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PROJECT Y2

DEVELOPMENT PROPOSAL

A. V. ROE CANADA LIMITED

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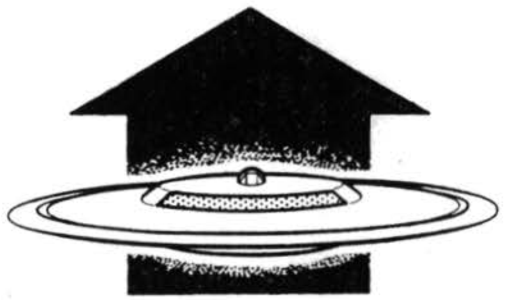


PROPOSAL

PROJECT Y2

DESIGN AND DEVELOPMENT PROGRAM

JUNE 1954 - JUNE 1956

A. V. ROE CANADA LIMITED

AIRCRAFT DIVISION

MALTON - ONTARIO

PROJECT Y-2 VERTICAL TAKE-OFF

SECRET

INTRODUCTION

World-wide study in the field of aircraft research is currently directed to the possibilities of a supersonic, gas turbine powered aircraft having a thrust/weight ratio greater than one. Using jet-lift, such an aircraft would be capable of vertical take-off and landing on unprepared bases, and free-air hovering, characteristics usually associated with the slow moving helicopter. The advantages of an aircraft combining these qualities with supersonic performance are obviously great. Furthermore a gas turbine powered aircraft having a very high thrust/weight ratio will also reap the performance benefits of faster climb, higher ceiling and improved supersonic economy.

The basic idea of the original Project Y aircraft was to rearrange a large gas turbine power plant for radial flow instead of axial flow, thereby providing an engine of thrust/frontal area far in excess of that obtainable with the conventional layout; it also provided gyroscopic stability and simplicity of manufacture. The radial flow power plant takes in air at the centre and distributes the jet outwards to the periphery and in the Project Y2 aircraft this characteristic is applied to produce the remarkable ground cushion effect.

From the military point of view, the radial flow engined aircraft now proposed can be visualized in roles as varied as pursuit, reconnaissance, ground support, ambulance, bomber or transport.

A separate brochure has been prepared describing in detail the basic Y2 aircraft. The work which has already been undertaken and the results of the development to date are covered in technical detail in a number of reports.

This proposal outlines a program of further design, test and development considered necessary to continue with the project and to produce a successful aircraft, together with the estimated costs of such a program.

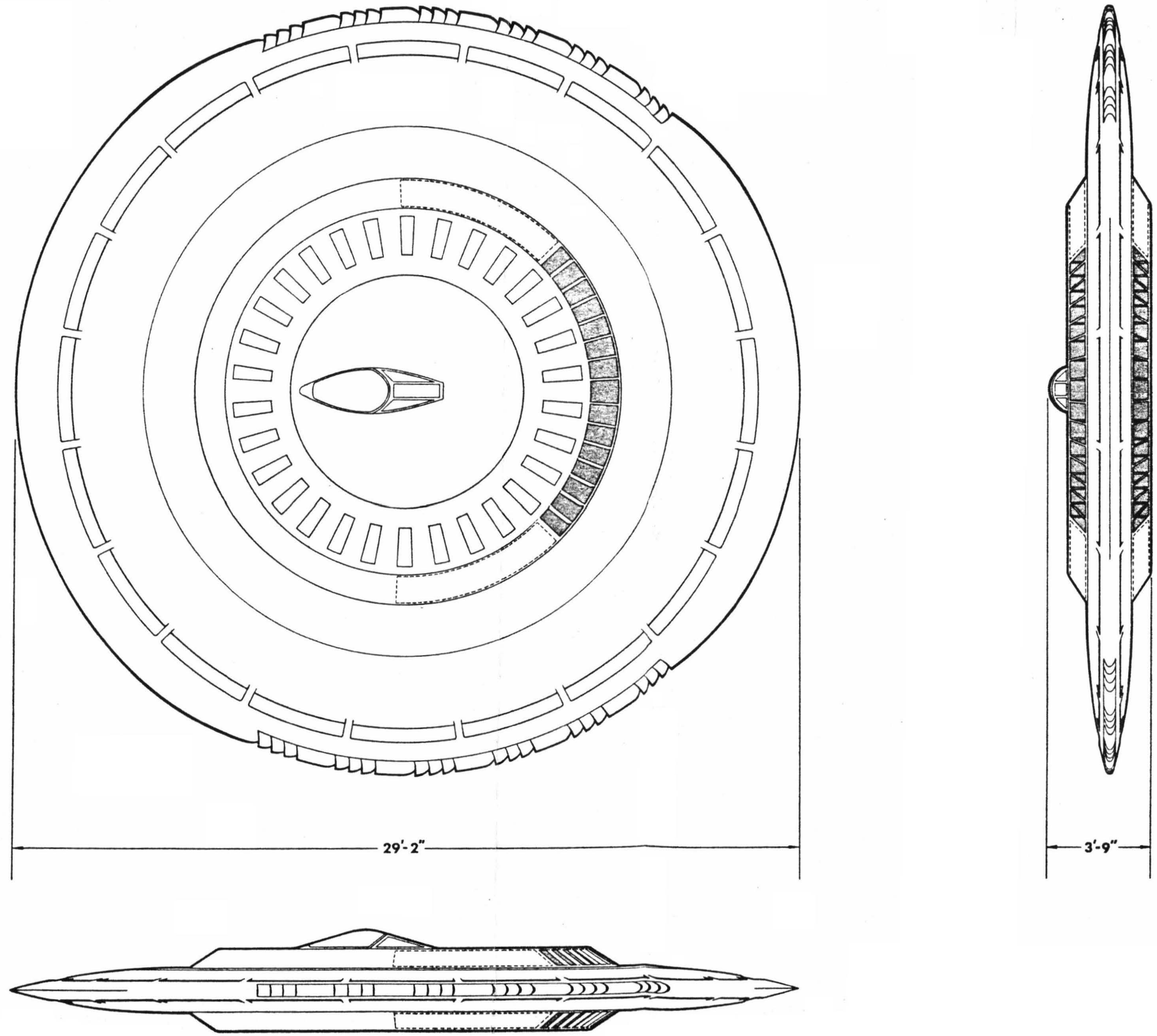


FIG. 1 3 VIEW GENERAL ARRANGEMENT OF RESEARCH AIRCRAFT

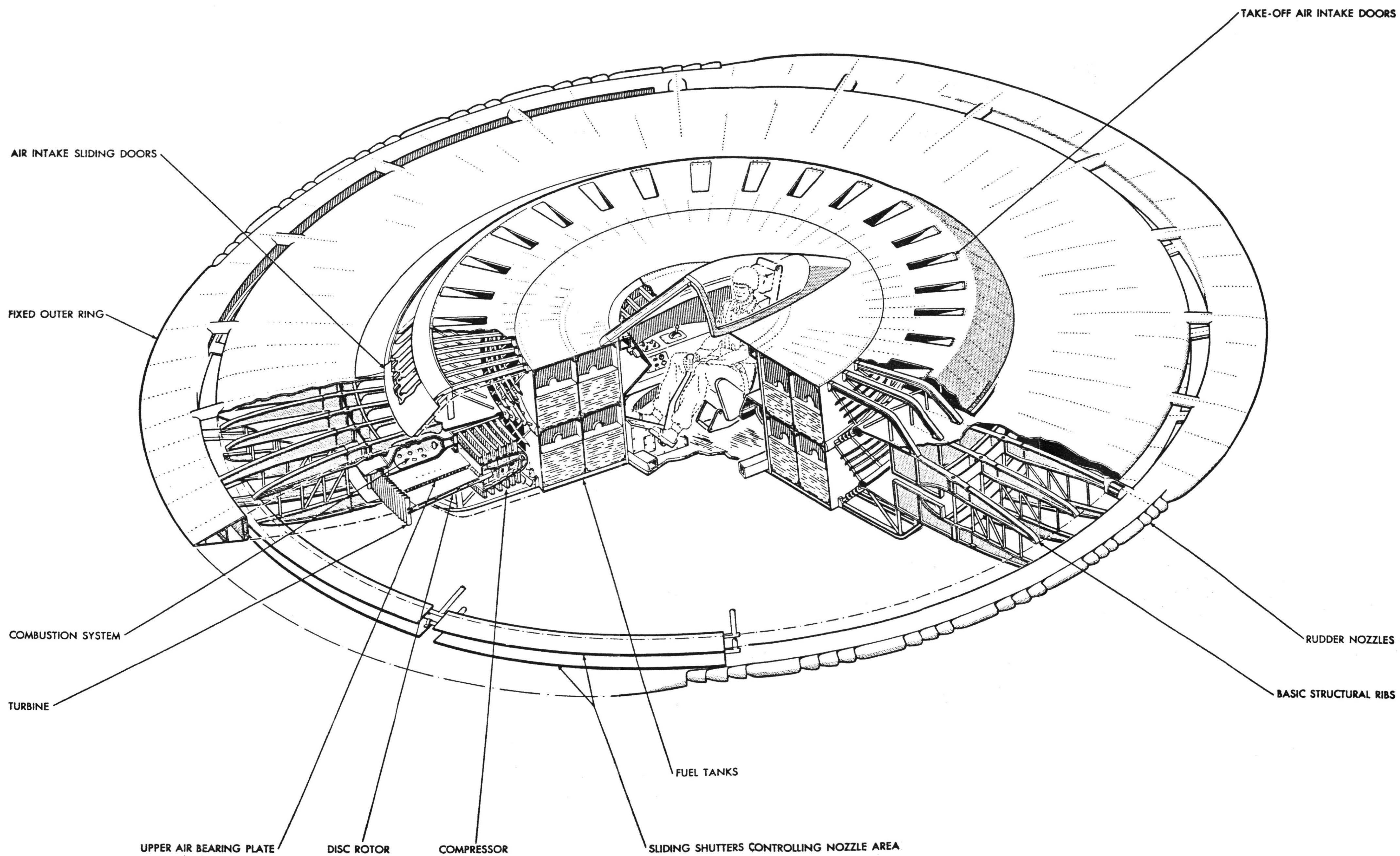


FIG. 2 SECTION CUTAWAY OF RESEARCH AIRCRAFT

Y2 - GENERAL DESCRIPTION

The Y2 aircraft is of lenticular form (i.e. lens-shaped), circular in plan, with a cockpit centrally located, the orientation of the cockpit determining the fore and aft axis of the aircraft and the normal direction of flight. Contained within the aircraft and encircling the cockpit, are the fuel cells and a radial flow gas turbine power plant. It is possible, and for the initial test stages perhaps desirable, to build this power plant of a number of small conventional axial engines, radially disposed in the aircraft like the spokes of a wheel. Ideally however, the power plant should comprise a single double-sided radial flow unit, having a disc rotor disposed in the chord plane of the aircraft (see Figs. 1 and 2) and rotating about the cockpit and fuel cells so that the plane of the rotor is edge-on to the line of flight, reversing the attitudes assumed by the rotor plane and axis in the conventional axial flow gas turbine power plant installation. The gyroscopic effect of the rotor will be the dominant factor in the stability and control of the aircraft.

The radial flow power plant is served by air intake structures encircling the cockpit on both upper and lower surfaces of the aircraft. Forward facing air intakes are provided in both upper and lower air intake structures to serve the power plant in forward flight. For vertical take-off or landing and free-air hovering, these intakes are closed and all the air required by the power plant is drawn through relieving doors on the top surface of the upper intake structure.

When taking-off, hovering or landing, air from the upper intake is passed through the compressor stages, the combustion chambers and the turbine of the power plant, and the exhaust is deflected downwards at the periphery of the aircraft to give vertical jet-lift, the jet thrust being regulated to control the rate of ascent or descent. A particularly important feature of the radial flow design using the peripheral jet is the powerful ground cushioning effect it provides, whereby the effective thrust has been shown to be several times the actual engine thrust when the aircraft is close to the ground.

In forward flight, air from the forward facing air intakes on both the upper and lower surfaces of the aircraft is passed through the compressor stages, the combustion chambers and the turbine of the power plant and is carried around the exhaustor duct and expelled through annular nozzles on the upper and lower surfaces at the rear and through backward facing nozzles on both side of the aircraft. The aircraft is controlled in flight by regulating the operation of sliding shutters, which control the jets exhausting through the annular nozzles for pitch and roll control,and the jets exhausting through the backward facing nozzles at the sides of the aircraft for control in yaw. (Actually, the annular nozzles extend circumferentially around the aircraft and can be controlled to propel the aircraft in any desired direction, but this refinement need not be discussed at this time).

Thrust forces in some form must be used to control the aircraft in the hovering condition and the use of jet control at all times obviates difficulties associated with hinged flap controls at high indicated speeds, such as flutter and the necessity of overcoming heavy forces on the controls due to hinge moment. Furthermore, the jet control applies a distributed load directly onto the aircraft structure avoiding stress concentration at hinge points. Though much data remains to be assembled, the results of initial testing in this direction appear most promising.

The range of speeds of most interest for the type of aircraft under consideration is from Mach 1.7 to 2.7 as it will have a supersonic cruising speed of around 1.7 to 2.0 and a top speed well in excess of this without reheat. At the present stage of the art, aerodynamic heating may limit the cruising speed to about Mach 2.0.

The higher thrust/weight ratio obtainable will give very fast climb and high ceiling with a time from sea level to 60,000 ft. in the order of two to three minutes, and an operational ceiling of around 70,000 ft. without reheat. For the aircraft under consideration the supersonic still air range will be approximately 600 miles.

As a matter of interest, it may be observed that though the radial flow power plant can be provided by a single disc double-sided radial engine or, with loss of gyroscopic stabilization, by a number of radially disposed conventional axial engines of total equivalent power, nothing could be gained by arranging conventional axial engines in line of flight and in fact, the vertical take-off characteristics would be lost.

The very shallow lenticular form of the Y2 configuration presents a much smaller frontal area for a given thrust in forward flight than a conventional gas turbine aircraft. Admittedly, from the standpoint of maximum installed thrusting efficiency in forward flight, the radial flow power plant is bound to be below the optimum represented by the straight through flow of a conventional gas turbine power plant installed in line of flight in some hypothetical aircraft, but for maximum overall efficiency, a complete aircraft/engine combination must be considered. In view of the thrust/frontal area of 900 lb./sq.ft. without reheat achieved by the Y2 aircraft it appears improbable that a higher thrust/frontal area can be achieved from an aircraft using a line of flight power plant installation. To illustrate the thrust/frontal area achieved by the Y2 power plant installation, the diagram at Fig. 3 shows the frontal areas of the conventional engines required to produce the same thrust as the radial flow power plant, superimposed upon a frontal view of the aircraft. These engines are of the most advanced conventional design, the bare engines having a thrust/frontal area of 1200 - 1300 lb./sq. ft.

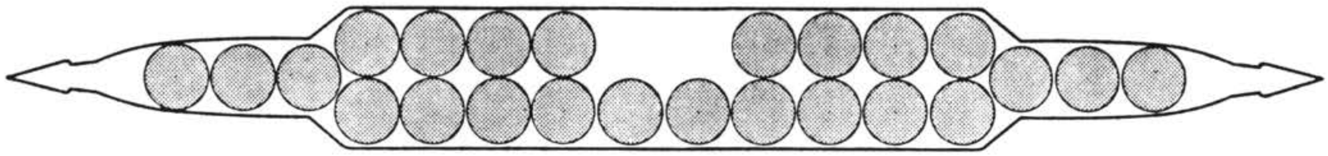


FIG. 3 FRONT VIEW OF AIRCRAFT WITH SPANWISE SUPERIMPOSED GAS TURBINE ENGINES

SUMMARY

The advantages of the Y2 aircraft may be summarized as follows:

A ground cushion effect is provided by the peripheral jet when taking-off or landing. As graphically illustrated in Fig. 4, a plain jet thrusting downwards from an aircraft of lenticular form taking-off edge-on to the ground would give no ground cushioning effect; a jet deflected to form a cylindrical 'waterfall' around the periphery of the aircraft provides a cushion effect when landing or taking-off, the effective thrust being several times greater than the actual engine thrust when the aircraft is close to the ground; while on the other hand, a jet exhausting centrally downwards from the aircraft has a reduced effective thrust in the presence of the ground. The comparative behaviours of the last two aircraft with engines discharging radially and axially (see Fig. 4), is worthy of note. In the case of a vertical approach to the ground from free-air hovering the throttle is eased back to give a thrust/weight ratio just less than one. The aircraft with the peripheral discharge sinks to about 40 ft. and then begins to decelerate, finally settling into its ground cushion at about 19 ft., but the aircraft having a jet exhausting centrally downwards sinks to about 25 ft. and then accelerates until it crashes onto the ground; the central jet actually produces a suction between the aircraft and the ground below a critical altitude.

In the case of take-off the aircraft with the peripheral discharge will leave the ground under perfect control at a very low engine thrust/weight ratio (considerably less than one) and, as the throttle is advanced, hover at a gradually increasing height, but an aircraft having a jet exhausting centrally downwards and supported, for instance, 6 ft. from the ground would not take-off until the thrust/weight ratio reached 1.75, thereupon leaving the ground with an uncontrolled acceleration.

Because the Y2 aircraft will hover at a considerable distance from the ground at a thrust/weight ratio of less than one it is probable that larger aircraft of this type which theoretically must have a maximum thrust/weight ratio of less than one, can be designed and will take-off from the ground cushion. The Y2 aircraft could land on such a difficult area as a small ship or submarine that is rising and falling in the ocean swell, because it will rise and fall cushioned at a fixed height above the deck. As the throttle is closed, it will settle on to the deck. The Y2 aircraft, which has a convex undersurface, has a stable tendency in the ground cushion and it is more than probable that a satisfactory transition to forward flight can be made directly from it.

