertical gust coming over the bow. This record shows that the disturbance in pitch causes a steady state error of approximately 0.3 degrees. In other words, the change in pitch attitude is negligible for a forty knot gust. When the disturbance is removed, the aircraft returns to its original pitch attitude. The second record shows a similar disturbance applied in the roll axis plane. The roll disturbance produces a somewhat larger steady state error of approximately 0.6 degrees, which is also negligible. The third record shows a disturbance applied in the ZX plane. This disturbance consists of a one hundred and fifty pound force tending to increase the height. This force is equivalent to a 40 knot gust acting on the complete aircraft at take-off incidence. The disturbance produces only approximately one half of one foot change in height. The fourth record shows the regulator response of the system when the output of one engine, the nose port engine, is increased by one hundred and fifty pounds or 3% change in engine thrust.

The second analogue graph "Response to Aircraft to Commands" shows the servo response characteristics. The first record on the left hand side of the page shows the response to a pitch attitude command. The second record shows the response to a roll attitude command. The third record shows the response to a pitch attitude and roll attitude command applied simultaneously.

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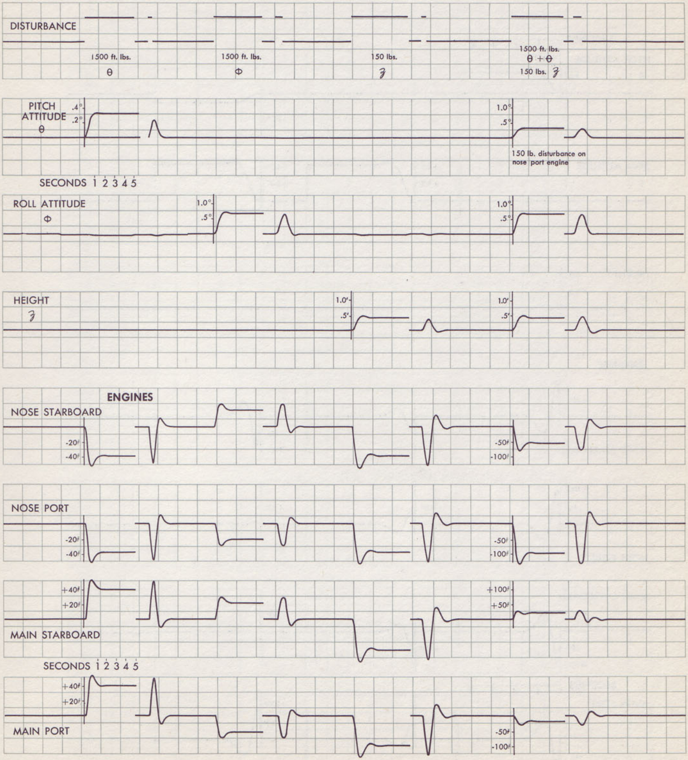
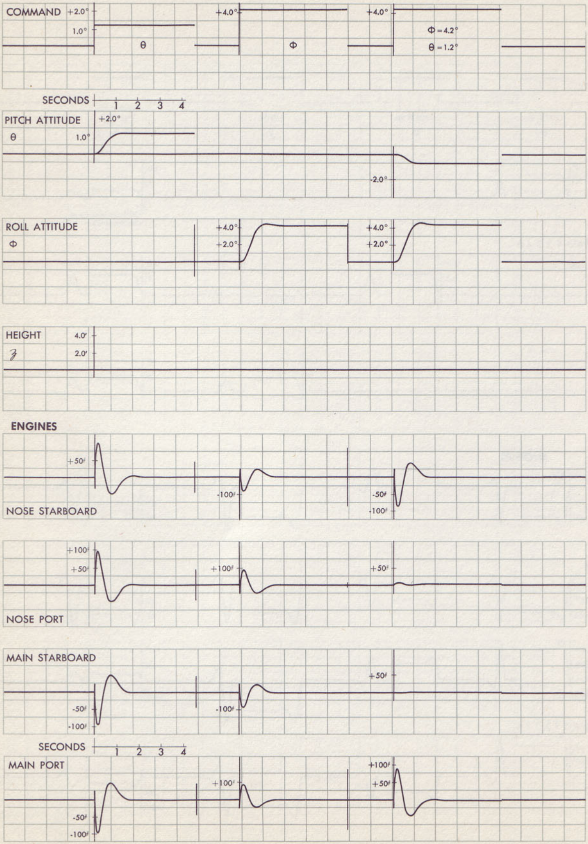


1870-VTOL-1

FIG. 40 RESPONSE OF AIRCRAFT TO SHORT AND LONG TERM DISTURBANCES

1828-VTOL-1

AVRO VTOL



1869-VTOL-1

FIG. 41 RESPONSE OF AIRCRAFT TO COMMANDS

1840-VTOL-1

The time for the commanded output to reach ninety per cent of its final value is less than one second in all cases. The step commands used make greater demands on the engines than normal pilot commands, since normal pilot commands would more likely correspond to ramp inputs. One particular feature of the comparison between regulator response and servo response, that is of particular interest, is the favourable ratio between the angular displacements due to gusts and the commanded attitude change available. This indicates that, compared to the expected displacements caused by gusts and turbulence, the rate of change of commanded attitude is adequate. These records also show the form of response of the engines. This depends on the gain of the system. Thirty per cent overshoot was considered reasonable, and with this overshoot, reasonable servo response times are obtained.

No specific analysis of an automatic yaw control system was carried out because it was felt that relatively steady disturbances would occur in this plane. Since the aircraft would tend to weathercock into the wind a control force would be required. With thrust tilt available in the zy plane control of yaw is obtained by connecting the engine thrust-deflecting device to the rudder bars. Since the yaw axis displays essentially the same stability as the pitch and roll axis, but without the weight terms, the root loci diagrams indicate that only automatic damping should be provided. With automatic damping the pilot would operate the rudder bar in much the same way as he would during taxiing of a normal aircraft.

The most critical period of the flight from a control point of view is the initial and terminal hovering conditions with zero forward speed. This was the primary condition investigated in the analysis. For the hovering condition, with increasing forward speed, the aircraft is flown with small angles of attack to reduce the drag. This means that only small amounts of aerodynamic lift are created. However, the aerodynamic damping increases with speed. In order to compensate for this increase in system gain, the artificial damping sensitivity may require scheduling. At the present time it appears that this could be avoided by proper choice of parameters. However, further investigation should be carried out in order to finally establish this fact. For normal flight a conventional control system is used. The damping system and the autopilot for this flight mode could be designed to use the rate gyros, and vertical gyro, required for the terminal control system. In this way, the control equipment needed, in addition to that carried by a normal aircraft, would be minimized.

7.5.6 CONCLUSIONS

The stabilization system described provides attitude control of pitch and roll, and regulated height, with commanded rate of ascent and descent. The system provides adequate regulation of gust disturbances, and uneven engine performance, and at the same time gives sufficient speed of response to commands, with minimum overshoot. The system as proposed minimizes the demands made on the pilot and, therefore, should require a relatively short training period and reduce pilot fatigue.

7.6 ENGINE TRIMMING IN HOVERING FLIGHT

7.6.1 STATIC TRIM CONDITIONS

To supplement the analogue investigation of the hovering performance, an analytical investigation of the engine requirements was carried out. The results are presented in Figs. 42 to 45.

Fig. 42 shows the nomenclature used in the analysis, a perspective view being given in Fig. 42a and a plan view in Fig. 42b. Engine thrusts are denoted by T with subscripts N for noseplane engines, M for mainplane engines, and P or S for port or starboard respectively. The wing spans are denoted by "b" with the relative subscript, and β is the ratio of the noseplane span to the mainplane span.

Fig. 43 gives the trim conditions. In Fig. 43a, it is assumed that T_o is the engine thrust when

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all the engines are perfectly matched, and that ΔT is the out-of-match thrust of the starboard noseplane engine. Then a static analysis shows that the aircraft can be trimmed by altering the other three engine thrusts as shown. When the value of T_0 is increased by ΔT_0 to give the aircraft a vertical acceleration, the comparable condition is as shown in Fig. 43b. These results have been combined in a non-dimensional form in Fig. 43c, where the trimmed conditions for the three matched engines, in relation to the variation in thrust of the out-of-match engine, are compared. Fig. 43d shows the thrust required per engine to give positive upwards acceleration. Superimposed are the maximum available engine thrust and the required engine thrust to satisfy the specification. It is seen that a minimum of 181 lb. thrust/engine is available for control moments.

7. 6. 2 DYNAMIC CONDITIONS

The dynamic investigation assumed that the throttle input signal, in both pitch and roll, would be a function of the angular displacement from the trim condition, and of the angular velocity. Because the time lag was small, it was neglected, as was also the time integral of the position angle.

The results indicated that if the gain settings in pitch are twice as large as those in roll, both results collapse to give a single curve for any parameter. This fact has been used in plotting Figs. 44 and 45, and suggests that to give equal rates in roll and pitch the available throttle setting in pitch must be twice that in roll. Referred to the excess thrust available for control of 181 lb. (Section 7. 6. 1), this gives 2. 6% of the engine thrust for pitch and 1. 3% for roll. Superimposing these values on the period versus throttle setting curves (Fig. 44), and time and cycle to half amplitude curves (Fig. 45), it is seen that if the gain is chosen to give 60% of the critical damping, the periods will be 5. 1 seconds, time to half amplitude 0. 75 seconds, and cycles to half amplitude 0. 15, in each case. This is with a vertical acceleration of 0. 1 'g', and would be the worst condition, larger control moments being available when lower engine thrusts are required with smaller vertical accelerations.

7. 7 LINKAGE OF HOVERING AND AERODYNAMIC CONTROLS

7. 7. 1 CONSIDERATIONS GOVERNING CONTROL LINKAGES

The basic considerations governing control synchronization between two different methods of controlling an aircraft, are that there should be no discontinuity in control force or sense during transition from one type of control to the other, and that wherever possible conventional cockpit installations should be used to cover both phases of the flight. These criteria have been rigidly adhered to in the following installation, and a control system is provided which should permit a conventionally trained pilot to operate this aircraft with a minimum of additional training. Aerodynamic control is retained throughout the hovering period, so that in the event of engine loss of thrust during hovering flight with forward speed, aerodynamic forces may be used to help support the aircraft.

7. 7. 2 PROPOSED INSTALLATION

A schematic layout of the proposed installation for hovering flight is given in Fig. 46. This figure indicates the control linkage as sensed by the pilot, and neglects all automatic control installations for clarity. It is seen that basically in hovering, a forward movement of the control column will decrease the thrust on the forward engines, while increasing it on the aft engines (Fig. 46a). This differential thrust will provide a pitching moment on the aircraft. At the same time the elevator will move up so that any aerodynamic moments on the aircraft will assist the engine moments. A similar condition holds for lateral movement of the control column (Fig. 46b), a movement that lowers the starboard aileron and raises the port aileron, produces an increase in thrust in the starboard engines and a decrease in the port engines, so that the thrust and aerodynamic moments are in the same sense. The rudder pedals are connected to deflection

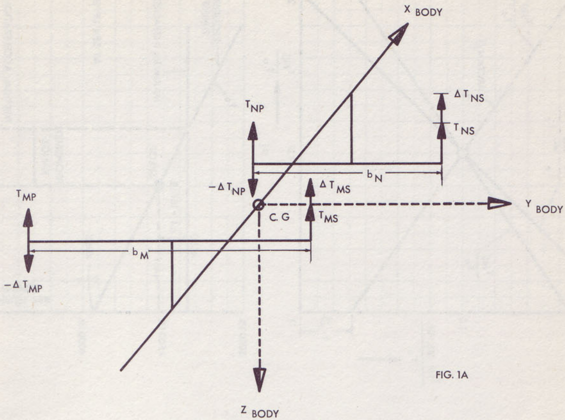
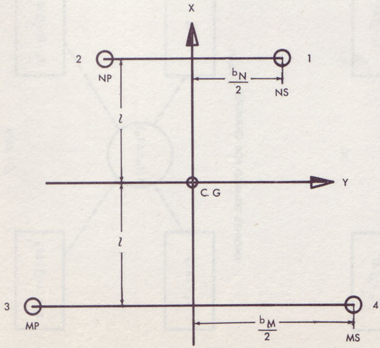


FIG. 1A



XY-PLANE

FIG. 1B

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FIG. 42 NOMENCLATURE USED IN ANALYSIS OF HOVERING FLIGHT.

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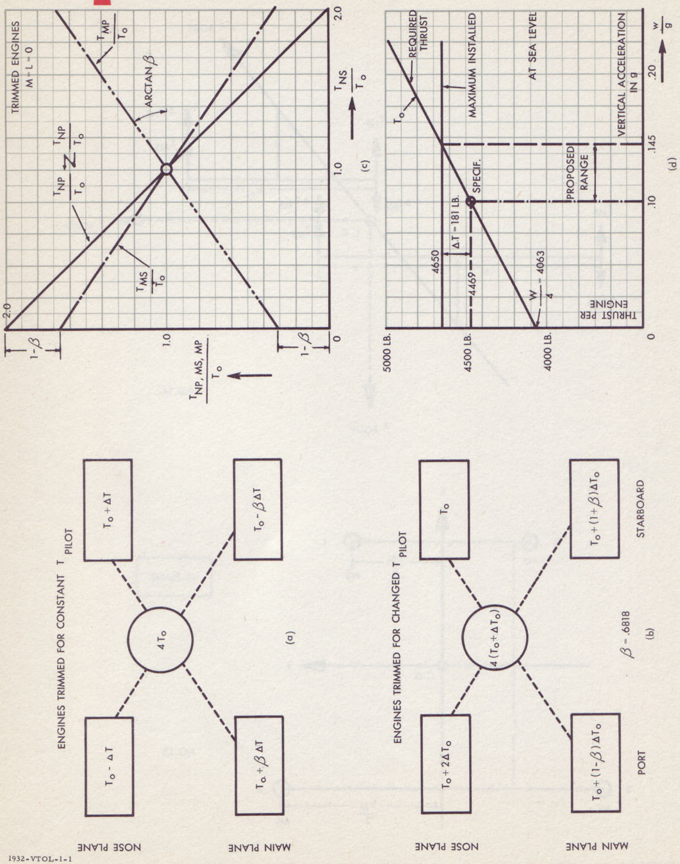

1932-VTOL-1-1

FIG. 43 EFFECT OF LOSS IN THRUST OF ONE ENGINE ON THE TRIM THRUSTS OF THE REMAINING ENGINES.

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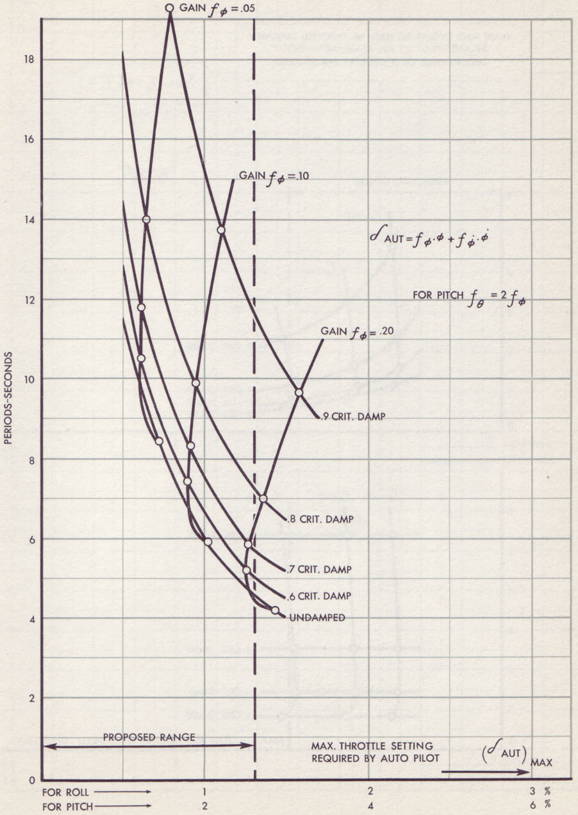

1933-VTOL-1-1

FIG.44 PERIODS VS MAX THROTTLE REQUIRED BY AUTOPILOT AT AN ANGULAR-VELOCITY DISTURBANCE OF 3 DEGREES PER SECOND

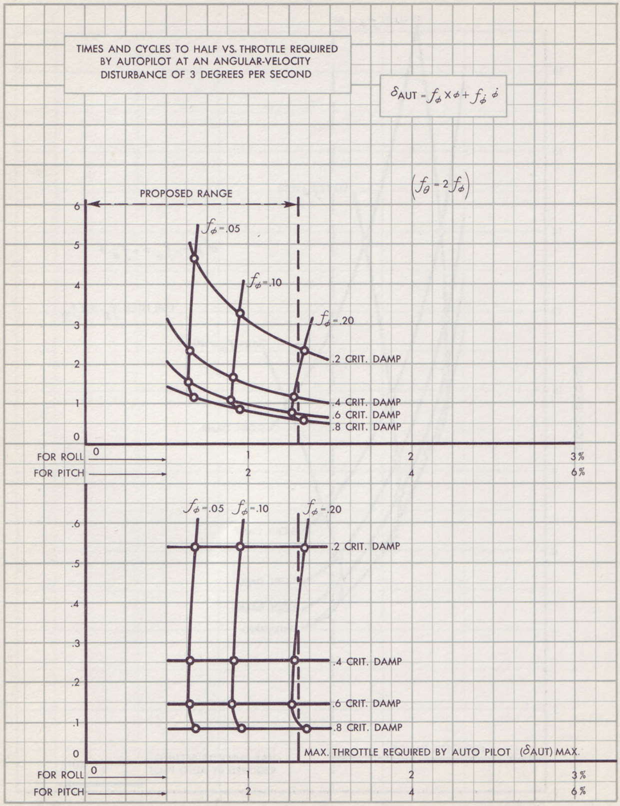

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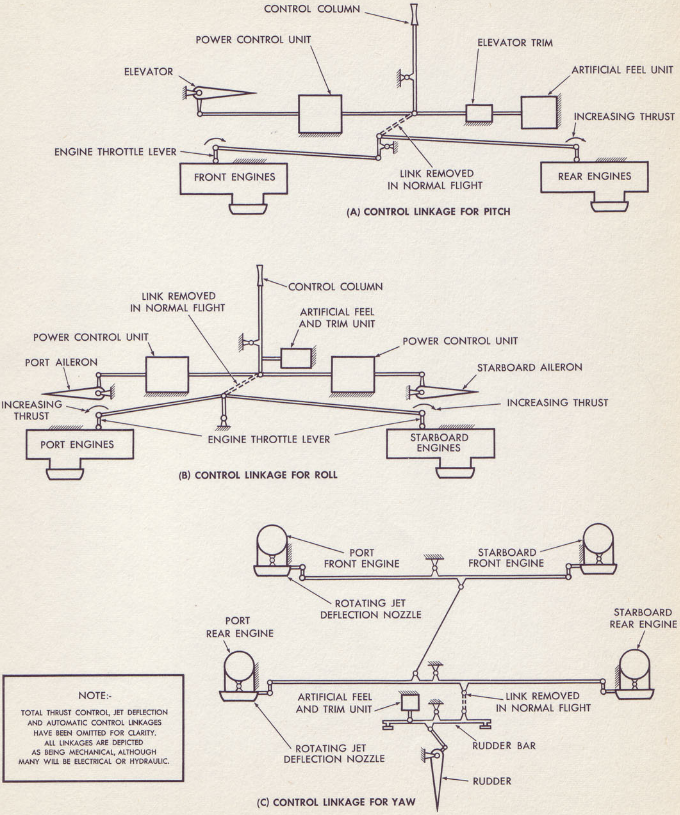
FIG. 45 TIMES AND CYCLES TO HALF VS THROTTLE



NOTE:-

TOTAL THRUST CONTROL, JET DEFLECTION AND AUTOMATIC CONTROL LINKAGES HAVE BEEN OMITTED FOR CLARITY. ALL LINKAGES ARE DEPICTED AS BEING MECHANICAL, ALTHOUGH MANY WILL BE ELECTRICAL OR HYDRAULIC.