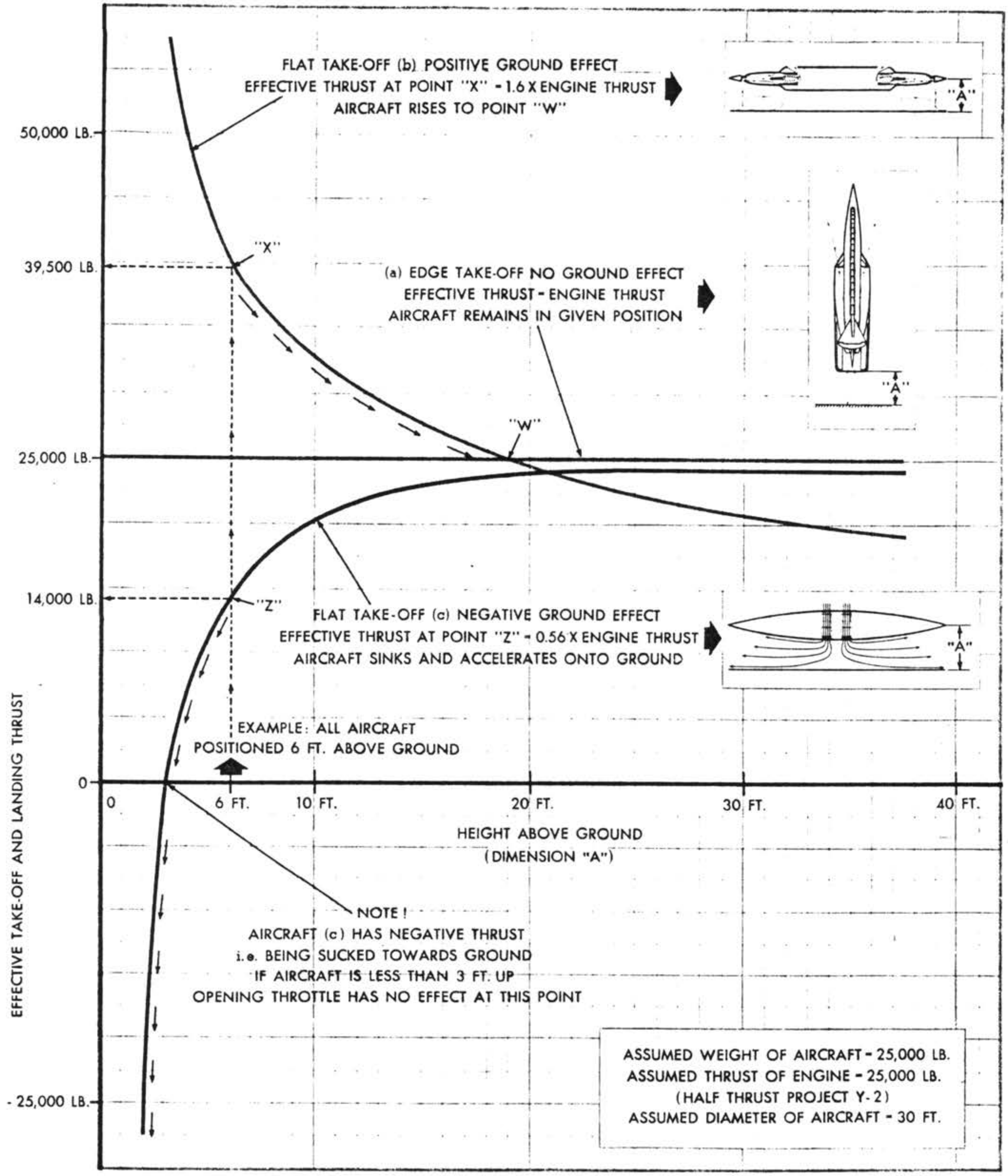


FIG. 4 COMPARISON OF GROUND EFFECTS, FOR FLAT TAKE-OFF AND EDGE TAKE-OFF AIRCRAFT

Though the lenticular form can use conventional engines disposed radially like the spokes of a wheel, it does lend itself to the installation of a single radial flow engine, with inherent gyroscopic stabilization.

The engine rotor of the single-disc, radial flow power plant, gives the same stability to the aircraft as that which would be provided by a gyroscope of similar size revolving within the envelope of the aircraft. Outweighing the restriction it imposes on the rate of manoeuvre of the aircraft are the advantages which this gyroscopic stability confers; it allows the designer to choose the position of the centre of gravity of the aircraft within limits imposed only by the trim force available; it provides stabilization in the absence of aerodynamic forces; and it can be expected to provide a steady flight regime and eliminate all sudden changes, especially in the transonic region.

Striking manufacturing simplicity results from the radial spoke structure of the aircraft and the use of the radial flow engine as may be seen from Fig. 2. The symmetrical circular configuration of the structure of both airframe and power plant makes it possible to use numerous identical components, resulting in a minimum of manufacturing operations, with consequent reduction of costs. This is exemplified in the design of the compressor and turbine blades, which require no aerodynamic twist and can be manufactured by simple extrusion.

DETAILS OF TESTING AND DEVELOPMENT PROGRAM

A partial program of testing has already been carried out, but a continuation of the test program is planned to solve the problems posed by the development of the aircraft, which is a radically new design. This is especially necessary to facilitate a full investigation of the problems of stability, control and power plant layout.

The program is divided into three phases. Phase 1 covers those tests which will solve the problems associated with the aerodynamics and manufacture of the airframe and power plant. This phase will culminate in the building and testing of a sixth segment of the test aircraft and the building and running of a half scale single-disc radial flow gas turbine power plant with air bearing. Phase 2 covers the development and manufacture of a test aircraft and the development of reheat for its multi-engine installation. Phase 3 will be the manufacture of a Y2 aircraft powered by a single-disc radial flow gas turbine power plant. The details of the program are described hereunder.

Phase 1 - Testing and Development

The aerodynamics and airframe aspect of Phase 1 comprises thrust recovery and control tests, ground effect tests, subsonic and supersonic wind tunnel tests and the building of the test aircraft segment. The power plant development, consisting of air bearing rig tests, combustion development, one half scale bearing and engine tests and a simulator study to investigate gyro stability and control will proceed simultaneously.

Aerodynamics and Airframe

(1) Thrust recovery and control tests

A two dimensional test specimen - in course of manufacture - has been designed for accurate test of the efficiency of thrust recovery and control at a Mach No. of 1.7, close to the estimated supersonic cruise speed.

The thrust recovery and control scheme proposed is based on the so-called Coanda effect whereby a high aspect ratio jet can be deflected through large angles by means of a curved surface in contact with its edge on one side. A test rig on which preliminary investigation of this effect was carried out is shown in

Fig. 5. This comprises a diffuser, with splitters (1), mounted on a stand (2) and exhausting an airstream through a narrow slot (3) across a 1.25 in. dia. x 6.0 in. long cylinder (4) at its end. This method of jet-bending appears to give very good efficiency statically but it cannot be used for take-off because statically the process breaks down at large values of height-of-orifice/radius-of-cylinder. At high forward speeds, however, the bending may be greatly assisted by the dynamic pressure of the oncoming air so that it appears possible to obtain good bending efficiency around a radius small enough to fit the aircraft nozzle system.

Preliminary tests at very small scale, in which large errors should be expected, particularly from boundary effects, show the proposed scheme to be workable; and control forces of about twice the gross thrust moment which would be obtained by assuming the line of action of the thrust normal to the surface, were realized.

The rig for these tests is illustrated in Figs. 6 and 7. A plexiglas channel (1) was constructed and a single sided two-dimensional nozzle designed for a Mach No. of approximately 1.45 produced a half inch square supersonic stream, having a free upper boundary, along the floor of this channel. Approximately one inch downstream of the expansion nozzle, a 0.045 in. wide slot with radiused edges, fed a secondary flow into the main stream at right angles to it, as illustrated in Fig. 8.

The channel was mounted on a swinging arm thrust balance (2), as shown in Fig. 6, and the thrust of the channel balanced on the scale pan (3). The normal jet (4) was then turned on and its thrust at right angles to its exit direction was measured by adding weight to the scale pan and increasing the supply total pressure (5), until the arm began to move; this was registered by breaking an electrical contact (6). In this way, the variation of thrust of the normal jet in an 'axial' direction, with pressure ratio, was obtained.

Pipe reactions were isolated from the balance system by supplying air for the two jets at the pivot: the main flow for the channel was fed downwards from the top through an air bearing arrangement (7), consisting of two 12 in. dia. plywood discs butting together with the supply tube ending flush with the face of the top disc. When air is turned on, the discs are drawn together (to a distance of approximately 0.02 in.) and a small leak takes place between them. The secondary flow (8), was fed upwards through the hollow pivot, and the neck in the centre of the bearing spindle. The pipe reactions with pressure 'on' and the exits blanked off, were nil.

The channel was then removed and remounted, turned through 90°, so that the main stream reaction passed close to the centre whilst the normal jet exhausted tangentially on the balance system, as shown in Fig. 7.