1929-VTOL-1

FIG. 46 SCHEMATIC DIAGRAM SHOWING INTERLOCKING OF AERODYNAMIC AND THRUST CONTROLS IN HOVERING FLIGHT

1828-VTOL-1

nozzles on the jet pipe (Fig. 46c), the deflected jets giving a lateral horizontal component of thrust,which provides a yawing moment in the same sense as the rudder yawing moment.

Total thrust will be controlled by a separate thrust control, which will have combined with it the jet deflection control mechanisms so that the pilot can select horizontal, vertical, or deflected vertical thrust without removing his hand from the control. When horizontal position is selected the engine controls will be disconnected from the aerodynamic control linkage.

7.8 PROPOSED TAKE-OFF AND LANDING PROCEDURES

7.8.1 GENERAL

To demonstrate the simplicity of the proposed control system in hovering flight a typical flight programme will be considered in take-off and landing. It will be assumed that automatic vertical velocity control is not installed, as this will give the most difficult conditions.

7.8.2 TAKE-OFF PROCEDURE

Initially, the aircraft will be at rest on the deck with its engines idling and the jets horizontal to avoid overheating the deck.

To take-off, the pilot will deflect his jets to the vertical position and trim the aircraft for zero lift. Deflecting the jets selects automatic attitude control. By increasing engine thrust, until it is greater than the all up weight, the aircraft will rise vertically with no change in attitude. Once the aircraft leves the deck maximum thrust will be selected and the aircraft will accelerate vertically. At a height of about 50 ft. the pilot will deflect the jets 15^{o} aft of the vertical, and accelerate forward until he has obtained a transition speed of about $1.15\ V_{stall}$.

During the period of forward acceleration, the aircraft will continue to climb, and no change of trim will be required by the pilot. Flight path, however, can be controlled by the control stick, forward movement increasing, and aft movement reducing, the forward acceleration. Lateral movement of the control column will give the aircraft lateral accelerations in the direction in which the column is moved.

At transition, the pilot will reduce his thrust, at the same time pulling his stick back so that the loss in thrust is balanced by the aerodynamic lift. When the engine thrust has dropped to its idling value he will select horizontal jet thrust, open the engine throttles, and trim the aircraft in the normal manner. The loss of height during transition is less than 250 ft. and the height at the beginning of normal flight about 1000 ft.

An alternative procedure at transition is to first trim the aircraft for gliding flight at that speed, thus increasing the vertical acceleration. Then, on reducing engine thrust, the aircraft will enter a glide with no further trim application required from the pilot until the jets are deflected horizontally and the throttles opened.

Time from take-off to aerodynamic flight depends very largely on the piloting techniques used, but following the above procedure will require approximately 1 minute.

7.8.3 LANDING PROCEDURE

A suitable landing procedure is for the aircraft to approach in the trimmed condition at a height of about 300 ft. with engines idling, airbrakes out, and a speed of $1.1\ V_S$. Then the jet is deflected to 15^{o} forward of its vertical position, and the thrust increased until the loss in height is reduced to about 300 ft./min. This rate of descent is maintained throughout the initial deceleration,by increasing thrust until the wings are stalled at about $0.5\ V_S$,by a change of trim when the rate of descent may be increased to about 600 ft./min. (10 ft./sec),until the aircraft has lost

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all forward speed. The jets are then deflected to the vertical position, and the aircraft brought down vertically, the rate of descent being reduced as the aircraft approaches the ground.

Time for landing, neglecting manoeuvring into position, is expected to take about 3/4 of a minute from the deflection of the jets at transition. As with the take-off case, however, this will depend very largely on the procedure adopted by the pilot.

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8.0 AERODYNAMIC DATA SUMMARY

8.1 AREAS, WEIGHTS, MOVEMENTS AND TIMES

GROSS WEIGHT 16,250 LB

COMBAT WEIGHT 14,250 LB

EMPTY WEIGHT 10,280 LB

THRUST/WEIGHT RATIO at sea level at take-off gross weight on Navy tropical day.		1.1
MAXIMUM SPEED AT 60,000 FEET with maximum thrust (B.Or.11 engine), level flight at combat weight		2.0M
TIME TO CLIMB TO 60,000 FEET at Mach No. varying from .9 at sea level, to 1.75 at 60,000 ft. with maximum thrust.		7.94 min.
ACCELERATION IN LEVEL FLIGHT AT 40,000 FEET with maximum thrust and combat weight from Mach No. 1.0 to Mach No. 1.75.		3.21 min.
LOITER TIME on station during combat radius mission		16 min.
FORWARD WING AREA	70.0 sq. ft.	Based on area inboard of nacelles
AFT WING AREA	145.0 sq. ft.	
FIN AND RUDDER AREA	52.5 sq. ft.	Based on the net fin
TOTAL AILERON AREA	28.75 sq. ft.	
TOTAL ELEVATOR AREA	11.70 sq. ft.	
RUDDER AREA	14.43 sq. ft.	
TOTAL SPEED BRAKE AREA	12.0 sq. ft.	
AILERON MOVEMENTS	$\pm 25^\circ$	
ELEVATOR MOVEMENTS	$+30^\circ$ -10°	
RUDDER MOVEMENTS	$\pm 25^\circ$	
SPEED BRAKE MOVEMENTS	67°	
ASPECT RATIO - FWD. WING	1.94	
ASPECT RATIO - AFT WING	2.40	

9.0 TESTING

9.1 OBJECT OF TEST

One-seventh scale model tests were undertaken to determine the thrust coefficient of the deflected nozzle. The tests were to establish minimum values of thrust coefficient, on which the take-off performance depends directly, and to provide rapid and cheap deflection nozzle development.

9.2 EQUIPMENT AND FACILITIES USED FOR TESTING

The ejector test rig at the Orenda Engines Nobel Test Plant was modified for these tests. Fig. 47 shows a schematic layout of this rig. Fig. 48 is a 3/4 front close-up picture of the bottom of the rig with one of the deflector nozzles mounted, and the thrust measuring apparatus in view.

The ejector rig supplies high pressure air at room temperature through a venturi to the deflector nozzle assembly, and this contains a rake to determine the inlet total pressure. The deflector nozzle turns the air through a nominal 90° and exhausts it to atmosphere. The resulting thrust drives the rig backwards but it is restored to its original position by loading a carefully calibrated proving ring (Morehouse). The measured distortion of the proving ring then gives the thrust measurement. At the top of the rig is a thin-walled flexible length of pipe designed to keep the rig restraint to a minimum.

9.3 TEST SPECIMENS

Due to the speed with which results were required, and in order to have a sound starting point, very simple models were constructed. The approach duct represents the Orpheus jet pipe at 1/7th scale. An opening of the same size was cut into one side and a nozzle with two spacer plates provided. If the nozzle length is measured from the rear side of the "jet pipe", the use of the spacer plates allowed nozzles of 3, 7 and 12" full scale lengths to be tried. The "jet pipe" was then provided with a sealing plug at 90° to the approach flow. Two inserts were provided, one flat and set at 30° to the nominal exit flow and the other resemling this but having a radius curve on the air-flow side. After the original tests, the model approach duct was reworked to blend out the sharp intersection between the "approach" cylinder and the "exit" cylinder.

Fig. 49 is a cross section of the deflector nozzles and Fig. 50 and 51 pictures of the models.

9.4 METHOD OF TESTING

Each nozzle was tested at approach total pressure (from rake) to ambient pressure ratios of 2.0, 3.0 and 4.0. The flow and temperature were obtained from upstream measurements. Thrust was obtained by measuring, electrically, the deflection of a proving ring which was used to restore the rig, with nozzle, to its static rest position, as shown by the dial indicator seen in Fig. 48. Since the thrust and flow coefficients depend on ratios of nearly equal quantities, the test scatter was appreciable. In order to minimize this effect each point was retested up to 10 to 15 times. Faired curves of the primary results were used to obtain the final coefficients.

9.5 RESULTS OF TESTING

The ejector rig is presently fairly stiff. Although the test method largely eliminates this effect, small movements of the rig can cause a ± 5 or 10 pound error in thrust which tends to be constant for a particular test. Although the large number of tests substantiates the general value of the thrust coefficient, the differences between configurations may not be exactly correct. Modifications to the ejector rig are scheduled to start before October, 1956. These should effectively eliminate rig stiffness thus making future testing much faster and allowing the effect of quite small model changes to be assessed.

In the first series of tests the 30° insert was tested with 3, 7 and 12 inch (full-scale length to jet pipe) nozzles and the 12 inch nozzle was tested with no insert, the radius curved insert and, as previously, the 30° insert. Figs. 52 and 53 show the measured thrusts and flows for the 12 inch nozzle with the 30° insert and the radius-curved insert. The symbols indicate the two completely separate series of tests. The agreement is generally very good although the scatter in thrust will be reduced in future testing. The scatter indicated is typical of all the test results although some of the earliest tests showed constant thrust errors, thus necessitating the large number of repeats.

All tests were carried out at nozzle pressure ratios greater than choking. The nozzle flow coefficient has been defined as the ratio of actual flow to the theoretical choking flow for the full nozzle area. Fig. 54 shows the flow coefficients obtained from the mean curves. With a 30° insert increasing the nozzle length from 3 to 7 or 12 inches, increases the flow by one or two percent. With a 12 inch nozzle-length, there is very little difference in flow between the 30° insert and no insert while the radius curved insert has the largest flows. The existence of upstream throats of various sizes but generally smaller than the exit area accounts for most of these differences.

Theoretical thrust was calculated one-dimensionally for each nozzle on the basis of the measured pressure ratio and flow. This thrust was assumed to be for an ideal nozzle with an actual area equal to the measured effective area. The measured thrust falls below this due to pressure losses between the measuring station and the exit nozzle and due to non-axial exit flow velocities. Fig. 54 shows the measured to theoretical thrust coefficients obtained. The shortnozzle shows much lower thrusts due to its very poor inlet passage and underturning in the very short length. The 7 inch nozzle with 30° insert and 12 inch nozzle with radius-curved insert show the best résults with a thrust coefficient of about 92%. The best obtained from ordinary jet pipes, of the model diameter, is of the order of 98% (c.f. NACA RM E. 52L24, E. 53J13, and E. 54A18 as indicated in Fig. 54).

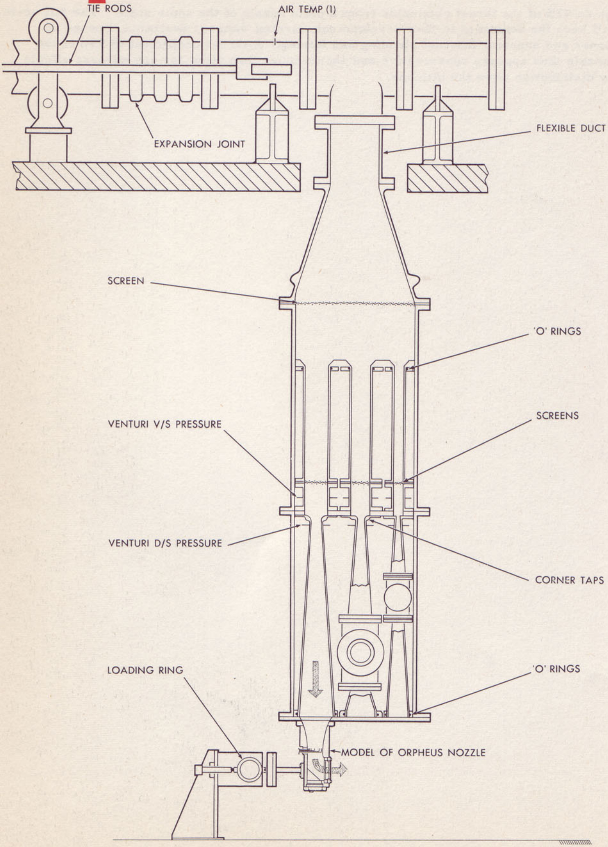
After these tests the nozzle body was modified. The sharp dividing line between the two cylinders (the "approach" pipe and the "exit" pipe) was blended out. This increased the throat area slightly and eliminated one cause of pressure loss. The nozzle body was then tested with 7 and 12 inch nozzles with both radius-curved and 30° inserts. There was time for only 2 test points for each configuration at each pressure ratio. The results are smooth, however, and show a 1 to 2% increase in effective area and a 2 to 3% increase in thrust coefficient. The 7 inch nozzle with radius-curved insert gave the best results with the similar 12 inch nozzle a little poorer. Visual checks indicated that the 12 inch nozzle was over-turning hence its drop in performance.

9.6 FUTURE TESTING

The ejector rig is being actively developed and before the end of October, 1956 a major modification to its suspension should be installed. This should remove almost all of the scatter on thrust readings and make model testing and development very rapid. Shadowgraph equipment is now available and Schlieren should shortly be available. This equipment will allow turning angle studies to be made cheaply and quickly.

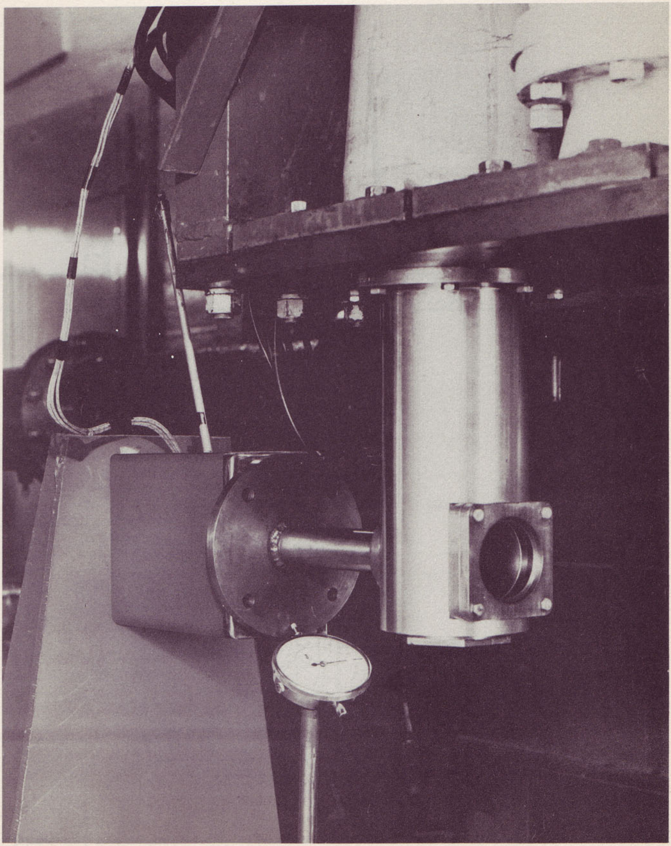
9.7 CONCLUSIONS

Tests of one seventh scale models show effective areas of 85 to 90% and thrust coefficients as high as 95% for very simple nozzles. The test results were good enough to show up small differences and smooth trends among the configurations tested. The results obtained already run as high as 97% of the thrust obtainable from a plain nozzle of the same scale. The full scale version will have the benefit of further development, turning vanes if desirable, opened up turning corner and superior internal blending and fairing. A thrust coefficient of 95% of the standard nozzle thus appears conservative and should more than allow for any adverse effects due to flow distribution down the tailcone.