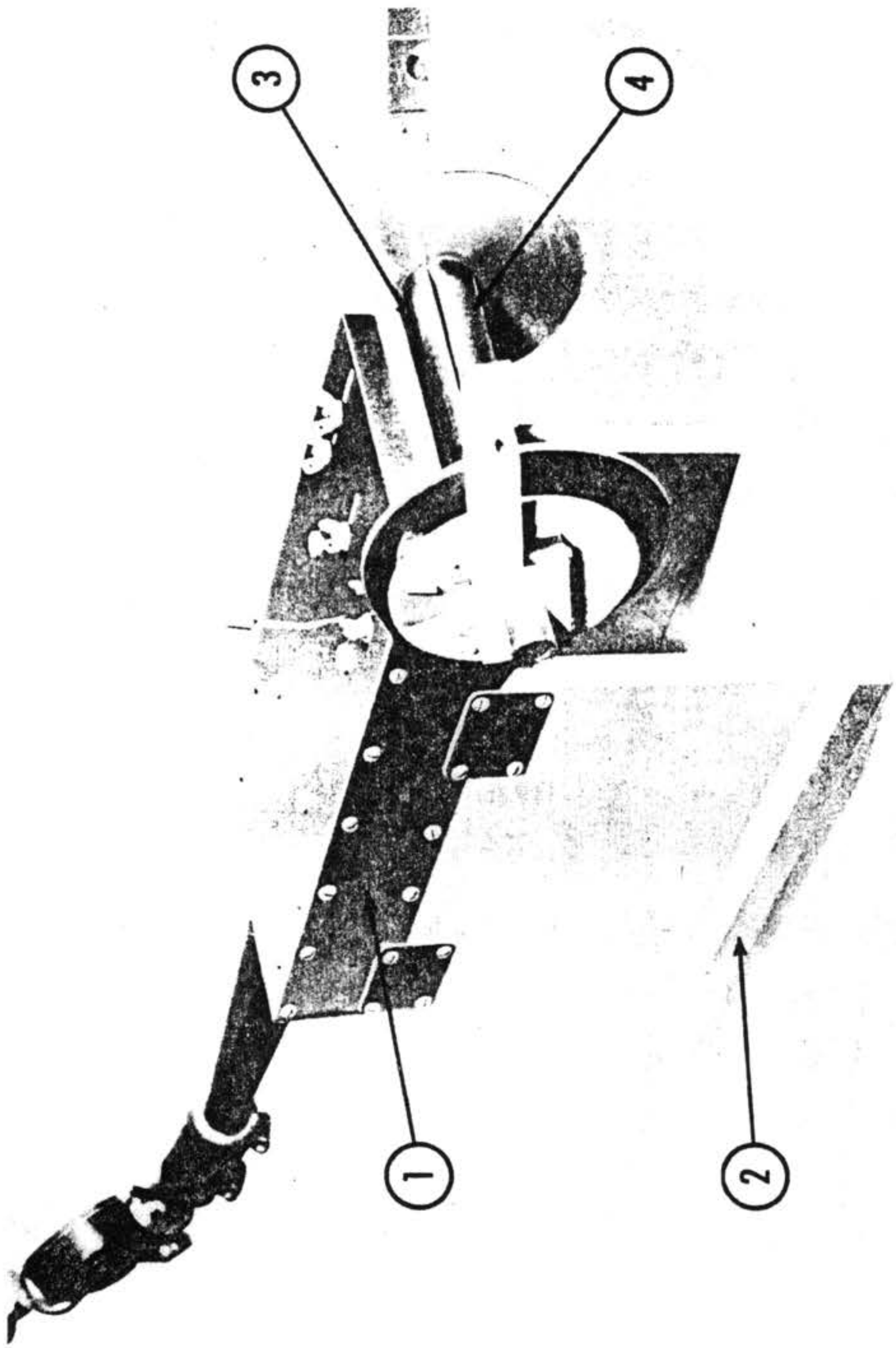


FIG. 5 COANDA EFFECT TEST RIG

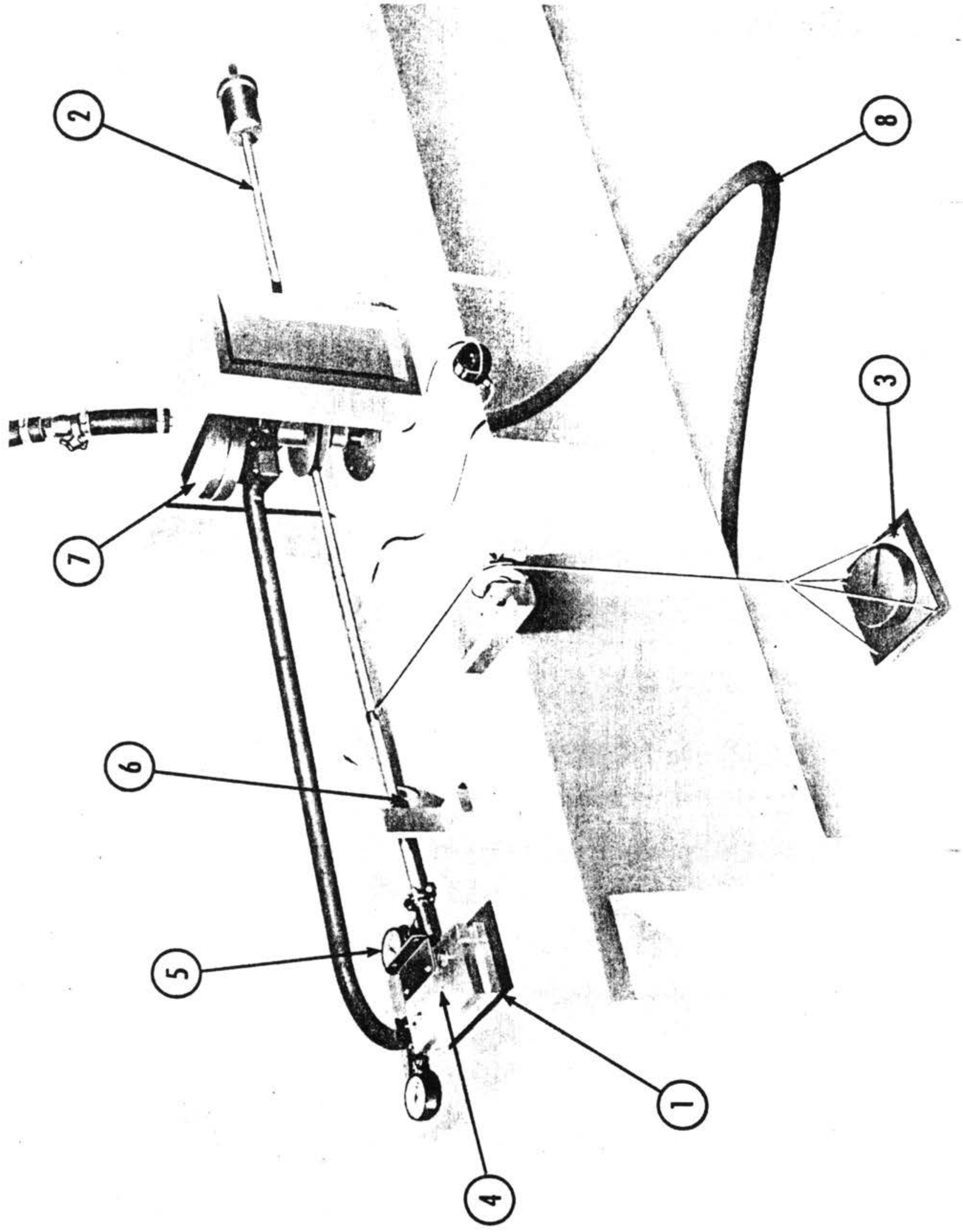


FIG. 6 THRUST RECOVERY AND CONTROL TEST RIG

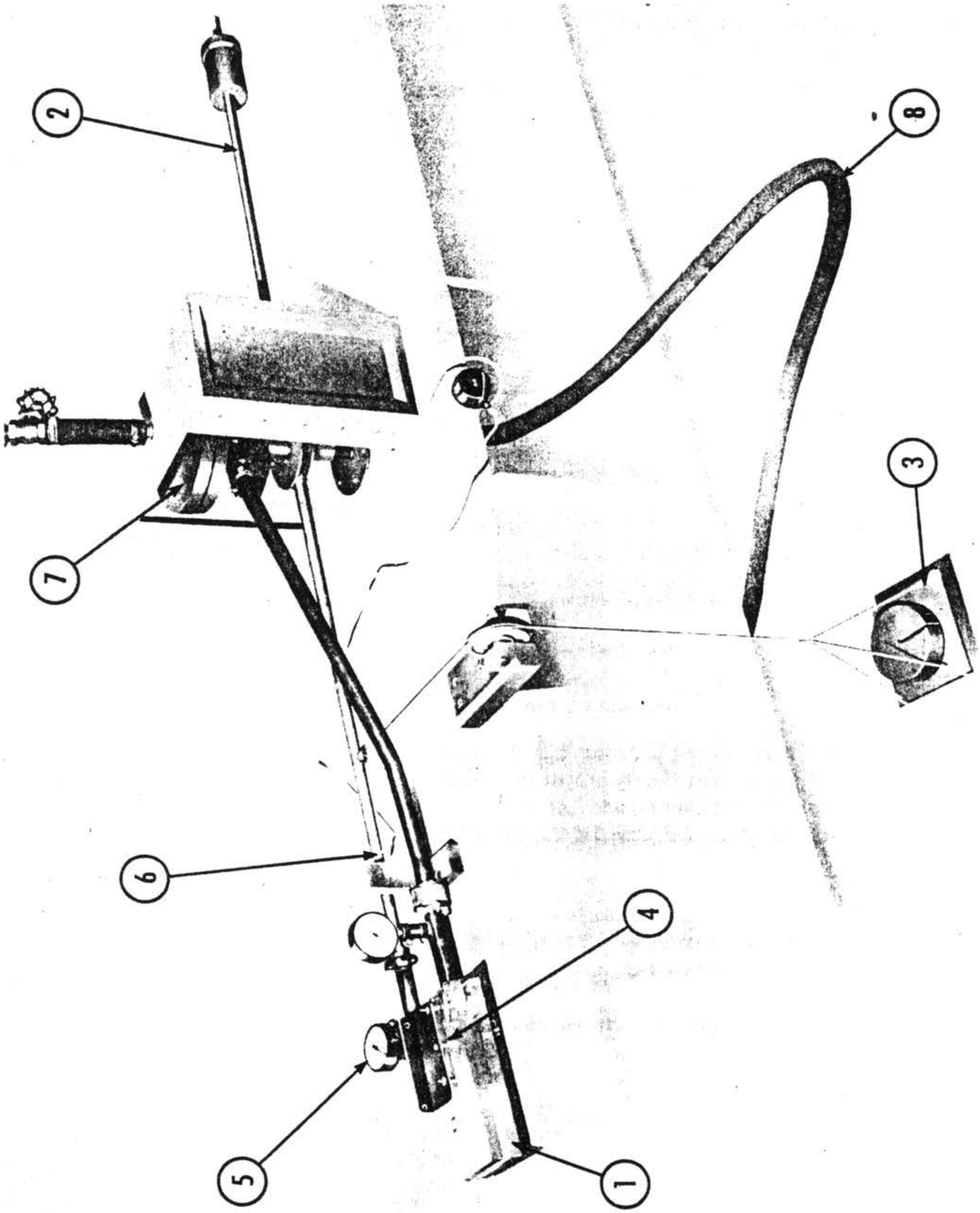


FIG. 7 THRUST RECOVERY AND CONTROL TEST RIG

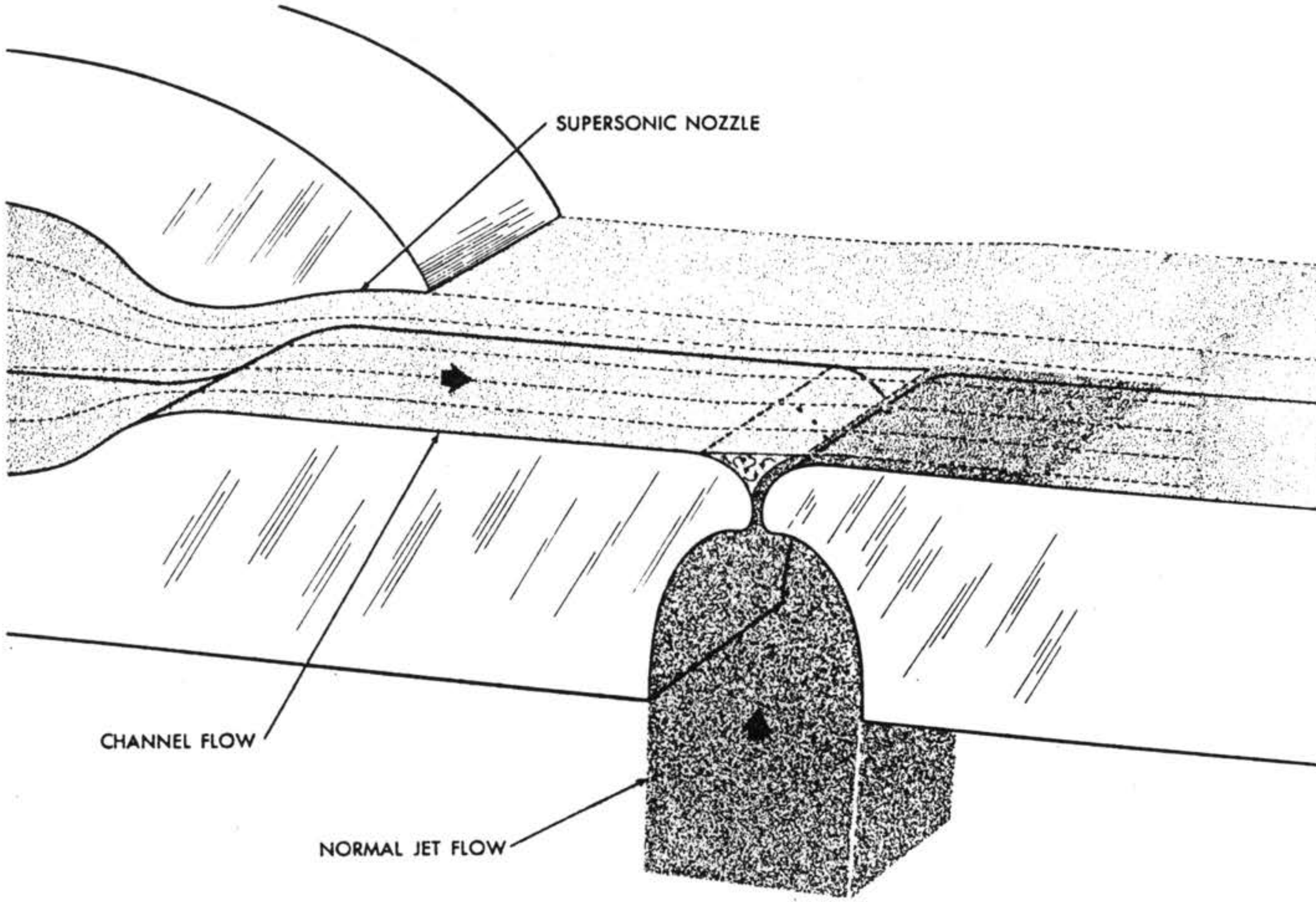


FIG. 8 THRUST RECOVERY FLOW DIAGRAM

With the channel flow off, the reaction of the normal jet (4) was measured against pressure ratio, for comparison with the previous thrust increase streamwise. Then, with the channel flow on, the tangential force due to the deflection of the main stream by the normal jet was taken in order to obtain an idea of the control forces available.

With the channel not mounted on the thrust balance, surface pressure measurements around the flare of the normal jet orifice were also taken. A high pressure was developed on the upstream side, and a low pressure on the downstream side, indicating how the thrust is recovered on the surface. Significant also was the fact, that as the channel air was turned down the negative pressure became more negative as the positive pressure was reduced, maintaining the same thrust.

Thrust recovery of around 95% was obtained from integration of the pressure measurements in a streamwise direction, and over 70% on the thrust balance.

Owing to the large errors expected, neither of these results is the least conclusive, but they give indication of the need for further work which the rig was

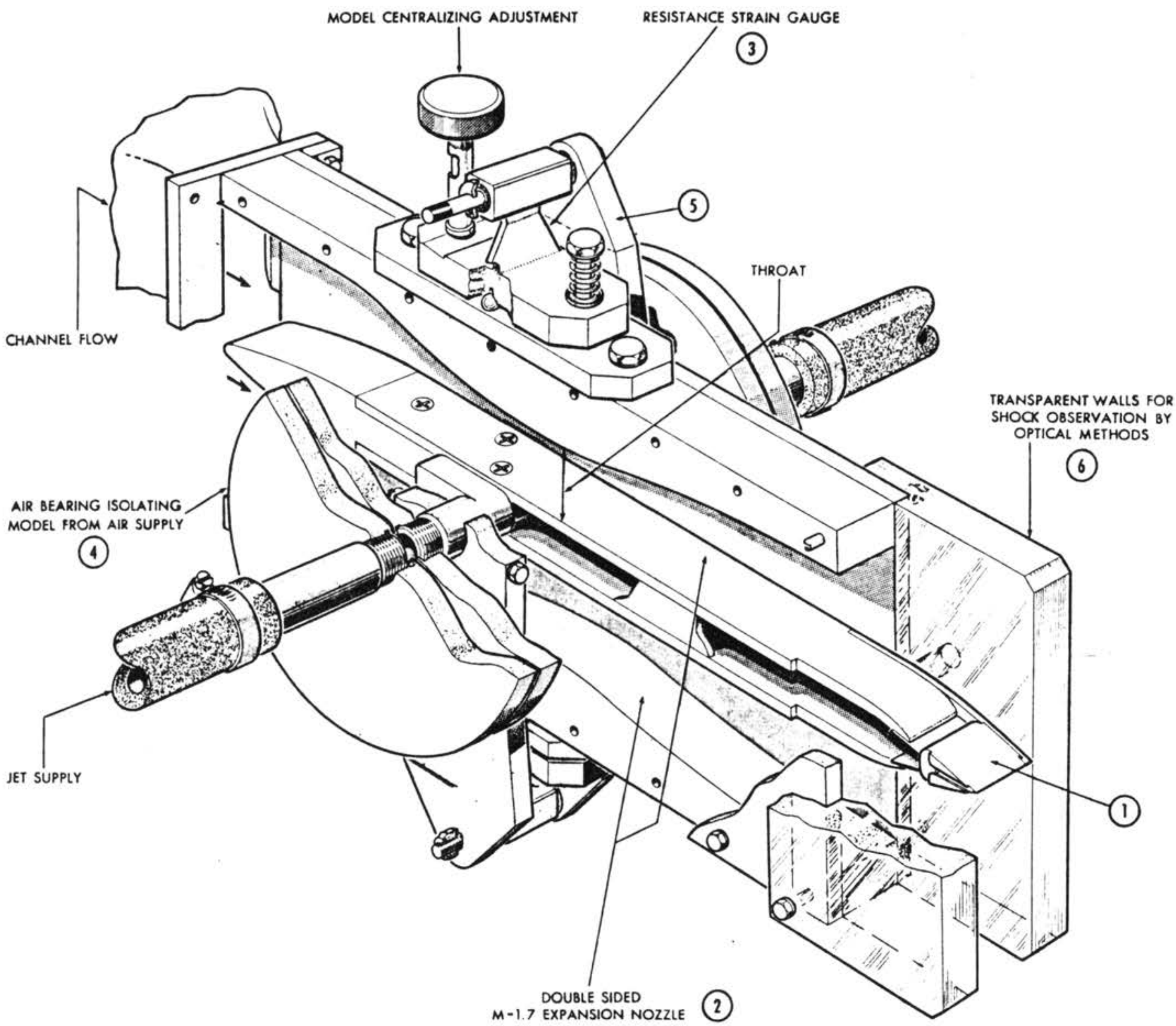


FIG. 9 PROPOSED THRUST RECOVERY AND CONTROL TEST CHANNEL

designed to achieve. The test now proposed, will determine the efficiencies accurately. Fig. 9, illustrates the specimen, which was designed for the A.V. Roe Nobel air supply, and requires 15 lb./sec. at 100 psi gauge to operate. This rig, consists of a two-dimensional test piece (1), representing the conditions at the rear of the aircraft, which is mounted in the centre of a supersonic nozzle (2). The characteristic line from the nozzle exit is designed to clear the specimen by a comfortable margin, and measurements of drag and pitching moment will be recorded on a resistance strain gauge balance (3). An air bearing arrangement (4), similar to that described above, will isolate the balance (5). Shock observations through the transparent end walls (6), and the measurements of drag and pitching moment will be taken.

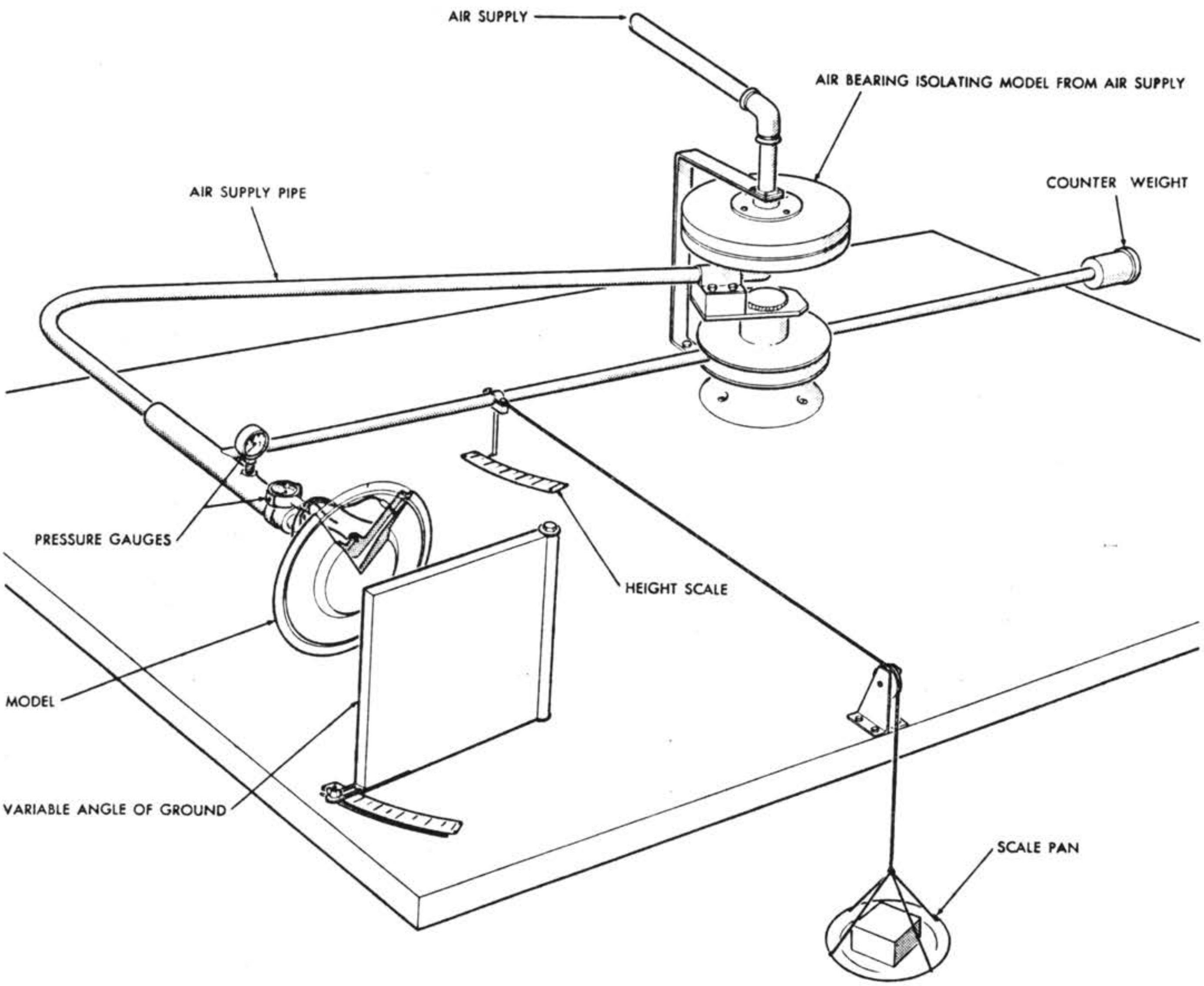


FIG. 10 GROUND EFFECT TEST RIG

(2) Ground effect tests

It is proposed to continue tests on the ground effect using the thrust balance already manufactured and illustrated in Fig. 6. Fig. 10 is a sketch of the proposed test set-up. Test specimens are being designed to evaluate -

(a) Effect of jet aspect ratio.
(b) Effect of angle to ground of chord plane.
(c) Effect of having discrete jets.
(d) Effect of moving the centre of thrust away from the centroid of area.
(e) Effect of planform and undersurface contour.
(f) Effect of scale.
(g) The reverse ground effect.

It is anticipated that the models will be about 10 in. dia. and that a 2 lb./sec. air supply at 100 psi gauge will be satisfactory.

A small dynamic rig will also be built to gain preliminary information on the transition from hovering to forward flight. This will consist of a 6 to 10 ft. dia. rotating arm, with a ground effect model, and counter-balance at its ends. Air will be supplied through an air bearing at the pivot, to the ground effect model, which will contain a swash plate to adjust the nozzle: the arm will be free to flap and the model may be fixed at various incidences and wing loadings, and the flight path observed.

(3) Subsonic tunnel models

A subsonic wind tunnel model will be required, incorporating a simulated exhaust jet flow and possibly an intake flow. This has not yet been designed, but from previous experience, it is envisaged as a half-plane model at approximately one-tenth scale. Variable nozzles will be required for control data and for conducting tests on the transition from hovering to forward flight. Measurements of lift, drag and pitching moment, for the latter condition, will take priority on this model. At least 5 lb./sec. air flow at 100 psi gauge is desirable.

It is also envisaged that useful preliminary information on lift, drag and pitching moment can be obtained from a preliminary model, omitting intake and jets, which can be very quickly manufactured.

A further model will be required to develop the internal flow in the intake. A half-plane model to as large a scale as possible is required. Pressure measurements will be taken, but force measurements on the model will not be necessary.

(4) Supersonic tunnel models

A supersonic tunnel model will also be required, at an early stage, to investigate the characteristics of this type of aircraft at supersonic speed. Jet simulation is essential; however, if a low pressure tunnel is employed, a useful range of Mach numbers can be covered by simply opening the exhaust duct to atmosphere. This model will also need to be designed with variable nozzles so that data on control and thrust recovery can be obtained. The scale will depend on the size of tunnel available.

If, as is likely, it is impossible to incorporate the intake on the above supersonic model, due to scale and air supply difficulty, an intake pressure recovery model will be required on which the front half of half the aircraft will be represented to as large a scale as possible. Pressure measurements and shock observations will be taken but force measurements are not required.

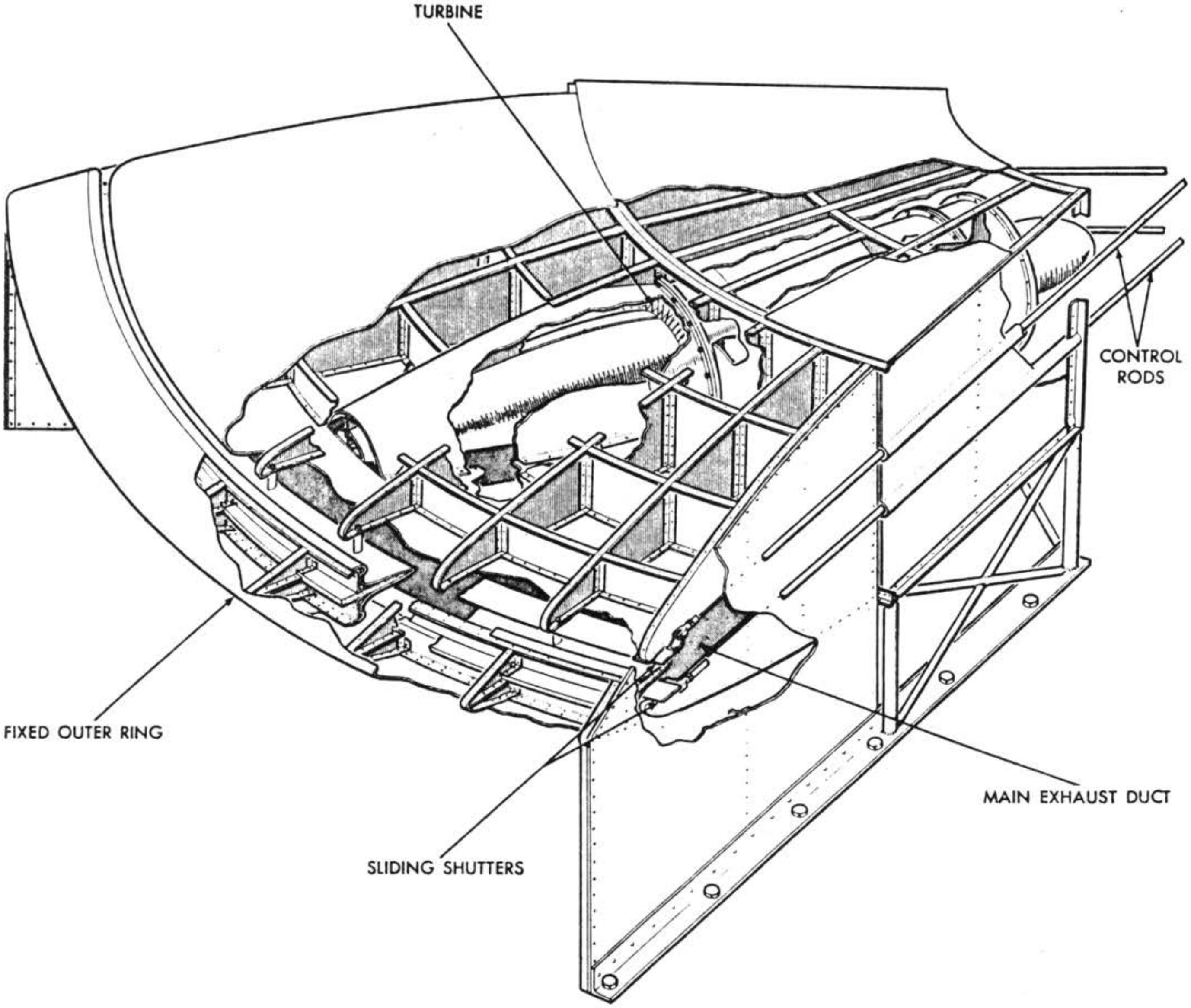


FIG. 11 TEST SEGMENT WITH VIPER ENGINE

(5) Sixth segment of test aircraft

A sixth segment of the six-engined test aircraft, described in Phase 2, will be constructed.

The only unusual mechanical problems, which can be foreseen in the construction of a multiple engine version of this aircraft, will be those associated with the control shutter mechanism and the ducting of hot pressurized exhaust gases. Apart from these, the design is simpler than usual.

The test segment will include one engine and everything on the aircraft in that segment outboard of the engine. Six of these segments, plus the central compartment, will form the complete aircraft wing.

Fig. 11 illustrates the proposed test piece with a Viper engine installed. A great deal of testing can be carried out on this rig; a short list of typical tests is as follows -

(a) Control functioning.
(b) Control functioning with loaded structure.
(c) Control power and sensitivity evaluation.
(d) Structural integrity under flight and landing loads.
(e) Exhaustor loss check - take-off case.
(f) Limit and ultimate exhaust pressure check.

Engine

(1) Small scale bearing rigs

The test work started on three existing air bearing rigs should be completed. These rigs are -

(a) The static rig, see Fig. 12, which is basically a flat steel plate (1) with air supplied from underneath through a single central orifice of variable size. A test series of flat steel discs (2) and (3) of varying diameter, on which the pressure distribution in a radial direction may be measured, are pressed down on to the baseplate by a screwjack (4). Bearing air is supplied at the central orifice to separate the upper disc from the baseplate, the air escaping at the rim of the former. The pressure points (5) are connected to a mercury manometer and force is measured by a resistance strain gauge (6); contact between the disc and the baseplate - which occurs when the maximum capacity is exceeded - being registered electrically by a lamp (7) in a low voltage circuit.

(b) A dynamic rig, see Figs. 13 and 14, which consists of a 3 ft. dia. steel disc (1) spun to a max. speed of 3600 rpm by an electric motor through a slipping clutch and brake transmission. Small discs (2) of varying under-surface contour and design, with air supplied at various points on their surfaces are loaded against the face of the big disc by a crank and weight system (3) and pressure distributions across the discs are again measured with a mercury manometer (4). This rig will also be used to test complete small scale bearing rings instead of discs, the latter being to scale with the actual engine bearing area available. In Fig. 16, the manometer records a pressure distribution across the face of the disc being tested. Fig. 15 is a longitudinal section through the test disc: air is supplied at points A and B, recorded by the first and seventh mercury columns from the left, and is compressed by rotation of the main disc to many times the supply pressure, building to a maximum in front of the self-lubricating pads, as seen on the manometer.

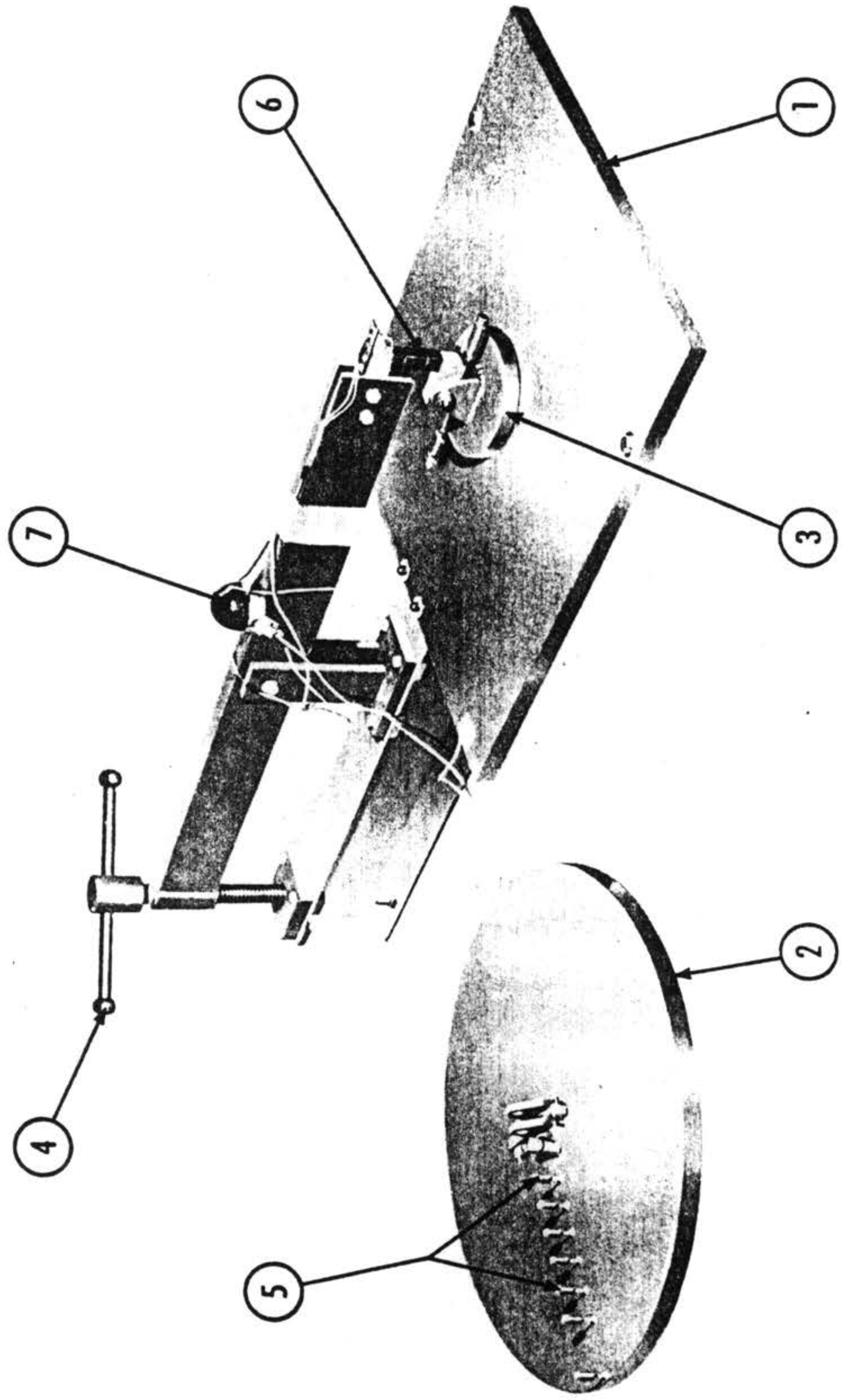


FIG. 12 AIR BEARING STATIC TEST RIG

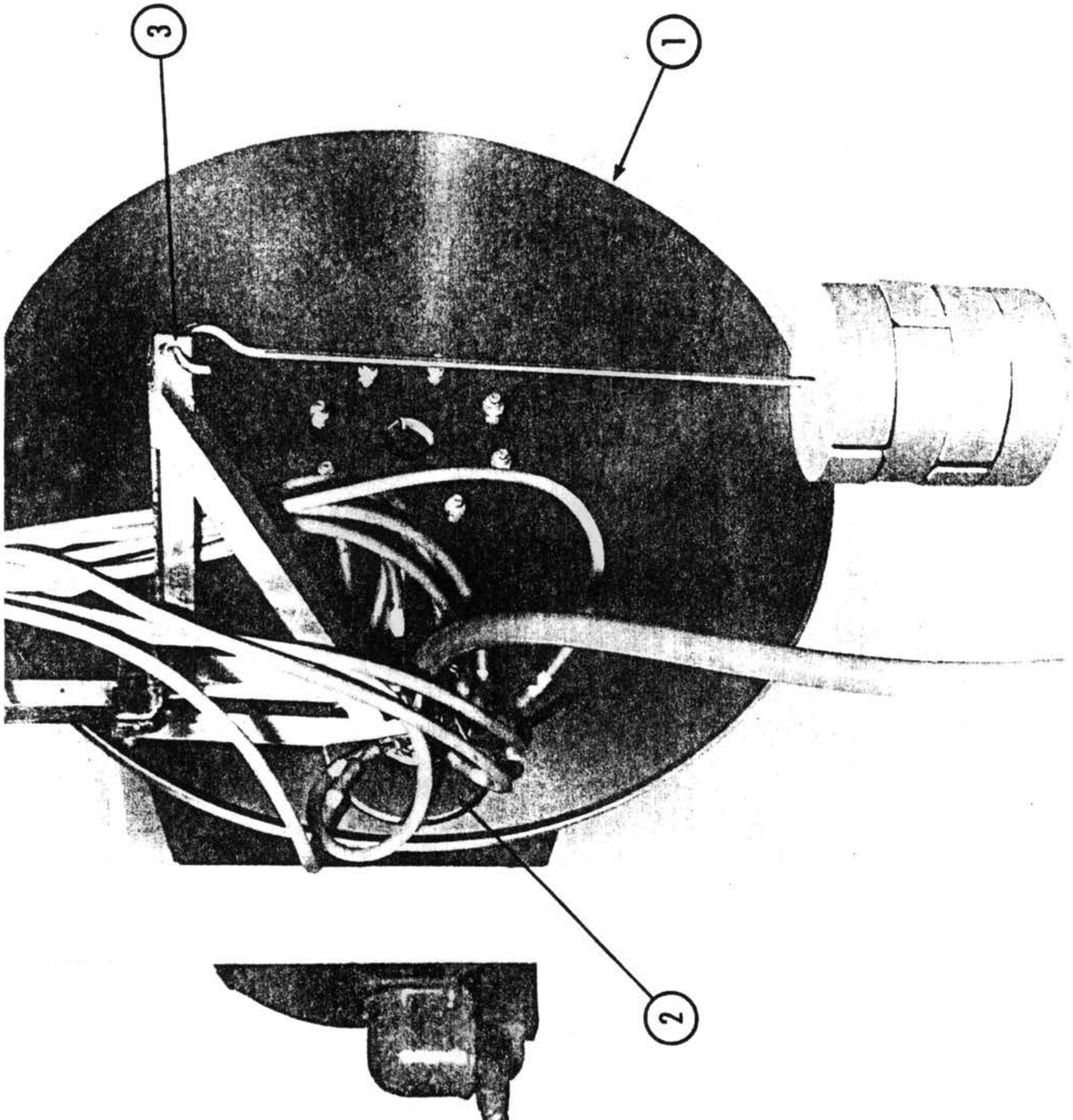


FIG. 13 AIR BEARING DYNAMIC TEST RIG

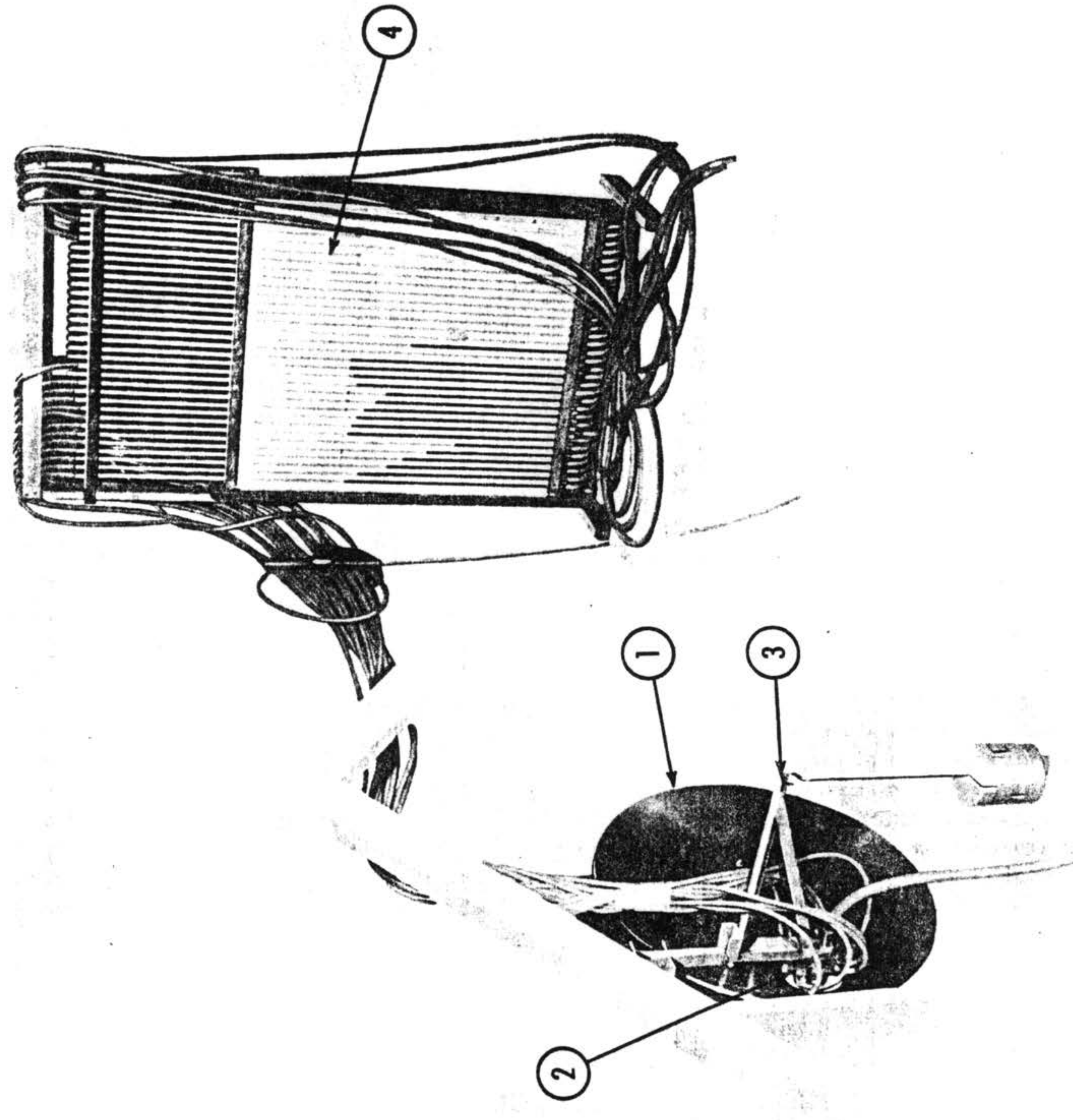


FIG. 14 AIR BEARING DYNAMIC TEST RIG

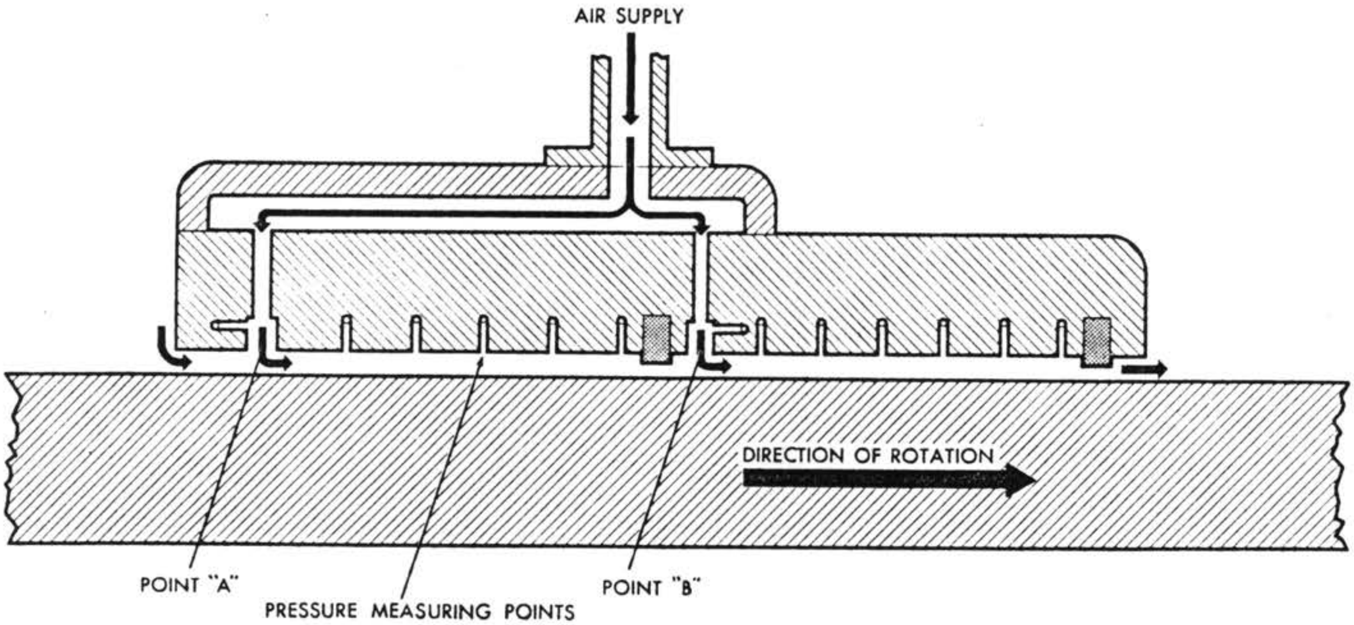


FIG. 15 SECTION DIAGRAM OF BEARING PAD

(c) A one-tenth scale bearing rig, see Fig. 16, which is representative of bearing pressures and areas on an actual engine. It consists of a solid steel ring rotor (1), artificially ten times too thick at one-tenth scale in order to represent the same bearing load per unit area as full scale. This rotor is fitted in a stiff steel housing (2) and air is supplied through two galleries (3) and plain discrete holes to the face of the rotor. The 'vertical' bearing (the top of the T) is independently supplied. The disc, which is cut as an elementary pelton wheel, is spun by high speed air jets (4), directed at its periphery. Rpm is measured by a magnetic pick-up which counts the teeth passing it and registers on a Cathode Ray Tube display (not shown). Contact between the rotor and housing is again registered electrically. The rig is designed to be swung backwards and forwards through a 90° arc at a controlled rate, by a pneumatic jack (5) with automatic valve reversal, to evaluate its capacity with simulated precessional loadings.