1903-VTOL-1

FIG. 47 SCHEMATIC LAYOUT OF TEST RIG



1904-VTOL-1

FIG. 48 SIMULATED ORPHEUS NOZZLE TEST RIG

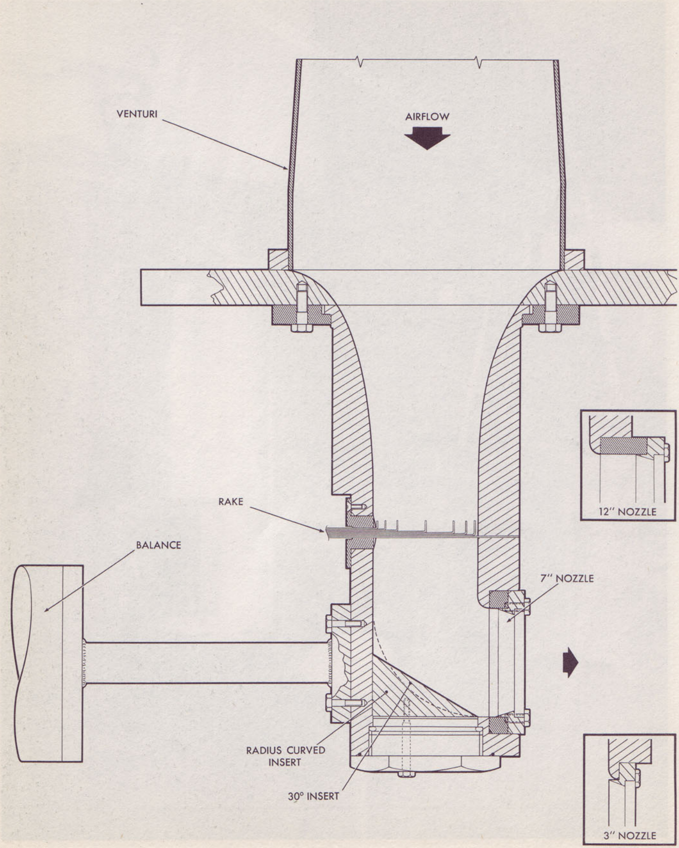


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FIG. 49 1/7TH. SCALE MODEL SIMULATED ORPHEUS NOZZLE

1840-VTOL-1

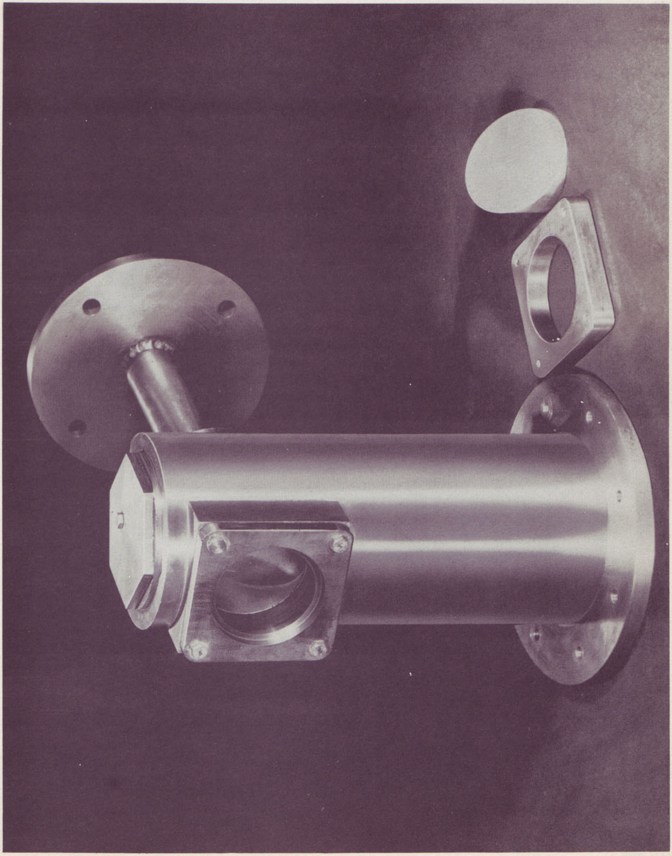

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FIG. 50 MODEL OF SIMULATED ORPHEUS NOZZLE

AVRO VTOL

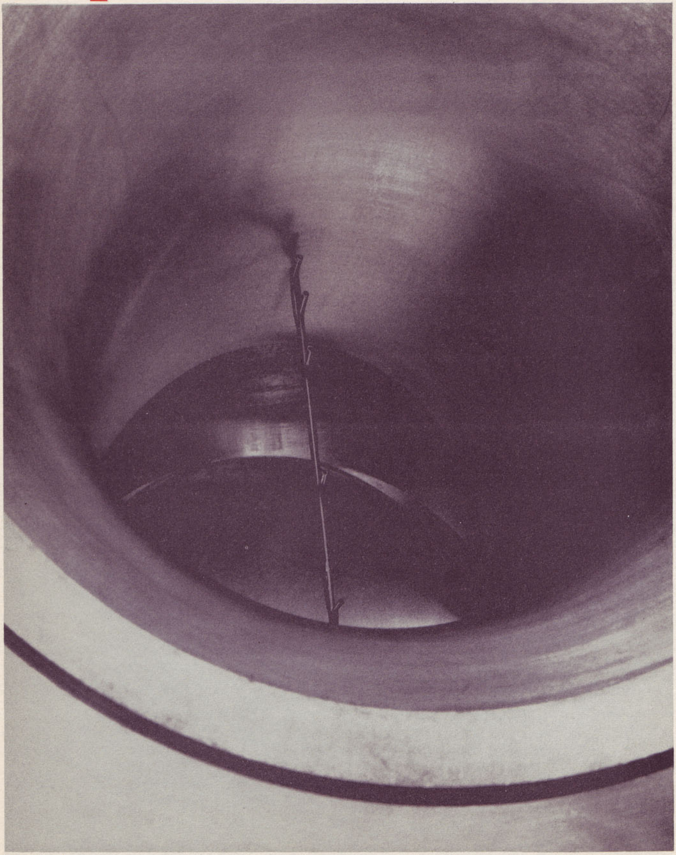

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FIG. 51 VIEW ON INLET OF SIMULATED ORPHEUS NOZZLE

AVRO VTOL

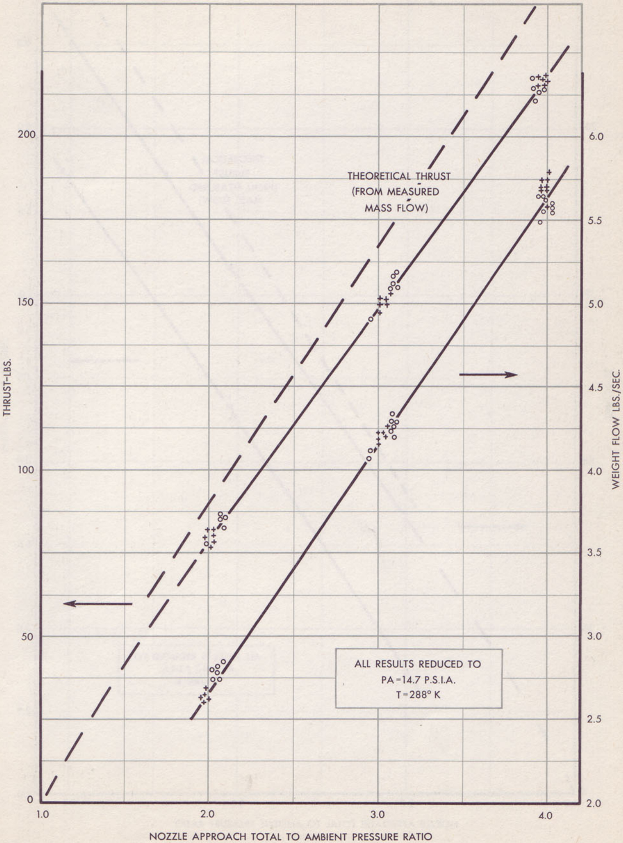

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FIG. 52 MODEL TEST RESULTS – 12 IN. LONG NOZZLE WITH 30° INSERT

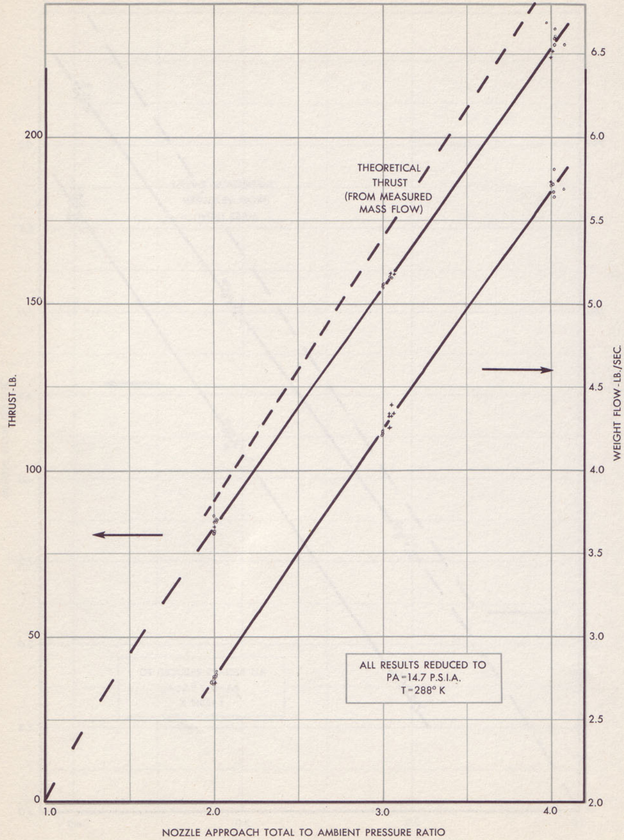

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FIG. 53 MODEL TEST RESULTS-12 IN. LONG NOZZLE WITH RADIUS-CURVED INSERT

AVRO VTOL

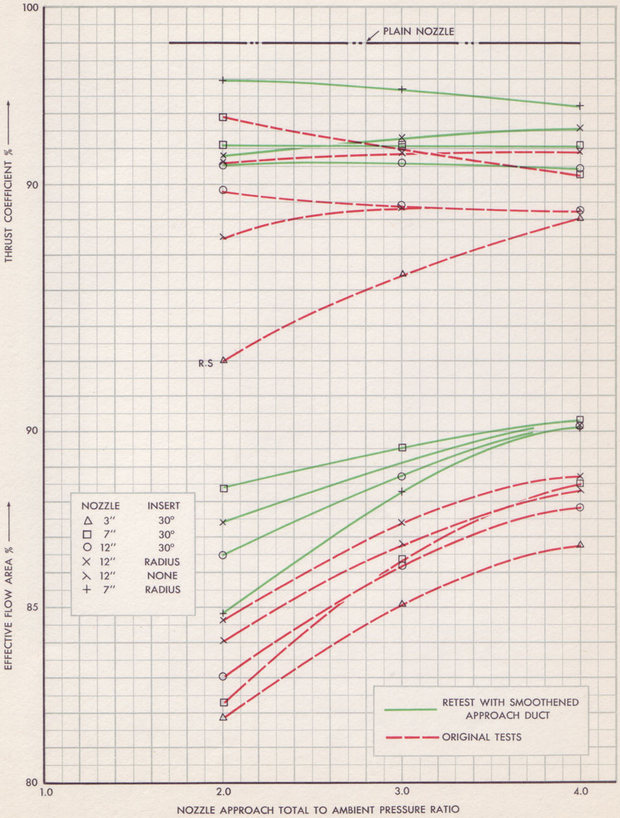

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FIG. 54 MODEL TEST RESULTS

AVRO VTOL

10.0 DETAIL DESIGN

10.1 WINGS

10.1.1 GENERAL

The forward and aft wings are almost identical in their method of construction, both having simple, multi-spar structures with machined skins. The aft wing structure, being typical, is shown in figure no 55. The forward and aft wings are each attached to the fuselage by four bolts, there being no connection between the fuselage skin and wing skin. The gap between these two parts is sealed.

The forward and aft spars, closing the torsion box, are machined from 24 ST aluminum alloy, the flanges and control pick-ups being machined integrally with the spars.

The root ribs, at the fuselage side, and the tip ribs, are forged and machined from 24 ST aluminum alloy, in one piece from leading edge to trailing edge. The wing-to-fuselage attachment lugs, and stringer joints, are part of the root rib machinings. The engine pod attachments are machined integrally with the tip rib.

There are 7 intermediate ribs in the forward wings, and 9 in the aft wings. These are intercostal between the spars, and are pressed from 24 ST aluminum alloy sheet.

There are detachable panels in the leading edges of all wings. These permit access to, and inspection of, electrical cables, engine controls, fuel lines and so on.

A trailing edge box machining is attached to the rear spar, and houses all controls situated between that spar and the control surface. The box is machined in two halves, top and bottom, and incorporates all necessary hinge fittings. The control is completely removable to facilitate simple and overall servicing of the controls. Service openings are provided in this boxto allow routine checks to be made on the control system. The box itself, having this type of structure, becomes an efficient load-carrying part of the wing, taking torsional loads, and some bending loads.

10.1.2 FORWARD WING

The above paragraphs have described the features common to both wings. There are, however, some differences between the wings, apart from size.

The upper skin and stringers of the forward wing are integrally machined, and the panel is in three pieces, port, centre, and starboard, with a kink at the fuselage side. The integral stringers are broken at the fuselage side, with stringer end loads being transferred through the machined root rib.

The lower skin is made up of two panels, joined at the aircraft centre-line, and has no stringers, these being replaced by the 24 ST spar webs. Each stringer on the top surface is connected to the lower surface by a spar web attached to it.

The method of assembly is simple, conducive to fast production, and requires an absolute minimum of tooling. The ribs and spars are riveted to the top skin and stringer combination, the lower skin is then blind riveted and blind bolted to the above parts. The trailing edge box is attached, and, the control surface pushed into place, being attached by pins inserted vertically through the trailing edge box.

10.1.3.AFT WING

The aft wing structure is similar to the forward wing structure, and there is no need to repeat the above detailed discussion. Instead, the differences will be given.

The lower skin and stringers are integrally machined, with the stringer end loads again being transferred through the joint at the root rib.

The upper skin, having two panels, has no stringers, and is connected to the lower skin by means of spar webs.

The method of assembly is identical to that for the forward wing.

10. 1. 4.SUMMARY

A great deal of the wing structure is machined, with flanges, stringers, control pick-ups, as integral parts. The skins are tapered, there are no cut-outs whatsoever between the front and rear spars, and riveting is kept to an absolute minimum. The multi shear webs greatly increase the permissible working stresses of the upper and lower skins. Such a structure is economical in weight, and having comparatively few parts, lends itself to fast and economical production. The large detachable panels afford complete access to all parts requiring routine servicing.

10. 2. FUSELAGE AND TAIL

10. 2. 1.GENERAL

The fuselage and tail fin are an integral unit, and their structure is shown in figure no 56. Machined skins, with integral stringers, are used throughout. The main frames are forged and machined from 24 ST aluminum alloy, and the intermediate formers are pressed from 24 ST aluminum alloy sheet. The structure is very simple, and, by virtue of the integral machining, it lends itself to fast and economical production and low weight.

10.2.2. FORWARD FUSELAGE

The radome is plastic, and a metal-skinned portion, housing the electronics, joins this part to the forward cockpit bulkhead.

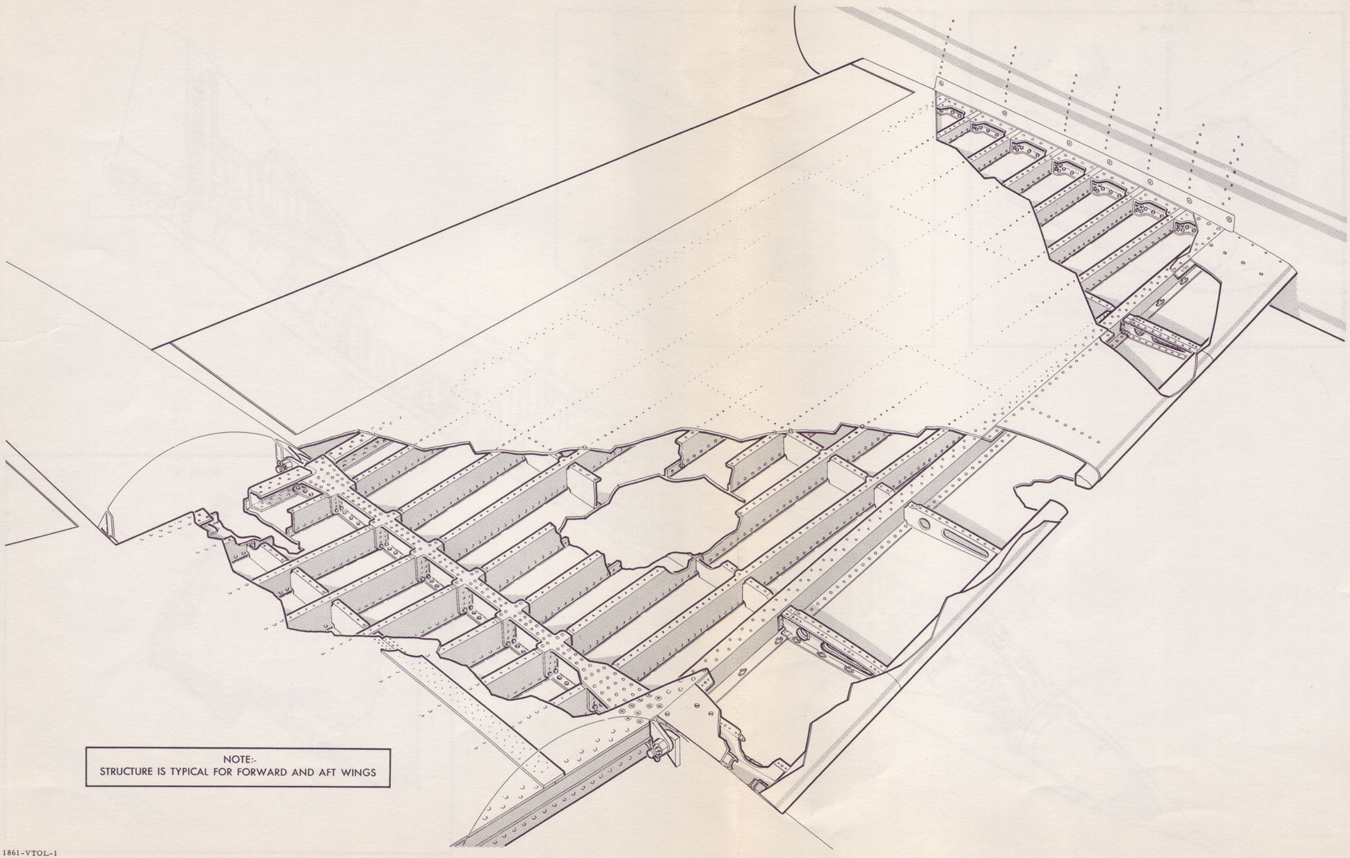
The cockpit region, bounded fore and aft by machined formers, is circular in cross-section, and has a narrow floor, thereby having excellent pressurizing properties.