(2) Combustion development

The combustion system design for Project Y is due to the well known combustion specialists Messrs. Joseph Lucas. Since the combustion system lends itself readily to rig test, being essentially a number of near-square section combustion chambers with circular flame tubes; and because, in view of the low engine pressure ratio and necessity for minimum length and volume, it is designed to operate at higher intensity than is usual on present day engines; development should be put in hand at an early stage. Tests will include water flow tests of a single combustion chamber and high altitude tunnel tests, through the range of conditions expected, to determine flame stability limits etc. It is not considered

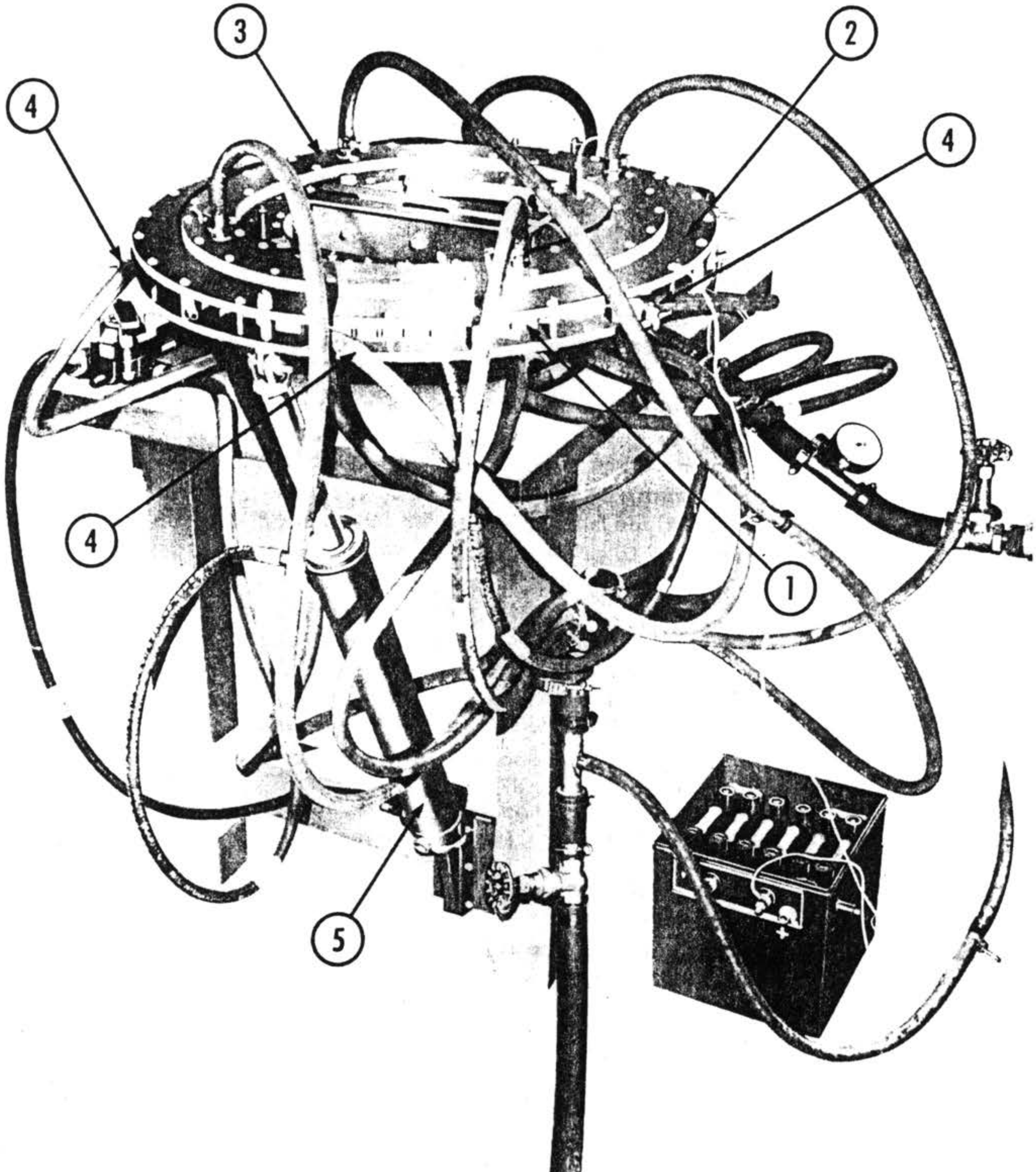


FIG. 16 AIR BEARING TEST RIG OF 1/10 SCALE MODEL

necessary to test a segment containing more than one flame tube. Initial development will have to be at the scale first of all required, but the differences to be expected on scaling up must be considered and checked. The following figures are applicable to the design, at full scale -

Area combustion intensity	5.2×10^6 CHU/sq. ft./hr./atm.
Mean velocity	70.7 ft./sec.
Pressure loss factor	46.5

Messrs. Lucas are expected to be able to begin development as soon as a decision can be made to proceed; and to supply the first few sets of combustion 'hardware'.

(3) Half scale engine and bearing

A considerable saving of time and money will result if data on the characteristics of the radial flow engine can be obtained at smaller scale. Construction of a static test facility and static running costs will be considerable at full scale. Although the aerodynamic design of the radial flow engine is considered to be conventional, nevertheless it is new and new problems should be expected. Furthermore, the solution of the mechanical problems can largely be discovered from smaller scale tests. We conclude that the wisest course is to make the engine at half scale, which is judged to be approximately as small as the compressor and turbine blades can be made and the engine still run efficiently at S.L. The last stage compressor blades will be 0.4 in. chord. This policy should produce fast and worthwhile results.

The small scale bearing tests have produced satisfactory results to date. It appears that the only likely cause of real difficulty with the bearing will be that the required tolerances will not be maintained at larger scale and with a more flexible structure: it is considered that half scale will show up difficulties of this kind, enabling a representative flexible structure to be designed.

Again at half scale it will be relatively easy to oscillate the test engine and structure to simulate precessional loads.

Fig. 17 is a skeleton of the proposed test rig. The aircraft structure, around the engine, will form an integral part of it, but the intake will simply be an open hole with a large vertical duct. Exhaust will not be collected from the radial direction, thrust being calculated from pressure measurements in a short dummy jet pipe.

(4) Simulator study

The other big development which must be linked to the engine is the analysis and test of gyro stabilized aircraft.

Considerable analysis of the gyroscopic stability conferred by the engine rotor

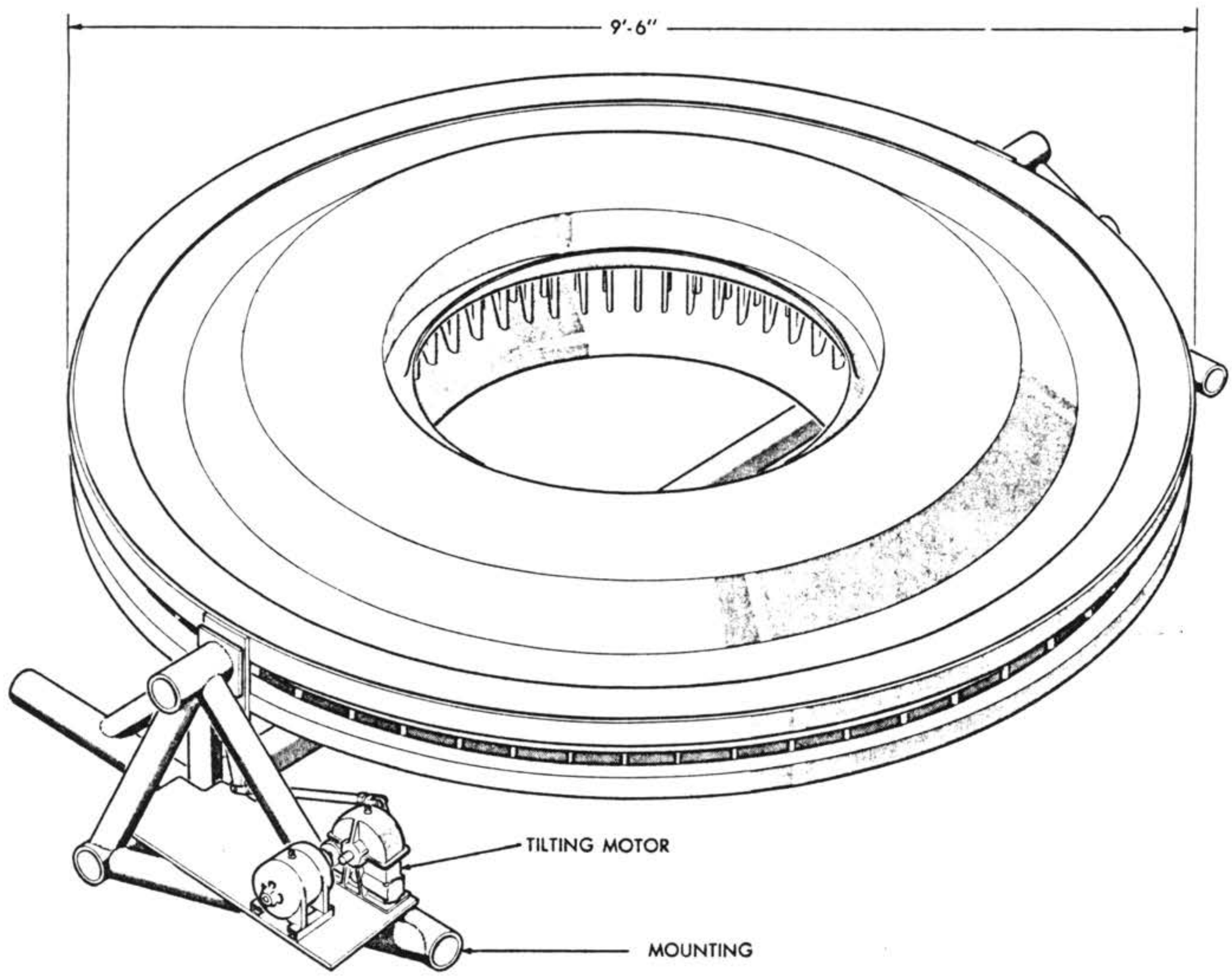


FIG. 17 HALF SCALE ENGINE AND BEARING TEST RIG

has already been carried out and the important stability modes have been identified and explained. The addition of the gyroscopic reactions to the normal stability equations is exact, since these reactions depend on accurately known physical quantities such as the rotor rpm. However, the arithmetic involved is cumbersome, since the gyro cross couples longitudinal and lateral stability, normally treated separately, and an eighth power stability polynominal results.

Mathematical analysis is greatly helped in problems of this sort by the use of electronic computing devices and work has already been done on a Reeves analogue. However, much remains to be done, particularly on the response of the aircraft to its controls in all conditions of flight. Its controllability also needs to be tested by a pilot. This will be possible, if a specially designed pilot's seat and control system is connected to a standard computer, such as the analogue mentioned above. The pilot's reactions in attempting to maintain a given display will then be fed into the equations and his comment on change of response, with variation of all the parameters under the designer's control, will be the guide to the design of the control system.

The manufacture of the equipment needed for fitment to a standard Reeves analogue will be undertaken and the necessary analysis will be made. The analogue will be obtained on a rental basis.

Phase 2 - Development of Multi-engine Test Aircraft

The second phase of the program will utilize the knowledge and experience gained in Phase 1, and will culminate in the manufacture of the multi-engined test aircraft and development for re-heat. The manufacture of the aircraft will be preceded by the manufacture of a cockpit mock-up and the development of an artificial stabilizer.

(1) Mock-up

A cockpit and front aircraft mock-up will be required to evaluate view, and to develop cockpit layout in the usual manner.

(2) Artificial stabilizer

A simple artificial stabilizer will be developed to help the pilot control the test aircraft in the hovering condition. This is believed to have been found necessary in certain other vertical take-off development work, and although it may well be that the stable tendency in the ground cushion will allow flight without it, it is considered a necessary assistance.

In view of the great simplicity of this type of aircraft and its performance potential when fitted with conventional engines, development of the stabilizer will be pursued on the test aircraft by progressively removing ballast from the nose, so as to render it naturally unstable in forward flight, and gradually increasing the negative margin, to assess the capability. It may be noted that an alleviating factor in the design, is the low inertia that the control shutters will have, and the rapid control response that may be expected. Clearly, the stabilizer must have a systematic reliability at least as good as that of the now accepted full-power control.

This development, would open the way to a multiple power plant operational aircraft which could be designed to be normally stable by a desirable margin supersonically, in the condition in which it is intended to operate, and would rely on some simple form of pitch stabilizer in subsonic flight. The magnitude of the unstable margin required cannot be estimated, as there is little data on aerodynamic centre position, furthermore, the trailing edge to wing tip jet flow may well have a significant effect. The tunnel tests proposed in Phase 1 will enable an assessment to be made.

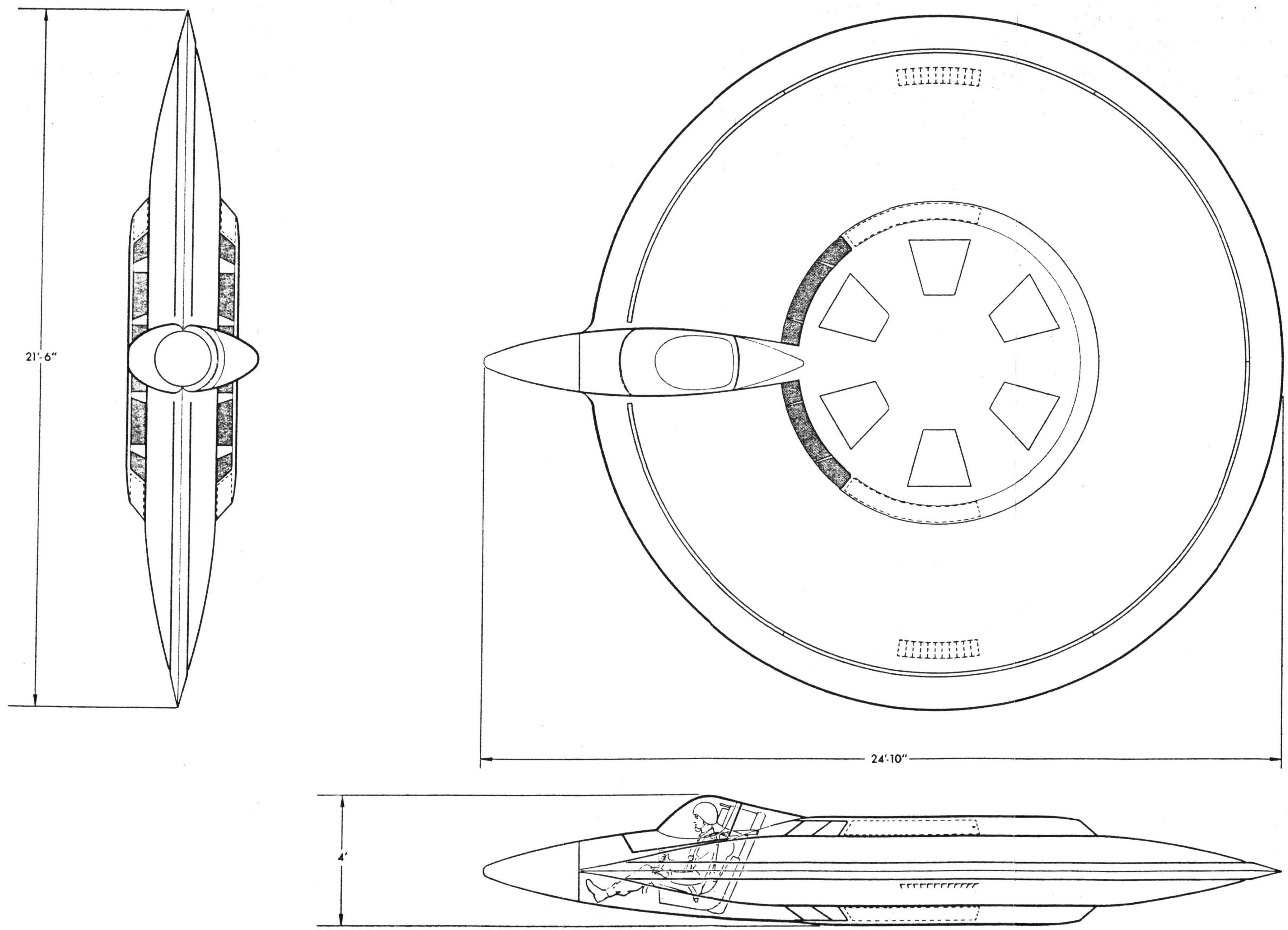


FIG. 18 3 VIEW GENERAL ARRANGEMENT OF TEST AIRCRAFT

Experiment and analysis, will show whether there is a future development of this system that will enable a moderate unstable margin to be tolerated supersonically. It will be appreciated that if this can be demonstrated, then the system will confer all the advantages of gyroscopic stability without the disadvantages of slow rate-of-roll and change of control response.