The cockpit layout is in accordance with U. S. Navy Specification MIL-S-18471 (Aer), and incorporates an ejection seat. Some consideration was given to the installation of an ejectable capsule, and this can be accomplished by breaking the cockpit at the aft cockpit bulkhead.

The cockpit skin is specially machined to suit the canopy edge member, an arrangement which helps to overcome many of today's problems prevalent at such joints.

The forward glass panel of the windscreen is flat, and has, therefore, suitable optical qualities for the use of a gunsight. The side panels are made of glass, and have single curvature. The frame, housing these portions, is made from a magnesium alloy casting.

The canopy is of glass, and affords excellent all-round visibility. It is suggested that it should be of the fore and aft sliding type, to permit it to be opened in hovering flight. This would give the pilot direct and improved vision downwards when taking-off or landing in rain or bad visibility. At night, distracting reflections in the canopy from instruments or deck lighting would be eliminated. The aft extension of the canopy is made of sheet metal, and houses some of the electronic equipment.



NOTE:-
STRUCTURE IS TYPICAL FOR FORWARD AND AFT WINGS

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FIG. 55 AFT WING STRUCTURE

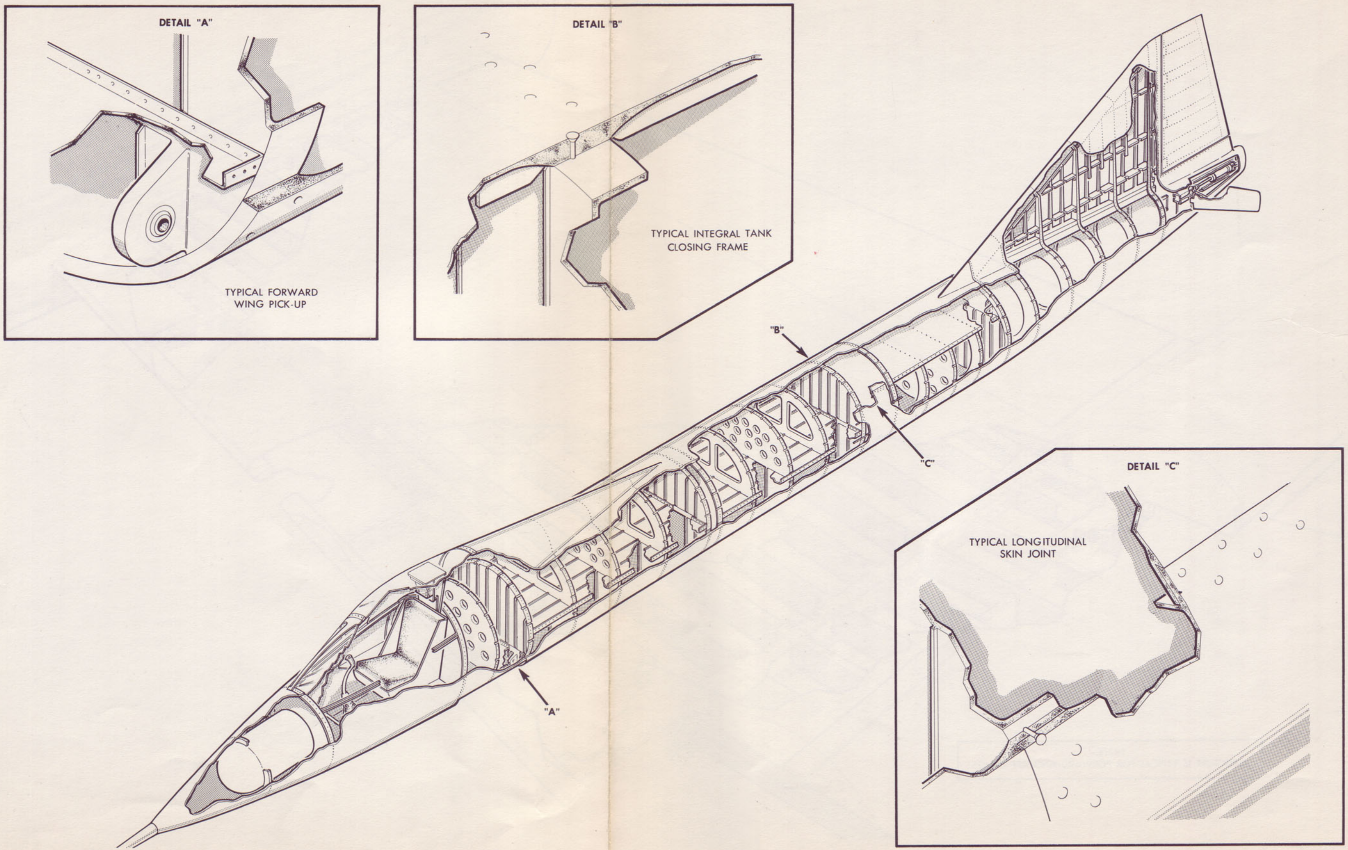

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FIG 56 FUSELAGE AND FIN STRUCTURE

10.2.3. CENTRE FUSELAGE

This part is circular in cross-section, and has a constant diameter throughout its length. It extends from the aft cockpit bulkhead to the aft mainplane bulkhead.

The centre fuselage contains four integral fuel cells, armament, air conditioning equipment, electronics, the armament control system, and the aft landing gear, and has cut-outs for the forward and aft wings.

The forward fuel cell is situated immediately above the forward wing, being separated from the wing by a machined floor.

The two centre fuel cells are situated immediately above the armament bay and forward equipment bay, being separated from the armament and equipment by a machined floor. The armament operating jack is mounted between these cells, being attached to one of the machined frames.

The aft fuel cell is situated immediately below the aft wing, being separated from the wing by a machined floor, and bounded fore and aft by machined frames.

The aft landing gear is mounted between the rear and centre fuel cells, being mounted on the aft machined frame.

There are a number of pressed aluminum alloy baffles suitably positioned inside the fuel cells.

Longerons run fore and aft along the fuselage skin, and act as edge members to the large opening in the bottom of the aircraft. This opening provides access to the forward equipment bay and armament bay.

10.2.4. AFT FUSELAGE AND FIN

The aft fuselage is built integrally with the fin, and has 5 machined frames. The fin spars are integral with these frames.

There is a stiff, machined, rib at the base of the fin. It is intercostal between the spars, and has an "H" - type cross-section, the side flanges acting as the connecting medium for the fin skins and the fuselage skins.

The fin skin are taper machined and have integral stringers between the spars, both skins being identical.

The machined rear spar of the fin has integral pick-ups for controls, rudder jack, and speed brakes.

A machined box is attached to the fin rear spar. It is similar to the box described in paragraph 10. 1. , and houses the controls.

The speed brakes are mounted at the aft end of the fuselage. There is a keel structure between the brakes, with a cut-out for mounting the speed brake jack, the jack being attached to the aft end of the cut-out.

10. 3. ENGINES

10.3.1. GENERAL

From the aerodynamic and weights analysis the engine thrust and weight requirements were established. The basic concept, and the logical analysis of the concept, postulated that a number of small engines were preferable to one or more large engines.

The above criteria, together with the specified requirements, were the basis for choosing the Bristol Orpheus B. Or. 11 jet engine as the most suitable power unit available for this project.

A survey was made of American and British engines, but the majority were either "on the drawing boards," or classified to such a degree that very little information was available. The Orpheus 11 was the only suitable engine beyond the design stage, with a background of successful engines in the same series, and a great deal of information on it was made available to the company.

The Orpheus 3 is a forerunner of the Orpheus 11, and is immediately available. Both engines have the same diameter, and differ only in length, the Orpheus 11 being somewhat longer. Although the Orpheus 3 has not sufficient thrust to meet the requirements of this specification, it can be used for all development testing of jet deflection, hovering mechanisms, as discussed in para 12.0.

As stated in Appendix "A" of this brochure, the Orpheus engine could be manufactured by Orenda Engines Limited, an associate company of Avro Aircraft Limited, thereby ensuring that the entire production of engines, airframe, armament and equipment is kept on this continent to satisfy the North American defence philosophy. An obvious corollary to the above is that the airframe and engines would be manufactured by closely integrated associate companies with their manufacturing establishments adjacent to each other - an ideal relationship from the standpoints of engine-airframe integration and liaison, engine development to meet vertical take-off requirements, and time, both companies being able to work together in an attempt to resolve all the mutual problems in a speedy and efficient manner.

Like most engines, the Orpheus is not perfect. It has a large frontal area, and the accessories are badly placed. Both of these faults are being investigated by this company and by the engine manufacturers, and in the proposed engine installation the accessories have been rearranged.

10.3.2. ENGINE DATA

The aircraft is based upon the use of four Bristol Orpheus B. Or. 11 engines, and the following data has been supplied by the manufacturers, Bristol Aero Engines Limited, Montreal, Canada:-

Rated thrust at sea level (static)	5760 lbs.
Rated thrust at 36,000 ft. (M=. 90)	2005 lbs.
Fuel consumption at sea level	1.014 lb. /lb. /hr.
Fuel consumption at 36,000 ft. (M=. 90)	1.260 lb. /lb. /hr.
Weight (dry)	960 lbs.
Weight (Installed)	980 lbs.
Diameter (without accessories)	32.4 ins.

10. 3. 3. ENGINE DEVELOPMENT

It is known that Bristol Aero Engines are working on further developments of the Orpheus, and other companies are designing engines based on today's knowledge of the state-of-the-art. However, lacking reliable information on the performance characteristics of these engines, we have been unable to consider them in this brochure.

10.4. NACELLES

Each engine nacelle can be broken down into three separate monocoque assemblies, bolted

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together to form the complete unit, see figure number 57. Each assembly is made up of an inner and outer skin and formers. The 24ST aluminum alloy inner skin forms the engine by-pass air tunnel, and the outer skin, of the same material, forms the contour of the nacelle. There are pressed aluminum alloy formers between the two skins, being fastened by A17 ST rivets to the outer skins, and by blind rivets to the inner skins. The transport joint formers are made of 75 ST forged aluminum alloy rings.

The forward nacelle assembly, attached to the centre portion by four equally spaced bolts, has several small inspection doors. By removing the four attachment bolts, the forward nacelle can be detached, thus giving complete access to the compressor casing, forward engine mounts, and ancillary equipment.

The centre nacelle carries the three engine suspension fittings, and would be the first unit to be attached to the engine, the other parts of the nacelle being moved forward and aft from this portion in a dismantling operation.

The rear nacelle, attached to the centre nacelle by eight bolts, carries the wing pick-up forging, and has six formers lining up directly with the wing spars. These transfer the engine loads to the wing in the most efficient manner. The end rings of this group are forged from 75 ST aluminum alloy. The four centre formers are pressed from 24 ST aluminum alloy. At the extreme aft end of the nacelle the inner skins are shaped to form a convergent-divergent nozzle, and will be made of titanium. The rear unit also carries the jet-deflecting nozzle shroud casting and a pair of clam-shell doors to close that nozzle when it is not in use. These doors are spring-loaded and depend upon the jet blast to open them.

It is pertinent to mention at this point that an investigation was carried out to compare the relative merits of rotating the engine nacelles or using jet deflectors, as means of obtaining vertical thrust. The rotating engine gave the aircraft a vertical thrust without any loss in thrust. It was, however, complicated by virtue of the engine controls, fuel lines and electrical controls having to pass through a rotating joint. The undercarriage would be longer with a rotating engine scheme and it would be impracticable to install an afterburner in such an installation, as the nacelle would be prohibitively long. In view of the above, the rotating engine mechanism was discarded in favor of a jet deflection mechanism.

10.5. JET DEFLECTION MECHANISM

The mechanism is extremely simple, and has very few moving parts. It is illustrated in figure no 58. The arrangement may be compared to a simple pipe "tee" connection, slightly deformed to provide a proper seating for the flapper valve when it is in either position.

The only moving part, inside the jet pipe, is the flapper valve. This is made in two pieces, being concave forged steel plates, having simple stiffening ribs parallel to the flow of the gases. This valve is hinged from the lower surface of the jet pipe, and is controlled by two hydraulic jacks, one mounted on either side of the jet pipe.

There are five fixed cascade vanes mounted inside the vertical deflection nozzle. These help in the control of the gases as they pass through the elbow, ensuring that the angle of deflection of the jet stream is maintained constant throughout the range of throttle settings.

The only moving part outside the jet pipe is the nozzle, and this gives control of the gases in any direction through an angle of 15 degrees.

The nozzle is mounted on a large spherical bearing and is controlled by four small hydraulic jacks, two on either side of the engine. By controlling the volume of fluid to each pair of jacks, rotation may be accomplished in any direction.

AVRO VTOL

10.6. ALIGHTING GEAR

10.6.1. GENERAL

The proposed landing gear illustrated in Fig. 59, meets the specified strength requirements, and as wheels are not required on this V. T. O. L. aircraft, they have been replaced by simple universally jointed, circular pads. This saves weight and requires a smaller stowage space.

The landing gear is made up of three legs, two forward legs are mounted on the inboard side of the forward engine nacelles, and the aft leg is mounted between the centre and rear fuel cells.

The legs are of the air-oil type, with a long stroke commensurate with landing requirements. The stroke on the legs is 16".

10.6.2. FORWARD LANDING GEAR

The forward legs are attached to integrally machined lugs on the forward nacelle formers, and can be dismantled by merely removing two bolts connecting the legs to the lugs.

The linkage is simple, and the retraction gear is hydraulic. The undercarriage doors are hydraulically operated, the first part of the jack stroke being used to open the door locks. The up and down locks are also hydraulically operated, the up lock being incorporated in the hydraulic jack, and the down lock being a hydraulically operated pin attaching the leg to the former in the down position.

The outer leg casing is made from aluminum alloy, and the piston, pad, and universal joint, are made of steel.

The piston incorporates an integrally machined collar for the attachment of ground equipment for lifting the aircraft.

10.6.3. AFT LANDING GEAR

The aft landing gear is attached to the forward frame of the rear fuel cell by two bolts and a strut, and can be dismantled by removing these bolts and the hydraulic fittings.

The leg is telescopic, and consists of three sleeves. The outer sleeve is attached rigidly to the frame. The second sleeve is operated by a hydraulic jack, mounted between the outer and second sleeves. Inside the second sleeve is a third sleeve, and this is the actual leg of the landing gear, having a circular pad at the end, and an air-oil shock absorbing system. This leg is connected to the second sleeve by two cables, one attached to the leg, and the other running through a pulley, mounted on the second sleeve, to a pick-up point on the outer sleeve. When the jack is extended, the second sleeve moves, and the inner leg is moved by the action of the cables and pulley. The movement of the leg relative to the outer sleeve is doubled-up by virtue of the pulley effect.

The piston incorporates an integrally machined collar for the attachment of ground equipment.

This system is comparatively light by virtue of the absence of doors, linkages, and jack pick-ups. It is also extremely economical in its space requirement.