(3) Test aircraft

The proposed prototype will be a small test aircraft built to resolve the problems of flat vertical take-off and jet control.

It is designed around six Armstrong-Siddeley Viper engines, a well proven engine which has been on the market some years and will no doubt be available immediately; the aircraft is expected to fly in not more than two years from the date ordered.

Fig. 18 is a three-view general assembly drawing of the test aircraft and Fig. 19 is an exploded view. Referring to Fig. 18 it will be seen that the aircraft is circular in plan except for the cockpit. The cockpit has been installed at the front of the aircraft to permit the engines to be placed closer to the centre and to simplify balancing problems. Since this is a relatively low speed test aircraft, aerodynamic heating and vulnerability do not matter.

The engines are radially disposed at regular 60° angles, one facing directly outboard to the left and one to the right. The intake is a double-sided plenum chamber with forward facing doors - the aircraft standing on the closed bottom intake (no undercarriage being required) for take-off, and drawing its air through the ample relieving door area on the top surface straight into the central intake sink, whence it flows radially through the engines and into the exhaustor duct which forms the remainder of the wing. The flow in the exhaustor duct, in the take-off case, is illustrated by Fig. 20. The main duct is a parallel-sided slot, in the centre of the hot outer structure, which cuts across the middle of the turbine annuli of the engines, dividing them into four segments. The horizontal segments, are short and more or less oblong and flow directly into the exhaustor duct: the upper and lower segments, however, are approximately 120° arcs and continue to diffuse from the turbine annulus along a contracting part conical fairing (above and below the exhaustor duct) into which the gas flows through radial slots at the bases of the part conical annuli.

The double annular nozzle at the perimeter of the exhaustor duct is controlled by a shutter system, as described in the accompanying brochure. In the take-off case the top shutters are closed and the bottom annulus is one inch wide all round the periphery, exhausting the flow downwards for jet-lift.

The flow in the duct in forward flight on the other hand, is represented on

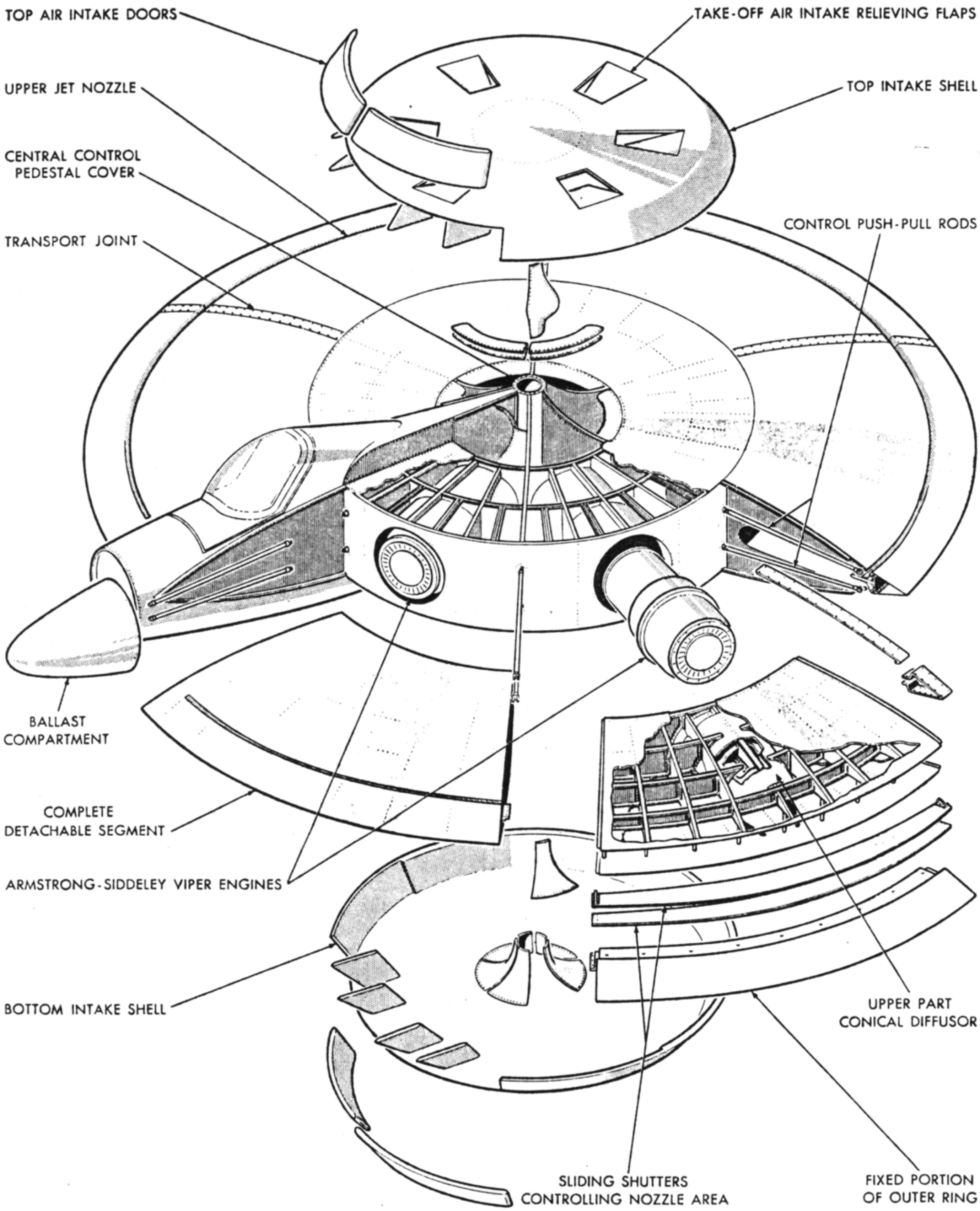


FIG. 19 EXPLODED VIEW OF TEST AIRCRAFT

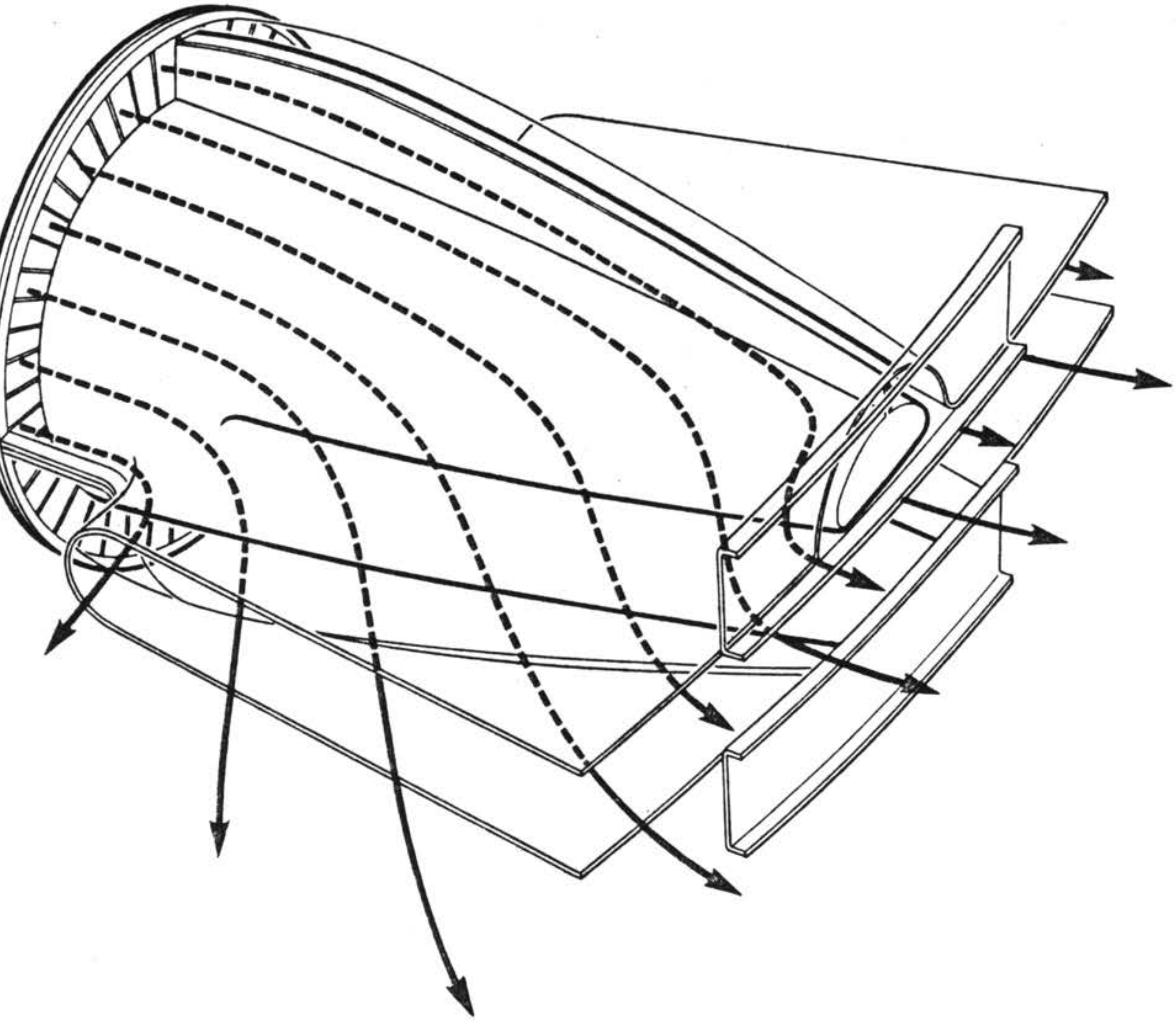


FIG. 20 TAKE-OFF FLOW IN EXHAUSTOR DUCT

Fig. 21. It is similar to the take-off condition except that the horizontal segment will exhaust into a largely circumferential flow and be carried around radii at their edges, whilst the flow from above and below, into the exhaustor, will similarly be carried around radii in a largely circumferential direction, blending to straight radial at the rear.

It is visualized that the exhaust pressure will be contained by the outer skin of the exhaustor duct which has the advantage of being well cooled. The whole of the outer wing will be of steel and its weight is estimated to average about 14 lb./ sq. ft.

The main centre-body structure of the aircraft with its panhandle cockpit, and the piece-of-cake segmental construction of the outer wing, is illustrated in

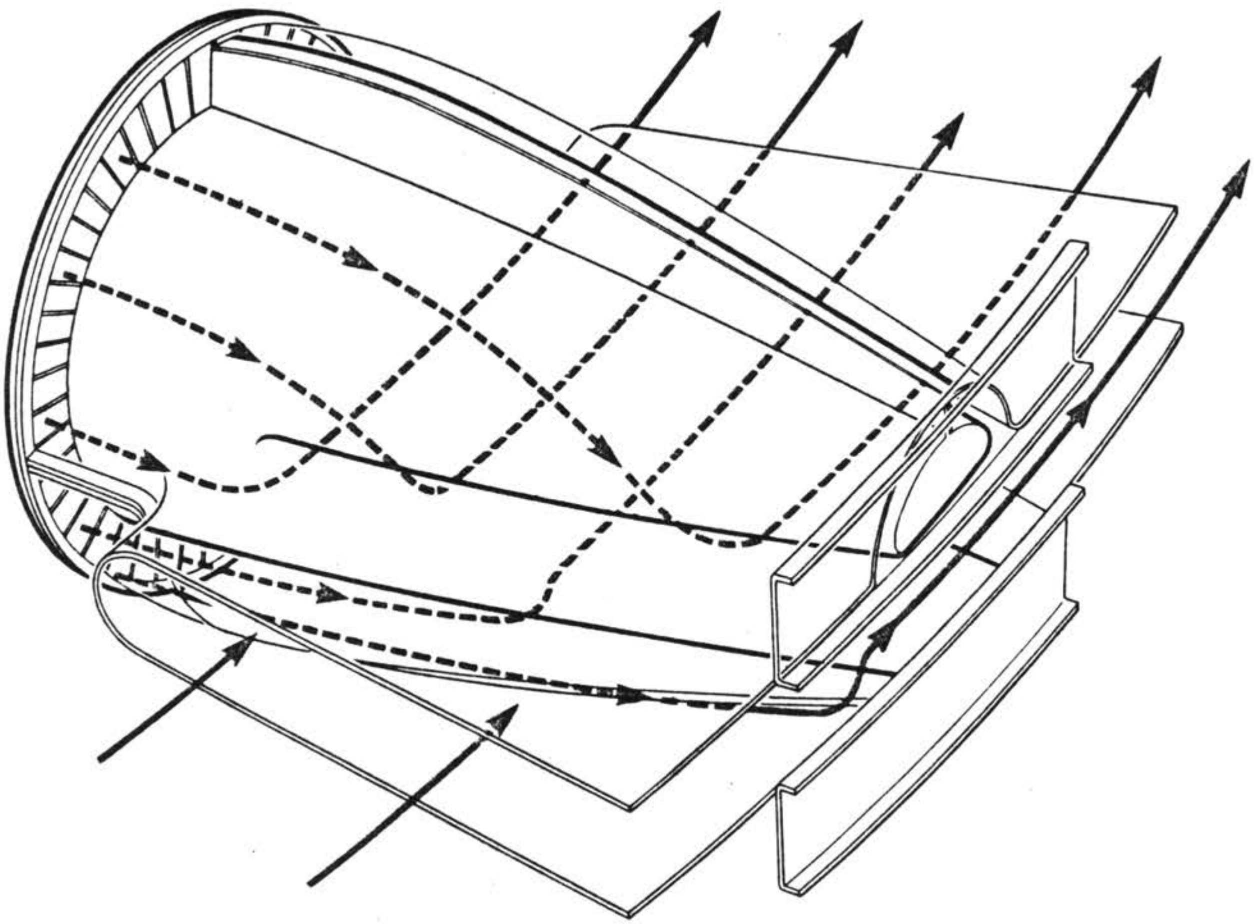


FIG. 21 EXHAUSTOR DUCT FLOW IN FORWARD FLIGHT

the exploded view, Fig. 19, which also gives a good impression of the basic simplicity of the design.

The main centre-body, intake shells and cockpit will be of light alloy construction. The nose of the fuselage forms a ballast compartment to trim the C.G. to acceptable limits.

It has not been possible to estimate whether a fin will be required to maintain directional stability, since the stability contribution of the wing in yaw is not known. If the wing contribution is zero, a small fin of approximately 8 sq. ft. is estimated to overcome the destabilizing fuselage and provide some positive directional stability. It is expected however, that the wing and intake base contribution will be decidedly stabilizing with the C.G. at 25-30% SMC and therefore a fin has not been shown.

Fuel tankage will be integral with the central structure, fuel being carried in the space between the engines in the forward 180° segment of the aircraft; capacity will be approximately 360 U.S. gal. The fuel system will be a simple connection of each engine to a central flow control unit. Fuel may be used indiscriminately as the fuel C.G. will be close to the aircraft C.G. The tankage will be lightly pressurized so that only the H.P. fuel pumps on the engines will be required. Fig. 22 is a plan view showing the disposition of fuel and engines.

Each engine cell consists of a large diameter steel tube fixed in the structure. Installation is arranged so that any engine may be withdrawn radially in a simple operation. It is envisaged that for the test aircraft, the engines will be installed as self-contained units each having their own auxiliaries, i.e. flow control unit, H.P. fuel pump, etc. The standard generator and drive will be removed from the engines and alternators will be fitted on two forward engines only. The engines will be started with compressed air from a ground truck, air being blown on the turbine rotors through the special holes provided for this purpose in the turbine casing.

Cockpit equipment consists of the normal pilot's flying controls, to which are added a trim wheel for setting the aircraft to forward flight, i.e. moving the shutter rings forward, and a hovering control (probably a double push-button on the stick) whose function, is to close the top shutters and simultaneously open those at the bottom, and vice-versa. The rudder pedals are connected to small shutters, controlling backward-facing rudder ports on the under-surface of the wing in the exhaustor duct at each side of the aircraft.

One throttle lever will control all the engines simultaneously, but there will be separate shut-off buttons controlling the solenoid operated shut-off cocks. The latter will be fitted to each engine, for rapid shut-down of a particular engine in the event of trouble.

Instruments will consist of the normal blind flying panel, a six dial rpm indicator, a jet pipe temperature indicator, a fuel contents gauge and a gyro compass. It is envisaged that no fire extinguisher equipment will be carried, but fire warning indication from each engine bay will be provided. The cockpit indicators will be situated close to the fuel shut-off cocks.

In view of the simple objective of the first test aircraft, which will be produced to resolve problems of V.T.O. and jet control, all equipment not strictly necessary will be omitted and the cockpit will not initially be pressurized. No radar, DF or other navigational or landing aids are required, since initial flying will be in VFR and within reach of the airfield. However, standard VHF air-to-ground R/T will be installed and oxygen for up to one hour will be carried.