10.7. FLYING CONTROLS AND CONTROL SURFACES

10.7.1 GENERAL

Aileron, elevator and rudder controls are provided, and are operated conventionally by the

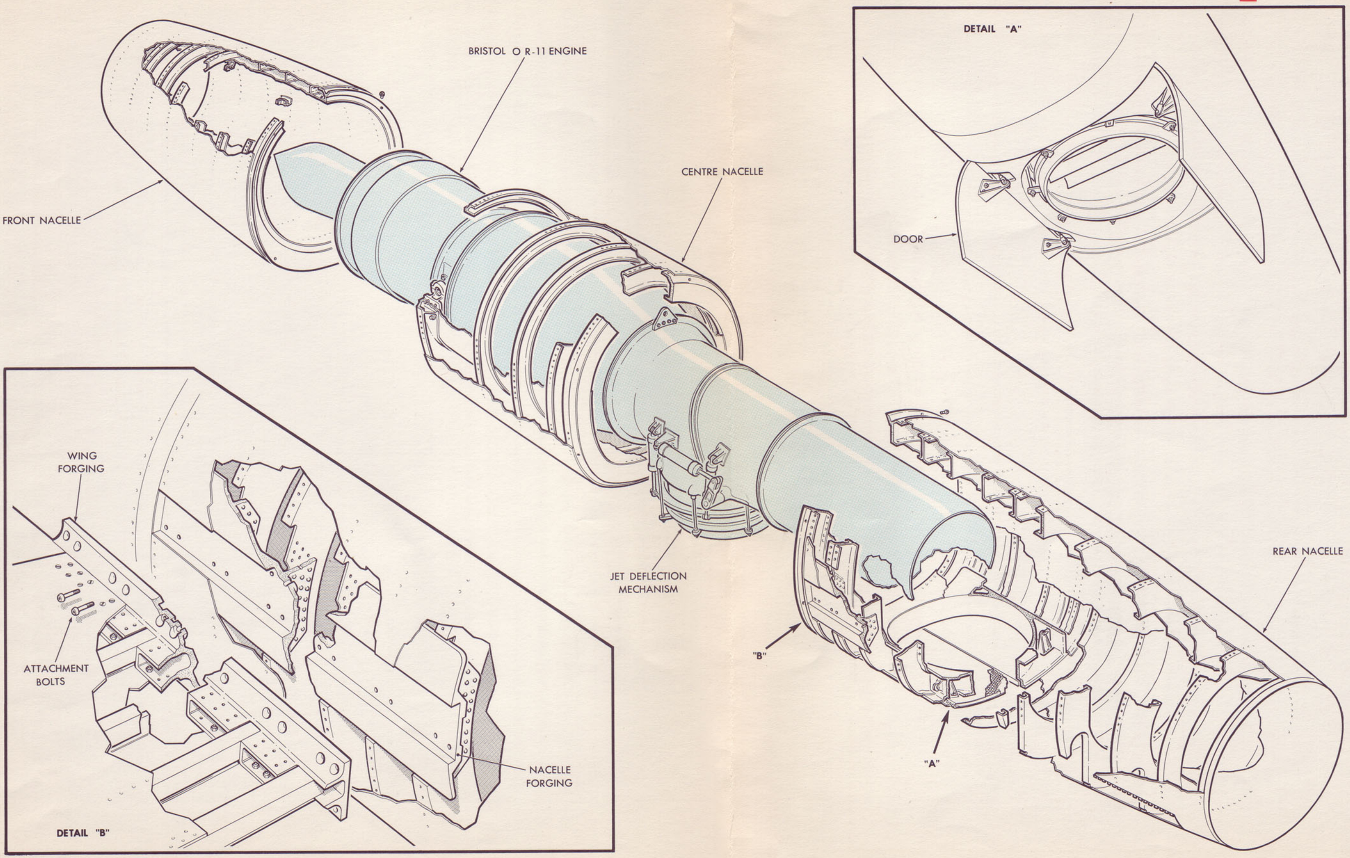

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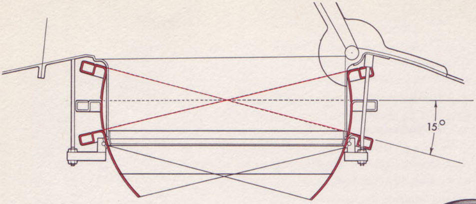
FIG 57 ENGINE INSTALLATION



NOZZLE MOVEMENTS

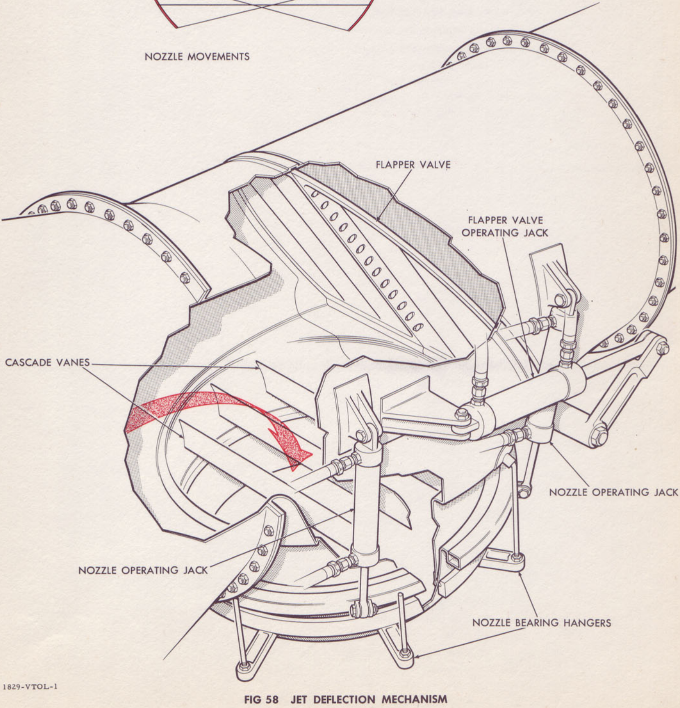


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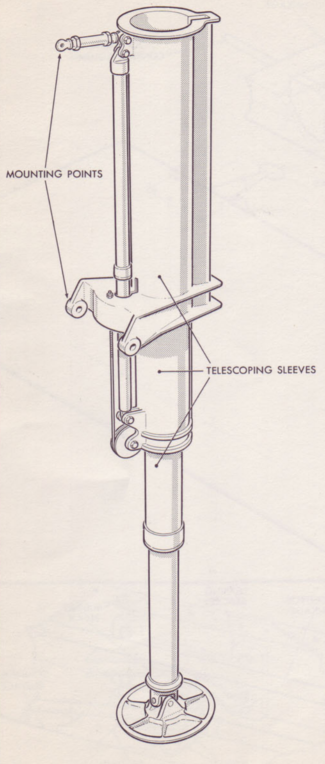
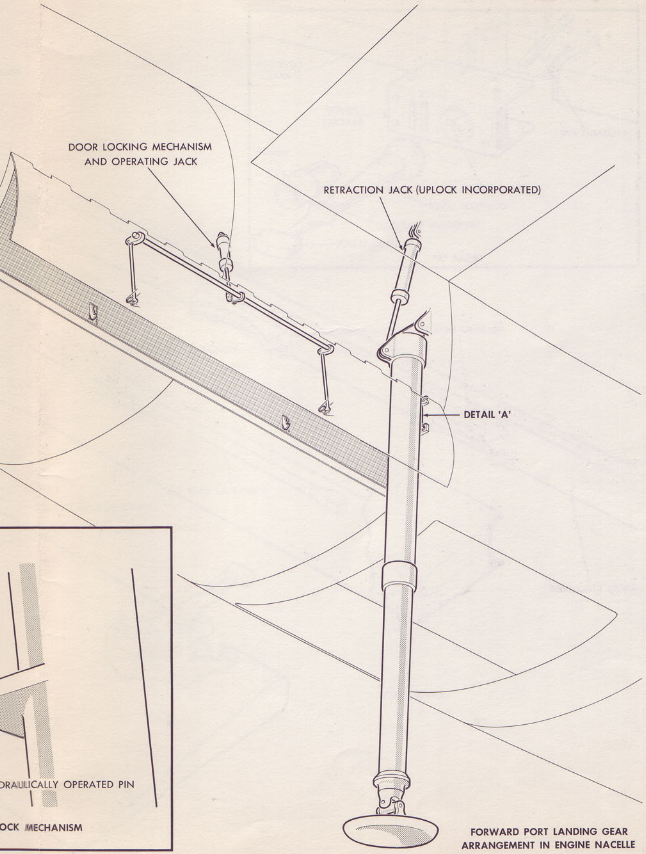
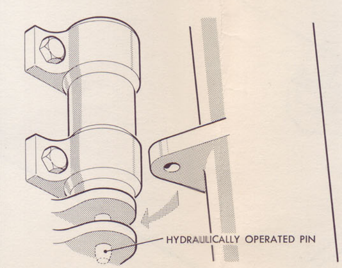
FIG 58 JET DEFLECTION MECHANISM



REAR LANDING GEAR ARRANGEMENT IN FUSELAGE





DETAIL 'A' DOWNLOCK MECHANISM

FORWARD PORT LANDING GEAR ARRANGEMENT IN ENGINE NACELLE

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FIG. 59 ALIGHTING GEAR

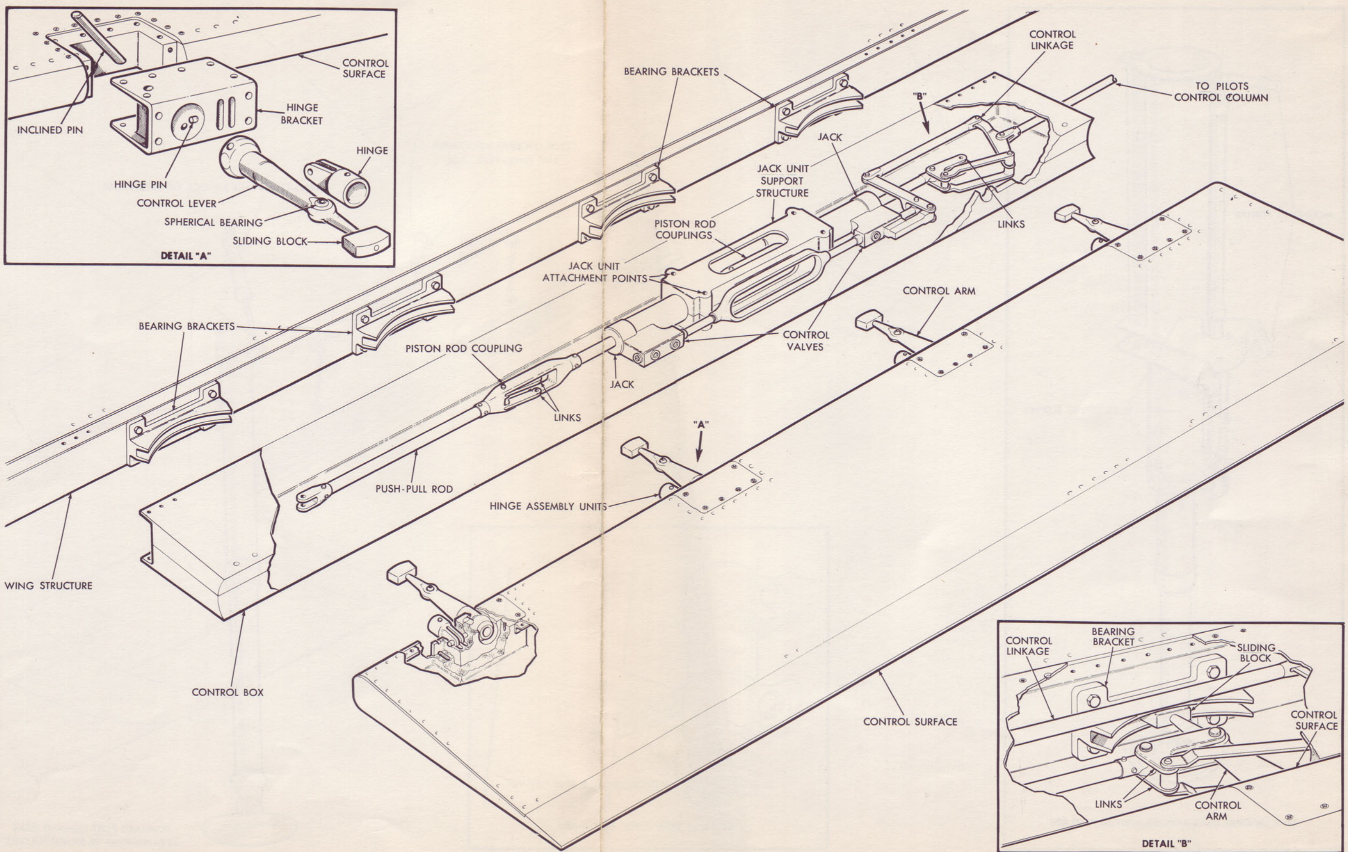

FIG. 60 FLYING CONTROLS

control column and rudder bar situated in the cockpit. The typical system and structure is illustrated in Fig. 60, the ailerons, elevators and rudder being almost identical in their system and structure.

10.7.2 CONTROL SYSTEM

This project is equipped with a fully powered flying control system, and the system is irreversible, and thus there is no feed-back of forces to the pilot.

Hydraulic means of transmitting power to the control surface actuators has been chosen as the most desirable; a 4,000 p.s.i. operating pressure has been selected, to achieve, within practical limits, the most compact design. Double jacks, operating together, are used for the control of each surface, each jack being on an independent circuit, either, or both, of which can actuate that surface, and thus the malfunctioning of one hydraulic pump will not cause a failure in the control system.

10.7.3 DESIGN OBJECTIVES

(a) To provide the ailerons, elevators and rudder, as required for full manoeuverability, with maximum rates of movement of 30°, 40° and 40° per second respectively.

(b) To provide duplication of the flying control hydraulic system to allow for failure of one system, still maintaining adequate power to control the aircraft without severe speed limitations.

(c) To provide a high degree of systems reliability consistent with design for small size and low weight.

(d) To provide the lowest possible degree of vulnerability to all anticipated types of aircraft damage.

(e) To provide a system utilizing components which require a minimum of inspection and maintenance.

10.7.4 METHOD OF CONTROL

Four machined fittings are attached to the spar of each control surface. Each fitting incorporates a hinge pin and a control pin. The short hinge pin is inserted into the fitting in a transverse direction, and the control fitting is inserted at an angle of 45 degrees to the chord plane.

In the sub-assembly stage, the hinge fittings are attached to the pin, and the control links are attached to the control surface by the insertion of the angled pin.

The control surface is then ready for assembly to the box - a very simple procedure. The surface, with the hinge fittings and control links protruding forward, is pushed into position. The forward end of the hinge fittings are circular in cross-section and slide into suitable receptacles mounted on the wing or fin, thus locating the surface with respect to the wing or fin. The forward end of each control link slides into a guide mounted on the wing or fin spar, allowing the links to rotate in the transverse direction only. The control surface is then prevented from sliding out of the wing by inserting a bolt through the hinge fitting at each end of that surface.

The control links are interconnected by double jacks and push-pull rods, either or both jacks being able to operate the control surface. The aileron and elevator jacks, with valves and "feelback" linkage is housed immediately forward of the control surface, in the control box. The rudder jack, with its valves and linkage is housed below the rudder, being mounted within the fuselage behind the machined fin post frame.

When the jacks are actuated the interconnecting rods move in a transverse direction, rotating the links in that same direction. The links are connected to the control surface by a pin inserted at 45° to the chord plane, and this transverse movement of the link will, therefore, cause a movement in the control surface.

The control system is thus simple, economical in space, easy to assemble, and has few bearings.

10.7.5 CONTROL SURFACES

Being a canard type aircraft, the elevators are mounted on the forward wing and the ailerons are mounted on the aft wing, these, and the rudder, all being full-span surfaces.

Their structure follows the general trend of simplicity, low weight, and adaptability to fast and economical production. The forward spar, ribs, trailing edge member, and lower skin, are all integrally machined from aluminum alloy, and the top skin is attached separately by blind rivets and bolts.

10.8 SPEED BRAKES

Speed Brakes are fitted to the aft end of the fuselage, the hinges being machined as an integral part of the rear fuselage frame. The brakes open in a spanwise direction through an angle of 67 degrees. They are operated by means of a hydraulic jack mounted to the keel member in the aft fuselage. The total time required for extension or retraction is well within the requirement of three seconds.

The speed brakes are capable of decelerating the aircraft from maximum level flight speed to 0.8V max. in less than 16 seconds. The speed brake hydraulic pressure may be dumped in case of emergency.

10.9. ARMAMENT

As required in the specification, adequate space has been provided for the installation of either 2.0" rocket launcher clips, Aero x 15A and Aero x 16A, or four folding fin "Sidewinder" guided missiles.

The armament is carried internally, in an armament bay situated in the centre under-belly of the fuselage. Such a position allows an unobstructed trajectory for the armament, and with engines mounted on the wing tips, they will not be affected by the weapon exhaust gases.

The armament is housed in a package, and is lowered into the firing position by a hydraulic jack mounted directly above the bay. This allows the lowering operation to be carried out quickly and efficiently.

Adequate space is provided above, and on either side of, the armament for all necessary controls.

The armament bay door is an integral part of the armament bay package, so obviating the need for special door hinges, releases, catches and so on. This saves weight and complication, and is more reliable.

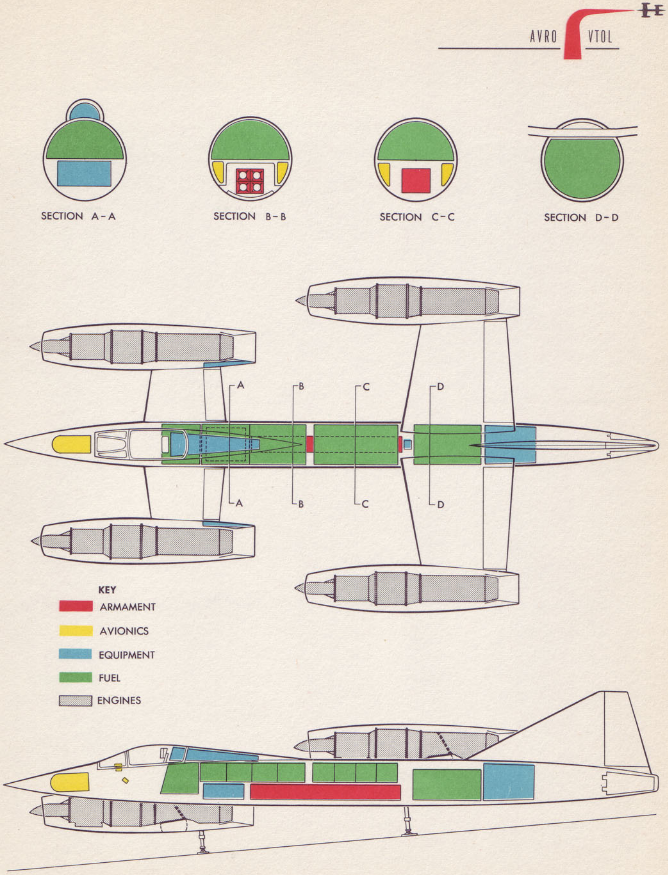

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FIG. 61 GENERAL ARRANGEMENT OF EQUIPMENT

10.10 SYSTEMS

10.10.1 OXYGEN

The oxygen equipment, consisting of two liters of liquid oxygen, is installed in the cockpit. It is a low pressure system conforming to Spec. MIL-C-9082 and MIL-E-5272, Procedure 1, and is in accordance with WADC Technical Memorandum Report WCRD 53-73. A capacitance type quantity gage is provided, and a shut-off valve is situated in the console between the converter and the composite disconnect. This disconnect leads to a GFE emergency oxygen back-pan.

10.10.2 FUEL SYSTEM

The fuel system is in accordance with Spec. MIL-F-17874 (Aer). Single point pressure fuelling is provided, suitable for a working pressure of 60 p. s. i. and surge pressures up to 120 p. s. i. It is capable of operating at maximum performance with JP-5 fuel, although JP-4 fuel is a suitable alternative.

All fuel is carried internally, in four integral fuselage fuel cells, the total capacity being 736 U. S. gallons.

The refuelling point is situated on the fuselage side in the region of the aft fuel cell. This is an easily accessible position, and the fueling operating can be carried out without the use of ladders or steps.

Fuel lines are housed in the leading edges of the wings, taking the fuel from the fuselage cells to the engines, and all necessary fuel controls are conveniently positioned in the cockpit.