A simple hydraulic system with at least two hydraulic pumps on the forward

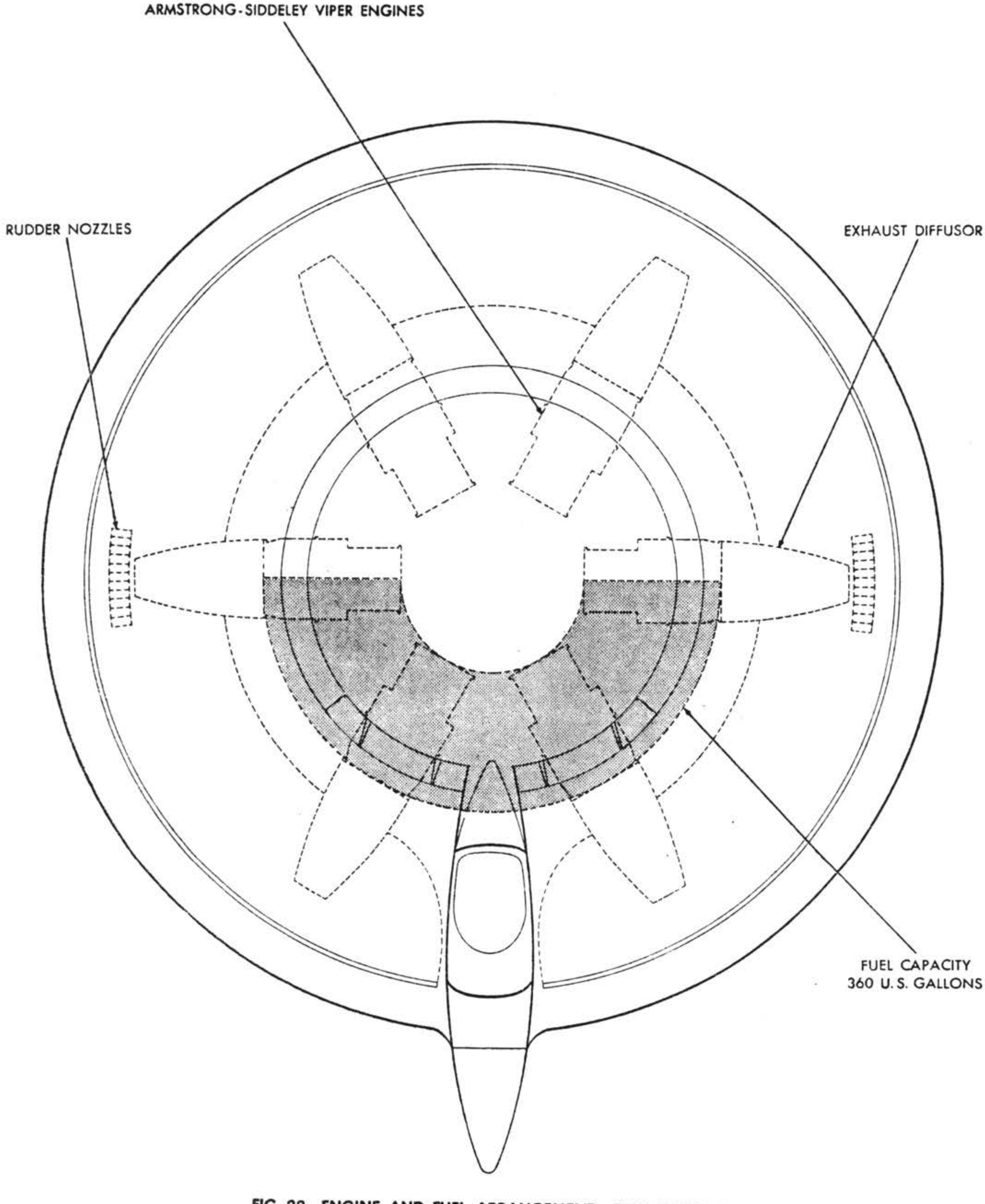


FIG. 22 ENGINE AND FUEL ARRANGEMENT - TEST AIRCRAFT

TABLE 2 - WEIGHT BREAKDOWN

Structure			
Cockpit	400		
Exhaustor duct	2690		
Halo ring	530		
Intake shells	320		
Central structure	1230		
Controls and gyro	290		
		5460	
Power Plant			
Six engines less generator	1870		
Mtg. 15 lb. per engine	90		
Two alternators	60		
Engine controls	40		
		2060	
Extra to Structure			
Radio	55		
Electrics and hydraulics	165		
Fuel and oil system	90		
Instruments	45		
Pilot's seat	30		
Oxygen	30		
		415	
Tare weight			7935
Disposable Load			
Pilot	200		
Fuel (360 U.S. gal.)	2440		
Oil	25		
Ballast	2300		4965
T.O. GROSS WEIGHT			12,900 lbs.

No reliable estimate of aerodynamic centre position can be made for an aircraft with the section and planform proposed in view of the paucity of data on anything approximating to it and also because the trailing edge to wing tip jet flow may well have a significant effect. The aircraft normal gross weight is here quoted at 30% SMC but the ballast compartment is adequate to bring the C.G. well forward of the quarter chord if necessary. The tunnel tests proposed in Phase 1 will enable an assessment to be made.

TABLE 3 - BALANCE DATA

Equ. to SMC $C.G./SMC = \frac{8.45-x}{16.9}$

x is distance in feet from centre of wing (positive forward)

	Wt./lb.	Arm ft.	Moment lb. ft.	CG/SMC
Aircraft tare weight	7935	0.70	5510	0.458
Add pilot	200	7.8	1560	
ballast	2300	12.1	27,800	
Wt. less fuel and oil	10,435	3.34	34,870	0.30
Add fuel and oil	2465	3.34	8230	
Aircraft gross weight	12,900	3.34	43,100	0.30

The aircraft is designed to make use of the thrust recovery principle outlined in Phase 1. The initial exploratory tests so far carried out do not enable a reliable estimate of the efficiency of this process to be made, and more rigorous simple tests are proposed. The speed and ceiling figures quoted in the following assume an efficiency of 90%.

Subsonically, the aircraft is not expected to have low drag, and in particular the contribution of the intake is expected to be high. The test aircraft is intentionally designed in a supersonic configuration and supersonically it is unlikely that any drag benefit would be obtained by fairing the rear of the intake; it is, moreover, undesirable from the structural point of view. Again, subsonically the low aspect ratio will give poor lifting qualities and a maximum lift/drag ratio of only about 6.5 is estimated. The following is a drag breakdown.

TABLE 4 - DRAG

Item	Drag in lb. at 100 ft./sec.
Basic wing friction	26.2
Intake base drag	47.9
Body friction	1.8
Cabin	0.5
	76.4
Interference	11.5
Total zero lift drag	87.9

$$C_D S = 7.38 \text{ sq. ft.}$$
$$C_{D_o} = 0.02$$

Subsonic efficiency factor is estimated at 0.85.

The following is a summary of performance:

TABLE 5 - PERFORMANCE

Performance in ICAN standard conditions with Armstrong-Siddeley Viper ASV. 6 engines. No reheat.	
Hovering height above ground	
Take-off gross weight	10 ft.
Landing weight (10% fuel remaining)	15 ft.
Max. level speed (36,090 ft.)	540 knots (M = 0.94)
Operational ceiling (1000 ft. min. rate of climb)	45,000 ft.
Approx. still air range at 40,000 ft. without allowances	400 naut. miles
Equivalent duration	50 mins.
Duration of flight at 100% max. rpm at SL	11 min.
Duration of flight at 90% max. rpm at SL	14 min.
(10% fuel reserve in each case)	

(4) Development for reheat

Investigations will be made into the possibilities of applying reheat to the test aircraft with a view to making it capable of supersonic flight. This will be considered as a separate development from the initial construction of the test aircraft.

Phase 3 - Design and Development of Single-disc Engine Prototype

A complete description of the single-disc engined aircraft is contained in a separate brochure. It is considered that the decision to proceed with this phase of the program shall be made later and Phase 3 is so scheduled that it is consistent with the technical progress achieved during Phase 1 and 2 of the development program, (see Chart 1 - Timing and Control).

TIMING AND COSTS

The following chart illustrates graphically the timing of the three phases of the proposed test and development program.

It will be noted that the timing of tests and manufacturing corresponds to the phases of the program detailed in the preceding pages. The parallel development of airframe and engine in Phase 1 will also be noted.

The forecast of costs for each phase of the program is shown in detail in the following pages.

FORECAST OF COSTS - PROJECT Y2

Phase 1 - Project Design and Development

AIRFRAME

Scope of Work		Design	Manufacture and Testing	Total
1 Thrust Recovery and Control Tests				
Hours		2460	8000	
Labour and Overhead	$	12,695	43,200	
Material & Direct charges		-	10,000	
Total	$	12,695	53,200	
Admin. Overhead		825	3,450	
Total Cost	$	13,520	56,650	70,170
2 Ground Effect Tests				
Hours		2460	8000	
Labour and Overhead	$	12,695	43,200	
Material & Direct charges		-	10,000	
Total	$	12,695	53,200	
Admin. Overhead		825	3,450	
Total Cost	$	13,520	56,650	70,170
3 Subsonic Tunnel Models & Testing				
Hours		6350	10,000	
Labour and Overhead	$	32,750	54,000	
Material & Direct charges		-	146,000	
Total	$	32,750	200,000	
Admin. Overhead		2,150	13,350	
Total Cost	$	34,900	213,350	248,250

Scope of Work		Design	Manufacture and Testing	Total
4 Supersonic Tunnel Models & Testing				
Hours		6550	10,000	
Labour and Overhead	$	33,800	54,000	
Material & Direct charges		-	100,000	
Total	$	33,800	154,000	
Admin. Overhead		2,200	10,000	
Total Cost	$	36,000	164,000	200,000
5 Segment of Test Aircraft and Testing				
Hours		12,100	40,000	
Labour and Overhead	$	62,500	216,000	
Material & Direct charges		-	85,000	
Total	$	62,500	301,000	
Admin. Overhead		4,100	19,500	
Total Cost	$	66,600	320,500	387,100
Additional Test and Development work not specifically covered above.				150,000
Total Cost Airframe Stage				$ 1,125,690
Fee @ 5%				56,300
Total				$ 1,181,990

ENGINE

Scope of Work		Design	Manufacture and Testing	Total
1 Air Bearing Rig Tests				
Hours		3700	12,000	
Labour and Overhead	$	19,100	64,800	
Material & Direct charges		-	12,000	
Total	$	19,100	76,800	
Admin. Overhead		1,250	4,800	
Total Cost	$	20,350	81,600	101,950

Scope of Work		Design	Manufacture and Testing	Total
2 Combustion Development				
Hours		2000	-	
Labour and Overhead	$	10,320	-	
Material & Direct charges		-	135,000	
Total	$	10,320	135,000	
Admin. Overhead		680	8,800	
Total Cost	$	11,000	143,800	154,800
3 Half Scale Bearing & Engine Tests				
Hours		36,840	150,000	
Labour and Overhead	$	190,100	810,000	
Material & Direct charges		-	400,000	
Total	$	190,100	1,210,000	
Admin. Overhead		12,400	78,650	
Total Cost	$	202,500	1,288,650	1,491,150
4 Gyro Stability Simulator Study				
Hours		8200	26,000	
Labour and Overhead	$	42,300	140,500	
Material & Direct charges		-	75,000	
Total	$	42,300	215,500	
Admin. Overhead		2,750	14,000	
Total Cost	$	45,050	229,500	274,550
Additional Test and Development work not specifically covered above.				250,000
Total Cost Engine Stage				$ 2,272,450
Fee @ 5%				113,600
Total				$ 2,386,050
TOTAL PRICE - PHASE 1				$ 3,568,040

Phase 2 - Development of Multi-engined Test Aircraft

Scope of Work	Design	Manufacture and Testing	Total
1 Cockpit Mock-up			
Hours	1440	5000	
Labour and Overhead	$ 7,450	27,000	
Material & Direct charges	-	5,000	
Total	$ 7,450	32,000	
Admin. Overhead	500	2,100	
Total Cost	$ 7,950	34,100	42,050
2 Artificial Stabilizer			
Hours	8200	26,000	
Labour and Overhead	$ 42,300	140,400	
Material & Direct charges	-	25,000	
Total	$ 42,300	165,400	
Admin. Overhead	2,750	10,750	
Total Cost	$ 45,050	176,150	221,200
3 Test Aircraft			
Hours	48,700	156,000	
Labour and Overhead	$ 251,300	842,400	
Material & Direct charges	-	400,000	
Total	$ 251,300	1,242,400	
Admin. Overhead	16,300	80,750	
Total Cost	$ 267,600	1,323,150	1,590,750
4 Reheat Development for Test Aircraft			
Hours	27,000	79,000	
Labour and Overhead	$ 139,300	426,600	
Material & Direct charges	-	250,000	
Total	$ 139,300	676,600	
Admin. Overhead	9,000	44,000	
Total Cost	$ 148,300	720,600	868,900
Additional Test and Development work not specifically covered above.			400,000
Total Cost			$ 3,122,900

SECRET

Fee @ 5%	156,150
Total Price Phase 2	$ 3,279,050
GRAND TOTAL PHASE 1 and 2	$ 6,847,090

Phase 3 - Design and Development of Single-disc Engined Prototype Aircraft

Until more knowledge has been gained from the development work planned in Phases 1 and 2 it is considered that a cost forecast for Phase 3, cannot be made. However, it is assumed that the costs of developing the full scale aircraft will be less than those of developing a conventional supersonic aircraft, including engines.

Summary

There are certain major factors which must be considered for this program and which may have a bearing on the total development costs. These are as follows:

(1) The half scale engine referred to in Phase 1 covers the development of this engine to the stage where it may be tested in the test cell facilities.

(2) Costs for Phase 2 include only the stages up to and including initial flight. Further development and testing costs would be additional to this figure for Phase 2.

(3) No provision is made for capital facilities such as test cells for engine testing etc.

(4) All costs are based on Canadian Cost Standards.

Expenditure of Costs by Fiscal Years (U.S. Gov't - July 1 - June 30)

1954/55 Expenditures by Quarters					1955/56	1956/57	Total
1	2	3	4	Total			
400,000	550,000	650,000	850,000	2,450,000	3,647,000	750,090	6,847,090

Note: Above cost forecast based on go-ahead by July 1, 1954.

SUPPLEMENTARY COST DATA

Design Costs

These figures cover all phases of the design of the aircraft and engine and include costs of engineers, draftsmen and other personnel directly engaged in this type of work.

Direct labour rates of $2.50 per hour have been applied to all the direct Design man hours. This figure is considered representative of the direct labour rate to be incurred over the duration of the work covered in the proposal.

Overhead rates applied to the direct engineering labour have been forecast at 105%.

An administrative overhead rate of 6.5% has been applied to the total design costs of each category of work. This overhead rate is developed to take care of administrative expenses and other indirect costs not specifically covered in the above.

Manufacturing and Testing Costs

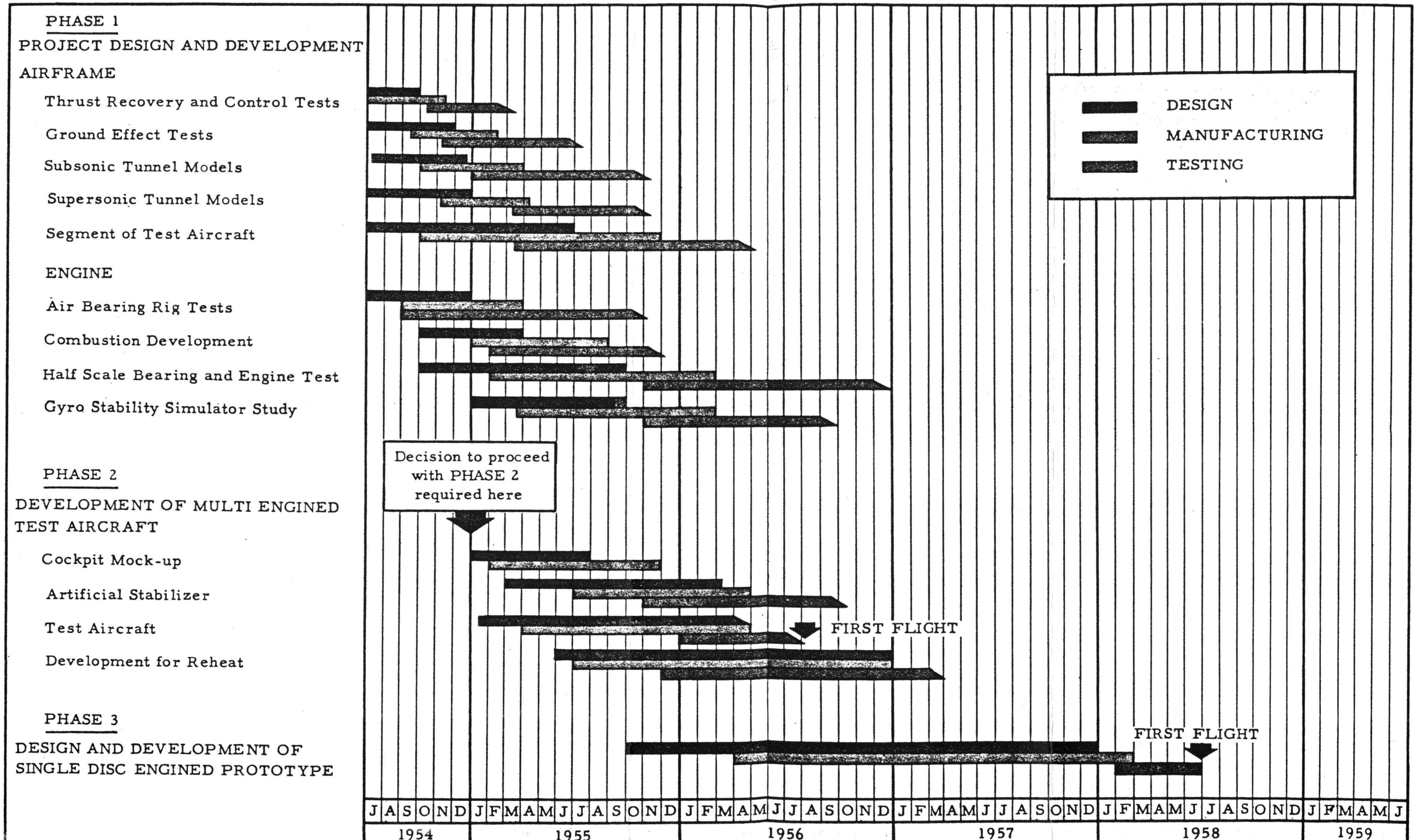
These figures include all other costs not contained in the Design figures. The costs include the manufacture of special tools, test rigs and apparatus, test parts, components and assemblies. The figures also include the testing costs for specimens, parts etc. and certain flight testing.

Wage rates used in developing the costs were estimated at $2.15 per hour and are intended to cover the costs of all trades engaged in this work for the period covered.

Shop overhead rates to be applied to direct labour were forecast at 150%. An administrative rate of 6.5% applied to total direct costs was also used.

Included under manufacturing and testing costs are certain specific direct charges the main items being (1) Combustion System Development and (2) Wind Tunnel Testing. No provision has been made for Government Supplied Materials.

The cost forecast for the above program is based on one shift, 40 hour week basis and all costs contained herein are predicted on Canadian Standards. The application of overhead rates is consistent with the general accounting practice carried out in the company at present.