10.10.3 INFLIGHT REFUELLING PROVISION

Space has been allocated for inflight refuelling, using a type MA-2 nozzle to USAF Spec MIL-N-25161. This company has recently investigated the installation of a flight refuelling system on the C. 105 aircraft. The nozzle would be installed in the side of the forward fuselage, giving it a short connection to the fuel cells.

10.10.4 STARTER SYSTEM AND CONTROLS

The starting system is pneumatic, the power source, and the controls, being external. The engine starter is housed in the engine bullet.

10.10.5 INSTRUMENTS

All instruments are installed in accordance with Spec. MIL-I-18373 (Aer) where applicable.

10.10.6 HYDRAULIC AND PNEUMATIC SYSTEM

The hydraulic reservoirs, controls, and pneumatic system (used for emergency operations only) are all housed in the aft equipment bay, conveniently centralized for all parts to be actuated, and ease of servicing. The parts actuated by the hydraulic system include the landing gear, the engine jet deflectors, armament pack, speed brakes, and all control surfaces.

10.10.7 ELECTRICAL

The electrical equipment, including alternator and storage batteries, is housed in the accessible aft equipment bay. Space is available for the alternate installation of an electrical primary system having 50 per cent excess power, and there is space, in the aft equipment bay, for the alternate installation of an electrical power-conversion system having 100 per cent excess power.

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10.10.8 RECEPTACLES

External power receptacles are provided. A fuel nozzle grounding receptacle, AN 3117, is provided. All such installations are in accordance with Spec. MIL-E-7080 and Dwg. AND 10439.

10.10.9 INSTRUMENT ELECTRICAL SYSTEM

Each instrument is individually connected to the power source by means of a fuse or circuit breaker to isolate the instrument in the event of a fault. A warning device is provided to show loss of power to essential flight instruments, and manual transfer is provided.

10.10.10 ELECTRONICS

A great deal of electronic equipment is installed in the aircraft, the positioning of that equipment being decided by the parts to be served by that equipment. The various packs of equipment are easily accessible, and are so grouped that the interconnecting cables are of minimum length, and therefore of minimum weight.

The AN/ARC-52 and AN/ARN-21 Communication and Navigation equipment are housed in the cockpit, in the aft canopy region, and in the forward equipment bay, depending upon the size and function of the individual items. The controls, and some indicators are in the cockpit, and the transmitter, network, and ancillary equipment are housed in the other two easily-accessible areas.

The weight and space analysis allowed for the heavier AN/ASQ-19 Integrated Electronics Central Equipment to replace the above items, and it was assumed that this would be used in the production aircraft. It would be stowed in the space previously allocated for the AN/ARC-52 and AN/ARN-21 equipment.

The AN/APS-67 search range radar is stowed entirely in the nose fuselage, forward of the cockpit, with all necessary controls and indicators situated in the cockpit. The nose fuselage, containing this equipment, is completely removable for ease of servicing and replacement of faulty parts.

The automatic stabilizing system, and the auto pilot, are housed in the cockpit, and the forward equipment bay, the controls and indicators being in the cockpit, and the rest of the equipment being in the equipment bay.

10.10.11 ARMAMENT CONTROL SYSTEM

The ACS Aero x 29A armament control system, consists of the AFCS Ex-16 and the AN/APS-67. The positioning of the radar has already been discussed in the above paragraphs. The remainder of the armament control system is housed very conveniently and efficiently in the easily-accessible space immediately alongside the armament bay, thus being at a minimum distance from the armament itself.

10.10.12 AIR CONDITIONING AND ANTI-ICING SYSTEM

The cabin is pressurized and air-conditioned in accordance with Spec. MIL-P-18927 (Aer), the necessary equipment being housed in the easily-accessible and adjacent forward equipment bay.

The only anti-icing equipment installed is that required for defrosting and defogging the cockpit enclosure. This is installed in accordance with Spec. MIL-P-18927 (Aer).

10.10.13 FIRE DETECTION AND EXTINGUISHING SYSTEM

The above systems are provided in accordance with relevant specifications.

10.11. FURNISHINGS

10.11.1 PILOT'S SEAT

An ejection seat is provided in accordance with Spec. MIL-S-1847 (Aer), as applicable.

An integrated parachute safety harness system is provided, as shown on NPU Dwg. X53-130, using a 28 ft. standard flat canopy in pin/cone conventional Pioneer back pack.

As mentioned in paragraph 10. 2, consideration has been given to the future alternate installation of an ejectable seat capsule.

10.11.2 ANTI-BLACKOUT SUIT PROVISION

Space and weight have been allocated for the provision of an anti-blackout suit.

10 11.3 PRESSURE SUIT

Air conditioning and pressurization of the pilot's pressure suit is provided in accordance with Spec. MIL-P-18927 (Aer). A GFE male receptacle half of the composite disconnect D-600 is installed on the left hand side of the ejection seat. All ancillary lanyards, tubing, hoses, brackets and so on are provided.

10.11.4 INSTRUMENT GLARE SHIELD

Space and weight have been allocated for the provision of an instrument glare shield.

11.0 OPERATION, HANDLING AND SERVICING

11.1 INTRODUCTION

The operating, handling and servicing of a naval aircraft are closely inter-related subjects and they are therefore dealt with together in the more general parts of this section.

The aim throughout has been to achieve the utmost simplicity and economy consistent with meeting the requirements.

It is assumed that, in order to obtain operational experience on V. T. O. L. aircraft, it would be of considerable advantage if existing aircraft carriers could be adapted to operate the V. T. O. L. aircraft concurrently with conventional naval aircraft. In such a composite ship, the V. T. O. L. complement would presumably consist of fighter or reconnaissance aircraft, and the conventional aircraft complement would consist of strike and/or early warning aircraft.

The proposed adaptation of the ship to permit V. T. O. L. operations could be made with the minimum expenditure of time and money and would not interfere with the operation of conventional aircraft.

Detailed proposals for the adaptation of other types of ships, including merchantmen, for V. T. O. L. aircraft operation could be made the subject of a separate study if required.

11.2 DISCUSSION OF PROBLEMS

11.2.1 SHIP MOVEMENT

The roll and pitch criteria given in Specification T. S. 140 make satisfaction of the landing and take-off requirements extremely difficult.

The provision of a platform stabilized in pitch and roll rotations only would not solve the problem, since the heaving motion and the horizontal translational motions (due to the height of the platform above the centres of pitch and roll) would remain unstabilized. Of these motions, the lateral translational motion due to roll is likely to prove the most embarrassing to the pilot. Yawing rotation of the ship can be neglected, as can the fore-and-aft translation due to pitch, but it might still be necessary to design a platform having four degrees of stabilization, viz: roll and pitch rotation plus vertical and lateral translation. Such a platform would be extremely heavy and complicated.

The design and construction of stabilized retrieving gear present similar problems, and the resulting complication, and rigidity imposed upon operations, are such as to rule out such methods as practical solutions, unless the ship and its machinery are specifically designed for V. T. O. L. aircraft operation.

The vertical motion due to pitch and heave can be minimized by locating the landing area in the vicinity of the null "line" on the deck. The major problems would then arise from the rolling motion of the ship. In the interests of simplicity and flexibility of operation, it is postulated that the pilot would only land during that portion of the rolling cycle which is found to be most favourable. At night the pilot could be given a visual or audio signal, controlled by a roll-sensing unit, to indicate the attitude and proximity of the deck.

Such a scheme of "free" landing would also have psychological benefits - no pilot likes to think that he is dependent on someone else's control while still airborne.

The main problem then is to supply the aircraft with adequate means of absorbing the vertical component of relative kinetic energy, and to provide means to stop the aircraft from sliding on deck or overturning.

Methods involving the use of simple deck equipment to render any sliding motion after touch-down "dead beat" have been considered, but have not been included in this brochure for simplicity.

During landing, the landing area will have been burned free of ice, water or patches of kerosine, and the normal dry coefficient of friction between the deck and the aircraft feet will be unimpaired. (This does not, however, apply to an aircraft spotted ready for take-off, or being moved along the pitching and rolling deck).

It is suggested, therefore that the coefficient of friction between the deck and the aircraft feet should be as high as possible, and the friction developed used to damp out any sliding motion. An investigation of the dynamics of the landing problem, and tests to determine the coefficients of sliding friction between various combinations of practical surfaces, have not been completed to date, but it is believed that the problem can be resolved without recourse to special arresting equipment. It is worth noting that the C.G. of this aircraft is very low compared with that of a tail sitter or helicopter, and the possibility of over-turning after touch-down is thereby reduced.

The problems of take-off are similar to, but of a lesser degree than, those of landing, except in one important respect: the coefficient of friction between the aircraft and the deck may be considerably reduced by the presence of ice, water, etc. Methods for preventing sliding under these conditions are suggested in Para.11.5.2.

11.2.2 MOBILITY

The aircraft is not provided with its own wheels. This considerably benefits the aircraft design, but leaves it immobile on deck. It is thus essential that it can be made mobile in the shortest possible time after landing and remain mobile until required again for take-off or parked and secured.

11.2.3 SERVICING

The modern fighter requires to be kept at immediate readiness for long periods. It must have the shortest possible turn-round time between missions. Its design must permit speedy replacement of engines, avionics, and other major and minor components, in order to ensure maximum availability for combat duty when operating from advanced bases, be they ashore or afloat.

11.3 RESOLUTION OF PROBLEMS

11.3.1 SITING OF TAKE-OFF AND LANDING AREAS

Since no stabilized platform or retrieving gear is employed, these areas should, ideally, be at the position of least motion on the flight deck, the landing area taking precedence. The part of the flight deck which has least vertical motion during heavy roll, pitch, and heave, is usually the area abaft the island superstructure and about 2/3 of the length of the ship from the bows and near the centre-line. Unfortunately the use of most of this area is precluded by the operation of conventional aircraft on the angled deck. The take-off area could be extended from the starboard edge of the deck in this region, as shown in Fig. 62, but when normal aircraft were being operated it would then be in the turbulent airflow in the wake of the island superstructure. However, this turbulence should not prevent the use of this area when it is required to launch a number of V.T.O.L. aircraft at short intervals. This auxiliary take-off area could also be used when green seas were breaking over the forward part of the flight deck, since conventional aircraft would not be operated in such weather, and it would also suffer less from ship motion.

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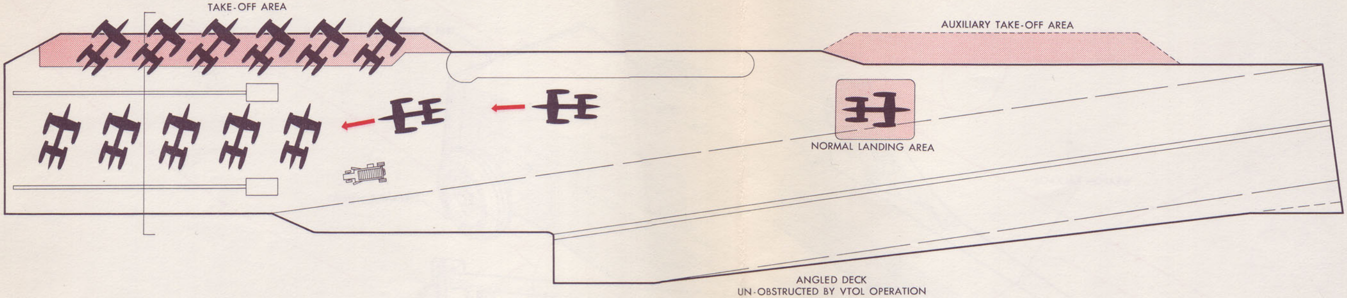

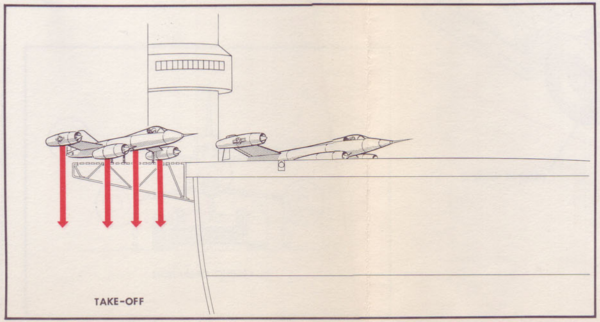

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FIG. 62 OPERATION FROM CVA

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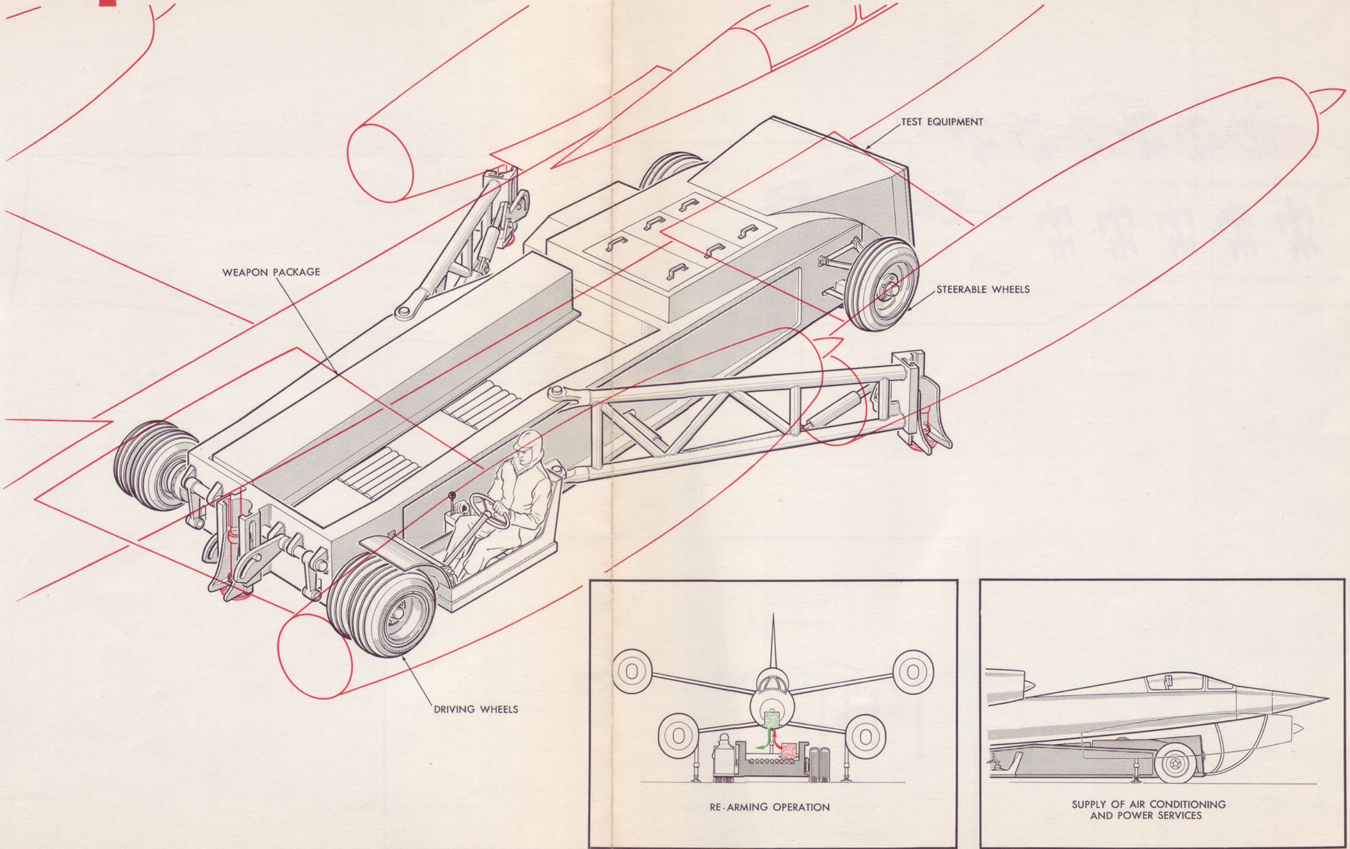

FIG. 63 TENDER VEHICLE

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The main take-off area could be situated forward of the island superstructure, (see Fig. 62). Since both take-off areas suggested are additions to the present flight deck area, they could be built as gratings, and the jet efflux could pass harmlessly over the ship's side.

11.3.2 MOBILITY

Various forms of trolleys and jury undercarriages have been considered but all suffer from a major disadvantage, viz: the time taken to make the aircraft mobile after landing is far too long.

A special Tender Vehicle has been designed which will combine with the aircraft to make a single self-propelled unit. Its special tires will give it maximum traction on a wet and sloping deck.

This vehicle is intended for shipboard use or for advanced bases ashore. In the hangar or on airfields ashore, the aircraft could be made mobile by fitting the jury undercarriage described in Para. 11.4, and tail towing by a normal tractor employed. The jury undercarriage could also be used in the hangar of a ship for spotting and for side-tracking.

11.3.3 SERVICING

The Tender Vehicle, in addition to lifting and moving the aircraft, supplies it with all the services required to keep it at immediate readiness. It also carries the equipment and stores needed during "turn-round", with the exception of fueling and engine starting.

11.3.4 ADVANCED BASES ASHORE

The Tender Vehicle can reasily be adapted so that it can be air-lifted by the V.T.O.L. aircraft and taken to the advanced base. For this mission the aircraft would carry no weapons and fuel would be reduced to the minimum required for the one-way trip.

11.4 <u>SPECIAL HANDLING EQUIPMENT</u>

11.4.1 INTRODUCTION

The special handling equipment proposed is integrated closely with the aircraft design and aims at:-

(a) safe and speedy movement of the aircraft under all weather conditions and with regard to the pitch and roll criteria of the specification.

(b) maintaining the aircraft at "immediate readiness" while parked in the take-off area awaiting the "scramble".

(c) making the aircraft mobile in the shortest possible time interval after touch-down.

(d) reducing the turn-round time between sorties to a minimum.

(e) dispensing as far as possible with the use of man-power for handling, thus easing the complement, accommodation, and victualling problems in the ship.

11.4.2 FUNCTION OF THE TENDER VEHICLE

The Tender Vehicle proposed is designed to perform the following functions:-

(a) It provides the aircraft with an undercarriage and makes it self-propelled on deck,

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cutting the ground handling time to an absolute minimum.

(b) It stays with the aircraft during the state of immediate readiness until immediately before take-off and supplies cockpit heating or refrigeration, weapon heating or cooling, electrical power for pre-running gyro instruments and pre-heating electronics, etc.

(c) It provides equipment and stores for all pre-flight and between flight servicing operations on the aircraft - such as re-arming, exchanging or re-charging oxygen and air bottles, avionics and weapon circuit testing, etc. - with the exceptions of re-fueling and engine starting (Fuel and pressure air would, presumably, be piped from the ship's systems).

The Tender Vehicle is a combination of tractor and fork-lift truck. It is designed to move in under the belly of the aircraft from ahead, engage its lifting forks with the aircraft feet, and lift them to a height which provides sufficient ground clearance. The Tender Vehicle can then move the aircraft in any direction required.

The T. V. would spot the aircraft and stay with it until the aircraft was parked and secured, or until immediately before take-off. After take-off it would stand-by in readiness for the return of its aircraft to the ship, unless required for another job.

If its aircraft is serviceable on landing, and required for quick turn-round, the T. V. is used for between-flight servicing. Otherwise, it would take its aircraft to the deck park or down the elevator into the hangar.

When not at flying stations, the T. V. 's could be stowed beneath the aircraft in the hangar and thus take up no additional space.

11.4.3 DESCRIPTION OF THE TENDER VEHICLE

As previously stated, the Tender Vehicle is a combination of tractor and fork-lift truck. However, unlike a normal tractor, this vehicle carries the weight on its back instead of dragging it behind and thus ballast weights on the chassis frame, to permit it to develop tractive effort, would not be required. Unlike a normal fork-lift truck, the Tender Vehicle would not require ballast weights in order to prevent it from tipping.

Thus the Tender Vehicle design should attempt to save weight wherever possible, especially if it is to be transported by air, using the V. T. O. L. aircraft as an aerial crane. It must, however, be rugged enough to do its job with adequate safety factors, and to withstand the normal use and misuse of service conditions.

Provision could be made to carry ballast weights and towing attachments should it be required to use the Tender Vehicle as a normal tractor for towing other types of aircraft.

The steering wheels have a large lock to permit the vehicle to manoeuvre in confined areas.

The vehicle incorporates full provision for re-arming the aircraft. It carries a re-fill weapon package to the aircraft and has the necessary machinery to facilitate re-arming. The empty weapon package is lowered from the aircraft into the tender vehicle and released. It is then moved sideways and the loaded weapon package moved simultaneously into place beneath the armament bay. It is then secured to the aircraft and hoisted into the stowed position.

The avionic and weapon test equipment is mounted behind the engine compartment of the vehicle, and when the vehicle is in place beneath the aircraft this equipment is conveniently located below the cockpit.

The air-conditioning unit, the compressed air and oxygen charging equipment, and other services, are located adjacent to the engine.

The driver sits within the wheel base and alongside the re-arming bay, from which position he has the best possible view along the deck to the rear of the aircraft and sideways under the rear wing and engine pod. The aircraft is normally moved in a tail first direction by the tender vehicle. (See Fig. 63).

11.4.4 FUNCTION OF THE JURY UNDERCARRIAGE

The jury undercarriage is intended for use ashore or afloat when it is desired to make the aircraft mobile but speed of handling is not of prime importance. Nevertheless, the jury undercarriage is specially designed for speed and ease of fitting to the aircraft, although when fitted, normal tractor towing is required.

The jury undercarriage can be used for side-tracking the aircraft for ease of stowage in the hangar, and its use as an auxiliary to the Tender Vehicle would permit a smaller number of the latter to be carried in a ship.

11.4.5 DESCRIPTION OF THE JURY UNDERCARRIAGE

The jury undercarriage proposed consists of three basically similar units, each being a tricycle type bogie, one wheel of which is steerable when required.

Each bogie contains a slipper which engages the aircraft foot (in a similar manner to the forks of the tender vehicle) and lifts it by the extension of two strong coil springs. Before fitting the bogie to the aircraft, these springs are compressed by hydraulic cylinders, (see Fig. 64), the hydraulic pressure being released when it is desired to lift the aircraft.

The bogies which are fitted to the two front legs of the aircraft have telescopic steering arms. These arms cannot be lifted above the horizontal until the steering is centered. When the steerable wheel is centered and the arm is raised above the horizontal, the steering becomes locked. The end of the steering arm is then inserted in a socket in the engine nacelle and held in place by a pip pin.

The steering arm of the bogie which is fitted to the rear leg is extended and attached to the tractor. The non-steerable wheels of the bogie are fitted with brakes, operated by the closure of the telescopic arm whenever the aircraft tends to over-run the tractor.

The aircraft is lifted by simultaneous release of the hydraulic pressure in all three units, and the aircraft can then be towed away tail first.

When it is required to remove the jury undercarriage, the lifting springs are compressed by closing the release valves and using the hydraulic hand pump fitted to each bogie.

11.5 OPERATION OF THE AIRCRAFT

11.5.1 NORMAL OPERATION FROM SHIPS

For take-off, the V.T.O.L. aircraft could be ranged on gratings on the starboard side of the flight deck (See Fig. 62). The outer portion of these gratings could be hinged, if necessary, in order to facilitate docking of the ship and its passage through canals. The grating area would provide space for twelve aircraft to be ranged for take-off, the next detail of six aircraft being ranged in the area between the catapults, and ready to fill the gaps as soon as the first detail has been launched.

Thus, eighteen V.T.O.L. aircraft could be launched at very short intervals and during their take-off, it would still be possible to launch conventional aircraft by free take-off, or to land-on four or five conventional aircraft and park them in line astern along the port catapult.

The Tender Vehicle would remain with its aircraft until the signal to start engines, when it would withdraw and bring up another aircraft.

After starting up, the un-deflected jets would be directed horizontally over the ships' side and after completion of cockpit checks they would be deflected to the vertical position immediately before take-off.

If conventional aircraft were also being operated, the relative wind at take-off would be fine on the port bow of the angled deck and, after take-off, the V.T.O.L. aircraft would fall astern outboard of the ship and climb to transition height.

Should low fuel states, or urgent tactical requirements, make it imperative to recover the V.T.O.L. aircraft in the shortest possible time, vertical landings could be made anywhere on deck, provided that the landing sites were kept clear of other aircraft and personnel, since no special retrieving gear is used.

The deflected jet nozzles would only be in close proximity to a small area of the steel deck for a very few seconds, and the engines would be stopped immediately after touch-down, since the use of the tender vehicle eliminates the need for the aircraft to taxi under its own power. It is anticipated that the ejector-type mixing of the jet streams with the much cooler surrounding air would be such that no damage would ensue to the aircraft or to the deck. The minimum distance required between two or more aircraft landing simultaneously could only be determined with any accuracy by actual trials.

When the aircrafts' engines are stopped, the tender vehicle would drive in, lift the aircraft in a matter of seconds, and remove it to the deck park or below to the hangar.

11.5.2 ROUGH WEATHER OPERATIONS

11.5.2.1 Take-Off

While waiting to take-off, the aircraft are prevented from sliding on the gratings by adaptor plates which are slipped under the aircraft feet as they are lowered. These plates have eccentric spigots and can be turned so as to engage with the holes in the gratings without the need for pre-alignment of the aircraft with the gratings. (See Fig. 65). The plates stack inside one another when not in use.

For take-off, the pilot would anticipate the most favourable attitude of the deck before opening up to take-off power. The adaptor plates would be left behind on the gratings as the aircraft left the deck, and could be used again for the next aircraft.

When the ship is rolling up to ± 10°, the operation of normal aircraft would be impossible. Under these conditions, the V.T.O.L. aircraft could use the gratings abaft the island for take-off, as already discussed in Para. 11.3.1.

11.5.2.2 Landing

For rough weather landing, the pilot would hover above the center of the selected landing area and make his touch-down during the period in which the deck is in the most favourable attitude. The aircraft feet, being universally jointed, would make full contact with the deck and the friction developed would absorb any relative kinetic energy due to drift, and bring the aircraft

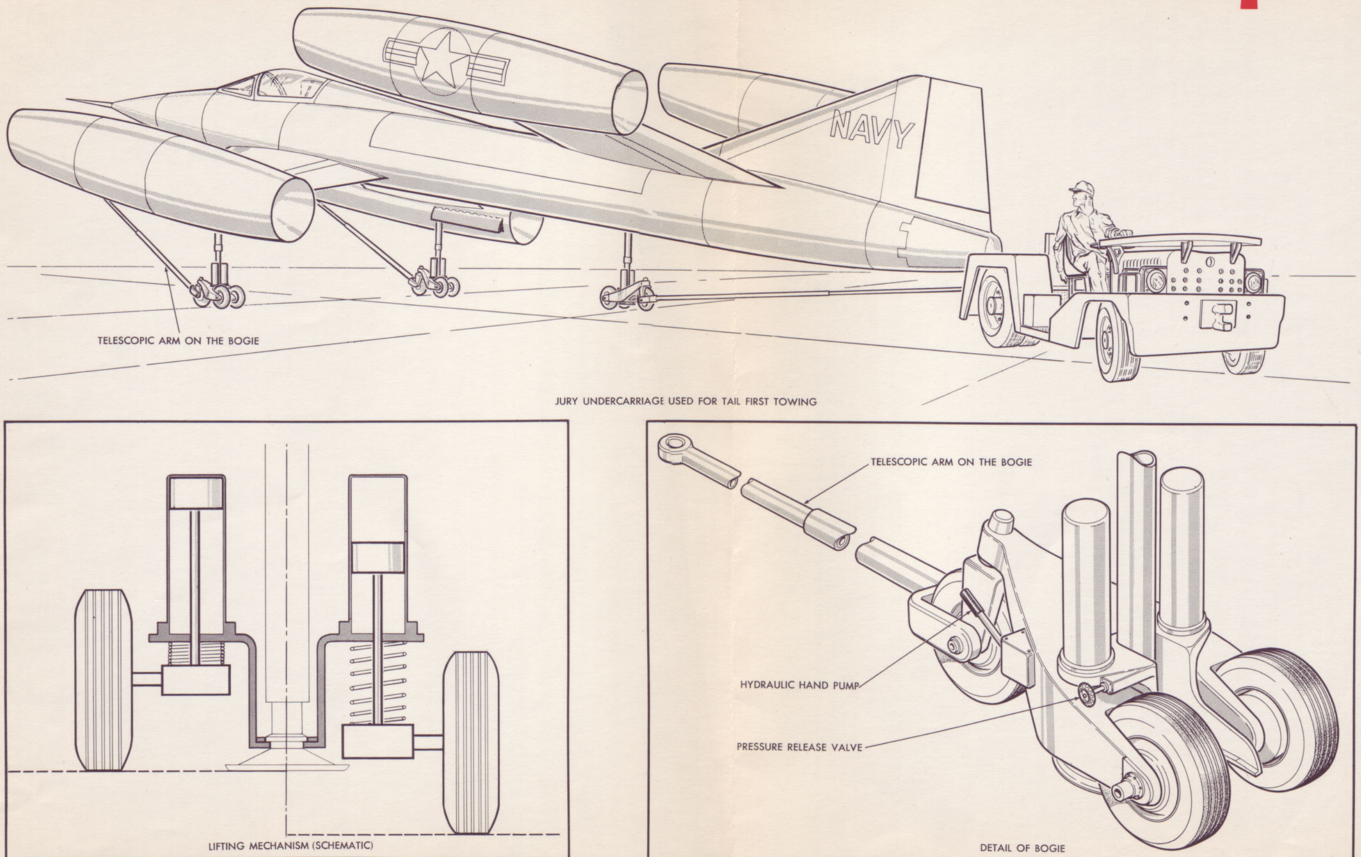

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FIG 64 JURY UNDERCARRIAGE

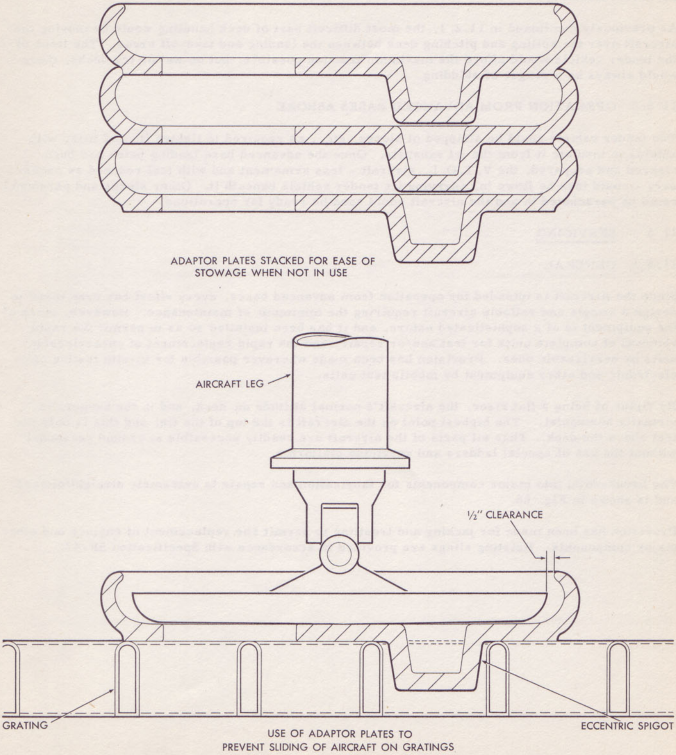

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FIG.65 ROUGH WEATHER OPERATION TAKE-OFF

to rest. The engines are stopped and the tender vehicle could then move in and pick up the aircraft.

11.5.2.3 Handling

As previously mentioned in 11.2.1, the most difficult part of deck handling would be moving the aircraft over the rolling and pitching deck between the landing and take-off areas. The tires of the tender vehicle would afford the maximum traction possible, but on wet or icy decks, there would always be a danger of skidding.

11.5.3 OPERATION FROM ADVANCED BASES ASHORE

The tender vehicle could be stripped of stores, etc., as required to lighten it, and fitted with shields to insulate it from the jet exhausts. Once the advanced base landing patch had been cleared and prepared, the V.T.O.L. aircraft - less armament and with fuel reduced as necessary - could then be flown in, carrying its tender vehicle beneath it. Other stores and personnel could be parachuted in and the aircraft could soon be ready for operations.

11.6 SERVICING

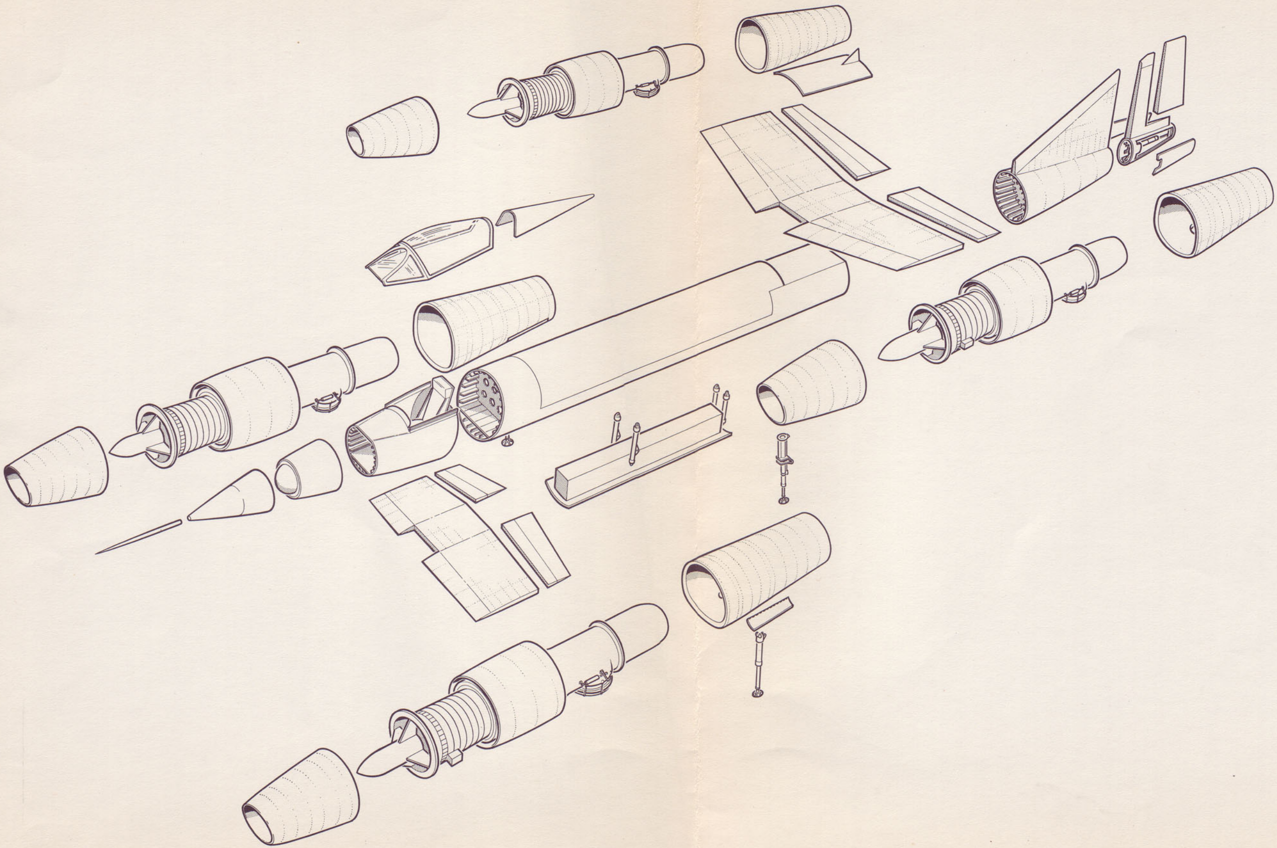
11.6.1 GENERAL

Since the Aircraft is intended for operation from advanced bases, every effort has been made to design a simple and reliable aircraft requiring the minimum of maintenance. However, much of the equipment is of a sophisticated nature, and it has been installed so as to permit the rapid removal of complete units for test and/or repair, and the rapid replacement of unserviceable units by serviceable ones. Provision has been made wherever possible for in-situ testing of electronic and other equipment by mobile test units.