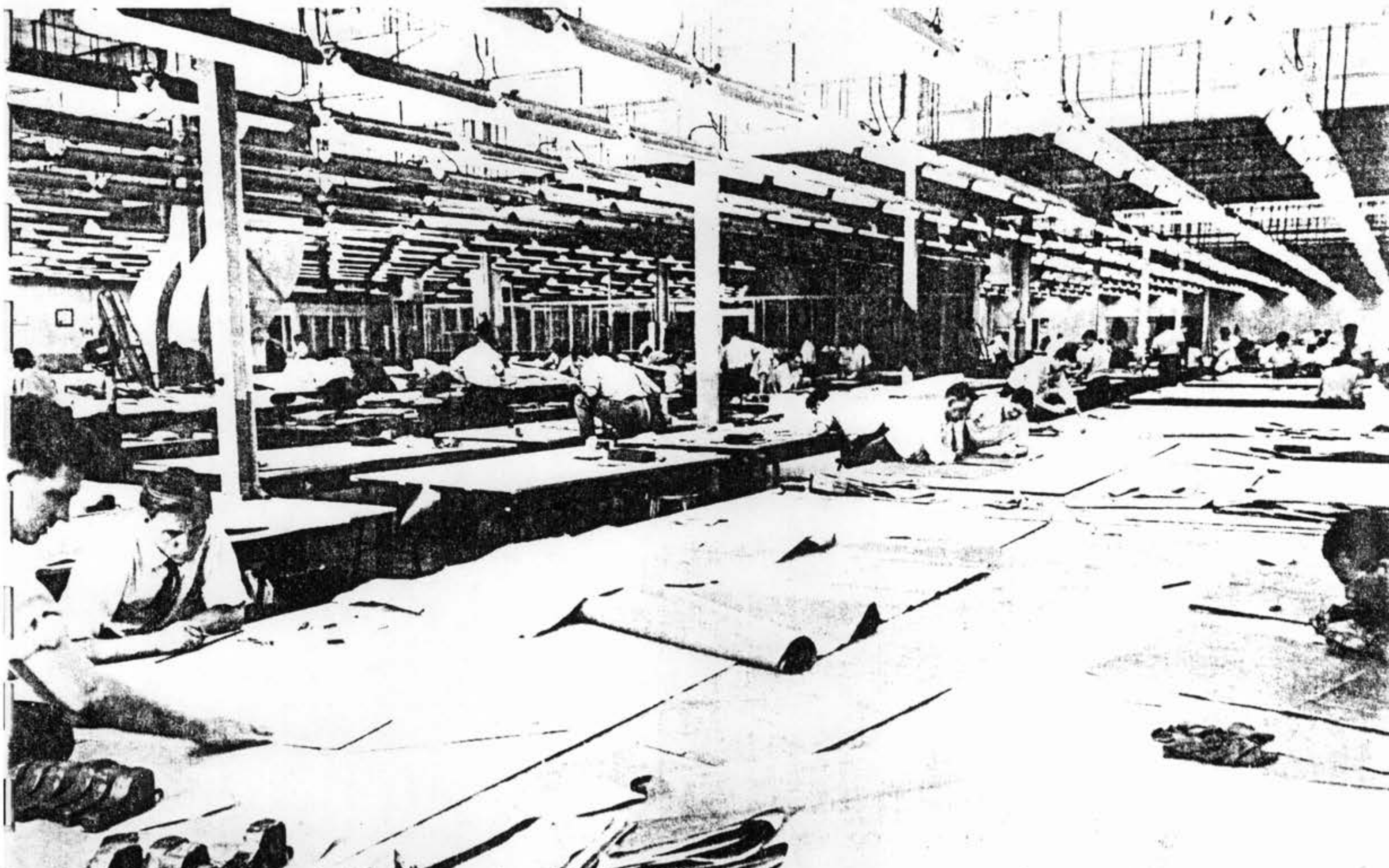
DESIGN AND DEVELOPMENT SCHEDULE

CHART NO. 1

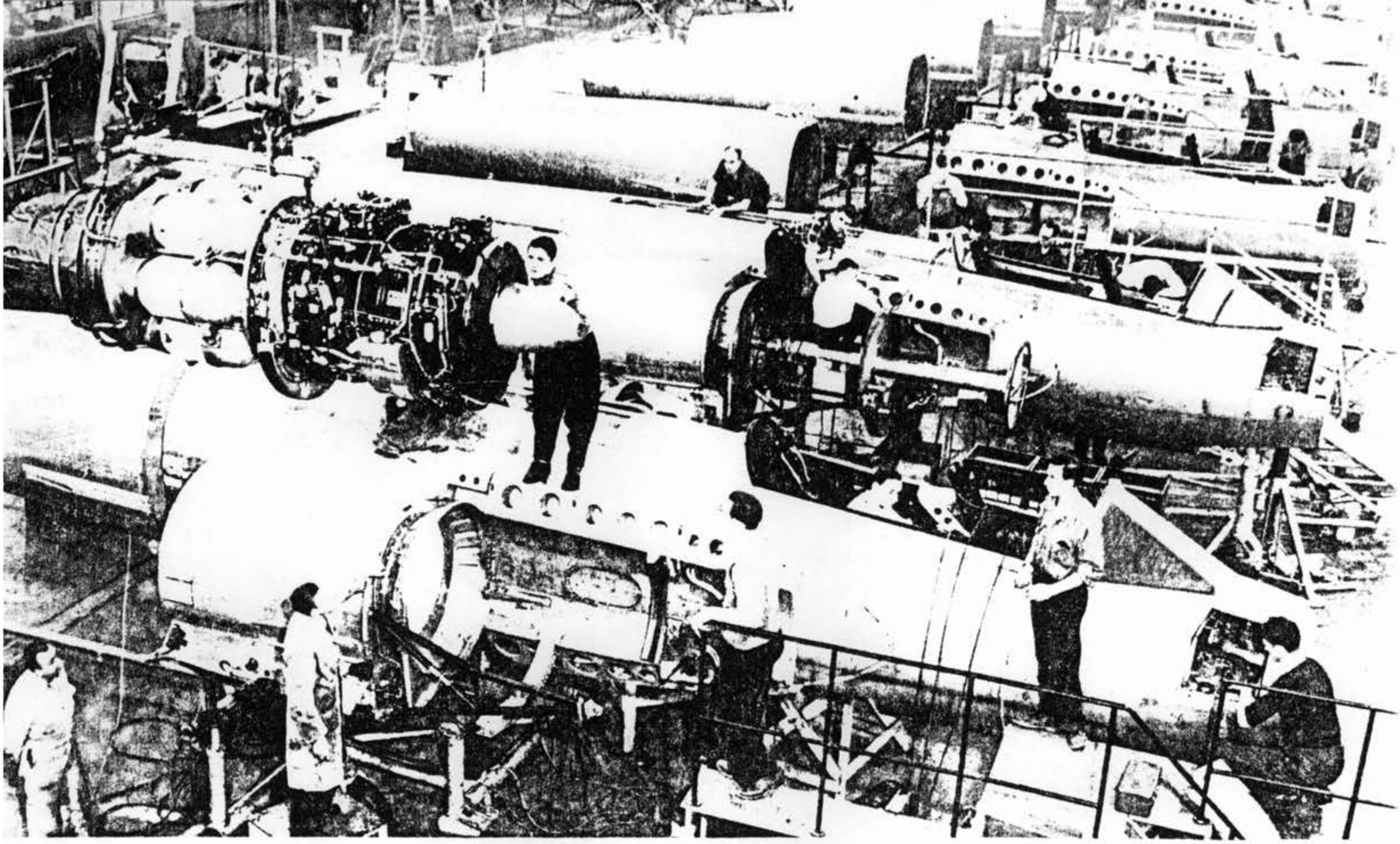
AVRO CANADA
Facilities

MALTON — ONTARIO

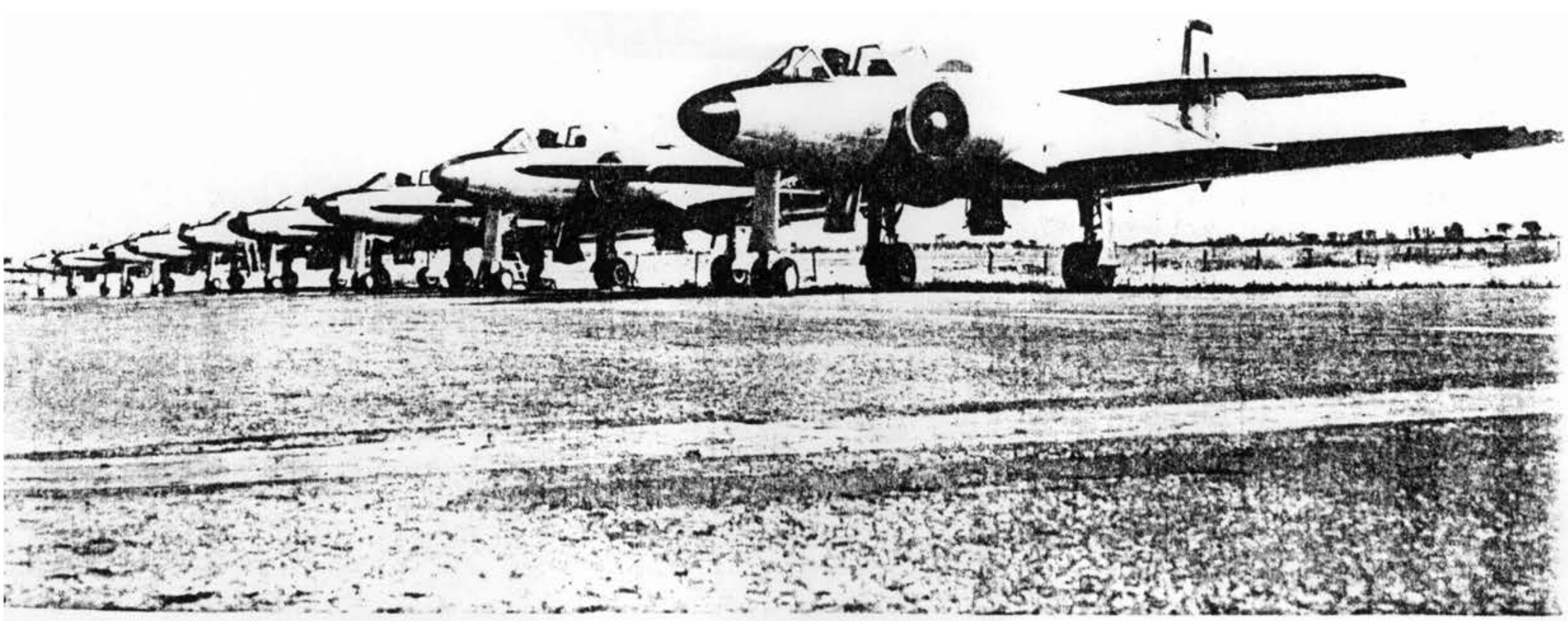
VIEW OF THE AIRCRAFT PLANT, SHOWING THE NEW FLIGHT TEST HANGAR IN THE BACKGROUND



THE AIRCRAFT DESIGN OFFICE

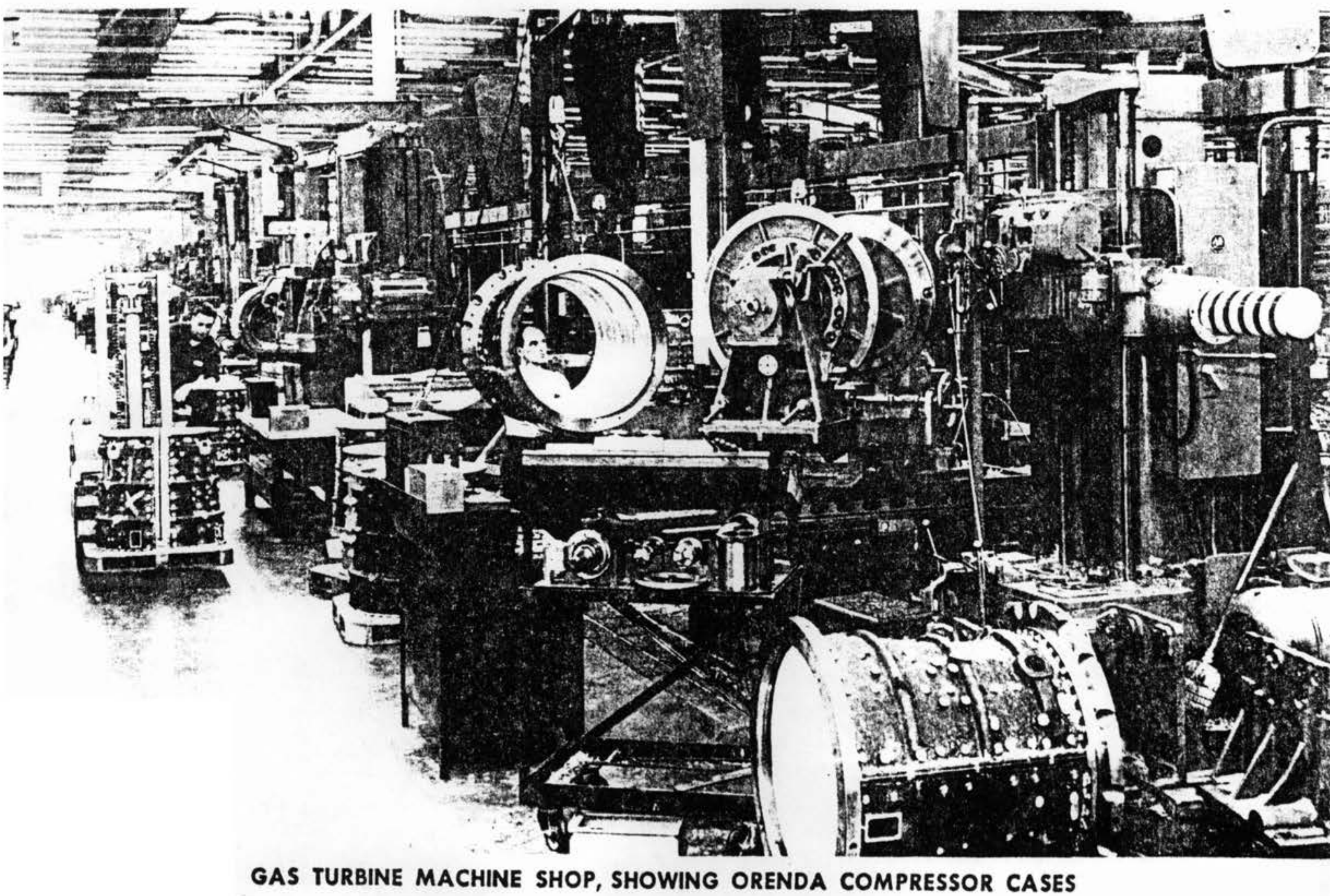


FUSELAGE ASSEMBLY BAY



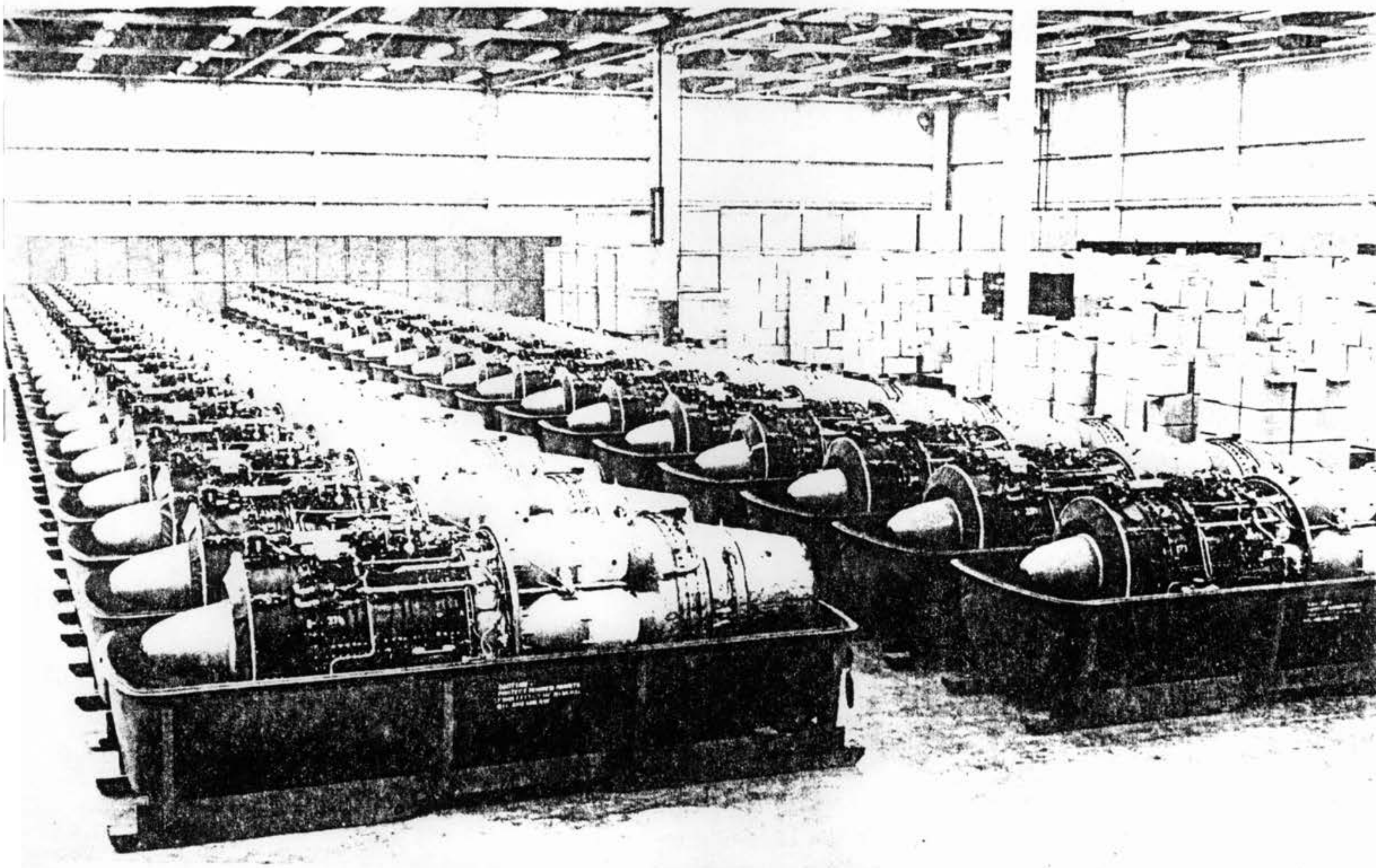
LINE-UP OF CF-100 MK. 3 A'S

VIEW OF THE GAS TURBINE DEVELOPMENT PLANT



GAS TURBINE MACHINE SHOP, SHOWING ORENDA COMPRESSOR CASES

GENERAL VIEW OF THE GAS TURBINE DESIGN OFFICE



ORENDA ENGINES AWAITING SHIPMENT