By virtue of being a flat riser, the aircraft's normal attitude on deck, and in the hangar, is virtually horizontal. The highest point on the aircraft is the top of the fin, and this is only ten feet above the deck. Thus all parts of the aircraft are readily accessible to ground personnel without the use of special ladders and servicing platforms.

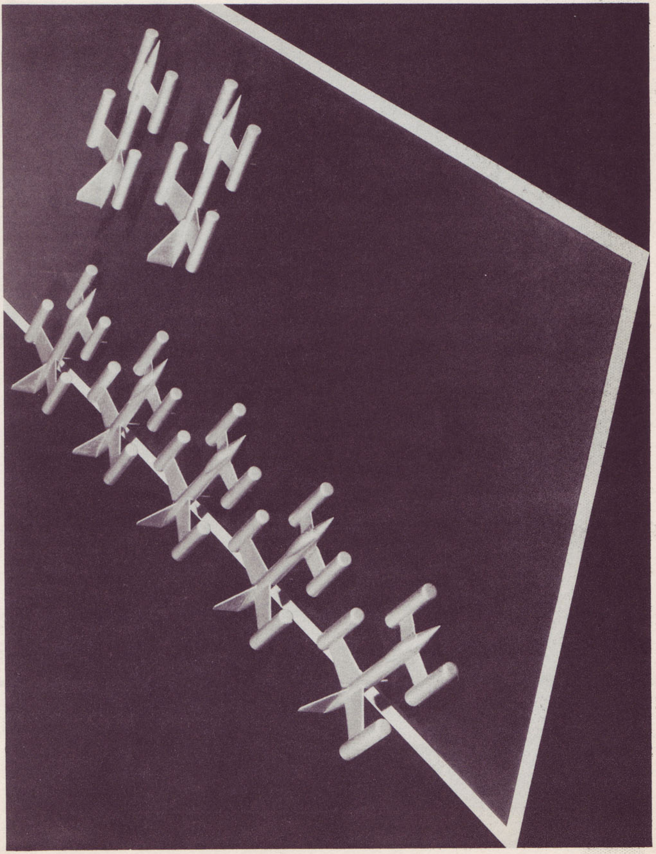
The break-down into major components for fabrication and repair is extremely straightforward, and is shown in Fig. 66.

Provision has been made for jacking and trestling to permit the replacement of engines and other major components. Hoisting slings are provided in accordance with Specification SR-47.



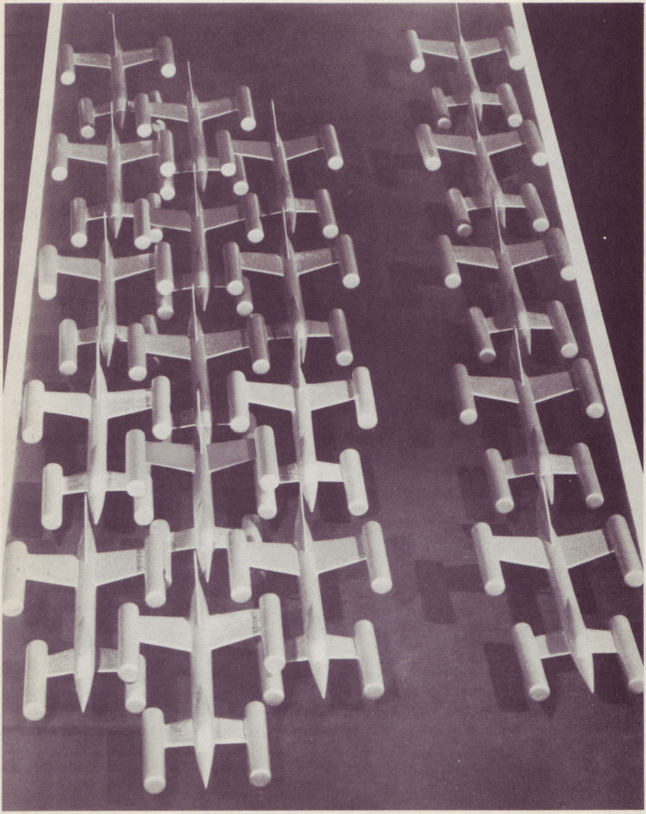
1879-VTOL-1-1

FIG 66 STRUCTURE BREAKDOWN



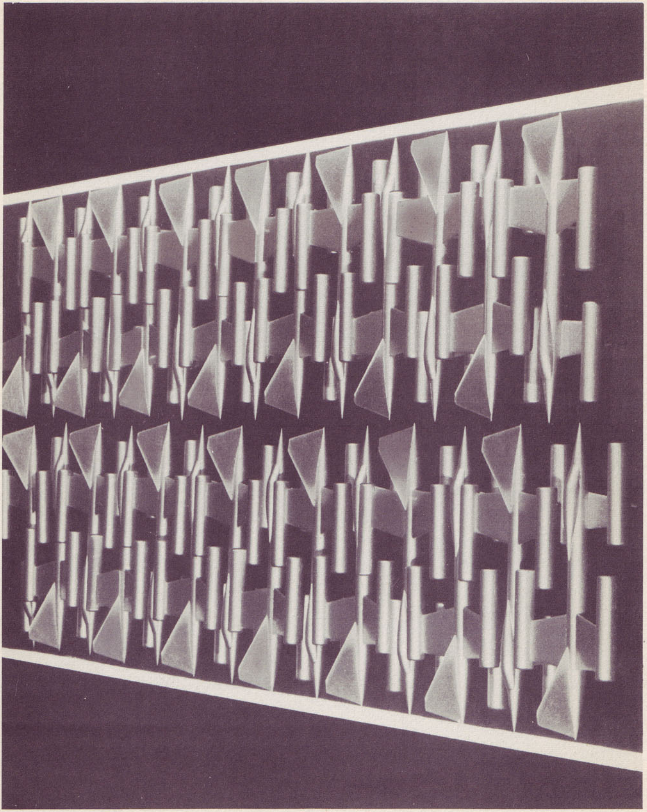
1936-VTOL-1

FIG. 67 AIRCRAFT RANGED FOR TAKE-OFF FROM FORWARD AREA



1937-VTOL-1

FIG. 68 HIGH DENSITY STOWAGE FOR OPERATIONAL USE. 28 AIRCRAFT CAN BE STOWED IN A DECK AREA 100 FT. BY 320 FT. LEAVING AN ADEQUATE GANGWAY FOR REMOVAL



1938-VTOL-1

FIG. 69 HIGH DENSITY STOWAGE FOR FERRYING.
28 AIRCRAFT CAN BE STOWED IN A DECK AREA 100 FT. BY 230 FT.

AVRO VTOL

12.0 ONE YEAR DEVELOPMENT PROGRAM

12.1 GENERAL

A one year development program is proposed, in accordance with specified requirements, to prove the feasibility and practicability of the chosen aircraft configuration. Being an unconventional type of aircraft, it has a number of unique features which must be proved before they are incorporated in a final design. In this program the parts involved will be designed, built, and tested.

12.2 FEATURES TO BE PROVEN

The salient features to be proven are as follows:-

(a) The feasibility and efficiency of jet-stream deflection in one plane through an angle of 84 degrees.

(b) The feasibility and efficiency of jet-stream deflection by means of eyelids within a cone of 12 to 15 degrees.

(c) The available thrust from one Orpheus engine fully mounted in a pod, including the proposed air intake, and fully representative jet pipe, with jet deflection arrangement and controlling eyelids.

(d) Rate of change of thrust, with both straight flow and deflected jet, which it will be possible to obtain with engine controls. This information will be essential to prove the controllability of the aircraft in hovering and transitional flight.

(e) Controllability of the aircraft in hovering flight when using a deflected jet with operating eyelids in conjunction with thrust variation.

12.3 TESTING

The feasibility of deflecting a jet through 90 degrees has already been proven, to a degree, in a series of initial tests carried out on a simplified model described in Para. 9.0. More representative models are being planned. They will be designed and tested to find the best configuration for the deflection valve chamber, and will incorporate a system of vanes and a movable nozzle. This will help to speed up the design of a full scale jet deflector planned for further tests.

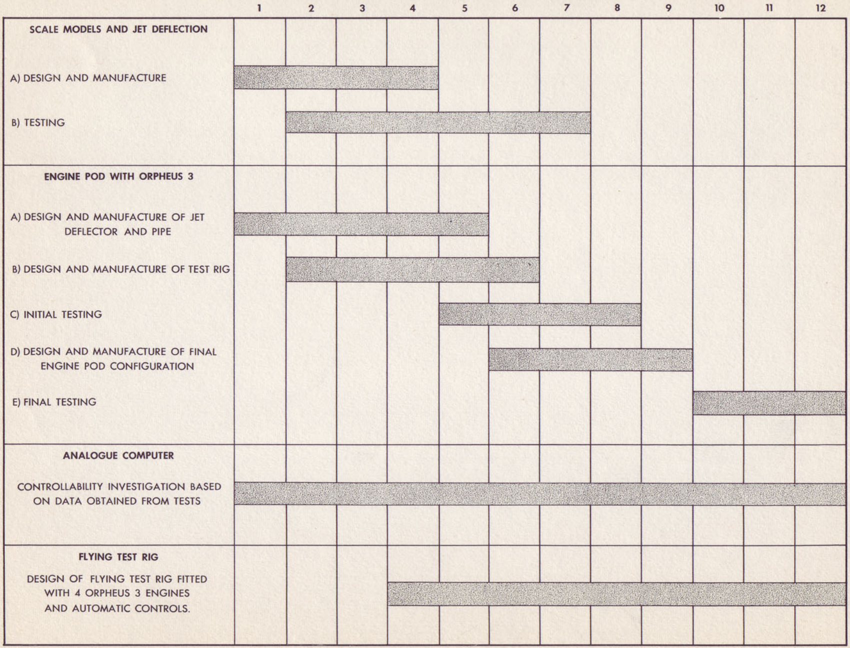
It is proposed that all full scale tests be carried out with an Orpheus 3 engine, this being immediately and readily available.

An Orpheus 3 engine will be mounted and tested, in a rig representing, as accurately as possible, the final engine pod. By this means the efficiencies of an installed engine with a straight jet-flow, as well as a deflected jet with movable eyelids, will be found. During these tests the rate of change of thrust, (the response of thrust to the movements of the throttle controls) will be invesitaged and measured. The basic information on rate of change of thrust will then be used for a more extensive analogue computer investigation of aircraft controllability. This should confirm, and enlarge upon, the initial analogue computer results described in Para. 7.0.

When the above tests have been successfully completed, the next logical step is to design, and build a flying rig. This will be equipped with four Orpheus 3 engines mounted in representative pods, their relative positions being the same as on the proposed aircraft, both in plan view, and in their distance from the ground. The flying rig will be operated by a test pilot and will dynamically simulate the proposed aircraft.

The automatic electronic controls required for hovering flight will be tested on this rig, it being ideally suited to this purpose. In spite of the face that the design and manufacture of the flying rig cannot be completed, and tested, within the first year, most of the work will be done in that year, and the design is therefore included in this program.

ONE YEAR DEVELOPMENT PROGRAMME - AVRO VTOL AIRCRAFT



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APPENDIX 'A'

FACILITIES AND EXPERIENCE

The closely integrated subsidiaries of A. V. Roe Canada Limited are unique in their ability to carry out all the design, research, manufacturing, and development on V. T. O. L. aircraft.

The close daily relationship between engine and airframe development teams is advantageous from the standpoint of cost and lead time required to successfully complete this project.

The facilities of the Orenda Engine Company, as a joint team, are available to Avro Aircraft and would be a great asset to this project. In addition to development work on the jet deflection mechanism, the Orpheus engine could be manufactured by Orenda Engines.

Relevant brochures describing facilities and experience of our companies are being enclosed with this brochure.

APPENDIX 'B'

AREA RULE

The area rule theory has been applied to this aircraft at Mach No. 1.32 and the results compare very favourably with the C_{D_W} as calculated by other formulae.

The area rule theory has been applied using Mr. R. T. Whitcomb's method. A twenty term Fourier series was used and the results were obtained from an I. B. M. electronic computing machine.

Five separate cuts were made through the aircraft at angles of rotation of $\theta = 0^o$, 45^o, 90^o, 135^o, and 180^o to give a good average curve, and a further cut was made at $\theta = 12\ 1/2^o$ to substantiate the trend of the curve at this angle.

Using standard N. A. C. A. practice of arriving at the total A/C drag, the results show that we have a wave drag as calculated by area rule of

C_{D_W} = .0568 plus subsonic drag

C_{D_S} = .0338 for a total drag

C_{D_O} = .0906

this compared to a total drag as calculated by other methods (see Para. 6) of C_{D_O} of .0920.

Further work is being done to reduce the slope of the area curves and it is felt that a substantial reduction is still potentially available with slight modifications to the fuselage and nacelles.

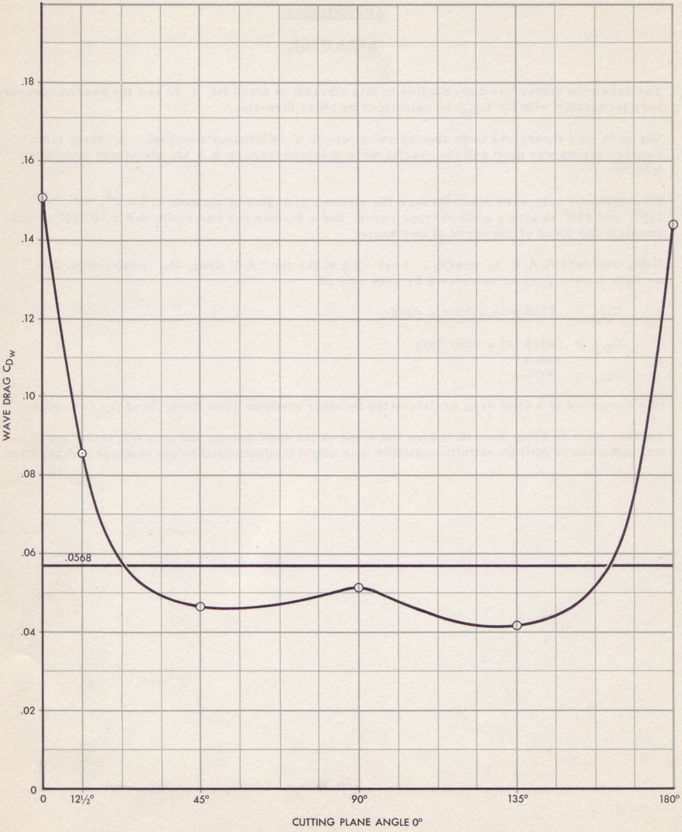

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FIG. 70 AREA RULE WAVE DRAG C_{D_W}

APPENDIX 'C'

ESTIMATED TIME CONSTANT OF ORPHEUS ENGINE

According to Reference 2, the turbojet can be considered a linear first order system in which the speed response to a step change in fuel flow is defined by the time constant

$$t = \frac{I}{K_q N_f}$$

Where I = polar moment of inertia of rotor

$$K_q = \frac{JC_P}{2\pi}\left(\frac{W}{N}\right)\left(\frac{\Delta T_c}{N^2}\right)$$

N_f = equilibrium RPM after step fuel flow change

N = RPM

J = 1400 ft-lb./CHU

C_p = specific heat of air at constant pressure

W = engine air flow

ΔT_c = compressor temperature rise

To obtain the polar moment of inertia of the engine rotor the following relationship is used,

$$\text{Gyroscopic torque} = I\, W \Omega$$

Where W = angular velocity about primary axis

Ω = precessional angular velocity

$$W = \frac{2\pi}{60}(9660) = 1011 \text{ radians/sec.}$$

Ω = 1 radian/sec.

$$I = \frac{2780}{1011} = 2.75 \text{ lb-ft-sec}^2$$

To obtain the compressor temperature rise assume an adiabatic efficiency of 0.85

$$T_c = \frac{T_1}{\eta_c}\left[R^{\frac{\gamma-1}{\gamma}} - 1\right]$$

Where T_1 = engine inlet absolute temperature

η_c = compressor adiabatic efficiency

R = compressor pressure ratio

γ = 1.40 for air

For a standard day, $T_1 = 288^\circ K$

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$$\Delta T_c = \frac{288}{0.85}\left[4.36^{0.286} - 1\right]$$

$$= 177^{\circ}C$$

For small changes, $N_f \approx N \approx 9660$ RPM

$$K_q = \frac{1400(0.24)}{2} \quad \frac{81.5}{9660} \quad \frac{177}{(9660)^2} \quad (60)^3$$

$$= 0.185 \text{ lb-ft-sec}^2$$

The time constant at 9660 RPM on a standard day is therefore

$$t = \frac{I}{K_q N_f}$$

$$= \frac{2.75(60)}{0.185\,(9660)} = 0.0923 \text{ second}$$

Summary

The time constant estimated by the NACA method for the response of the Bristol Orpheus engine to fuel flow changes is approximately 0.1 second. Although this is the RPM response, the desired thrust response will be nearly the same. Since the thrust during an acceleration will always be greater than that obtained by reading a steady-state Thrust vs. RPM graph, the thrust response is expected to be slightly faster than the RPM resonse.

References:

(1) NACA Report 1011 "Dynamics of a Turbojet Engine considered as a Quasi-Static System".

(2) NACA RM E51K21 "A Method of Estimating Speed Response of Gas Turbine Engines".

(3) NACA TN 2642 "Application of Linear Analysis to an Experimental Investigation of a Turbojet Engine with Proportional Speed Control".

(4) NACA TN 2826 "Simulation of Linearized Dynamics of Gas-Turbine Engines".

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